23rd ABCM International Congress of Mechanical Engineering
December 6-11, 2015, Rio de Janeiro, RJ, Brazil

EXPERIMENTAL INVESTIGATION OF SHOCK WAVE PATTERNS OVER A NACA 0012 PROFILE

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Abstract. *The transonic range is very important part of an aerospace vehicle flight envelope because of the very complex flow phenomena happening in this speed range, as for example shock wave occurrences and associated boundary-layer interactions. The shock waves formation patterns can be very different depending on the flow conditions, mainly regarding the Mach number. The present study is aimed at investigating the behavior of flow in the NACA 0012 profile in Mach number speed 0.6 and 0.8. The techniques used to experimental analyses were pressure taps and pressure sensitive paint (PSP). The experiment was conducted in the Pilot Transonic wind tunnel of the Instituto de Aeronáutica e Espaço (IAE).*

Keywords: *shock-wave, Transonic Wind Tunnel, NACA 0012 profile, transonic flow, PSP*

1. INTRODUCTION

Aerodynamics tests for obtaining the surface pressure distribution on a model surface are very important for structural analysis in complete vehicles models and also in specifics parts of model. Pressure measurements are very important also to study regions where the flow becomes very complex, are for example the transonic range, which is a critical part of the flight envelope of an aerospace vehicle because of the occurrence of complicated flow phenomena as shock wave/boundary layer interactions (Bell *et al*., 2001).

The pressure distribution over the model surface is traditionally obtained through static pressure taps measurements though its space resolution restricted because it is one-off measures.

The usual and most used method for wind tunnel surface pressure measurements on model surfaces is the technique of pressure taps, which consists of installing arrays of small holes on a model surface and connecting them through small flexible tubes to pressure transducers. This technique, relatively simple, provides results highly consistent, but has also some disadvantages. It can be costly in price and time to manufacture and to prepare a model with the hundreds of pressure taps necessary to provide a reasonable resolution, mainly when one is dealing with industrial tests using big models. Another disadvantage is that it is not possible to install pressure taps in very thin areas of the model. An alternative for these deficiencies is the rather recent technique of Pressure Sensitive Paint (PSP).

The PSP technique working principle is well documented in the literature (Liu and Sulivan (2005); Engler et al. (2000); Gregory et al. (2008)) and it is based on an oxygen-quenching process in which excited molecules are deactivated by oxygen, this phenomenon produces different degrees of luminosity on the model surface. The PSP paint is composed of an oxygen-permeable polymer binder containing luminescent oxygen-sensitive molecules. When illuminated with light at an appropriate wavelength, the luminescent molecules become excited electronically to an elevated energy state. These molecules can decay to the ground state through a radioactive process, luminescence, or non-radioactive way through release of heat. In some materials, oxygen can interact with the luminescent molecules such that the change to the ground state can be non-radioactive, and takes place by colliding with an oxygen molecule, a process known as "oxygen quenching" (Gregory, *et al*, 2008).. A PSP system is composed of PSP paint, an illumination source, an image detector, and a long-pass filter. The PSP paint is distributed over the model surface and then illuminated by the excitation source making the PSP to luminescer. The luminescent intensity from the PSP is recorded by the detector and converted to pressure using a previously obtained calibration. The final pressure map is obtained using complex image processing techniques. A detailed description of this technique is given in (Gregory *et al*., 2008).

In fact, PSP is the first global optical technique that is able to give non-contact, quantitative surface pressure visualization for complex aerodynamic flows and provide tremendous information on flow structures that cannot be easily obtained using conventional pressure sensors (Jahanmiri, 2009).

2. DESCRIPTION OF THE EXPERIMENTS

The experiments were conducted in the Transonic Wind Tunnel (TTP) of Institute of Aeronautics and Space (IAE) . TTP has a conventional closed circuit, continuously driven by a main compressor of 830 kW of power and maximum speed of 7600 RPM. Figure 1 shows a diagram of the TTP aerodynamic circuit. Its main components are: 1 to 4 – circuit corners, 5 – main compressor installed in corner 2, 6 – cooler, 7 – stagnation chamber where turbulence control screens are located, 8 – contraction, 9 – first throat, 10 – test section, 11 – re-entry flaps, 12 – second throat, 13 – injection chamber, 14 – high speed diffuser,15 – F.O.D screen (*Foreign Object Damage* – dotted line).

