Pressure coefficient evaluation on the surface of the SONDA III model tested in the TTP Pilot Transonic Wind Tunnel

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Abstract. A test campaign of the Brazilian sounding rocket Sonda III was carried out at the Pilot Transonic Wind Tunnel, TTP. The aim of the campaign was to investigate aerodynamic phenomena taking place at the connection region of the first and second stages. Shock and expansion waves are expected at this location causing high gradients in airflow properties around the vehicle. Pressure taps located on the surface of a Sonda III half model measure local static pressures. Other measured parameters were freestream static and total pressures of the airflow. Estimated parameters were pressure coefficients and Mach numbers. Uncertainties associated with the estimated parameters were calculated by employing the Law of Propagation of Uncertainty and the Monte Carlo method. It was found that both uncertainty evaluation methods resulted in similar values. A Computational Fluid Dynamics simulation code was elaborated to help understand the changes in the flow field properties caused by the disturbances.

1. Introduction
Scale models of aerospace vehicles are tested in wind tunnels in order to predict the performance of their full scale counterparts during actual flight.
This study presents the results of a test campaign of a half-model of the SONDA III sounding vehicle carried out at the Brazilian Pilot Transonic Facility, TTP.
The purpose of the tests is to evaluate changes in the airflow field properties due to the presence of shock and expansion waves in the inter-stage region of the vehicle.
The evaluated parameter is the pressure coefficient ($C_p$). Measured parameters are the total airflow temperature, $T_0$, total and static pressures, $p_0$ and $p_\infty$, and static pressure, $p$, taken at pressure taps located at 22 stations distributed longitudinally along the model surface.
Uncertainties in the measured parameters are evaluated and propagated to the pressure coefficient by using both the Law of Propagation of Uncertainty and the Monte Carlo method. Mach number of the airflow, $M$, and associated uncertainty are also evaluated.
Experimental data are compared to numerical data obtained throughout Computational Fluid Dynamics, CFD.

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1.1.1. The sounding rocket Sonda III

The aerospace vehicle Sonda III (figure 1) is a bi-stage vehicle developed by the Institute of Aeronautics and Space, IAE. It is one of the sounding rocket family named Sonda, which started with Sonda I, first launched in 1965. In 1996, the first stage of Sonda III was evolved to receive European experiments onboard. This new single stage vehicle was known as VS-30. A boosted version, the VSB-30, which contains the S31 booster motor, was developed in 2001. This bi-stage geometric conception has been utilized in many aerospace designs and a better understanding of the aerodynamic phenomena taking place around the vehicle is important for future design initiatives.

![Figure 1: The Sounding Rocket Sonda III.](image)

With this purpose, a half-model of the vehicle was tested in the TTP wind tunnel in order to study the flow field in the inter-stage region, where shock and expansion waves are expected. Shock waves arise on the vehicle surface when the airflow becomes locally supersonic. The flow properties change through the shock wave and undesirable effects which compromise the flight performance may occur. Therefore, an investigation of the behaviour of shock and expansion waves in wind tunnels is important to characterize their interactions with the model. The nominal Mach number values covered in the tests varied from $M = 0.20$ to $1.10$.

1.2. Shock and expansion waves

When a rocket vehicle is in a free flight condition at high velocity, the flow may become locally supersonic over the surface of the vehicle [1]. The freestream Mach number at which $M = 1$ is first achieved on the surface is called the critical Mach number. In supersonic regimes, oblique shock waves occur when supersonic flow encounters a concave corner and it is attached to the corner as long as the concave corner angle is below a defined limit. Above this limit a more complex detached shock wave occurs. In contrast, if the supersonic flow encounters a convex corner, expansion waves occur. Flow field properties change across shock and expansion waves, as shown in figures 2a and 2b.

