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6. AUTHOR(S)

Dr Yu Ermak

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Central Aerohydrodynamics Institute
Zhukovsky
Moscow 140160
Russia

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13. ABSTRACT (Maximum 200 words)

This report results from a contract tasking Central Aerohydrodynamics Institute as follows: The present report contains the description of the wind tunnel, the model, the test program, and the test results obtained.

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INVESTIGATION OF TEMPERATURE FACTOR INFLUENCE ON CONTROL SURFACE EFFECTIVENESS IN HYPERSONIC FLOWS WITH MODERATE INTERACTION

(Final report)

Project Supervisor: Dr.Yuri Yermak

CONTENTS

NOMENCLATURE

d_p	probe tube diameter
h_f	flap height
l_m	model length
l_p	probe tube length
M	freestream Mach number
q	freestream dynamic pressure
Re	freestream Reynolds number, $Re_\infty = f(T_\infty)$
Re_0	Reynolds number, $Re_0 = f(T_0)$
S_m	reference model base area
T_0	total temperature
T_w	model wall temperature
t_w	temperature factor, $t_w = T_w / T_0$
x_m	longitudinal model coordinate
$x_{c.m}$	conditional mass center position, $x_{c.m} = 2/3\ l_m$
C_x	drag coefficient, $C_x = F_x / (qS_m)$
C_y	lift coefficient, $C_y = F_y / (qS_m)$
Cm_z	pitching-moment coefficient, $Cm_z = M_z / (qS_m l_m)$
α	angle of attack
γ	azimuth angle in the transverse model plane
θ	cone half angle
$æ$	specific heat ratio
X	hypersonic interaction parameter

INTRODUCTION

For the flight of a hypersonic maneuverable vehicle at velocities of $M_\infty >> 1$ and altitudes higher than 40 km there arises the problem of increasing the effectiveness of control surfaces, especially when the control surfaces are near the vehicle base where a relatively thick hypersonic boundary layer develops on the vehicle surface when its thickness becomes of the same order as the control span. The effectiveness of the control surfaces in such a hypersonic boundary layer depends on local dynamic pressure, i.e., on relation $\sim 1/M^2_\infty$.

Another important point is that the hypersonic boundary layer flow condition is acted upon considerably by temperature factor in two ways. First, this parameter exerts an influence on the boundary layer thickness. Second, temperature factor influences the upstream propagation of disturbances and, thereby, the extent of the separation zone ahead of a control surface. Therefore, along with the Mach and Reynolds numbers, the temperature factor t_w is one more parameter influencing the control surface effectiveness.

In conventional hypersonic wind tunnels, the temperature factor is $t_w \sim 0.15$ to 0.3 and higher, while in full-scale conditions this quantity is less almost by an order. Consequently, it is impossible to evaluate reliably the effectiveness of the control surfaces of hypersonic maneuverable vehicles in conventional hypersonic wind tunnels although the parameter $M_\infty/\sqrt{Re_\infty}$ is simulated correctly.

TsAGI is in possession of hypersonic facilities capable of simulating not only the parameter $M_\infty/\sqrt{Re_\infty}$ but also the parameter t_w; among these is, for example, a cryogenic wind tunnel VAT-3 in which the model surface can be cooled down to cryogenic temperatures with temperature factor t_w ranging from 0.04 to 0.3.

A model in the form of a 5^0 -cone with a flap was designed and manufactured to test the control surface effectiveness in this wind tunnel.

The spoiler-type flap as a control surface was chosen based on the fact that in the tests the effectiveness of the flap, i.e., the property of producing aerodynamic loads, is mostly exhibited precisely at zero angle of attack since the model body in the form of a circular cone without a flap does not produce any normal force and longitudinal moment at $\alpha = 0$. In this case, the influence of temperature factor on the effectiveness of the flap itself can be judged from the behavior of variations in coefficients C_y and Cm_z of the model with varying parameter t_w. The present report contains the description of the wind tunnel, the model, the test program and the test results obtained.

1. Wind tunnel VAT-3

The vacuum wind tunnel VAT-3 is designed for investigating aerodynamic and thermal characteristics of models of such vehicles as aerospace planes and others (Figs. 1, 2).

This wind tunnel may be used for:

- evaluation of integral aerodynamic characteristics and disturbances caused by interference of vehicle-induced jets;

- evaluation of local aerodynamic and thermal characteristics of vehicle components;

- flow visualization by glow discharge method;

- physical investigations.

VAT-3 is an intermittent wind tunnel featuring cryogenic pumping of test gas. The cryopanels are cooled by gaseous helium being passed through a cooling gas-cycle unit.

The flow is produced by contoured and conical hypersonic nozzles. Test gas condensation is precluded by using Cowper electric heaters. The facility has two test sections to investigate external aerodynamics of winged vehicles and gas-dynamics of orbiter jets.

The wind tunnel parameters enable simulation of aerogasdynamic effects in orbital descent of such vehicles as aerospace planes from altitudes of 100 to 75 km.

1.1. Technical characteristics of VAT-3

VAT-3 is an impulse wind tunnel with cryopumping of test gas.

Test section dimensions:

diameter, m	1.0
length, m	1.7
volume, cub.m	1.5
Mach number range:	
contoured nozzle	M=12
conical nozzle	M=20
nozzle diameter, m	0.15 to 0.3
Reference model dimension, m:	
aerospace plane-type vehicles	0.1 to 0.2
orbiters	up to 0.3
Reynolds number range (per 1m):	
at M=12	$0.5*10^6$ to $1.5x10^6$
at M=20	10^4 to 10^5
Adiabatically retarded gas parameters:	
pressure, MPa	2 to 6
temperature, K	1000 to 2000

Gas flow rates, kg/s:	
maximum	1.0
nominal	0.2
Wind tunnel efficiency:	
run time at maximum gas flow rate, s	0.07
same at nominal gas flow rate, s	3 to 5
between-runs-interval, min	no more than 5
runs per cycle	up to 100
cycles per week	2 to 3
Consumed electric power, kW	up to 100

2. Model description

The hypersonic maneuverable vehicle model is a thin axisymmetric cone with a half angle of $\theta = 5^0$; one or two spoiler-type flaps were placed on the model afterbody. The geometric dimensions of the flaps, the model and its components are given in Figs. 3, 4.