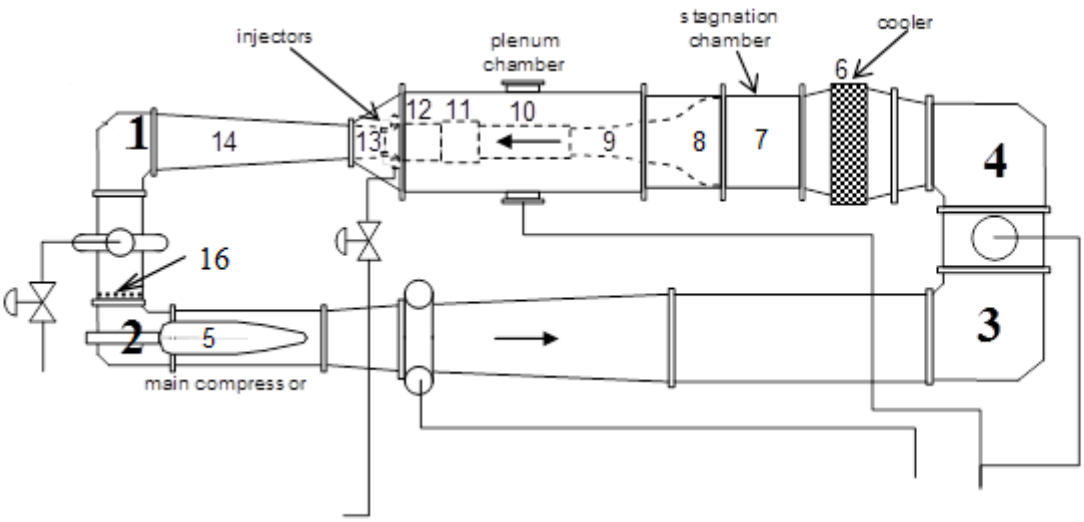
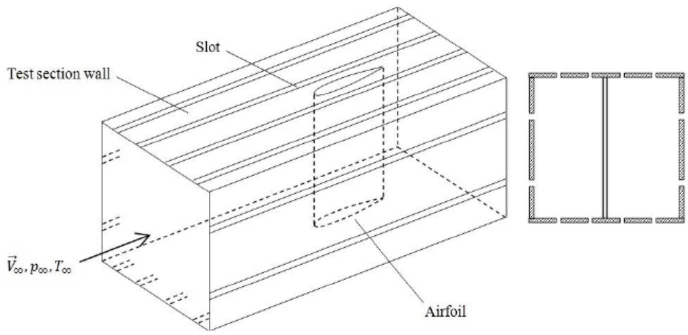


Figure 3. Diagram of the aerodynamic circuit of TTP.

The dimensions of the test section are 0.25 x 0.30 m^2 with a length of 0.8125 m, the longitudinal side walls are adjustable in angles divergent/convergent of 0.5 degrees. Figure 2a shows a sketch of the test section in which it is represented the four slots at the floor and four at the ceiling and to at the each lateral wall (Falcão Filho and Mello, 2002), and Fig.2b shows the airfoil NACA 0012 painted and installed in the test section of TTP . The TTP test section has automatic controls of pressure, from 0.5 bar to 1.25 bar, Mach number, from 0.2 to 1.3, temperature and humidity.



(a)

(b)

Figure 4. (a) Sketch of the test section (Goffert, 2012). (b) Airfoil NACA 0012 painted and installed in the test section of TTP.

In the TTP wind tunnel each experiment has some peculiarities regarding the models installation in the test section.