Tables and diagrams relating the angle of the corner ($\theta$), the angle of the shock ($\beta$), and the Mach number ($M$) can be found in [2].
1.3. The TTP wind tunnel
The Pilot Transonic Wind Tunnel, TTP, is located at the Aerodynamics Division of the Institute of Aeronautics and Space, Brazil (figure 3). The tunnel is continuously driven by an 830 kW main axial compressor and can also be operated intermittently by means of an injection system, which supplies airflow for around 30 seconds. It is a variable-pressure wind tunnel with control capability to independently vary Mach number, stagnation pressure, stagnation temperature and humidity. The test section is 0.25 m high and 0.30 m wide and has longitudinally slotted walls to favor the uniformity of the airflow. At TTP, the Mach number range can vary from 0.20 to 1.3. Some configurations can be changed in order to allow better flow control such as reentry flaps position and the rate of forced mass extraction, determined by the Plenum Evacuation System, PES.

![Figure 3: The TTP aerodynamic facility.](image)
1.4. Description of the Model

Figure 4 shows a 1:8 scale model conception, mounted on an Aluminum platform which will be fixed to the lateral wall of the tunnel test section. In figure 4 the model is configured with one fin perpendicular to the tunnel wall. Threads made in special places allow the model to change the direction in relation to the platform in order to adjust angles of attack of 0°, 5.5°, 8°, 12°, 13° and 15.5°. Between the half-model and the platform special spacers can be installed to create a gap between the tunnel wall and the model. This way the possible boundary layer formed on the tunnel wall can be eliminated and thus better represents the free flight situation. The total length of the model is 0.778 m and the diameter of the first stage is 0.070 m.

![Figure 4: Model conception of Sonda III with scale 1:8 mounted on an Aluminum platform.](image)

The Sonda III model with details of plastic tubes is presented in figure 5. The tubing connects the pressure taps and pressure instrumentation. There are a total of 154 measuring pressure taps located in 22 positions along the model surface. The positions are identified by letters A to V and are distributed in a region encompassing the inter-stage frustum cone (figure 6). The classification is in alphabetic order, departing from the base of the first stage in the direction of the nose of the second stage. Taps A to H belong to the first stage, this latter defining the limit border of the connection between both stages. The connection region has a frustum cone shape, with taps I, J, K and L. The remaining pressure taps, M to V are located in the second stage of the model.

![Figure 5: Sonda III model with detail of the tube connections.](image)
2. Methodology
This study describes results obtained with the half-model installed on the test section lateral wall of the tunnel. The aim of the wind tunnel tests is to determine the pressure distribution on the model surface. This section describes the data reduction of the estimated parameters. Firstly, the mathematical models of the estimated quantities, which are the negative pressure coefficient, $C_p$, and the freestream Mach number of the airflow, $M_\infty$, are presented. Afterwards, the Law of Propagation of Uncertainty is applied to the mathematical models of both parameters, to estimate the associated uncertainties. Information about the Monte Carlo implementation is then given, such as the number of trials and probability distributions of the measured quantities. The section ends with a description of some properties of the Computational Fluid Dynamics code employed in the numerical simulation.

2.1. Pressure coefficient
The pressure distribution on the model surface can be represented in terms of the pressure coefficient defined by [2]:

$$C_p \equiv \frac{p - p_\infty}{q_\infty}$$

where $p$ is the local pressure measured on each pressure tap station of the model, $p_\infty$ is the freestream static pressure and $q_\infty$ is the freestream dynamic pressure. The static pressure sensor used to measure $p_\infty$ is positioned on the upper part of the wall at the beginning of the wind tunnel test section. The freestream dynamic pressure, $q_\infty$, is defined by:

$$q_\infty \equiv \frac{1}{2} \rho_\infty V_\infty^2$$

where $\rho_\infty$ and $V_\infty$ are the freestream density and velocity, respectively.

