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RESULTS OF S5 CH TESTS WITH AN ARL
(Aerospace Research Laboratory) CASCADE

A. Fourmaux

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Technical Report RT 5/3568 EN, March 1984, pp. 1-31.



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SUMMARY

As part of a research agreement between Iowa State University of Science and Technology (ISU) and ONERA, a blade cascade designed and already tested in the United States was manufactured and tested in France.

The ONERA cascade wind tunnel and the mock-up are described. Attention is focused on defining the upstream conditions of the cascade. The main experimental results are then presented, as well as comparisons with tests conducted by Detroit Diesel Allison (DDA) in the United States.

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TABLE OF CONTENTS

Title and Authors	iv
Identification Card	
1. INTRODUCTION	1
2. TEST MEANS	1
2.1. Wind Tunnel	1
2.2. Mock-Up	2
2.3. Measuring Means	3
3. PRELIMINARY TESTS: ADJUSTMENT OF UPSTREAM FLOW	3
3.1 Theoretical analysis of upstream flow	3
3.1.1 Infinite cascade	4
3.1.2 Semi-infinite cascade	4
3.2 Application to the cascade under study	5
3.2.1 Theoretical aspect	5
3.2.2 Experimental aspect	5
4. TEST RESULTS	7
4.1 Overall characteristics derived from wake measurements	7
4.2 Quasi-bidimensional tests	9
4.3 Other tests	9
5. COMPARISON WITH THE AMERICAN RESULTS	10
6. CONCLUSION	10
REFERENCES	12
FIGURES	13

RESULTS OF A S5 CH TESTS WITH AN ARL* CASCADE
A. Fourmaux

1. INTRODUCTION

/6**

The need for experimental data on rectilinear cascades made up of modern airfoils led ISU and ONERA to sign a research agreement on this subject. DFVLR was also associated with this program.

Within the context of this cooperation, ONERA manufactured and tested a cascade designed in the United States and already tested by DDA [1].

The purpose of this report is to analyze this research, after the first ONERA test campaign.

The cascade was put into production during the first quarter of 1983 and was tested at the end of the year. The tests were conducted in excellent conditions.

After a quick description of the wind tunnel and mock-up, we will study the theoretical operation of the cascade to make it easier to interpret the test results, especially in regard to the delicate problem of defining the upstream conditions. The main experimental results are then presented, as well as comparisons between the American results and the French results.

2. TEST MEANS

2.1 Wind Tunnel

The ONERA cascade tests were conducted in the S5 Ch wind tunnel at Chalais-Meudon (Aerodynamic Directorate). A full

*Aerospace Research Laboratory.

**Numbers in the margin indicate pagination in the original text.

description of this installation is provided in reference [2]. Let us mention the main characteristics of the device used for tests on supersonic plane cascades.

-S5 Ch: return wind tunnel:

Generating pressure adjustable by 0.25 at about 1 bar (current value for cascade tests: 0.8 bar). Generating temperature about 300 K.

-Flow upstream from the cascade field.

It is provided by a supersonic nozzle on which the throat section can be slightly varied to obtain a "nozzle" Mach number (M_0).

-Flow downstream from the cascade (downstream periodicity, against pressure:

The periodicity is provided by mobile flaps issuing from the upper and lower airfoils of the cascade. The counter-pressure is adjusted by a sonic throat behind the cascade.

2.2 Mock-Up (figures 1 and 2)

The airfoils are arranged between two circular plates (windows) made of plexiglas, 100 mm apart. They have suction holes for impacts on the lateral boundary layers and thereby permitting an adjustment of the downstream-upstream convergence ratio of the stream layers.* These windows are also equipped with static taps upstream from the cascade (information on cascade field) and downstream (back pressure periodicity control).

*Note: A device with detachable wedges enables the location and size of the suction holes to be selected.

The set of windows + airfoils is mounted on rings which are then installed in the wind tunnel: it is thus possible to adjust the geometric setting of the cascade, which we will define as the angle formed by the horizontal and the cascade front.

The "ONERA ARL" cascade is made up of 7 blades. The scale with respect to the "Detroit Diesel Allison ARL" is 1.2. This gives a chord of 83.41 mm and therefore an aspect ratio of $100/83.41 = 1.199$ (DDA aspect ratio = 1.104).

The blades are numbered from 1 to 7 and the channels from 1 to 6, from bottom to top.

Blade no. 4 is equipped with 15 static pressure taps ($\emptyset = 0.4$ mm) on the suction face, blade no. 5 with 17 static vents on the pressure face.

2.3 Measuring Means

In addition to the static pressure taps on the windows and blades, a pitot tube with 5 holes is provided to determine the flow [3]. This tube is used to measure the wake, and to determine the flow upstream from the cascade.

Figure 3 shows the various reference marks and notations used in the tests.

3. PRELIMINARY TESTS - ADJUSTMENT OF THE UPSTREAM FLOW

3.1 Theoretical Analysis of the Upstream Flow

The definition of the upstream reference conditions (Mach number M_1 , direction β_1) is important for calculating the magnitudes such as the static compression ratio, the deviation and the convergence ratio.

This is why we will quickly mention the theoretical operation /8
of this type of airfoil.

3.1.1 Infinite Cascade (Figures 4 and 5)

Let us assume that $O_{n-1} A_{n-1}$ is the concave upper surface area affecting the upstream flow. Given the geometry of $O_{n-1} A_{n-1}$ and the surrounding flow, we will see a compression beam appear C_{n-1} which focalizes into a shock Ch_{n-1} . This compression passes in front of the leading edge of the upper cascade (upper surface E_n). An expansion (D_n) forms at this leading edge (which we are assuming is pointed).

The beam (D_n) may be divided into two beams (D_n^- and D_n^+), separated by the characteristic Δ_n : the characteristics of D_n^- encounter the shock Ch_{n-1} , those of D_n^+ the shock Ch_n .

Although these compressions and these expansions balance each other, a uniform flow can only appear at upstream infinity, strictly speaking.

However, calculations will show us that the mutual damping of the compressions and expansions may be considered to be obtained after 4 or 5 shock-expansion torques pass - while rising upward. The characteristics then become the Mach lines of the uniform infinite upstream flow (M_1, β_1).

Furthermore, it is obvious that the various choices of the A_{n-1} point determine other torques (M_1, β_1).

3.1.2 Semi-Infinite Cascade

In this case, the cascade field is not infinite. If the flow (M_o, β_o) upstream from the field is not adapted to the cascade, the leading edge of the first blade introduces a shock or expansion ($M_o \rightarrow M_1, \beta \rightarrow \beta_1$), so that the torque (M_1, β_1) satisfies the single incidence principle.

If the upstream flow is adapted to the cascade, the periodicity in the strict sense is reached only at infinity. However, the following points should be noted:

-the influence of a cascade is virtually felt only on 3 or 4 of the upper channels,

-the periodicity in the inter-cascade channels is assured as of the 2nd cascade if the entropy increase due to shocks is disregarded.

3.2 Application to the Cascade Under Study

/9

3.2.1 Theoretical Aspect

In the calculations, the uniform upstream flow is obtained very near the cascade front (figure 6). The shocks spread out due to the limit of the calculation method (meshing, shock capture on several meshes).

A few calculations determine the single incidence β_1 as a function of M_1 (figure 7).

3.2.2 Experimental Aspect*

We will try to minimize the perturbation (shock or expansion issuing from the leading edge of the first blade, or in other words to geometrically set the cascade with respect to the nozzle so that the cascade is adapted to the flow provided by the nozzle.

An exploration with the 5-hole tube upstream from the cascade gives the variation of the flow magnitudes along a straight line parallel to the nozzle axis and passing through the 4th channel (figures 3 and 8).

*This paragraph was written with the assistance of the S5 Ch wind tunnel team.

From downstream to upstream, we find according to the theoretical scheme (figures 4 and 5):

- compression C_4 and shock Ch_4 , then relaxation D_4 ,
- the expansion: shocks Ch_3 , D_3 , Ch_2 , D_2 , Ch_1 .

Upstream from the shock Ch_1 , we find:

-at the abscissa - about 12 mm, a perturbation issuing from the lateral wall: in effect, this perturbation disappears when one does the same exploration for another position along the span. Further, this perturbation does not change angle β , but angle γ (β and γ = angles defined in figure 3).

-at the abscissa - about 21 mm, a perturbation probably issuing from the leading edge of the first blade.

It is important to note that the impact of these two /10 perturbations on the impact pressure is totally insignificant.

Upstream from the abscissa - 21mm, one finds the values $M_0 = 1.615$, $\beta_0 = 58.2^\circ$ which correspond to the undisturbed flow issuing from the nozzle.

It is this pair of values which was initially retained to define the aerodynamics at the cascade entrance ($M_1 = M_0$, $\beta_1 = \beta_0$).

Nevertheless, the three following remarks allow us to conclude that one should adopt a value higher than 1.615 for M_1 :

-the expansion observed upstream from Ch_1 , greater than the theoretical expansion D_1 ,

-the slight gradient of Mach due to the divergence of the nozzle,

-the flow exactly upstream from the 4th channel, the pressure level of which is lower than that given by the calculation

at Mach 1.615.

This is why we finally retained the values $M_1 = 1.66$, $\beta_1 = 57.5^\circ$ to calculate the static compression ratios, the convergence ratios and deviations.

This point is slightly above the curve defining the single theoretical incidence (figure 7).

Two reasons explain this trend:

1) the blade thickening, due to the development of the boundary layer. This phenomenon may increase the single theoretical incidence by a few tens of degrees.

2) the thickness of the leading edge (here 0.2 mm) which causes a detached shock thereby increasing the angle β upstream with respect to the case of the pointed dihedral.

4. TEST RESULTS

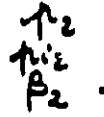
The location and size of the suction holes, the value of the suction pressure and the adjustment of the counter-pressure made it possible to obtain highly diverse operating points, from the energized state to the de-energized limit.

4.1 Overall Characteristics Derived From the Wake Measurement

The wake measurements are processed and analyzed using a method calculating the mean downstream flow having the same flow rate and same dynalpy (axial and tangential components) as the real flow.

We may deduce:

- the downstream static pressure
- the downstream impact pressure
- the direction of the downstream flow



The upstream flow is defined by:

$$M_1 = 1.66, \quad \beta_1 = 57.5^\circ, \quad p_{i2} = p_{i0} \quad (p_{i0} :$$

generating pressure given by the wind tunnel, retained as reference).