A base sting made of heat-insulated material to reduce heat transfer to the model was used to suspend the model. In the balance tests, the sting was attached to a strain-gage balance placed in a thermostatically-controlled container on a three-component traversing gear creating displacements along the x-axis up to 600 mm, along the y-axis up to 100 mm at angles of attack up to 45^0.

Temperatures of the side model surface and of the flap (a total of 5 points) were measured during the balance tests using thermocouples; the thermocouple wire diameter was not more than 0.1 mm to reduce the thermocouple influence on the balance test results.

The three-component semiconductor strain-gage balance was used to measure frontal drag F_x, lift F_y and pitch moment M_z over the ranges: 1 to 200 g for F_x and F_y and M_z < 200 g*cm. The balance was thermostatically-controlled at 10 ± 5^0C. The amplifier of the AHЧ-21 type was used to amplify signals; the measurement error was ~ 1%.

The model was tested at zero angle of attack (α=0); the maximum model deflection under the action of the model weight and aerodynamic loads was not exceed $\sim1^0$.

The pressure in the flow separation zone upstream of the flap was measured by a probe (Fig.5); its capillar part was made in the form of a tube with the wall thickness of 0.5 mm and the dimensions of $l_p * d_p$ = 50 mm * 2 mm. The probe was fixed on the traversing gear guide so that the capillar part (d_p = 2 mm) was in the longitudinal vertical model plane and oriented in a downward direction. The gap between the model surface and the probe was ~ 1 mm; a low-inertia pressure tap capable of measuring pressure up to 0.1 atm was used as a measuring element.

The model flow visualization system based on the glow discharge technique involved 2 copper electrodes with the potential of 3 to 10 kV from a high-voltage constant-current source (Fig.1). The most part of the plate surfaces was shielded to focus the discharge into the model location region.

The flow patterns over the model were recorded by a special photographic unit mounted on the side test section window. It consisted of a photo camera, an electromagnet with a pusher and a relay controlling the electromagnet switching-on delay. An impulse to initiate the photo camera operation was supplied by the data acquisition system during a run.

3. Test program

The evaluation of the effectiveness of a control surface in the form of a spoiler-type flap on a hypersonic maneuverable vehicle included two stages.

First, the free-stream hypersonic flow (Mach number $M_\infty = 12$) over the model with two flaps of different heights (h_f = 5 and 10 mm) was visualized by the glow discharge technique at different temperature factors t_w = 0.06 to 0.3. The model flow pattern, shock system configuration upstream of the flap, separation zone dimensions on the side model surface and aerodynamic characteristics were determined. The data obtained made it possible to improve the test procedure and test equipment operation and to choose the dimensions of a spoiler-type flap at which the temperature factor influence would be appreciable.

At the second stage, optical tests of the model with one flap (h_f = 10 mm), pressure measurements in the flow separation zone and then balance tests were conducted in a given range of the parameter t_w.

In contrast to the previously adopted test program, the pressure in the separation zone upstream of the flap was measured simultaneously with optical tests using a probe rather than on a single tapped model. This alteration introduced in the test program was caused, on the one hand, by a considerable rise in the manufacturing cost of the models themselves and, on the other hand, by a drastic rise in the cost of wind tunnel runs that would be required for testing a special tapped model. Unfortunately, the accuracy of static pressures in the separation zone measured by using a probe was, of course, lower than that in the case of applying a special tapped model; therefore, the model pressure distributions obtained are of a qualitative nature.

The test program for the 5^0-cone model is given in Table 1.

Table 1

Test stage	Test stage scope	Model configuration	Parameters measured	Results obtained
I	Optical and balance tests	Cone ($\theta = 5^0$) with two flaps ($h_f = 5.0$ and 10.0mm, $\gamma = 0$ and 180^0)	1)Flow spectra 2)Aerodynamic characteristics	Model flow pattern, aerodynamic characteristics
II	Optical tests, pressure measurements (probe), balance tests	Cone ($\theta = 5^0$) with one flap ($h_f = 10$mm, $\gamma = 0$)	1)Flow spectra 2)pressure distribution upstream of flap 3)aerodynamic characteristics	Evaluation of control surface(flap) effectiveness using balance measurements

Here γ is the azimuth angle in the transverse model plane

4. Test results

At hypersonic velocities, pressure disturbances for a flow over a flat plate propagate, in the condition of moderate or strong viscous interaction, upstream along the boundary layer for distances comparable with the body length [1-4]. As for weak viscous interaction, the disturbance influence is limited by a small vicinity of a disturbance source [1]. The disturbance propagation in flows over slender axisymmetric bodies in the case of weak interaction is similar to that for a two-dimensional case.

When viscous interaction is moderate or strong, the increase in the interaction level causes the cross dimension of the zone, through which the disturbances propagate, to tend to zero [5], and variations in flow parameters are concentrated in the vicinity of a disturbance source. Temperature factor t_w and specific heat ratio $\varkappa$ are also important parameters that exert some influence on the upstream disturbance propagation [6-8]. When tw tends to zero, the interaction region contracts to a disturbance source. A similar situation also exists as $\varkappa \rightarrow 1$.

The case when disturbances propagate upstream along the boundary layer is known as a supercritical flow condition in a hypersonic boundary layer [7]. This boundary layer is characterized by a Mach number averaged over the boundary layer thickness. If there is three-dimensional disturbance source in such a boundary layer, the disturbance propagation in this boundary layer begins in the azimuth direction inside characteristic disturbance lines, i.e., analogs of Mach numbers.