In this study, the air is considered as a perfect gas. Its density is calculated by:

$$\rho_\infty = \frac{p_\infty}{RT_\infty}$$

$p_\infty$: freestream pressure in pascal;
$R$: perfect gas constant, equal to 287 J/(kg.K) for normal air; and
$T_\infty$: freestream temperature expressed in kelvin.

The airflow Mach number, $M$, is the ratio between velocity, $V$, and the speed of sound, $a$:

$$M = \frac{V}{a}$$
In equation (4), $a$ represents the speed of sound travelling through the air, considered in this study as a perfect gas. Its value is estimated by:

$$a = \sqrt{\gamma RT}$$  \hspace{1cm} (5)

where

$T$: temperature expressed in kelvin; and

$\gamma$: ratio of specific heats, equal to 1.4 for air considered as a perfect gas.

Rearranging equation (1) by using (2), (3), (4) and (5), results in:

$$- C_p = \frac{p_\infty - p}{\frac{1}{2} \rho_\infty M^2_\infty}$$  \hspace{1cm} (6)

In regions of the flow where $p > p_\infty$, the pressure coefficient $-C_p$ will be a negative value.

Considering the flow in the test section as compressible flow, the freestream Mach number, i.e., the Mach number at undisturbed condition (before reaching the model) is given by:

$$M^2_\infty = \frac{2}{\gamma - 1} \left[ \left( \frac{p_0}{p_\infty} \right)^{\gamma/(\gamma-1)} - 1 \right]$$  \hspace{1cm} (7)

where $p_0$ is the freestream total pressure.

As already mentioned, a static pressure tap is located on the upper wall of the test section and connected to a pressure sensor to measure $p_\infty$. The total pressure sensor used to measure $p_0$ is located in the stilling chamber of the circuit.

### 2.2. Law of propagation of uncertainty

According to reference [3], the value of the uncertainty in measurement is the positive square root of equation (8):

$$u^2_c = \sum_{i=1}^{N} \left( \frac{\partial y}{\partial x_i} \right)^2 u^2(x_i)$$  \hspace{1cm} (8)

where $y$ is the output quantity and $x_i$ are the input quantity. Equation (8) is called the Law of Propagation of Uncertainty and its evaluation is known as the GUM uncertainty framework [4].

Applying equation (8) to (6) results in the values of the uncertainty in the pressure coefficient:

$$u^2_{-C_p} = \left( \frac{\partial C_p}{\partial p} \right)^2 u^2_p + \left( \frac{\partial C_p}{\partial p_\infty} \right)^2 u^2_{p_\infty} + \left( \frac{\partial C_p}{\partial M_\infty} \right)^2 u^2_{M_\infty}$$  \hspace{1cm} (9)

which leads to:
In equation (10), the uncertainties in the static pressure, $u_p$, and in the freestream pressure, $u_{p\infty}$, are estimated combining two components: the uncertainty declared in the calibration certificate of pressure instrumentation and the uncertainty evaluated as type A, quantified by calculating the standard deviation of the temporal pressure signal. To evaluate the uncertainty in the freestream Mach number, one applies the Law of Propagation of Uncertainty to equation (7):

$$u_{M_\infty}^2 = \left( \frac{\partial M_\infty}{\partial p_0} \right)^2 u_{p_0}^2 + \left( \frac{\partial M_\infty}{\partial p_{\infty}} \right)^2 u_{p_{\infty}}^2$$

which results in:

$$u_{M_\infty}^2 = \left( \frac{0.714285}{M_\infty} \right)^2 \left[ \frac{1}{p_0^2} \left( \frac{p_{\infty}}{p_0} \right)^{1.428571} \right] u_{p_0}^2 + \left( \frac{p_0}{p_{\infty}} \right)^2 u_{p_{\infty}}^2$$

(12)