The following relationships give the overall characteristics:

$$\begin{aligned} \pi_s &= p_2/p_1 && \text{(definition)} \\ \eta &= p_{i2}/p_{i0} && \text{(definition)} \\ \Delta\beta &= \beta_2 - \beta_1 && \text{(definition)} \end{aligned}$$

$$\tau_c = \frac{\cos\beta_1}{\cos\beta_2} \cdot \frac{\sum(M_1)}{\sum(M_2)} \cdot \frac{1}{\eta} \quad \left(= \frac{1}{A.V.D.R.} \right) \quad \text{(conservation of flow rate)}$$

with:

$$\omega(M) = \left(\frac{p}{p_i} \right)_{(M)} = \left(1 + \frac{\gamma-1}{2} M^2 \right)^{-\frac{\gamma}{\gamma-1}}$$

$$\sum(M) = \left(\frac{A}{A_c} \right)_{(M)} = \left(\frac{2}{\gamma+1} \right)^{\frac{\gamma+1}{2(\gamma-1)}} \cdot \frac{1}{M} \cdot \left(1 + \frac{\gamma-1}{2} M^2 \right)^{\frac{\gamma+1}{2(\gamma-1)}}$$

The table below gives the characteristics of the next test presented:

Test No.	π_s	η	$\Delta\beta$ (°)	τ_c	M_2
2623	2,67	0,882	4,9	1,017	0,809
2629	2,51	0,888	2,7	0,925	0,872
2624	2,35	0,877	2,8	0,929	0,923
2627	1,95	0,893	3,0	0,920	1,098
2634	1,67	0,903	2,5	0,924	1,226
2635	2,23	0,887	1,6	0,884	0,980
2630	2,00	0,903	2,1	0,884	1,086

4.2 Quasi-Bidimensional Tests

/12

Figures 9 to 12 show the results obtained for five counter-pressures. The convergence ratios of the tests are comparable.

Figures 9 and 10: static pressure distribution on the blade.

There seem to be two appreciably different states:

-Low pressure ratios: shock attached to trailing edge (impact not visible on suction face.

-High pressure ratios: shock in inter-blade channel.

This is established by a quick reconstruction of the flow. This is done by using the inviscid flow calculation method in inverse mode. In this type of calculation, one looks for the channel shape which would give in an inviscid flow the same pressure distributions as those obtained experimentally (figure 11).

Figure 12: wakes.

The wakes are measured in a plane 20 mm from the downstream cascade front.

4.3 Other Tests

Two other cases are also presented (figures 13 and 14). They correspond to a lower convergence ratio ($\tau_c = 0,88$). The most de-energized of the two gives a straight shock in the channel (suction face affected). The least de-energized gives a shock attached to the trailing edge (suction face not affected), as for the least de-energized states seen previously.

5. COMPARISON WITH THE AMERICAN RESULTS

It is not easy to find tests performed by ONERA and DDA in totally similar conditions.

However, two tests may be compared, they correspond to the following conditions (figures 15 and 16).

- | | | | |
|---------------|----------------|---------------------------|------------------|
| 1) DDA test | $M_1 = 1.616,$ | $\frac{p_2}{p_1} = 1.87,$ | $\tau_c = 0.970$ |
| 2) ONERA test | $M_1 = 1.66,$ | $\frac{p_2}{p_1} = 1.95,$ | $\tau_c = 0.920$ |

The agreement is satisfactory given the differences between these two operating points:

-Suction face: the ONERA tests, performed at a slightly higher Mach give an out of pitch pressure distribution,

-Pressure face: the shock seems more accentuated for the ONERA tests, more spread out for the DDA tests.

Furthermore, various DDA configurations are shown for the cases where the convergence ratios are comparable

6. CONCLUSION

The ARL-ONERA cascade, subject of a research agreement between ISU and ONERA (research in which DFVLR is also associated), was manufactured and tested in 1983.

The tests were carried out properly and the cascade was characterized for a large number of operating points.

The upstream Mach number, calculated to be 1.616, was actually somewhat higher (1.66).

The lateral boundary layer suction of the wind tunnel not only led to flow configurations close to de-choking (high back pressure), but also to vary the downstream/upstream convergence ratio of the stream layers.

Given the good results obtained, it is interesting to pursue and complete the initial program, for example, for other upstream conditions and by more detailed analyses of wakes for the analysis of losses.

REFERENCES

1. Fleeter, S.; Holtman, R.L.; R. B. Sinnet, G.T; "Experimental Investigation of a Supersonic Compressor Cascade", U.S. Air Force Systems Command. Aerospace Research Laboratories. ARL TR-75-0208, 1975.
2. "Etat actuel de la soufflerie de grille d'aubes de l'ONERA" ("Current Status of the Cascade Wind Tunnel at ONERA") (RT ONERA now in preparation).
3. Gaillard, R., "Etalonnage et Utilisation de Sondes "Cinq Trous" ONERA", ("Calibration and Utilization of "Five-Hole" ONERA Probes", Paper presented at Seventh Symposium on Measuring Techniques for Transsonic and Supersonic Flow in Cascades and Turbomachines, Aache, September, 1983.

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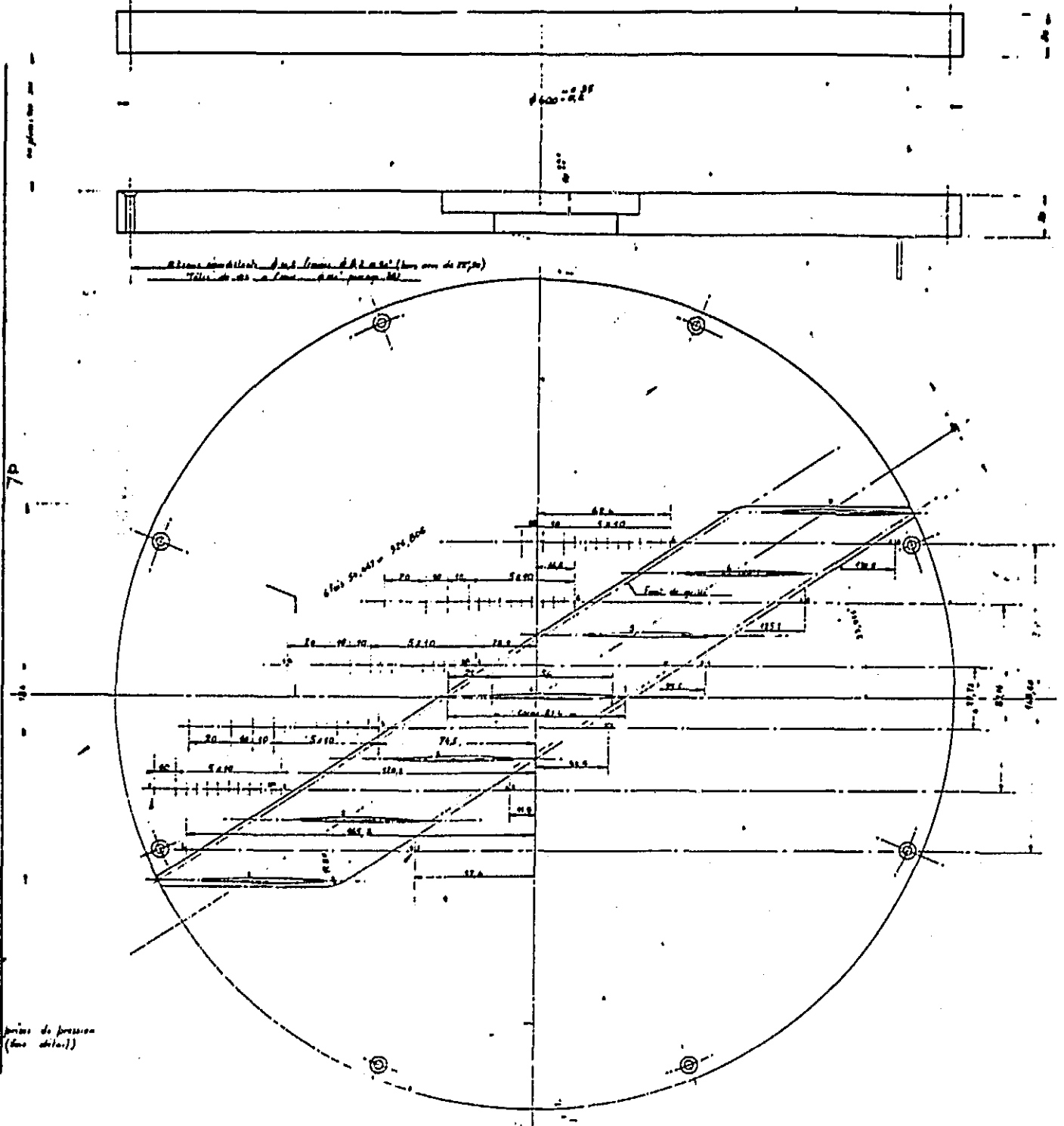


Figure 1 - View of cascade. Location of static pressure taps (window).