In this investigation, values of hypersonic interaction parameter $X=1/(\alpha^2 * \sqrt{Re_0})$ were close to 1 when $M_\infty = 12$ and $Re_0 = 1.6*10^4$ which indicated the existence of a moderate viscous interaction condition and the possibility of propagating appreciable disturbances in a flow over a flap both upstream and in the azimuth direction.

From the analysis of the results obtained in the present investigation for a 5^0 -cone with a spoiler-type flap at hypersonic velocities with temperature factor t_w ranging from 0.30 to 0.06 it is inferred that the flap effectiveness depends considerably on parameter t_w in the case of moderate interaction.

4.1. Optical tests

Figs. 6 to 8 present pictures of flow spectra for a 5^0 -cone model with one spoiler-type flap at three temperature factors: t_w = 0.06, 0.095, and 0.3.

These pictures show clearly a shock system characteristic of a flow over a body of revolution with a flap: a bow shock (N1,4), an oblique shock upstream of the flap initiating at the boundary layer separation point (N2), and a shock wake immediately upstream of the flap (N3).

In addition, one more oblique shock (N5) occurs on the model side opposite to that where the flap is placed (γ=180^0); the longitudinal coordinate of the shock onset corresponds approximately to the coordinate of the flow separation point upstream of the flap.

This shock is likely to be caused by a strong disturbance or even by the boundary layer separation on the opposite model side due to the extension of the influence of the separation zone upstream of the flap.

As a result of the treatment of the pictures obtained on a densitometer (of the Pericolour type) it is evident that the model boundary layer thickness decreases as temperature factor reduces, while the shock system configuration changes slightly (Figs. 9 to 12), but no appreciable variations in the boundary layer separation point position were observed.

4.2. Pressure distribution in the separation zone upstream of the flap

Pressure coefficients measured by a probe along the cone model generatrix upstream of the flap at two temperature factors t_w = 0.06 and 0.3 are given in Fig. 13. Possible influence of the probe itself on the pressure distribution over the side model surface implies only qualitative nature of the dependence $C_p = f(x, t_w)$. But it is clearly seen that pressure values in the separation zone upstream of the flap vary considerably with varying temperature factor, while the separation zone dimensions vary slightly.

4.3 Balance tests

Fig. 14 shows main aerodynamic characteristics of a 5^0-cone model with a flap, that is longitudinal and normal force coefficients and pitching moment coefficients as functions of temperature factor tw.

It is seen that these aerodynamic coefficients increase by 30-40% as parameter t_w decreases especially in the range of t_w =0.20 - 0.06.

It should be noted that in the case of moderate interaction, temperature factor influences insignificantly the total drag of a cone without a flap since the increase in its wave drag with a rise of the hypersonic boundary layer thickness due to increased temperature factor is counteracted by a respective friction drag decrease.

The behavior of the aerodynamic characteristics of a 5^0-cone with a flap observed in the experiment is explained basically by an increased interaction of the flap with the flow when the boundary layer thickness decreases as parameter t_w reduces.

The flap exerts the most influence on the aerodynamic characteristics of the model under consideration; therefore, the influence of variations in the flow conditions over the flap on the characteristics of the whole model as parameter t_W decreases is so considerable.

CONCLUSIONS

Based on the present investigation carried out at $M_\infty = 12$ using a 5^0-cone model with a flap the following tentative concluding remarks can be drawn:

1. The decrease in the hypersonic boundary layer thickness with reducing temperature factor makes a basic contribution to the effectiveness of a spoiler-type flap. For example, the longitudinal moment acting on the model increases by 30 to 40% when t_w varies from 0.3 to 0.065.

2. The variation in temperature factor in the range under investigation exerts a weak influence on the variation in the boundary layer separation point position upstream of the flap.

3. It is important that pressure distributions on the side cone surface and on the flap be investigated in more detail using a special tapped model.

4. It is desirable to continue the investigation of the temperature factor influence on the effectiveness of aerodynamic control surfaces with varying viscous interaction parameter χ and specific heat ratio æ.