2.3. Monte Carlo method

Reference [4] was used to supply Mach number and pressure coefficients values and associated uncertainties by using the Monte Carlo method. The input quantities for the freestream Mach number, $M_\infty$, are the total pressure, $p_0$, and the freestream static pressure, $p_{\infty}$ (equation 7). The input quantities for the pressure coefficient, $-C_p$, at each pressure station on the half SONDA III model, include $M_\infty$ itself and the local static pressure $p$, taken at pressure taps A to V, along with $p_0$ and $p_{\infty}$ of the airflow (equation 6). The mean value and standard deviation of the pressure signals supplied by the instruments during the runs of the wind tunnel tests are computed and this information is used to evaluate the measurement uncertainty by using the Monte Carlo method. The probability distributions for the input quantities are propagated through the measurement models expressed by equations (6) and (7), resulting in the estimated output quantities $-C_p$ and $M_\infty$ and the associated standard uncertainties, $u_{-C_p}$ and $u_{M_\infty}$ (equations 10 and 12). Codes in MatLab® were developed for the implementation of the Monte Carlo method. Gaussian distributions were assigned for all input quantities. The selected number of trials was 50,000.

2.4. Computational Fluid Dynamics code

The numerical code used to solve Reynolds-Average Navier-Stokes equations was the CFD++® solver. The wind tunnel walls were not considered in the simulation therefore the vehicle is in free flight condition. In this study, the model representation is confined to a dihedral angle of 10°, with origin at the symmetric axis of the model fuselage. This is possible because the simulated model is without fins and is positioned at null angle of attack. These conditions lead to axisymmetric flow and it is possible to limit the computational field. Pre-processing was carried out by using ANSYS ICEM® CFD meshing software. The airflow is considered viscous and the turbulence model is the Spalart Allmaras [5]. The resulting $y+$ value is less than 1, adequate for the investigation of boundary layer effects. Meshing is composed of hexahedrons and prisms. The code application for this kind of rocket geometry was validated by using a cone-cylinder model.
3. Results and discussion

The test campaign to analyse the pressure coefficient distribution on the Sonda III half-model surface carried out at the transonic wind tunnel TTP considered the nominal Mach numbers 0.20, 0.30, 0.40, 0.50, 0.60, 0.70, 0.80, 0.85, 0.90, 0.95, 1.00, 1.05 and 1.10. The angle of attack of the model is 0°.

The pressure signals obtained for total pressure, \( p_0 \), and freestream static pressure, \( p_\infty \), as a function of time, are shown in figures 7a and 7b respectively. The presented signals were obtained for nominal Mach number, \( M \), equal to 0.95.

When the desired regime settles down, an interval of the pressure signals is chosen and the average and standard deviation are calculated. The interval between 40 s and 120 s was chosen for \( M = 0.95 \), and the transient of the signal discarded. The transient is caused by the wind tunnel drier system, which controls the humidity of the airflow. The ratio \( p_0/p_\infty \) of average values of total and static pressures is computed. This ratio is included in equation (7) to supply the freestream Mach number value, \( M_\infty \). The same procedure is performed for static pressure values in taps A to V in order to obtain pressure coefficients (equation 6).

![Figure 7: Temporal pressure signals for: (a) total pressure, (b) freestream static pressure.](image)

In Figure 8 one can see the local Mach number at each pressure tap of the model, for all regimes covered by the wind tunnel tests. The origin of the x-axis coincides with the beginning of the frustum cone and is located between taps L and M. For nominal Mach number 0.80 there is a region on the model surface where the local Mach number is above 1, indicating that the critical Mach number is between regimes 0.70 and 0.80.

![Figure 8: Mach number at pressure taps A to V.](image)
Figure 9 and table 1 present the pressure coefficient values for nominal Mach number regime 0.90, obtained at pressure taps positions A to V of the SONDA III model, by using equation (6). The airflow direction is from left to right in the picture. Uncertainties estimated by using the GUM approach (equation 10) are compared to the Monte Carlo results in table 1. No differences were found when comparing both methods. Pressure coefficient data are presented in table 2 for regimes 0.60 and 1.10. Only Monte Carlo results are shown.