The NACA 0012 airfoil used is symmetric with 12% of thickness in relation to the aerodynamic chord. It has a chord of 83mm and with 250 mm of span, representing a blockage ratio of 3.32%. Five pressure taps were drilled locally perpendicularly to the surface with 0.5 mm of diameter at 6.0%, 25.3%, 39.8%, 56.3% and 70.5% alongside of the chord direction.

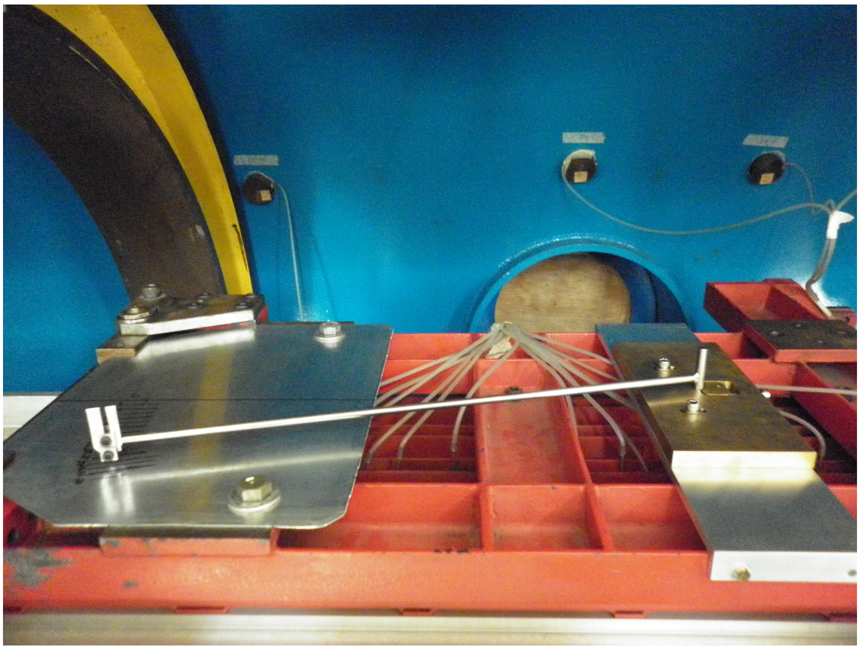


Figure 5. The upper region of the test section.

The profile was installed between the upper and lower wall of the section tests wall. The test section walls have holes of 8mm diameter where the shaft crossing the profile to fixed it. In the extremity they have a square geometry which allows the attachment of the profile without a gap. On upper region of test section an aluminum plate was installed for controlling of the angle of attack, as shown in Fig. 5.

Figure 6 shows schematically the PSP system used in the experiments. The illumination source, a source LED (*Light Emmiting Diode*) LM2X-400© emits radiation with wavelength referring to ultraviolet 400 nm to excitation molecules present in the PSP paint. The images are acquirement by a scientific CCD camera of high resolution among the time intervals determined by *gate* one e *gate* two.

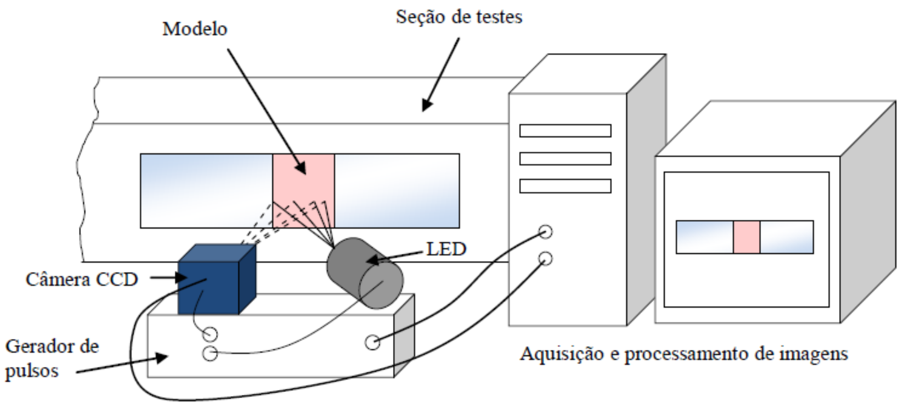


Figure 6. Aparato experimental da técnica PSP (Goffert *et al*., 2012).