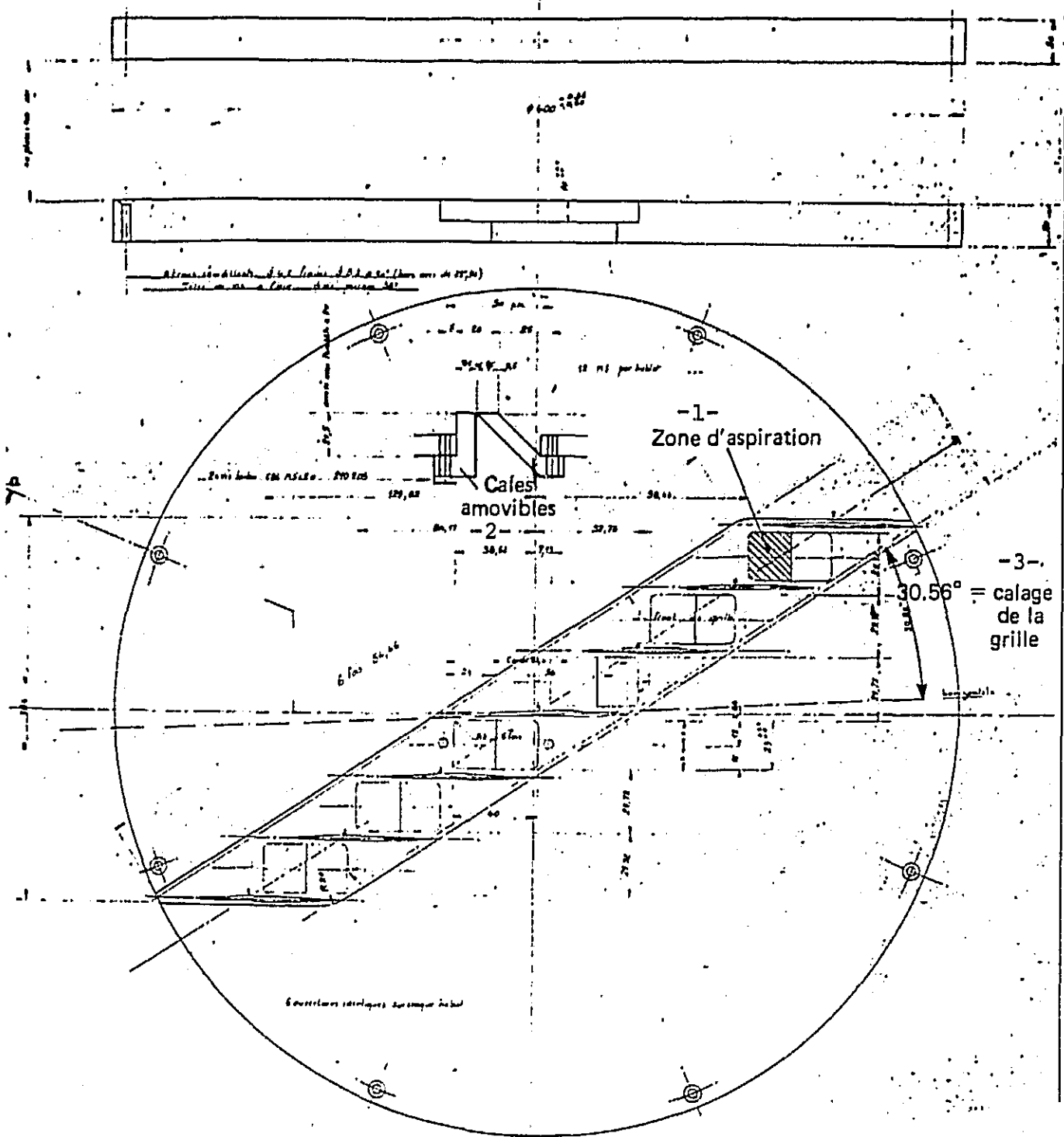
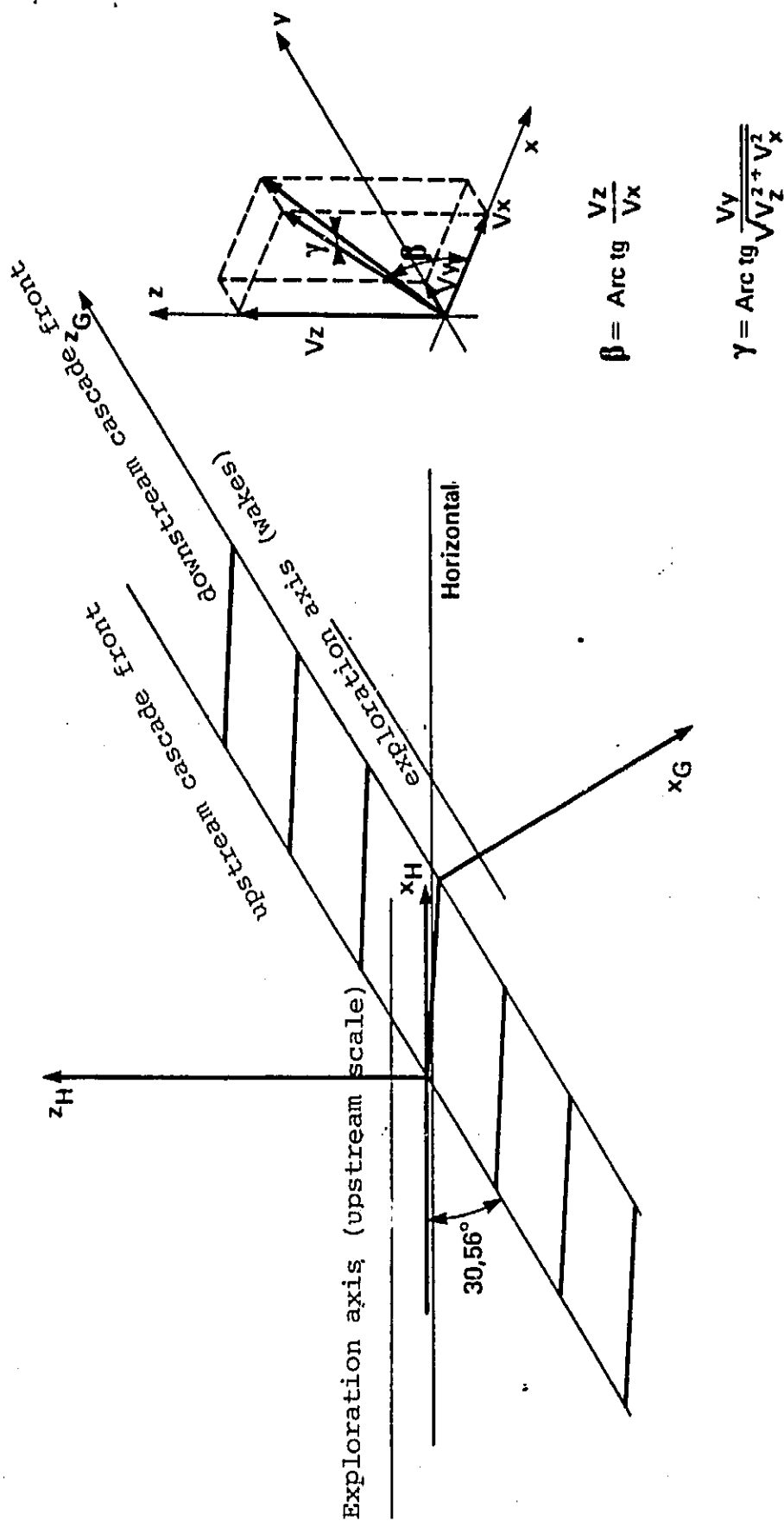


Figure 2 - View of A.R.L. cascade. - Cascade setting;
-Suction holes.

Key: 1-Detachable wedges; 2-Suction area; 3-Cascade setting.



$$\beta = \text{Arc tg } \frac{V_z}{V_x}$$

$$\gamma = \text{Arc tg } \frac{V_y}{\sqrt{V_z^2 + V_x^2}}$$

γ = angle of velocity vector with its projection to plane (x,z).

β = projection of angle of velocity vector to plane (x,z) with x-axis.

Figure 3 - Identification marks and notations

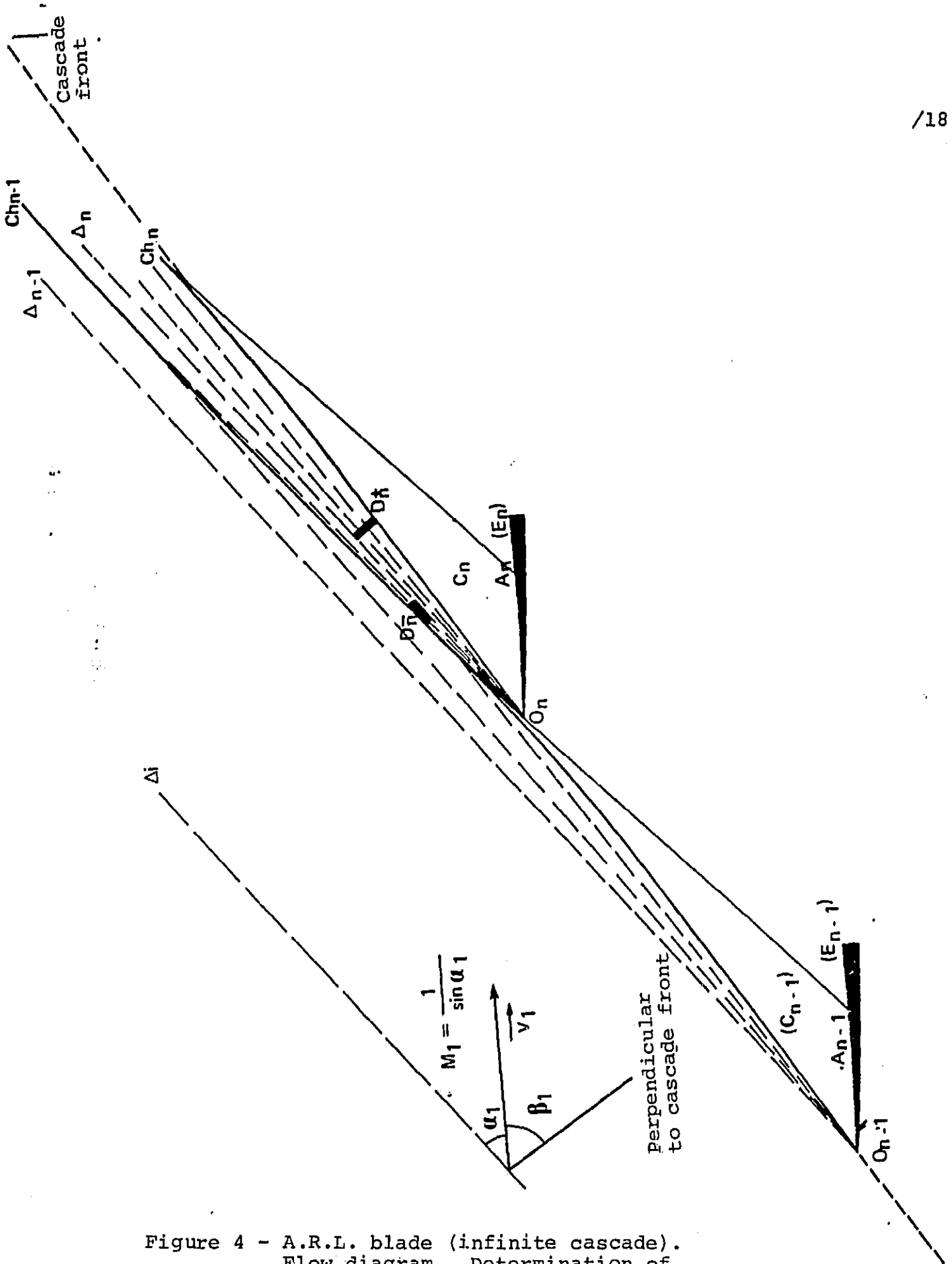


Figure 4 - A.R.L. blade (infinite cascade). Flow diagram. Determination of upstream conditions.