![Figure 9: Pressure coefficient and associated uncertainty at pressure tap locations. $M_\infty = 0.90$.](image_url)

| Pressure tap | A  | B  | C  | D  | E  | F  | G  | H  | I  | J  | K  |
|--------------|----|----|----|----|----|----|----|----|----|----|----|
| $-C_p$ GUM   | 0.11 | 0.09 | 0.05 | 0.00 | -0.07 | 1.50 | 8.99 | 0.75 | -1.72 | -2.92 | -3.72 |
| $u_{-C_p}$ GUM | 0.24 | 0.24 | 0.24 | 0.24 | 0.24 | 0.25 | 0.29 | 0.24 | 0.28 | 0.30 | 0.32 |
| $-C_p$ MC    | 0.11 | 0.09 | 0.05 | 0.00 | -0.07 | 1.50 | 8.99 | 0.75 | -1.72 | -2.92 | -3.72 |
| $u_{-C_p}$ MC | 0.24 | 0.24 | 0.24 | 0.24 | 0.24 | 0.25 | 0.29 | 0.24 | 0.28 | 0.30 | 0.32 |

| Pressure tap | L  | M  | N  | O  | P  | Q  | R  | S  | T  | U  | V  |
|--------------|----|----|----|----|----|----|----|----|----|----|----|
| $-C_p$ GUM   | -4.39 | -3.82 | -2.77 | -1.94 | -1.29 | -1.09 | -0.72 | -0.47 | -0.33 | -0.26 | -0.22 |
| $u_{-C_p}$ GUM | 0.34 | 0.32 | 0.31 | 0.27 | 0.26 | 0.26 | 0.26 | 0.25 | 0.25 | 0.25 | 0.25 |
| $-C_p$ MC    | -4.39 | -3.82 | -2.77 | -1.94 | -1.29 | -1.09 | -0.72 | -0.47 | -0.33 | -0.26 | -0.22 |
| $u_{-C_p}$ MC | 0.34 | 0.32 | 0.31 | 0.27 | 0.26 | 0.26 | 0.26 | 0.25 | 0.25 | 0.25 | 0.25 |
Table 2: Comparison of pressure coefficients and associated uncertainties for subsonic and supersonic regimes.

| Pressure tap | A   | B   | C   | D   | E   | F   | G   | H   | I   | J   | K   |
|--------------|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|
| \( -C_p \)  | 0.16| 0.18| 0.21| 0.30| 0.54| 1.19| 2.54| 3.89| -0.05| -1.66| -2.69|
| \( \mu C_p \) | 0.19| 0.19| 0.19| 0.19| 0.18| 0.18| 0.18| 0.19| 0.19| 0.20| 0.21|

Pressure tap | L   | M   | N   | O   | P   | Q   | R   | S   | T   | U   | V   |
|--------------|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|
| \( -C_p \)  | -3.69| -3.27| -2.21| -1.53| -1.03| -0.92| -0.63| -0.45| -0.34| -0.29| -0.25|
| \( \mu C_p \) | 0.23| 0.22| 0.23| 0.20| 0.19| 0.19| 0.19| 0.19| 0.19| 0.19| 0.19|

\( M = 0.60 \)

| Pressure tap | A   | B   | C   | D   | E   | F   | G   | H   | I   | J   | K   |
|--------------|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|
| \( -C_p \)  | 0.00| -0.05| 0.02| -0.62| 3.45| 4.36| 5.45| -1.66| -3.55| -4.45| -4.86|
| \( \mu C_p \) | 0.06| 0.06| 0.05| 0.19| 0.04| 0.04| 0.05| 0.07| 0.08| 0.08| 0.15|