The camera is manufactured by company PCO AG (model 1600) and has a resolution of 1600 x 1200 pixels, where each pixel has dimensions 7,4 x 7,4 µm. This type of camera is often used in PSP experiments because their efficiency is considered very high. A Quantum Composer© 9614 pulsed generator was used for the synchronism between the camera, illumination and the computer used for image acquisition. The programs used for data acquisition and were ProImage Application© e OMS Acquire©, usually, both from ISSI

Simultaneously with the PSP measurements, pressure taps were used as well for some punctual pressure measurements over the model. In spite of being simpler, this technique is more precise them PSP and it is used for comparison and also for adjustment of the PSP results, since these last ones are very sensitive to the temperature variation during the wind tunnel tests. For the PSP measurements, a transducer with 32 channels, where the volt signal acquired are convert in values of pressure through a calibration curve, were used. The mechanism to denominate the PSI (Pressure Sensitive Instrumnt) has a differential operation range of ±10 psi (Goffert *et al.*, 2012).

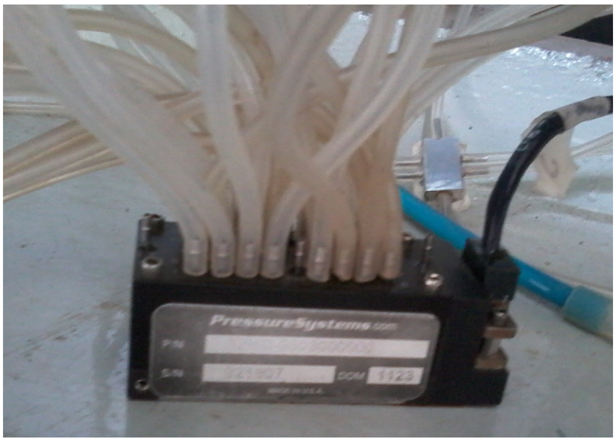


Figure 7. Aparato experimental da técnica PSP

The absolute pressure value at each measurement are obtain by Eq. 1, where the reference pressure was relative the atmosphere.

$$p = p_{amb} - \Delta p \tag{1}$$

where p is absolute pressure, p_{amb} is ambient pressure and Δp is the pressure obtained by PSI.

It was necessary to calibrate the PSI system. The calibration process was conducted based on the work of Souza *et al*. (2010), in which a linear relationship between the signal V and the pressure in Pa was used.

3. PARAMETERS OF INTEREST

Considering the pressure is a parameter that varies according to each situation, the parameter more convenient to compare different situations is the pressure coefficient, defined by (Anderson, 1984) as:

$$Cp \equiv \frac{p - p_\infty}{q_\infty} \tag{2}$$

where p is the local pressure, p_∞ is the undisturbed, or far field, pressure and q_∞ is the undisturbed dynamic pressure, defined by

$$q_\infty = \frac{1}{2}\rho_\infty V_\infty^2 \tag{3}$$

where ρ_∞, V_∞ are the undisturbed values of density and velocity.

The dynamic pressure can be expressed in terms of M_∞ as follows:

$$q_\infty = \frac{\gamma}{2}\rho_\infty M_\infty^2 \tag{4}$$

Substituting Equation (4) into Eq (2), we have:

$$Cp = \frac{2}{\gamma M_\infty^{\ 2}}\left(\frac{p}{p_\infty} - 1\right) \tag{5}$$

which is the parameter presented in the results shown in the next section.

4. RESULTS AND DISCUSSIONS

Table 1 shows the configurations tested in TTP. The data acquisition system gives the read values of the static pressure along the aerodynamic chord of the airfoil NACA 0012 and the results obtained are expressed in terms of the pressure coefficient, *cp*. These results are compared with the experimental work of Harris (1981) due to the high reliability of its results. It is important to note that different Reynolds numbers of experiments conducted in TTP are different from the Reynolds number values reported by Harris (1981). Consequently, some discrepancy must arise, mainly because the viscosity of the flow.