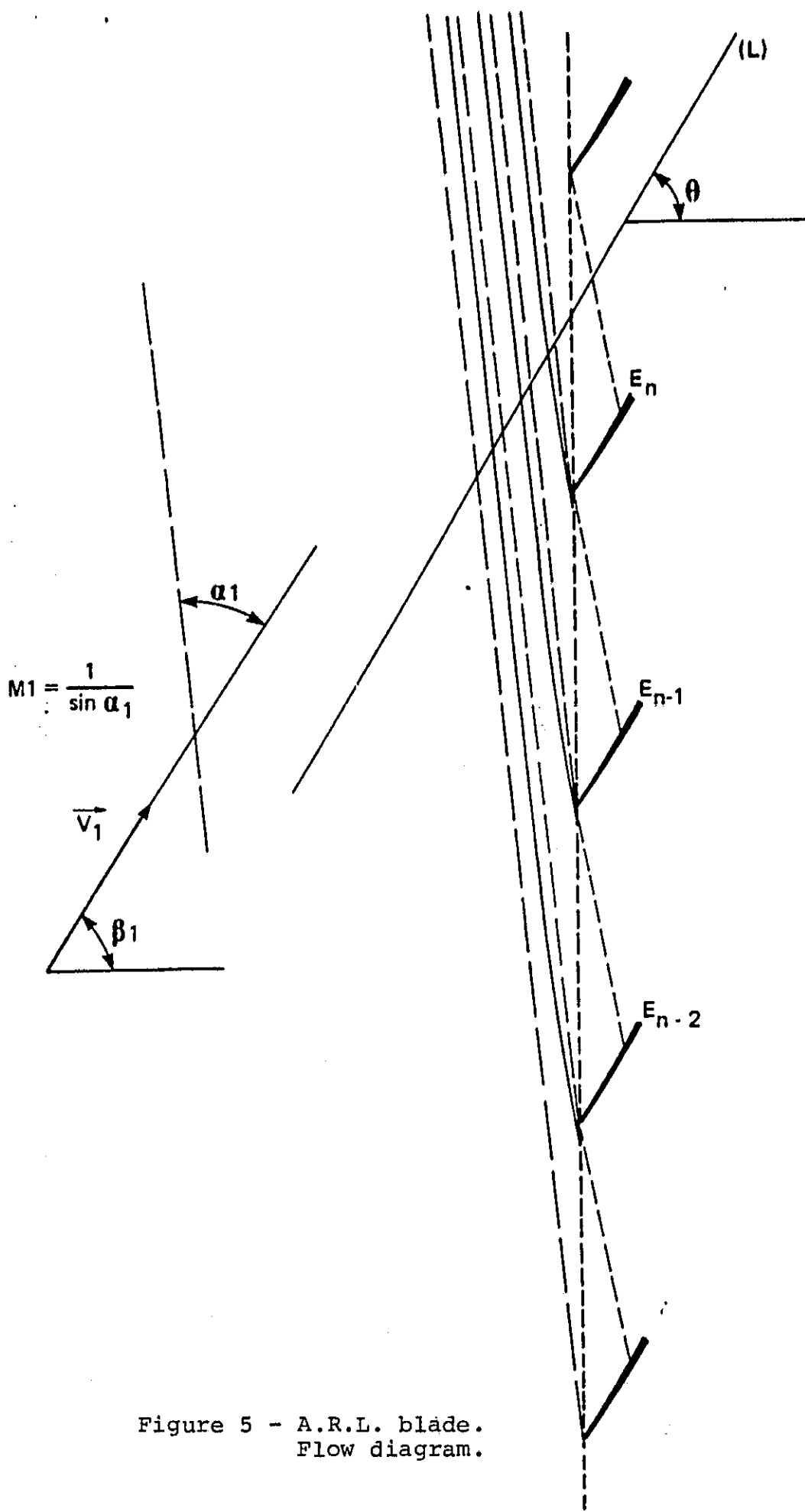


Figure 5 - A.R.L. blade.
Flow diagram.

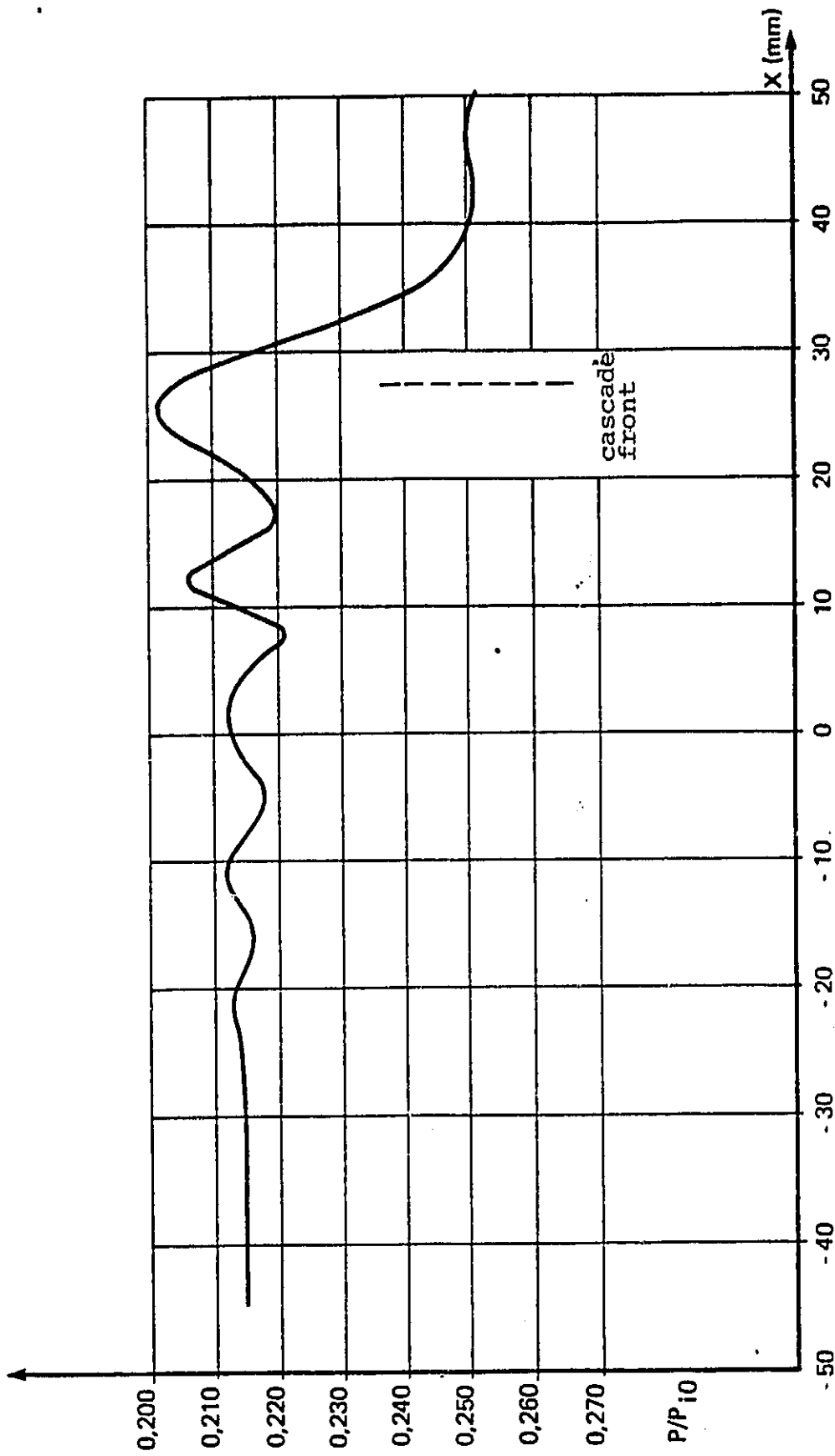


Figure 6 - A.R.L. blade. Upstream Flow. Calculation ($M_1 = 1.66$)

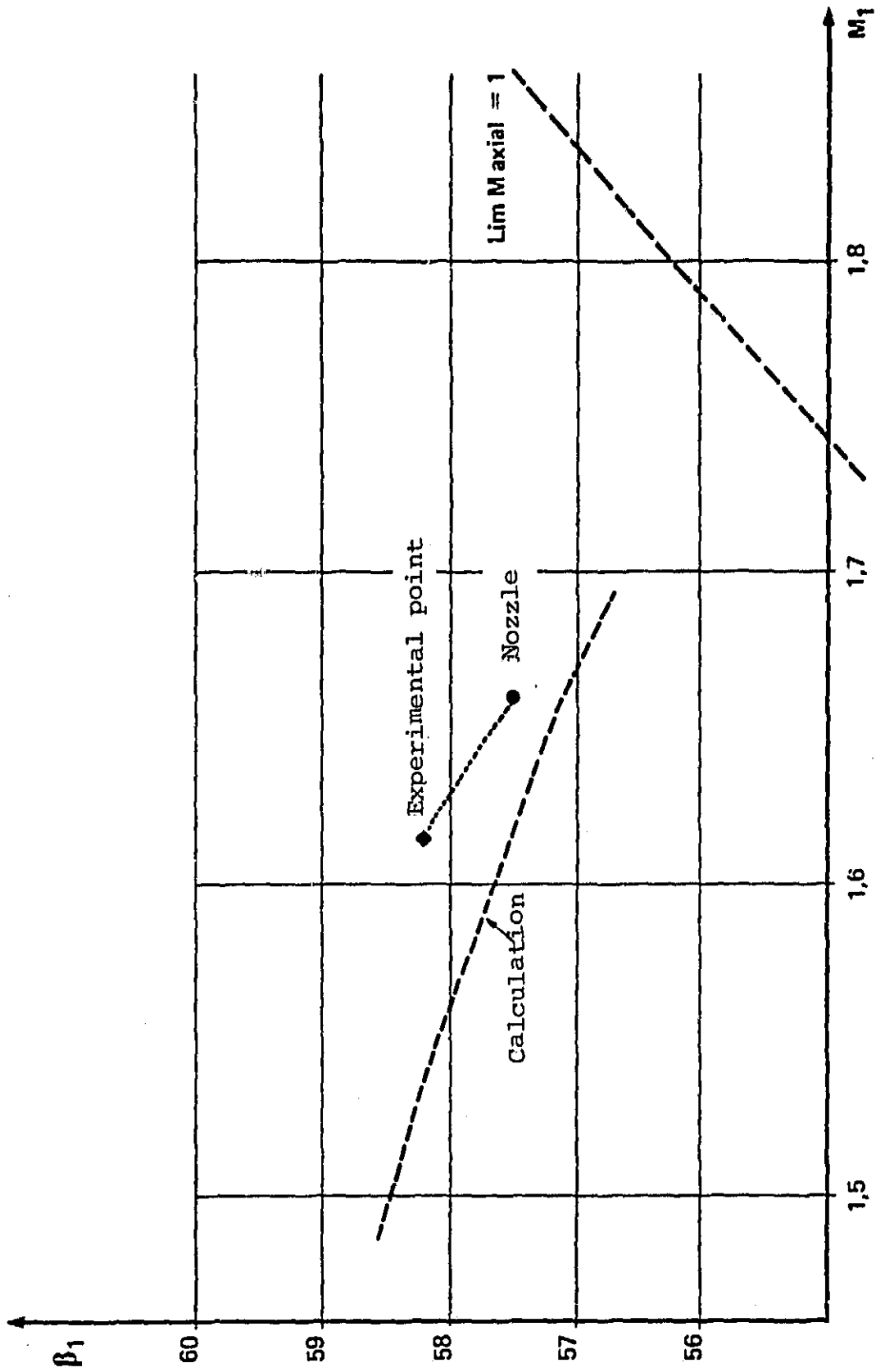


Figure 7 - A.R.L. blade - Single Incidence.