Pressure tap | L   | M   | N   | O   | P   | Q   | R   | S   | T   | U   | V   |
|--------------|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|
| \( -C_p \)  | -4.70| -3.95| -3.22| -2.19| -1.04| -0.35| 0.13| 0.16| 0.18| 0.27| 0.35|
| \( \mu C_p \) | 0.17| 0.13| 0.13| 0.08| 0.09| 0.08| 0.06| 0.05| 0.07| 0.06| 0.06|

\( M = 1.10 \)

A quantitative comparison between Computation Fluid Dynamics results and experimental data is plotted in figure 10 for nominal Mach number 0.90. The numerical and experimental pressure distributions agree to a certain extent, but considerable differences occur at points E and F. The departure from experimental data reveals that the simulation was not adequately capable of capturing details of aerodynamic phenomena occurring in this region. Uncertainty limits are not sufficient to explain the discrepancies between experimental and simulation data. Differences may have been caused due to the fact that the simulation considers the entire model, not the half-model attached to the wind tunnel wall. Therefore, the simulation is for a vehicle traveling in free flight. Half-model computational simulation would be desirable.

Figure 10: CFD and experimental results for freestream Mach number equal to 0.90.
Pressure coefficients obtained for all Mach number regimes are plotted together in figure 11. Uncertainty values were omitted for clarity. Vertical axis represents \(-C_p\). A schematic representation of the model region where the measurements were taken was also drawn at the bottom of the graph, highlighting the initial and the final sections of the frustum cone.

Following the wind direction in figure 11, in the region of pressure taps of the second stage identified by T, U and V, the static pressure is very close to \(p_\infty\), resulting in near zero values of \(-C_p\) (equation 6). This fact reveals that aerodynamic disturbances were not relevant at these points. Nevertheless, one can note a different behavior between subsonic and supersonic airflows, i.e., supersonic Mach number pressure distributions are apart from the subsonic ones. This also occurs after the frustum cone, in positions A, B and C.

Enlarged versions of these regions are shown in figure 12.

Figure 11: Pressure coefficient distribution on model surface. \(M_\infty = 0.20\) to 1.10.

Figure 12: Regions presenting pressure coefficient values near zero: (a) second stage, (b) first stage.

To better visualize the effects of the concave and convex corners in the inter-stage region and the presence of shock and expansion waves, graphics are ordered in increasing values of Mach number in figure 13.
One can see similar pressure distribution patterns for the subsonic ranges in figure 13a for Mach number up to 0.60. As a result of the disturbance caused by the concave corner around point L, the airflow static pressure starts to increase and reaches its maximum value at this point. From point L to I, which are located in the frustum region, the airflow local static pressure continuously decreases. This tendency remains up to pressure tap H, giving the minimum pressure value. From this point on, the pressure starts to recover the freestream condition.

The pattern is modified from figure 13b on. The local pressures after the area expansion change in such a way that the pressure coefficient distribution becomes more complex. From figure 13c on, the regime is supercritical. In figures 13c and 13d, it is possible to observe the successive growth of \(-C_p\) due to the expansion region around the convex corner at point H. The difference \((p_\infty-p)\) becomes more positive for Mach number equal to 0.90 and \(-C_p\) achieves maximum value. The \(-C_p\) did not continue to increase, as shown in figure 13e.

At freestream supersonic flow condition, a shock wave would be expected in the region of the pressure tap L due to the concave corner if the relation \(\theta-\beta-M\) (angle of the corner, angle of the shock and Mach number) was adequate. The local static pressure behind the shock increases in relation to \(p_\infty\), resulting in negative values for \(-C_p\). For Mach numbers 1.05 and 1.10, the airflow presents this
behavior even before reaching the concave region (see figures 12a and 13f). Apparently, the shock wave is detached due to the high angle of the frustum cone, and negative values start to occur earlier, characterizing a local subsonic regime.

Simulation by employing computational fluid dynamics was done for the Mach number regime 0.90. Some properties of the numerical code have already been mentioned in section 2.4. A diagram of the whole computational field defined by a dihedral angle of 10° is shown in figure 14, highlighting the central plan. The model is positioned at the bottom right and the distance is defined from the beginning of the first stage of the model.