Experimento	Mach number	Reynolds number ($x10^6$)	angle of attack (α)
1	0.5999 ± 0.0006	0.943	1.1.1.1.1.1.1.1.1.1 0^o
2	0.8004 ± 0.0005	1.106	1.1.1.1.1.1.1.1.1.2 0^o

The use of the PSP technique allows observing the results obtained throughout the model surface. Figure 8 shows the result of pressure coefficients, *Cp*, for Mach number 0.6 and zero angle of attack. For this configuration it can be observed that the transition of the static pressure values occurs smoothly, indicating a well behaved flow. In addition, the absence of large three-dimensional effects and the regularity of distribution in the chord direction indicate good quality of flow in wind tunnel test section. The position of the pressure holes are numbered from 1 to 5

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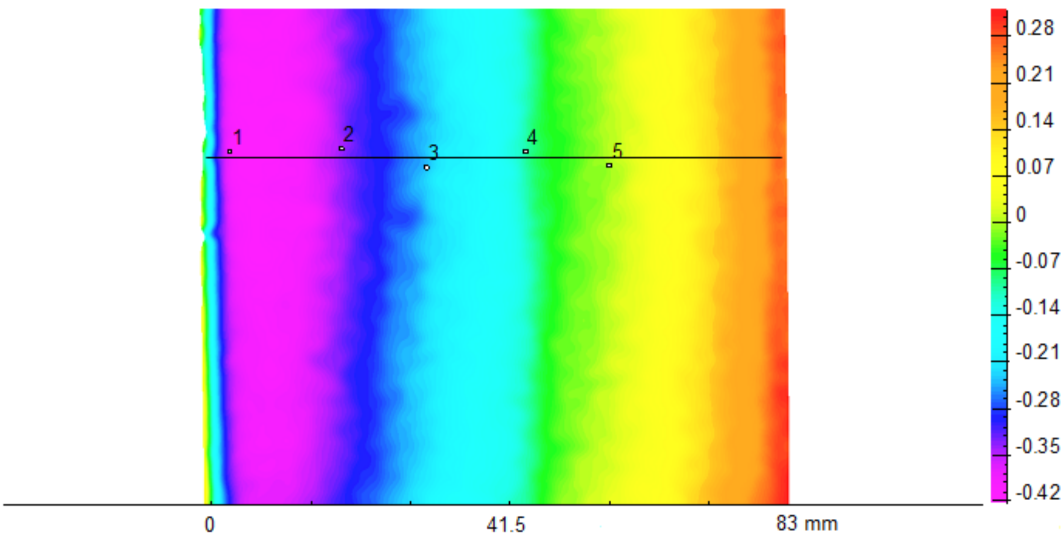

Figure 8. PSP Pressure coefficient distribution over the surface of the airfoil NACA 0012 model for Mach number of 0.6 and zero angle of attack.

In Figure 9 profiles of pressure cofficients for the Mach number value of 0.6 and zero angle of attack obtained in the present study are compared with the results reported by Harris work (1981), represented by hollow circles. The diamonds represents the pressure taps measurements and the dotted line is the profile obtained by the PSP technique corrected by the least squares method from the measurements of the module PSI. The PSP profile was extracted from Fig.8 along the line shown in Fig.8.