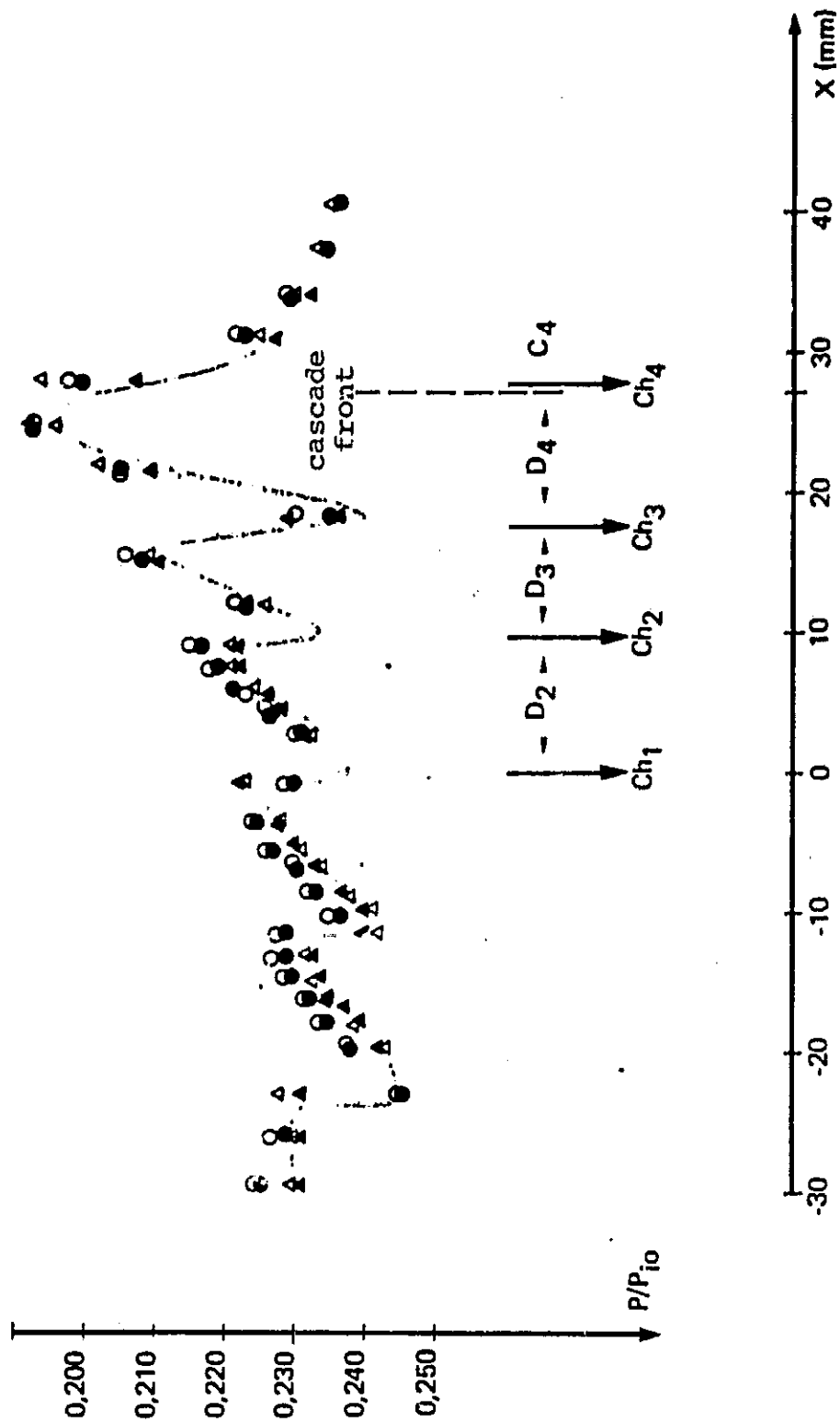


Figure 8 - A.R.L. cascade. Upstream flow (Test)

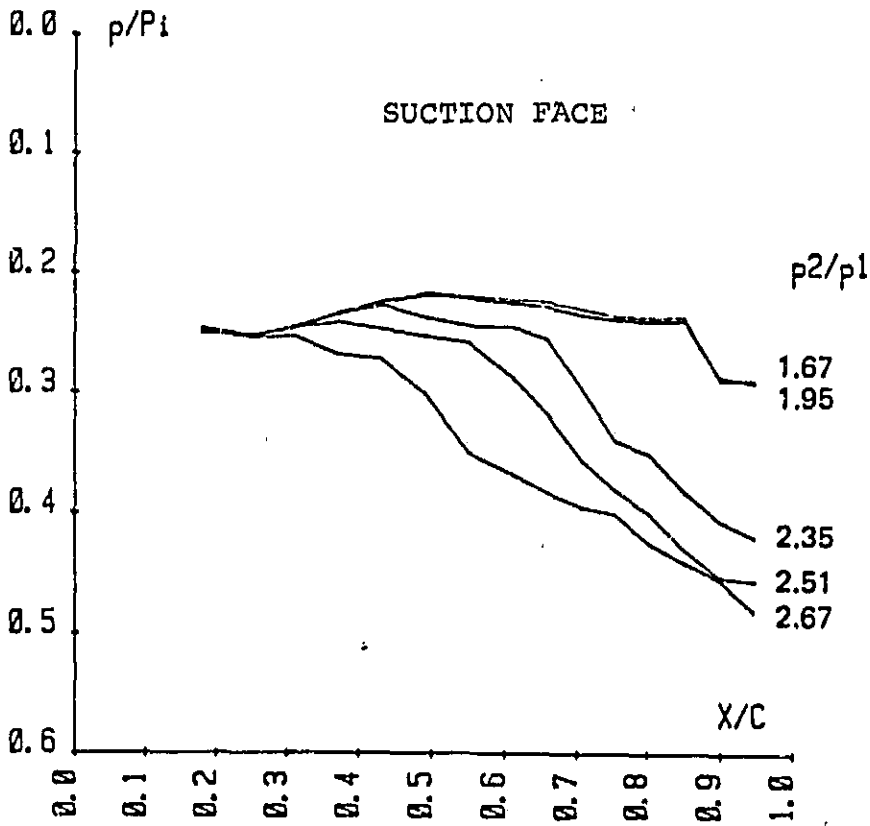
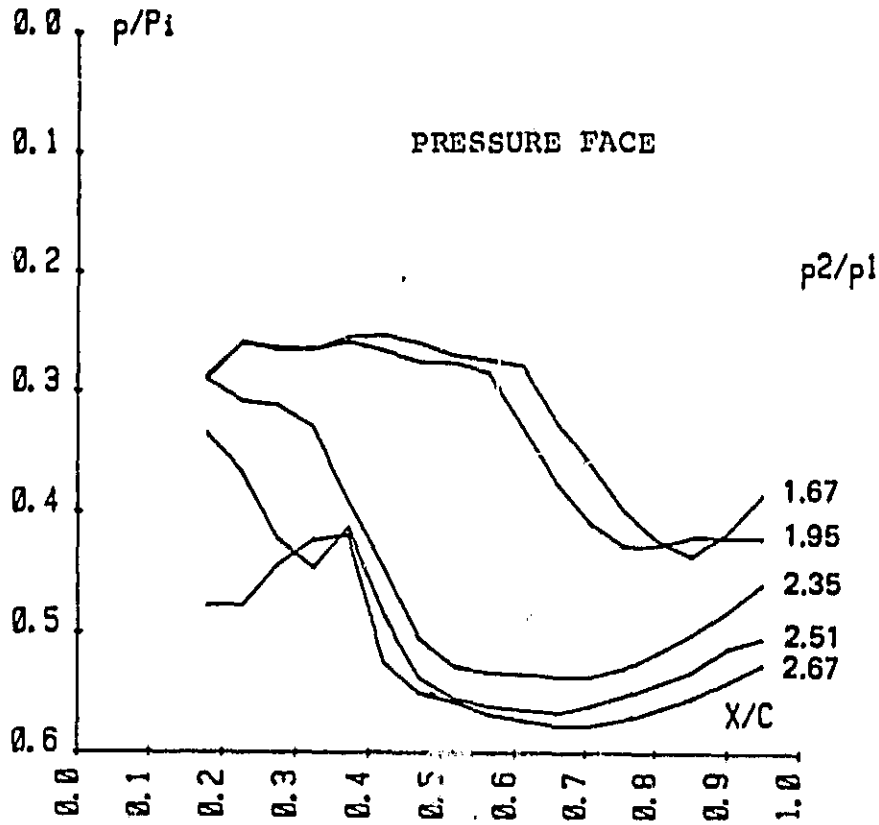
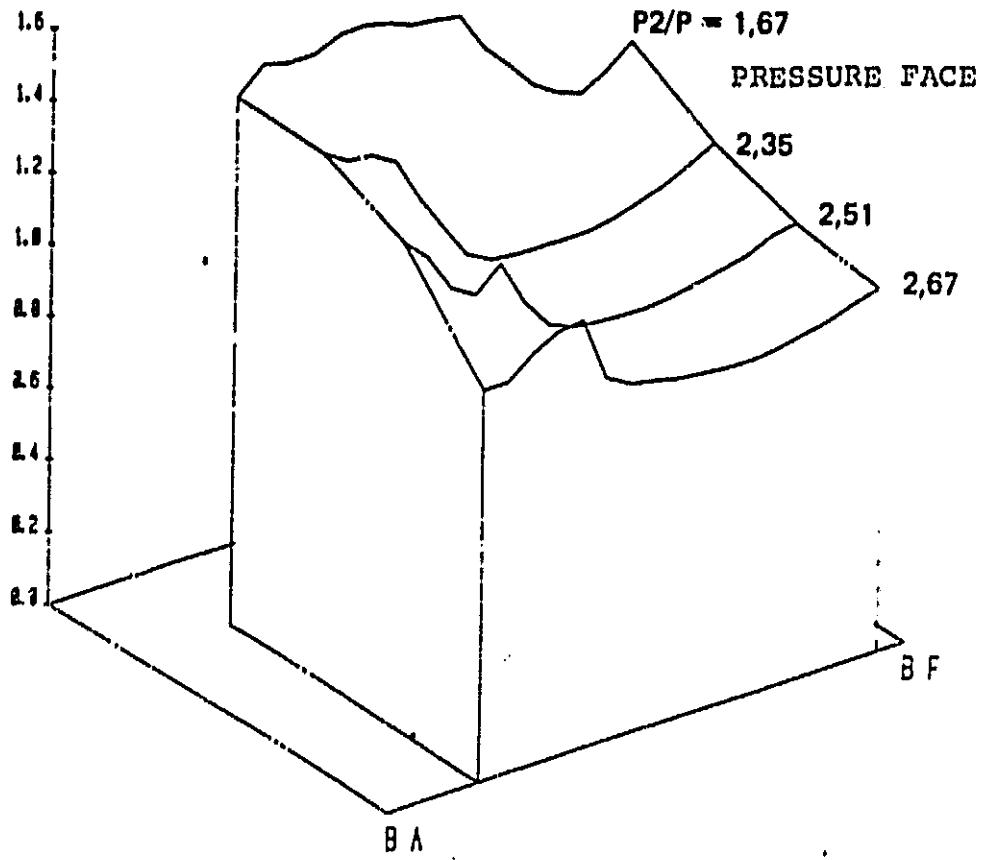
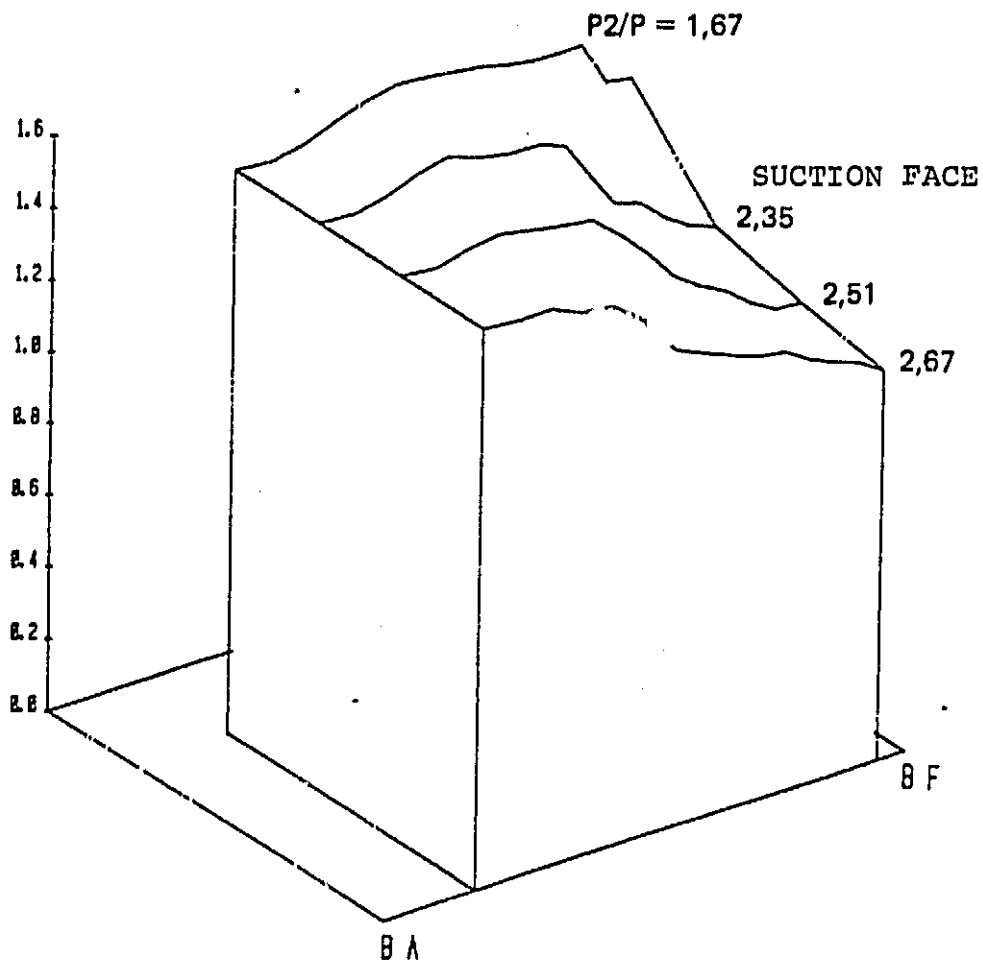
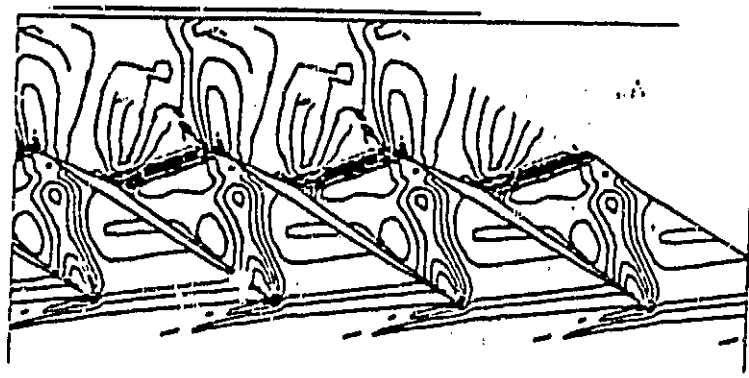