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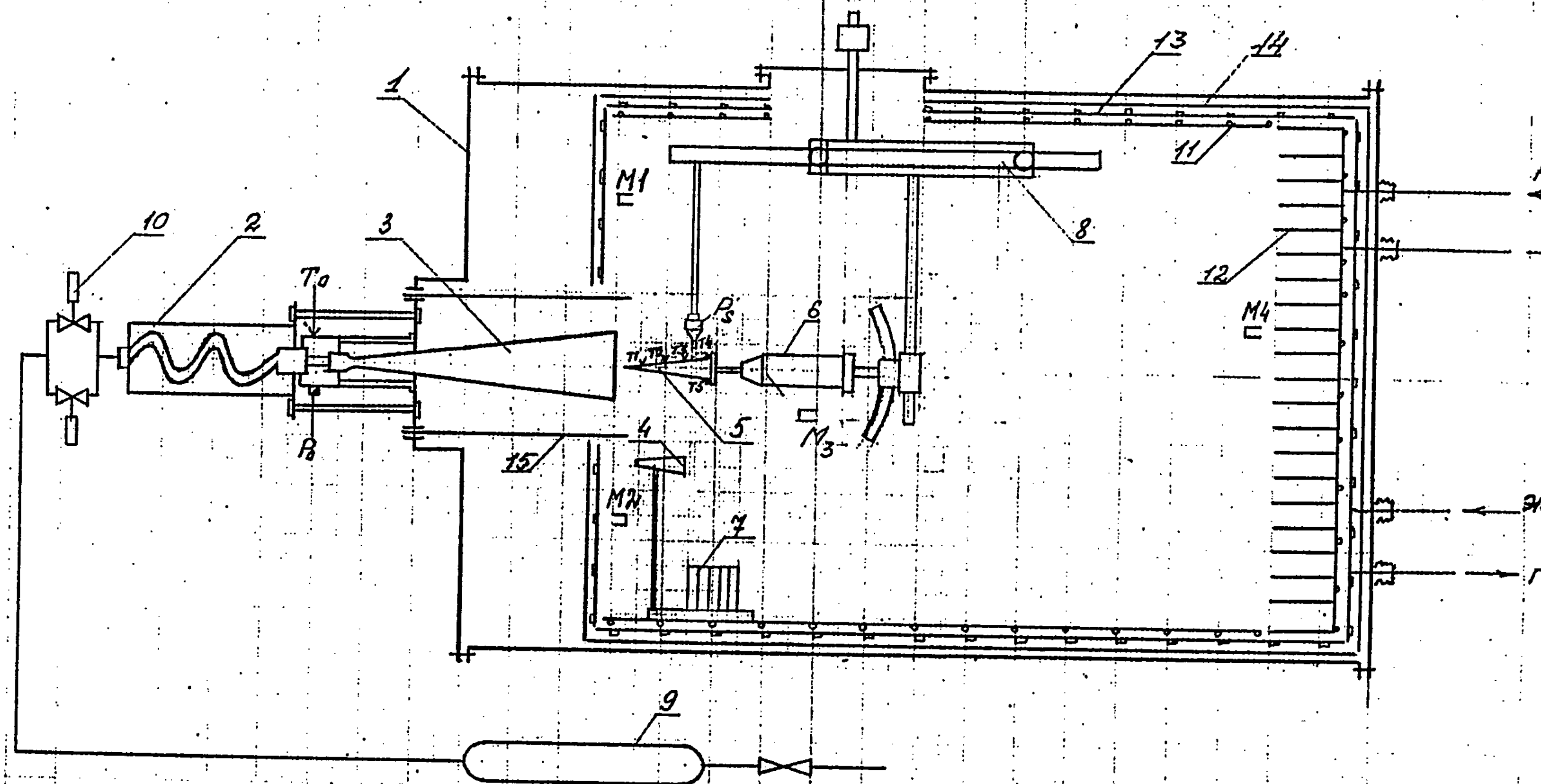

1- vacuum chamber; 2 - heater; 3 - nozzle; 4 - cooling device; 5 - model; 6 - three-component balance; 7 - cryostat; 8 - traversing gear; 9 - receiver; 10 - electric pneumatic valve; 11 - cylindrical cryopanel; 12 - end cryopanel; 13 - screens cooled by liquid nitrogen; 14 - uncooled screens; 15 - elecrodes;