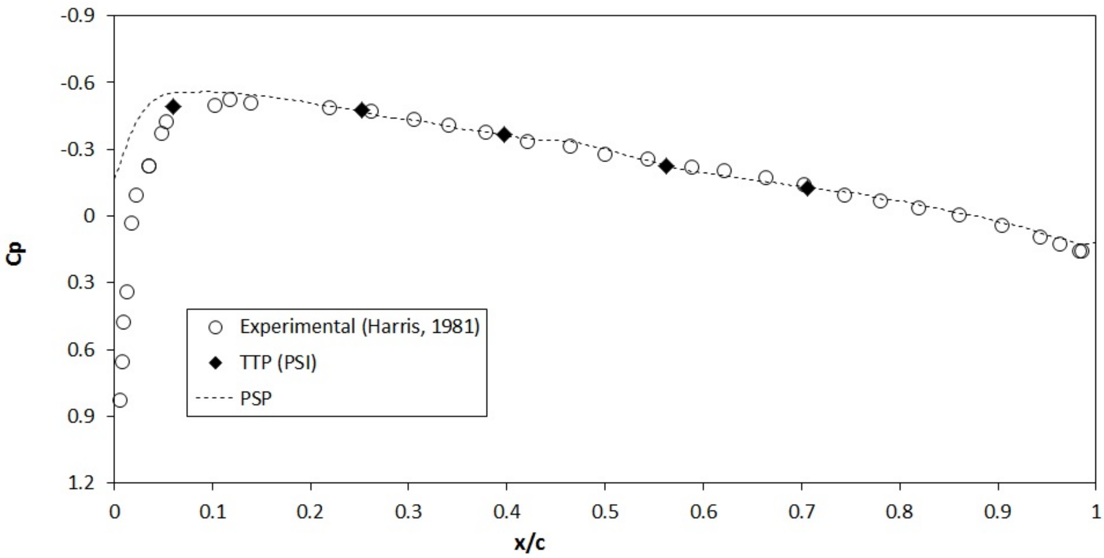

Figure 9. Comparison between the static pressure distribution results on the surface of the airfoil NACA 0012 using the PSP technique, pressure taps and the results from Harris (1981) for the Mach number of 0.6 and zero angle of attack.

It can be noticed from Fig. 9 that the flow on the model surface does not present large pressure gradients, and the results are in good agreement with the results from Harris (1981). The large different between the pressure coefficients appear in leading edge. The reason can be the more accentuated curvature of the model in this region, which makes more difficult the PSP measurements in this position.

Results for Mach number of 0.8 and zero angle of attack are presented in Fig. 10, which is an important configuration because of the appearance of a well-defined shock wave on the surface of the NACA 0012 airfoil in transonic flow.

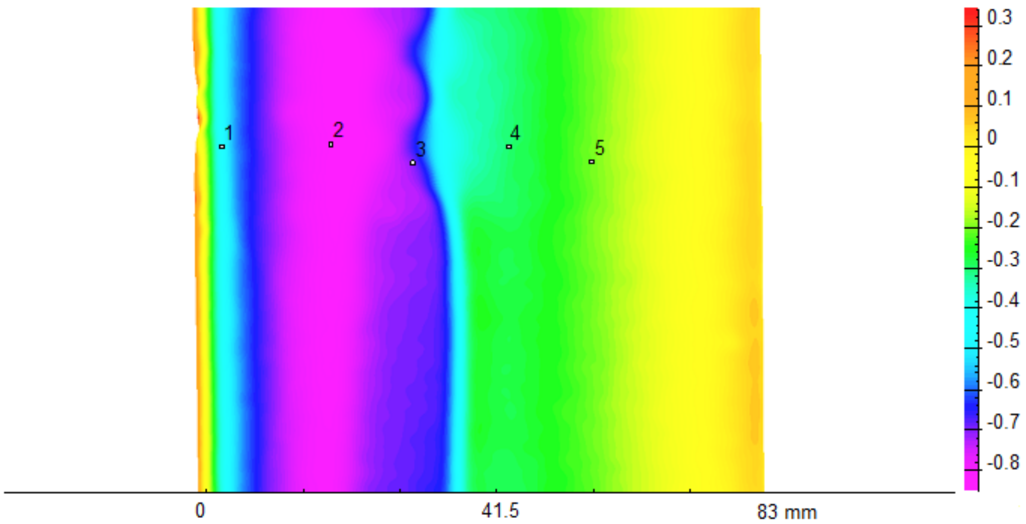

Figure 10. Results of the static pressure on the surface of the airfoil NACA 0012 PSP technique for using the Mach number 0.8 and zero angle of attack.