Figure 9 - A.R.L. cascade. ONERA tests of December 1983.

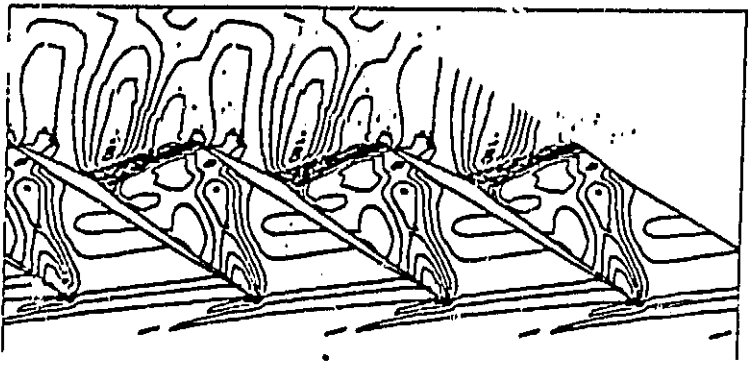


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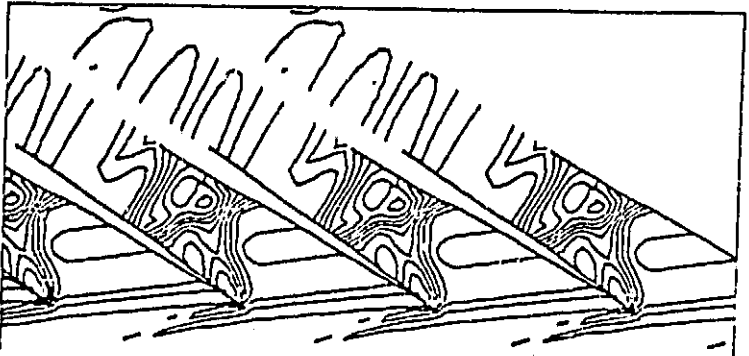


$$\frac{P_2}{P_1} = 1,67$$

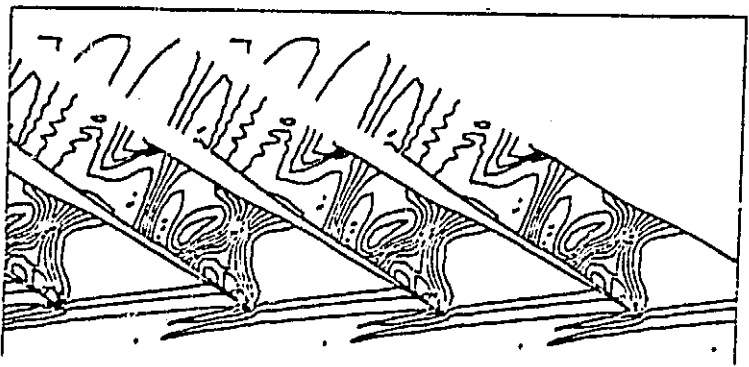


$$\frac{P_2}{P_1} = 1,95$$

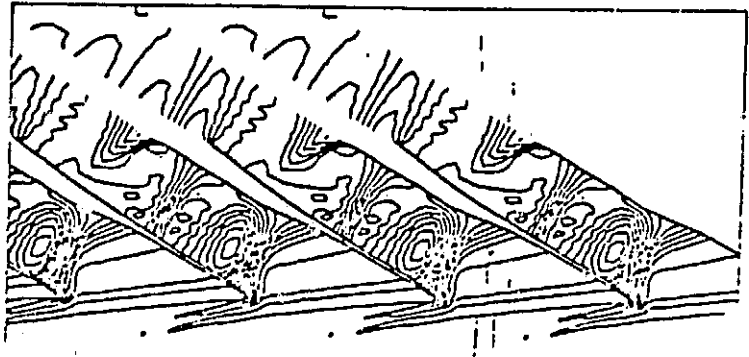
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$$\frac{P_2}{P_1} = 2,35$$



$$\frac{P_2}{P_1} = 2,51$$



$$\frac{P_2}{P_1} = 2,67$$

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Figure 11 - Flow reconstruction