Fig.1 SCHEMATIC OF WIND TUNNEL VAT-3

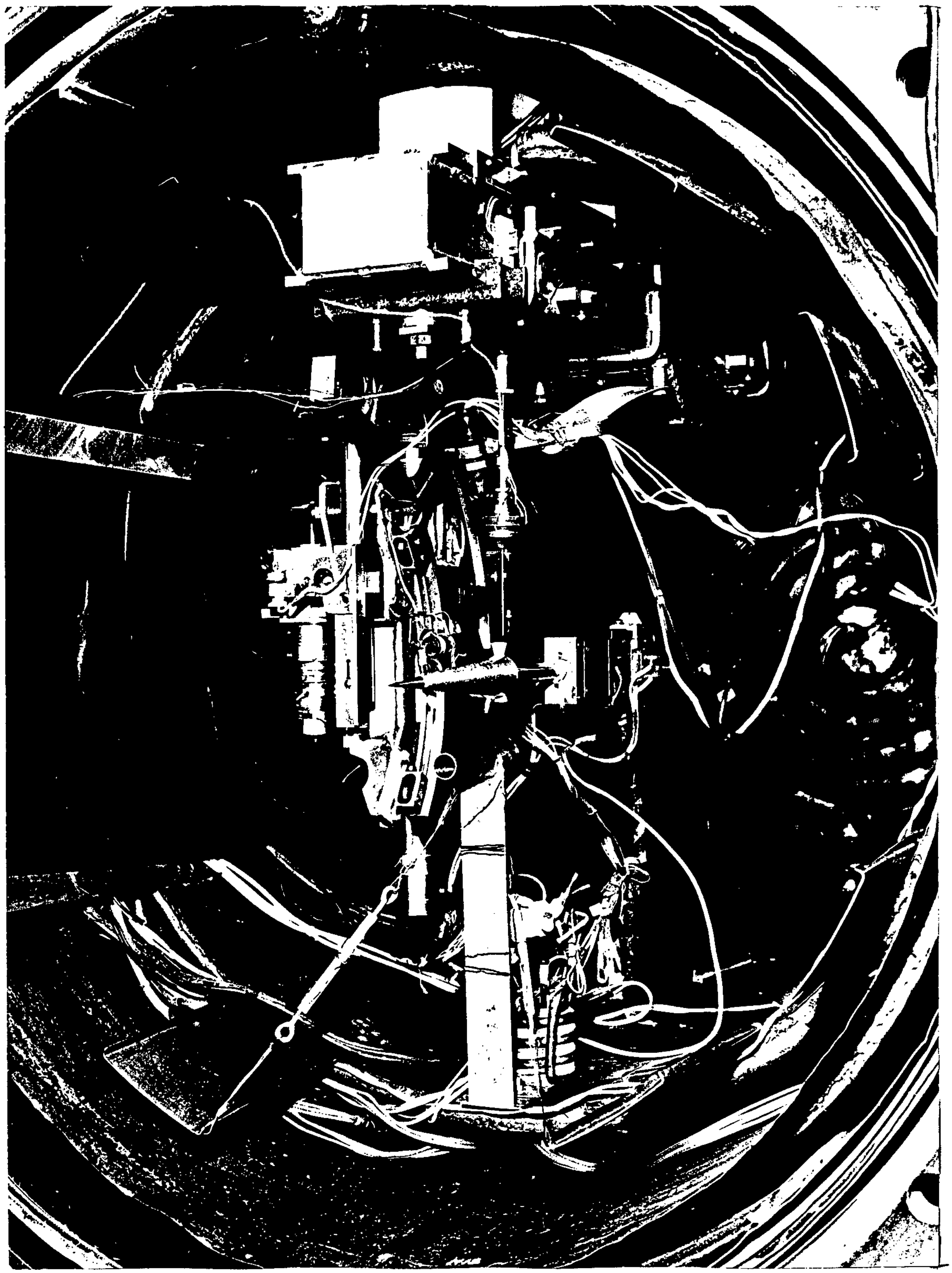

Fig 2   Test section of VAT 2

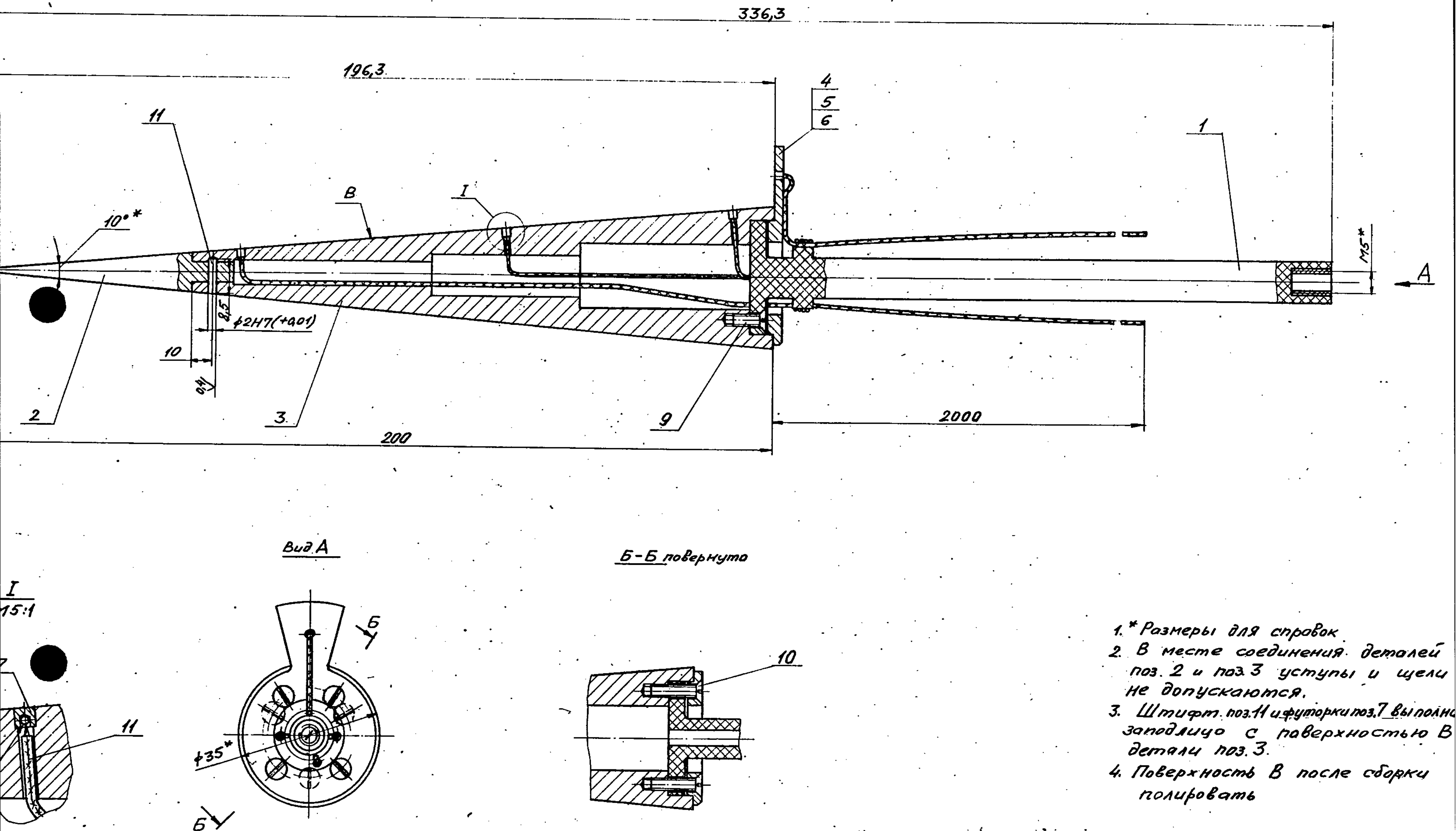

Fig.3 Schematic of 5°-cone model with flap

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