The computational grid topology around the vehicle for the central plane is shown in figure 15a. Note the grid clustering where a better anlysis of aerodynamic phenomena is desired, i.e., inter-stage region, nose and next to the surface. A detailed view of the point clustering at the inter-stage part can be seen in figure 15b. The very closely spaced grid in the vicinity of the model surface is necessary in order to accurately capture the boundary layer effects, according to the requirement \( y^+ < 1 \) of the Spalart Allmaras turbulence model [5]. There are 10 circumferentially and uniformly spaced planes (1° between planes). The whole field has 482,760 calculation cells.

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**Figure 14:** Diagram of the airflow field; x-axis unit: meter.

**Figure 15:** a) Computational meshing around the vehicle. b) A zoom view of the inter-stage region.
Figure 16 summarizes the phenomena which occur around the model at an angle of attack equal to $0^\circ$ in Mach number regime 0.90. The airflow parameter shown is static pressure. The freestream pressure is equal to 55,580 Pa. The flow is from left to right. One can observe the static pressure increasing before the frustum cone and the pressure decreasing after this region. After passing through the cone, expansion waves are present followed by a shock wave which is identified by the coalescing of isobaric curves. This shock wave interacts with the boundary layer in the vicinity of the model surface resulting in shock thickening. The boundary layer grows after the passage of the shock wave, because of the adverse pressure gradient. This fact is better seen in the Mach number field shown in figure 17. The supersonic region defined by the line corresponding to Mach number 1.00 starts almost at the end of the frustum cone and has an extension of 0.37 times the diameter of the first stage.

![Figure 16: CFD static pressure in the inter-stage region. $M_\infty = 0.90$](image1)

![Figure 17: Mach number field. $M_\infty = 0.90$.](image2)

Figure 18 is a detailed view of the shock-wave boundary-layer interaction region with velocity vectors. One can see the reduction of velocity profile (region in blue), caused by the shock wave impinging over the boundary layer. As a consequence, the boundary-layer external frontier is enlarged.
4. Conclusions

The paper has presented the results for a test campaign carried out at the Pilot Transonic Wind Tunnel. A half-model of the sounding rocket SONDA III, equipped with pressure taps around the inter-stage region, was tested in the subsonic and transonic regimes. Pressure coefficient distributions around the vehicle were estimated, as well as the associated uncertainties, to supply information about aerodynamic phenomena taking place in the airflow around the vehicle. Uncertainties were evaluated by using the Law of Propagation of Uncertainty and Monte Carlo method. It was found that both methods resulted in similar uncertainty values and these were not significant to indicate the presence of measurement disturbances caused by the aerodynamic circuit.

A Computational Fluid Dynamics numerical code was used to better understand the changes in airflow properties caused by the presence of the model. This paper detailed the simulation results for Mach number equal to 0.90. Experimental and simulation data did not agree completely and causes will be better investigated in future studies.

5. References

[1] Délery J and Dussauge J P 2009 Some physical aspects of shock wave/boundary layer interactions *Shock waves* (Springer-Verlag) 19 pp 453–68. DOI 10.1007/s00193-009-0220-z
[2] Anderson Jr J D 2007 *Fundamentals of Aerodynamics* (Mc Graw Hill) New York p 985
[3] BIPM/JCGM 100:2008 *Evaluation of measurement data* – Guide to the expression of uncertainty in measurement (GUM 1995 with minor corrections) p 134
[4] BIPM/JCGM 101:2008 *Evaluation of measurement data* – Supplement 1 to the Guide to the expression of uncertainty in measurement – Propagation of distributions using a Monte Carlo method p 82
[5] Wilcox D C 1993 *Turbulence Modeling for CFD* La Cañada DCW Industries p 460