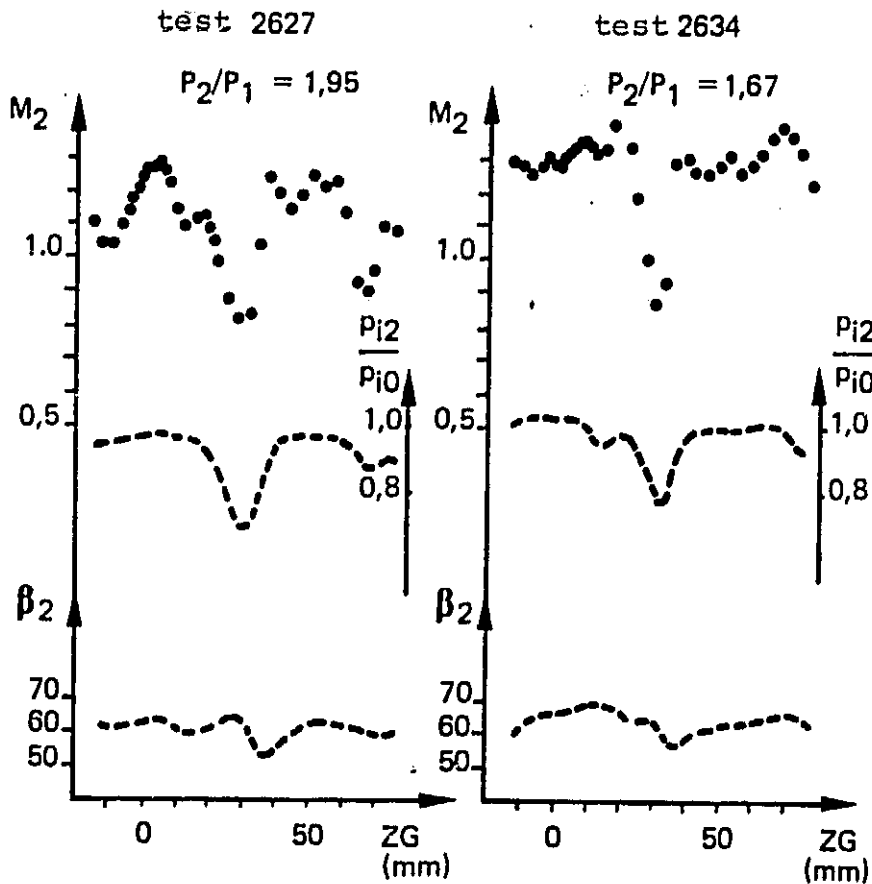
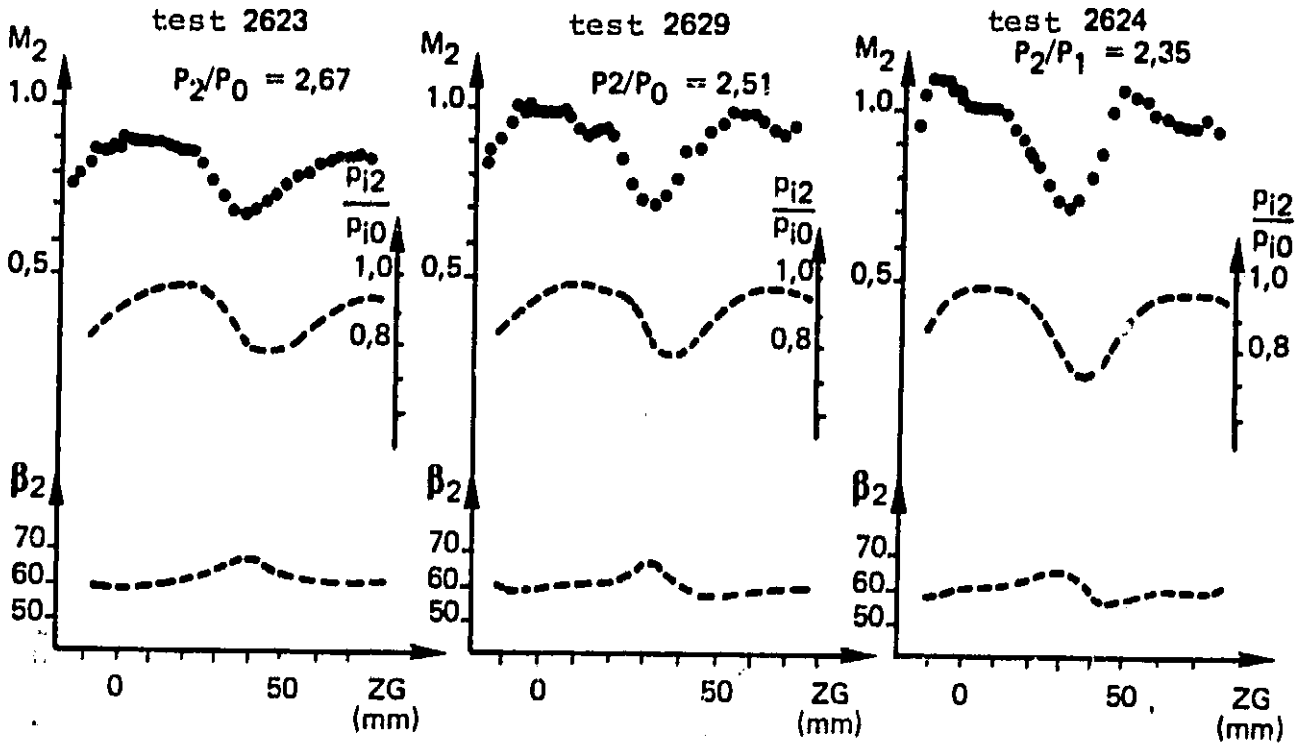


Figure 12 - Wakes

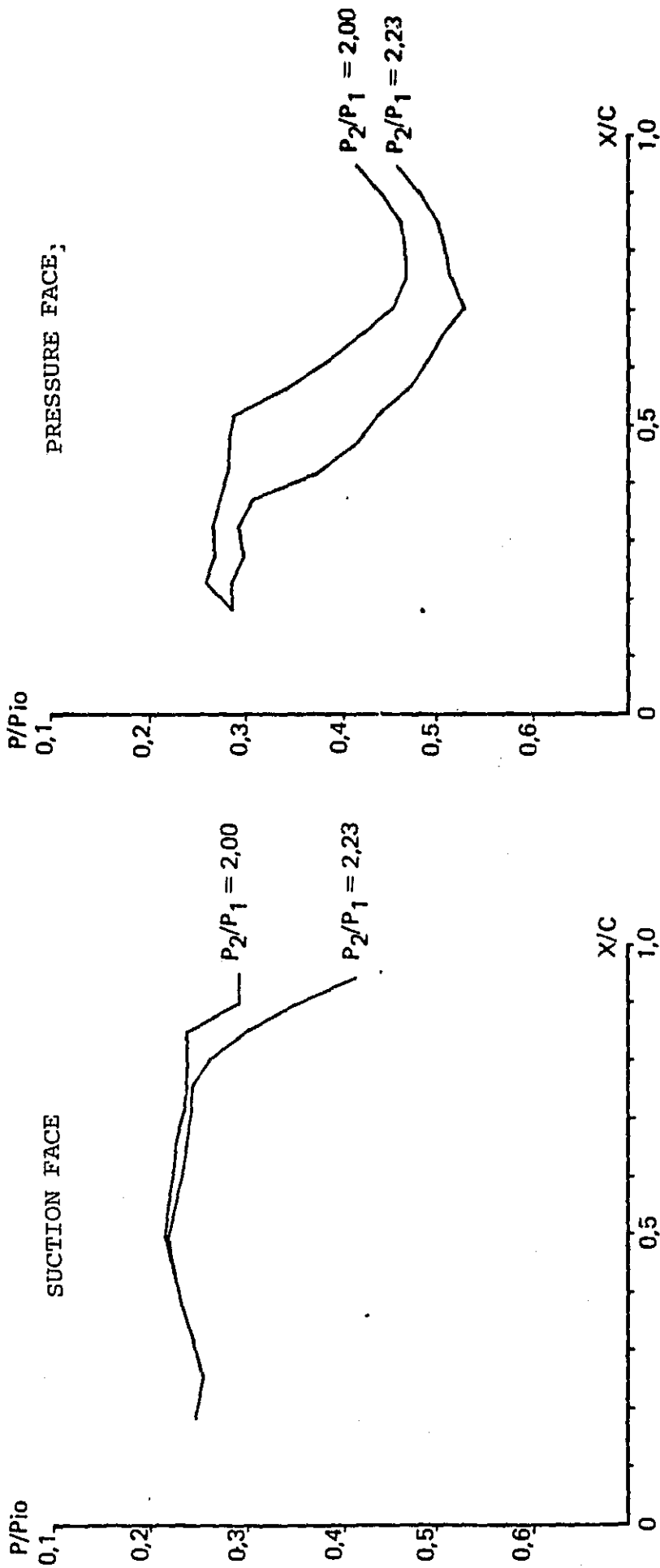
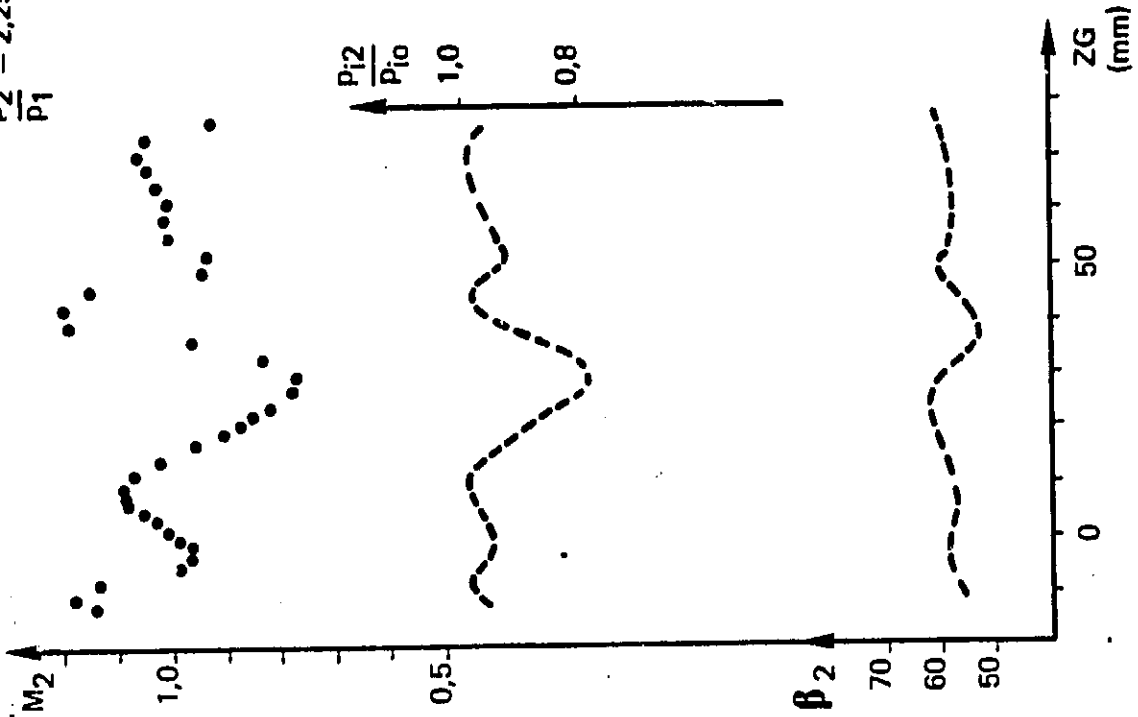


Figure 13 - ONERA tests December 1983 ($T_c = 0.884$)