In the literature we find a comparison as the position of the shock wave to test this configuration. Fig. 10 shows the mapping of static pressure obtained by the PSP technique on the surface of the model.

From Fig. 10 it is possible to observe the shock waves formation on a position close to 40% of the chord length. It is possible to identify the occurrence of three dimensional effects, which may be occurring due to existence of the test section walls.

Figure 11 shows the distribution of the static pressure coefficient along the surface in a single station where it possible to note also the occurrence of mixed shock waves effects (oblique and normal).

The position of the shock wave formation can vary from 35% to 55% (McCroskey, 1987) in relation to the aerodynamic chord for the configuration of Mach number 0.8 and zero angle of attack. According to Fig. 11 the normal shock is located at 42% of aerodynamics chord demonstrating good reliability of the TTP and of the methodologies used in the present analyses.

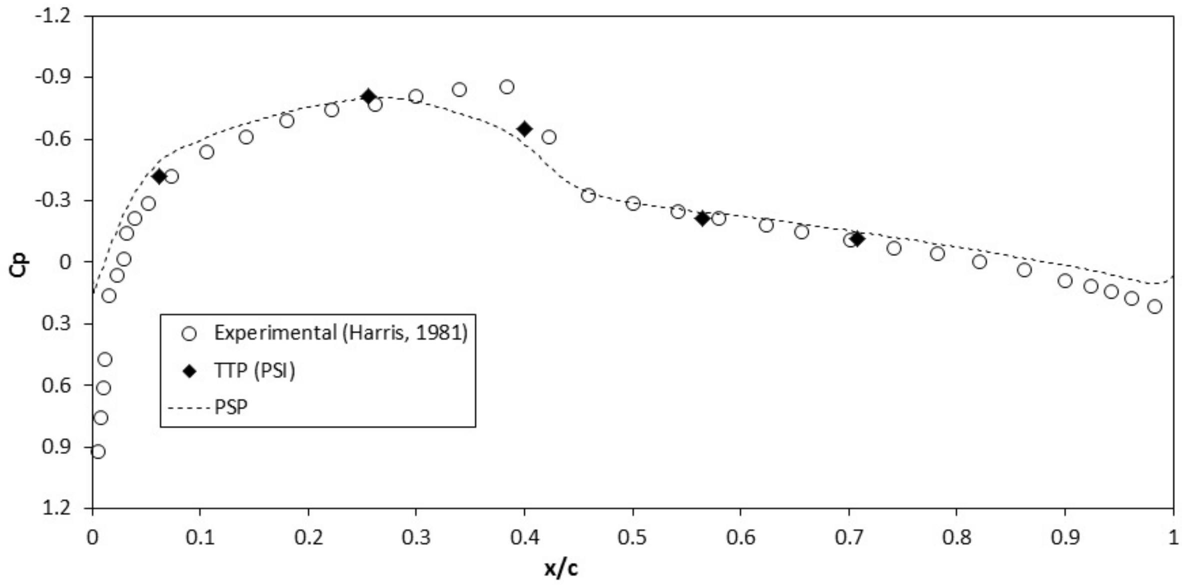

Figure 11. Comparison between the static pressure distribution results on the surface of the airfoil NACA 0012 using the PSP technique, pressure taps and the results from Harris (1981) for the Mach number of 0.8 and zero angle of attack

5. CONCLUSÕES

The experiments carried out with the NACA 0012 airfoil in transonic Pilot Wind Tunnel provided results that point to the reliability of the tunnel and the methodology used in the experiments when compared with results from literature, Harris (1981) and the The best agreement was verified for the subsonic cases, Mach number of 0.6. In the case of transonic condition Mach number of 0.8 and angle of zero attack the shock wave of the position was fully in good agreement with the literature as well.

6. ACKNOWLEDGEMENTS

The authors would like to express their gratitude to CNPq, the Brazilian Counsel of Research and Development for their help during the development of this work through grant **560200/2010-2,** 402233/2013-1 and the individual support number **38449/2013-8.**

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