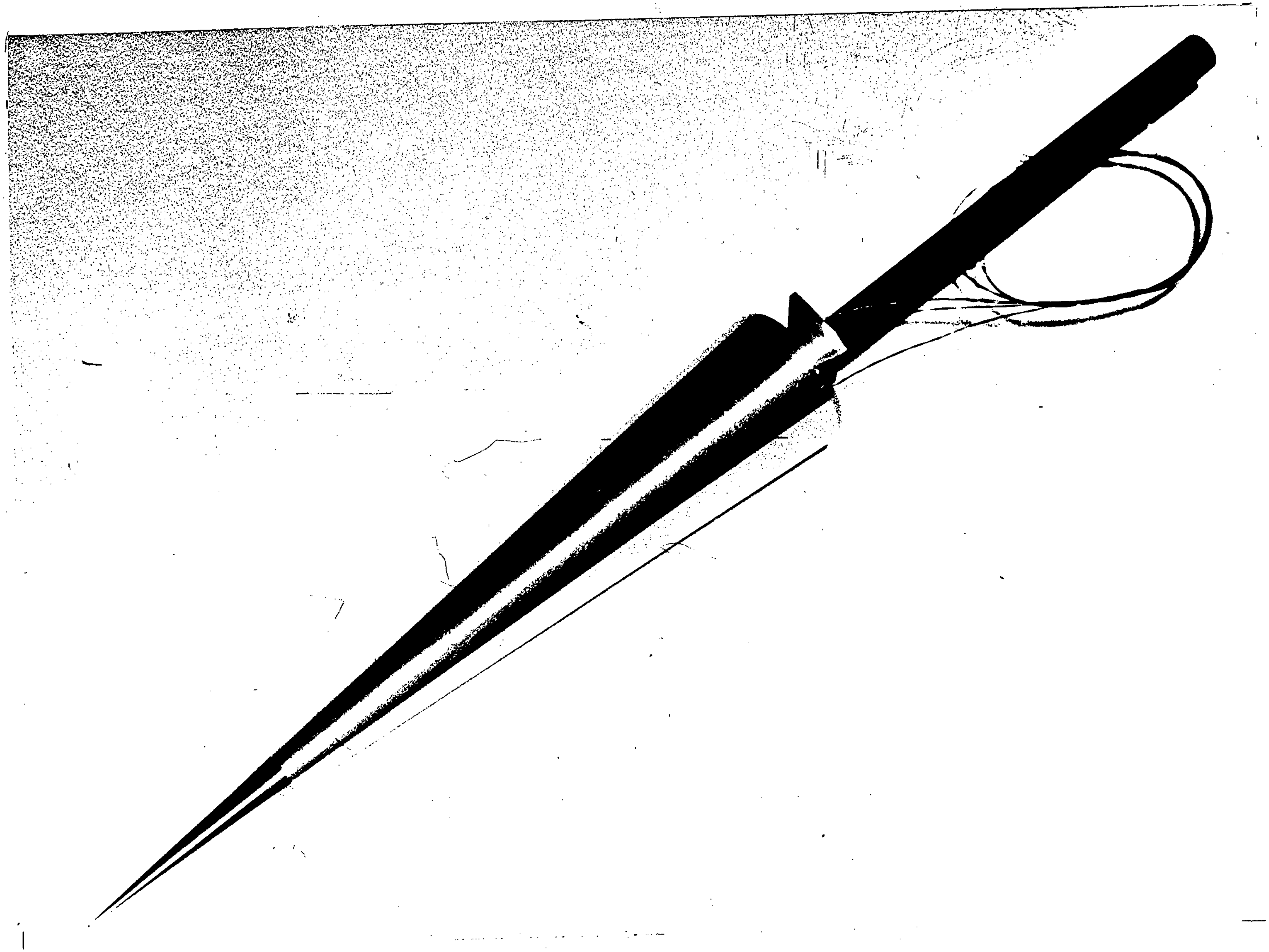

Fig. 4 Photo of 5°-cone model with flap

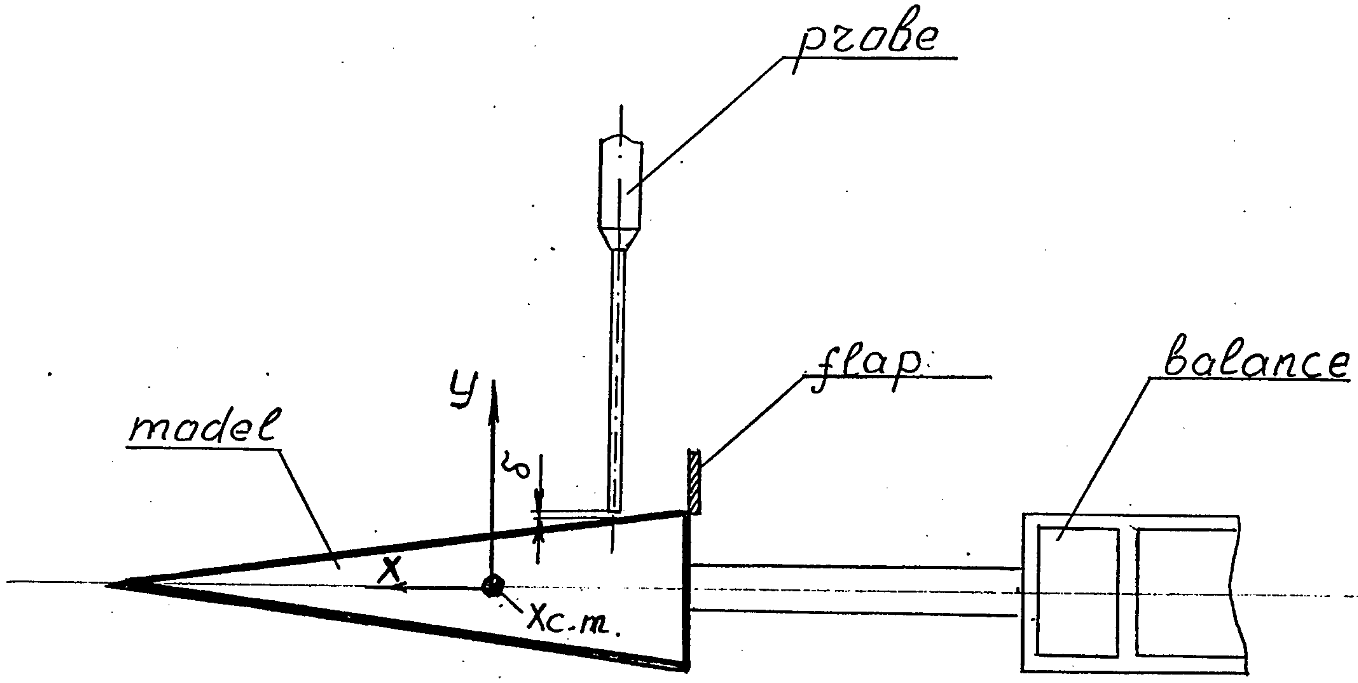

Fig. 5 Schematic presentation of technique to measure side model surface pressure upstream of flap using probe