test 2635

$\frac{P_2}{P_1} = 2,25$



test 2630

$\frac{P_2}{P_1} = 2,00$

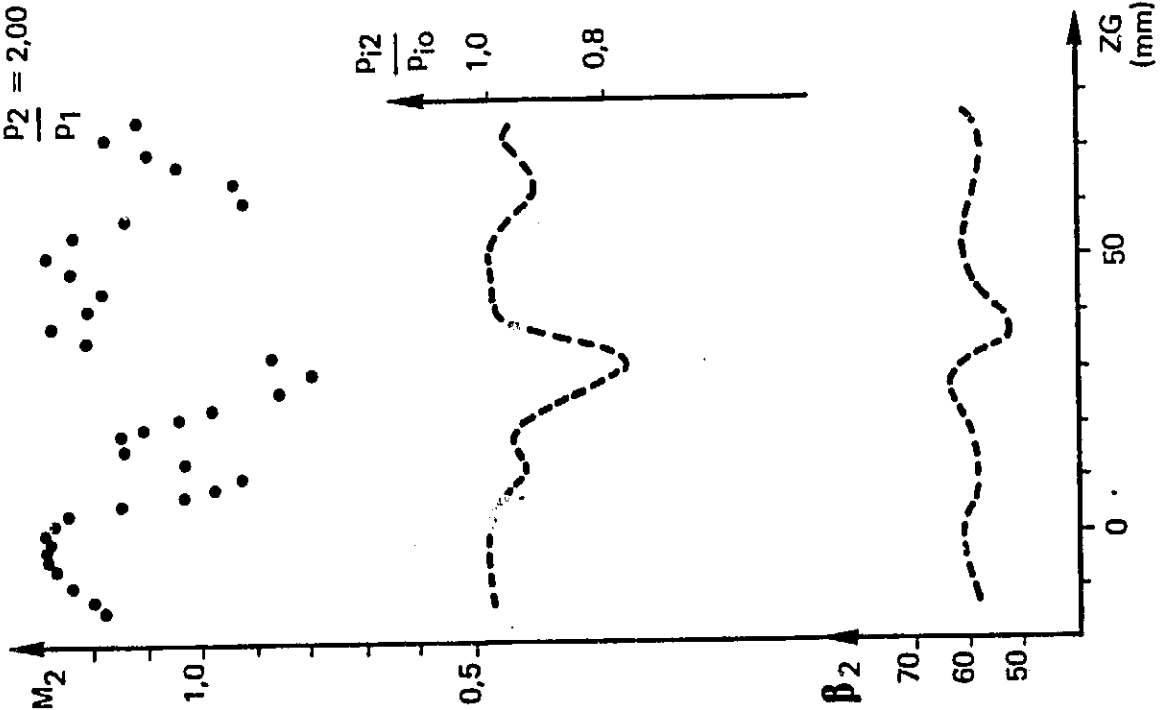


Figure 14 - Wakes

0.0 - p/P_i

ONERA

$p_2/p_1 = 1.95$

/29

0.1

— Suction face
-- Pressure face

0.2

0.3

0.4

0.5

0.6

X/C

0.0 0.1 0.2 0.3 0.4 0.5 0.6 0.7 0.8 0.9 1.0

ALLISON

0.0 - p/P_i

$p_2/p_1 = 1.97$

0.1

— Suction face
-- Pressure face

0.2

0.3

0.4

0.5

0.6

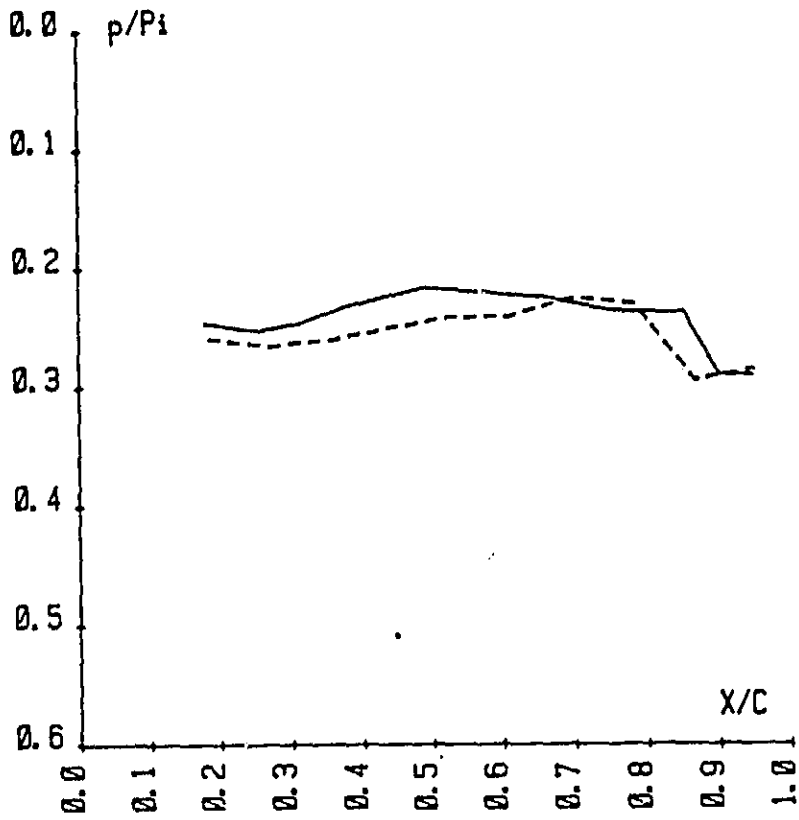
X/C

0.0 0.1 0.2 0.3 0.4 0.5 0.6 0.7 0.8 0.9 1.0

Figure 15 - A.R.L. cascade. ONERA and ALLISON test comparison

Suction face

/30



— ONERA $p_2/p_1 = 1,95$
 - - - ALLISON $p_2/p_1 = 1,87$

Pressure face

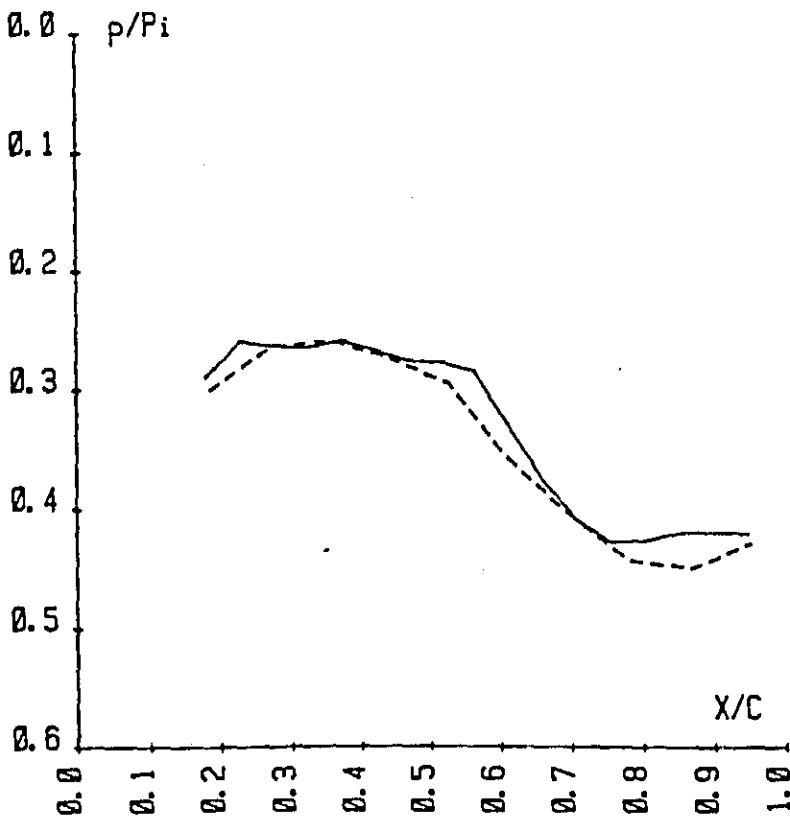


Figure 16 - A.R.L. cascade. ONERA and ALLISON test comparison.

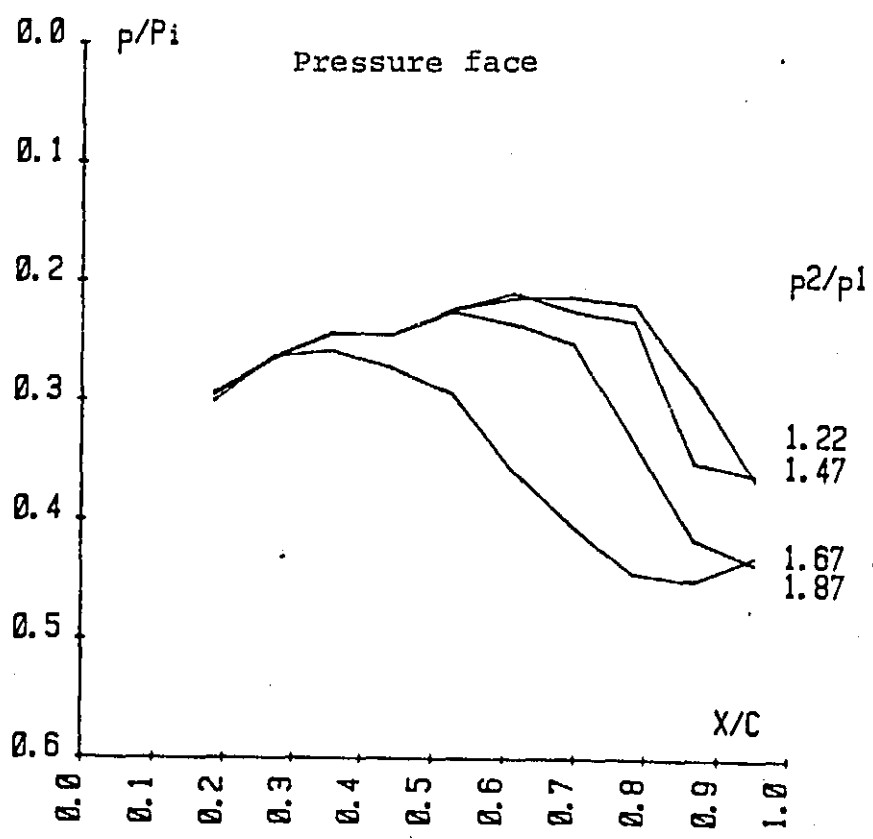
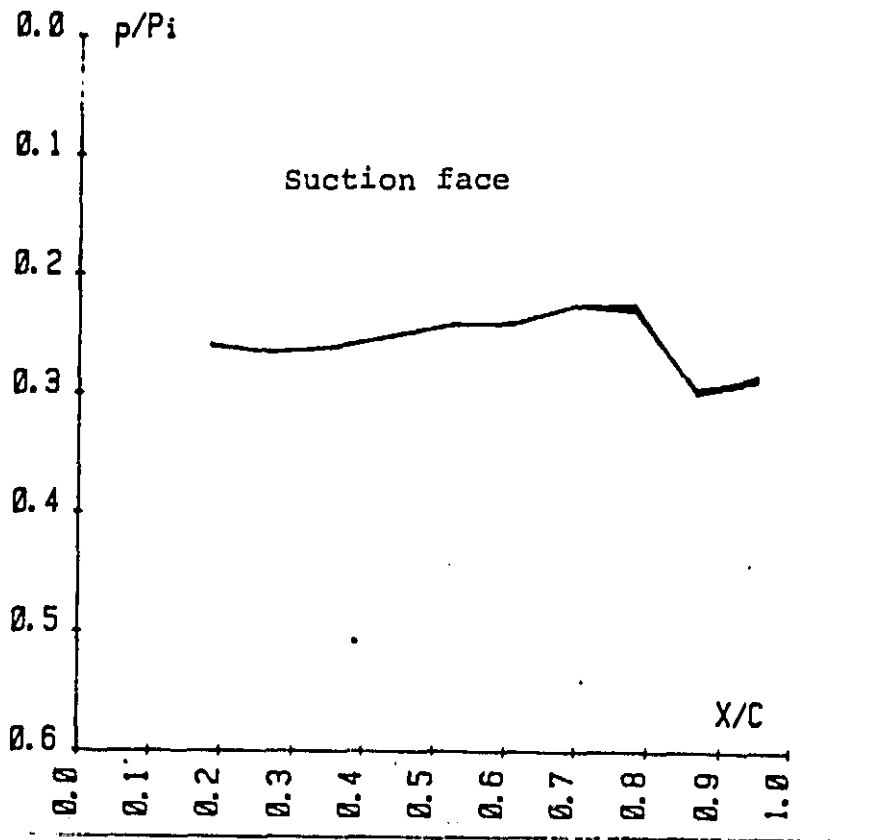


Figure 17 - A.R.L. cascade - D.D.ALLISON tests 1971