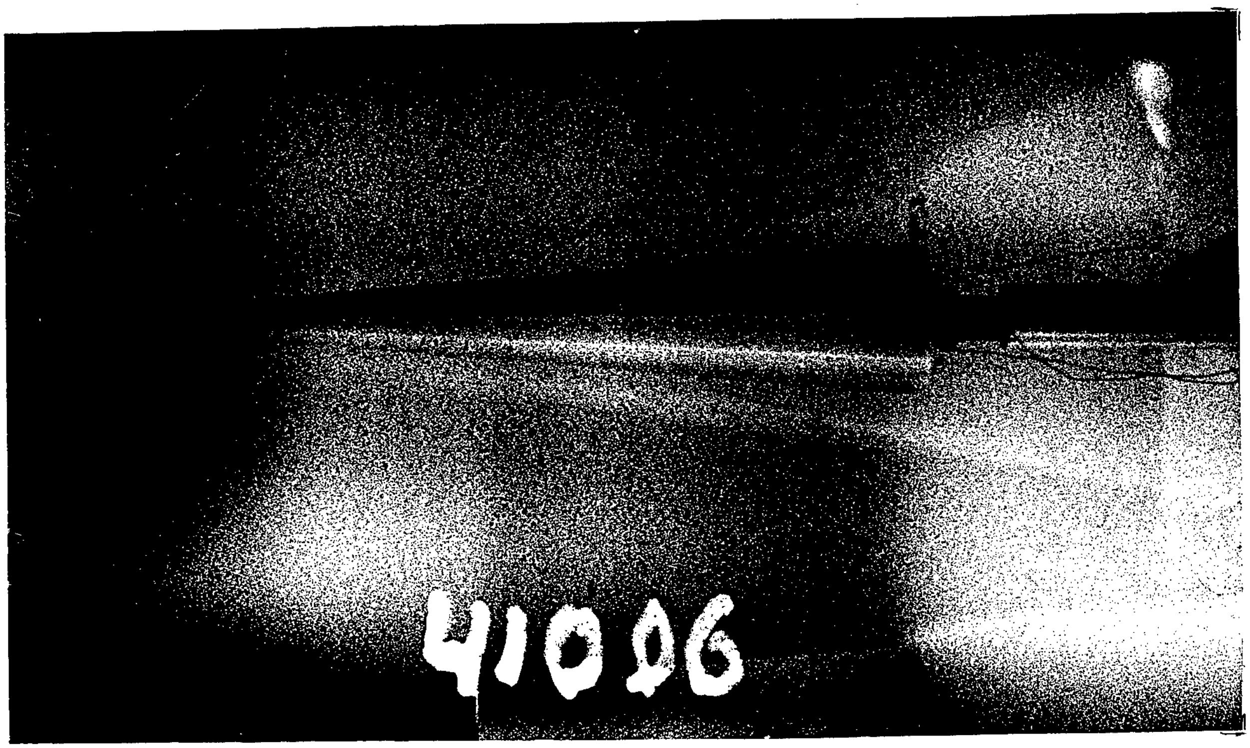

Fig.6 Flow spectrum over 5°-cone model with flap at $t_w = 0.3$

Fig.7 Flow spectrum over 5°-cone model with flap at t_w = 0.093

Fig.8 Flow spectrum over 5°-cone model with flap at t_w = 0.065

Flow pattern over 5°-cone model with flap at $t_w = 0.3$

N пуска 41006

N угла	φ, град.	
1	~7°	$M_\infty=12$
2	~13°	$Re_o=0.16*10^5$
3	– °	$Re_\infty=0.385*10^6$
4	~85°	$t_w=0.3$
5	~135°	

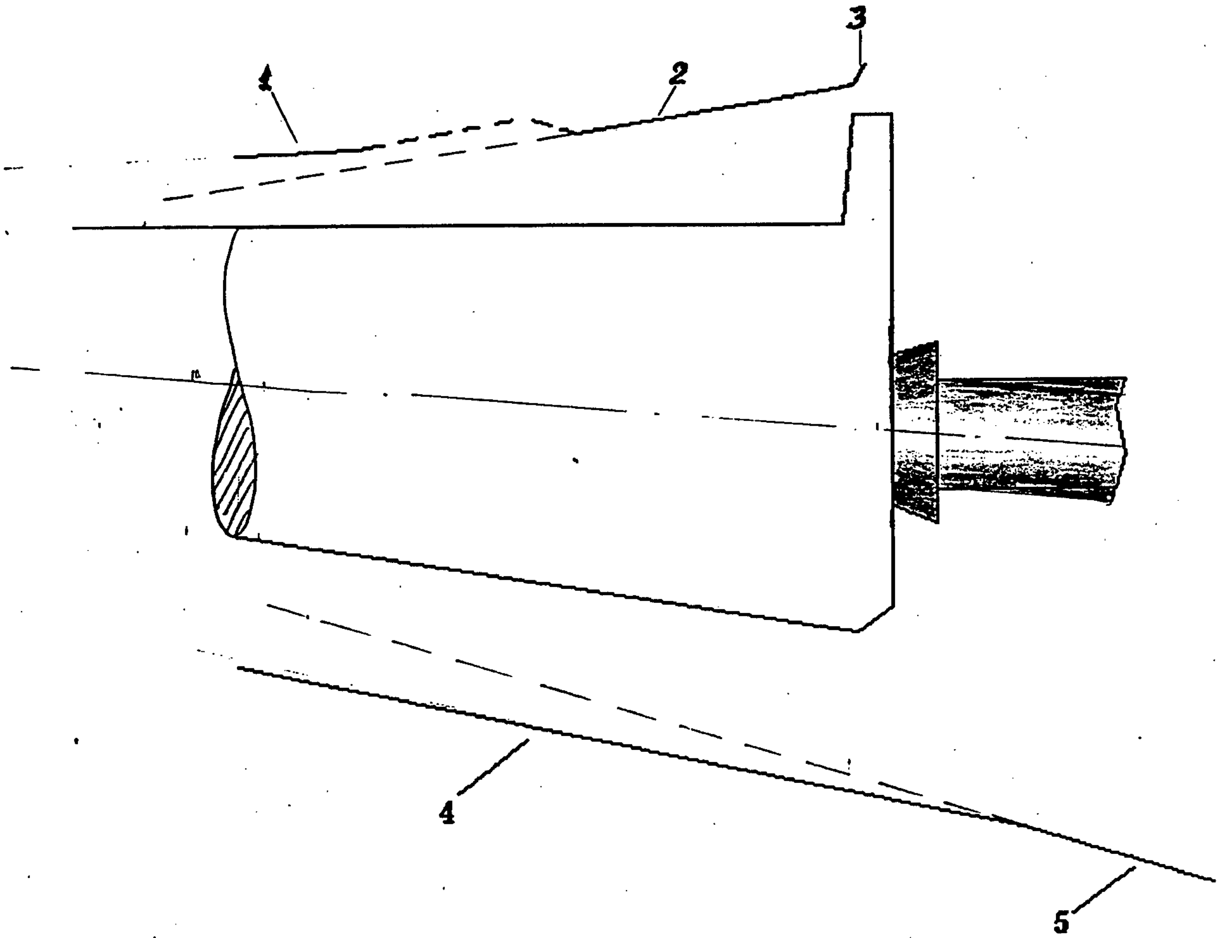

Fig. 9

Flow pattern over 5°-cone model with flap at $t_w = 0.093$

N пуска 41012

N угла	φ, град.	
1	~6.5°	$M_\infty = 12$
2	~12.8°	$Re_o = 0.16 * 10^5$
3	– °	$Re_\infty = 0.385 * 10^6$
4	~8 °	$t_w = 0.093$
5	~13 °	

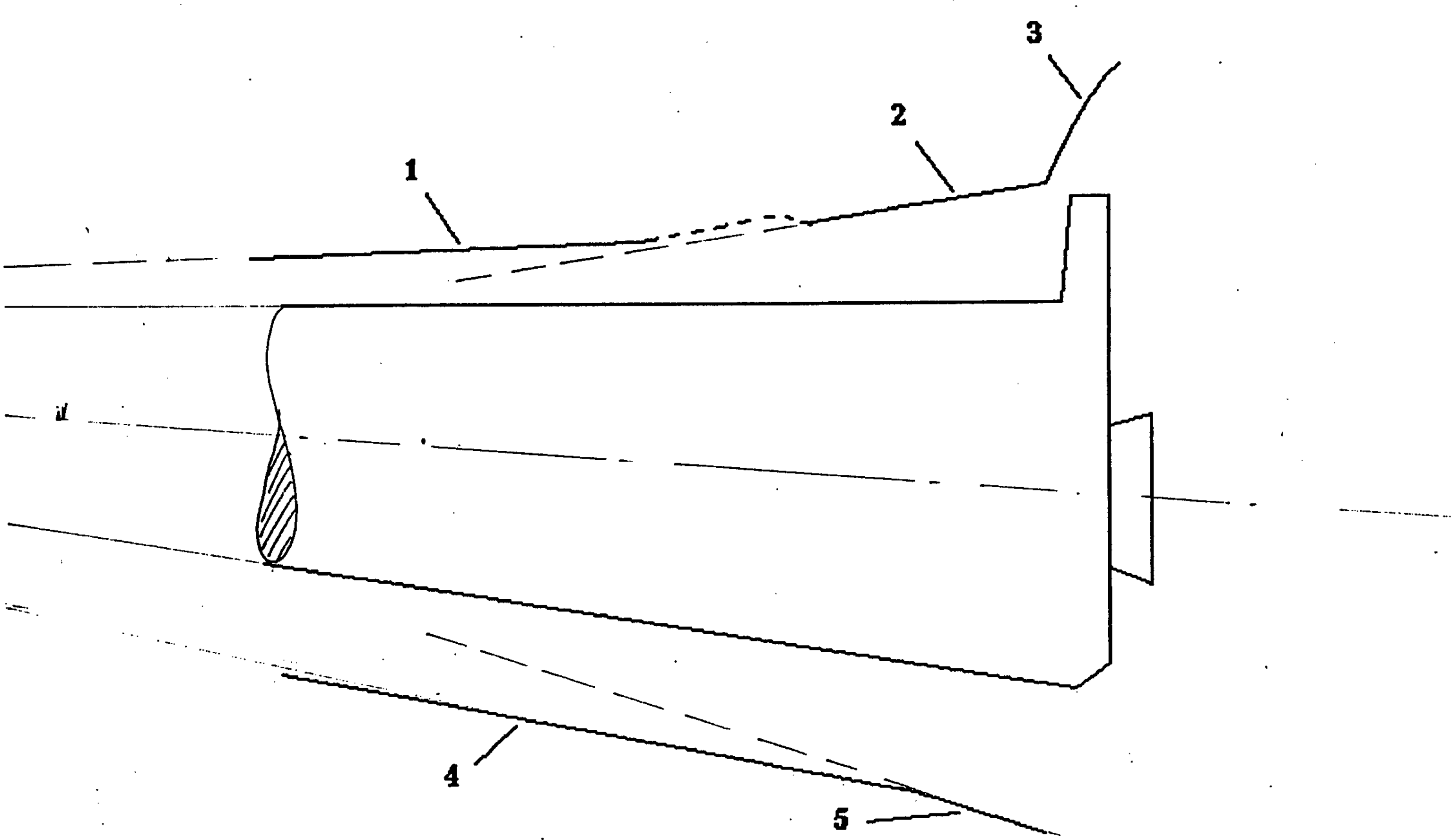

Fig. 10

Flow pattern over 5°-cone model with flap at t_w = 0.065

N пуска 41011

N угла	φ, град.	
1	~6°	$M_\infty = 12$
2	~12,5°	$Re_o = 0.16{*}10^5$
3	– °	$Re_\infty = 0.385{*}10^6$
4	~7,5°	$t_w = 0.065$
5	~12°	

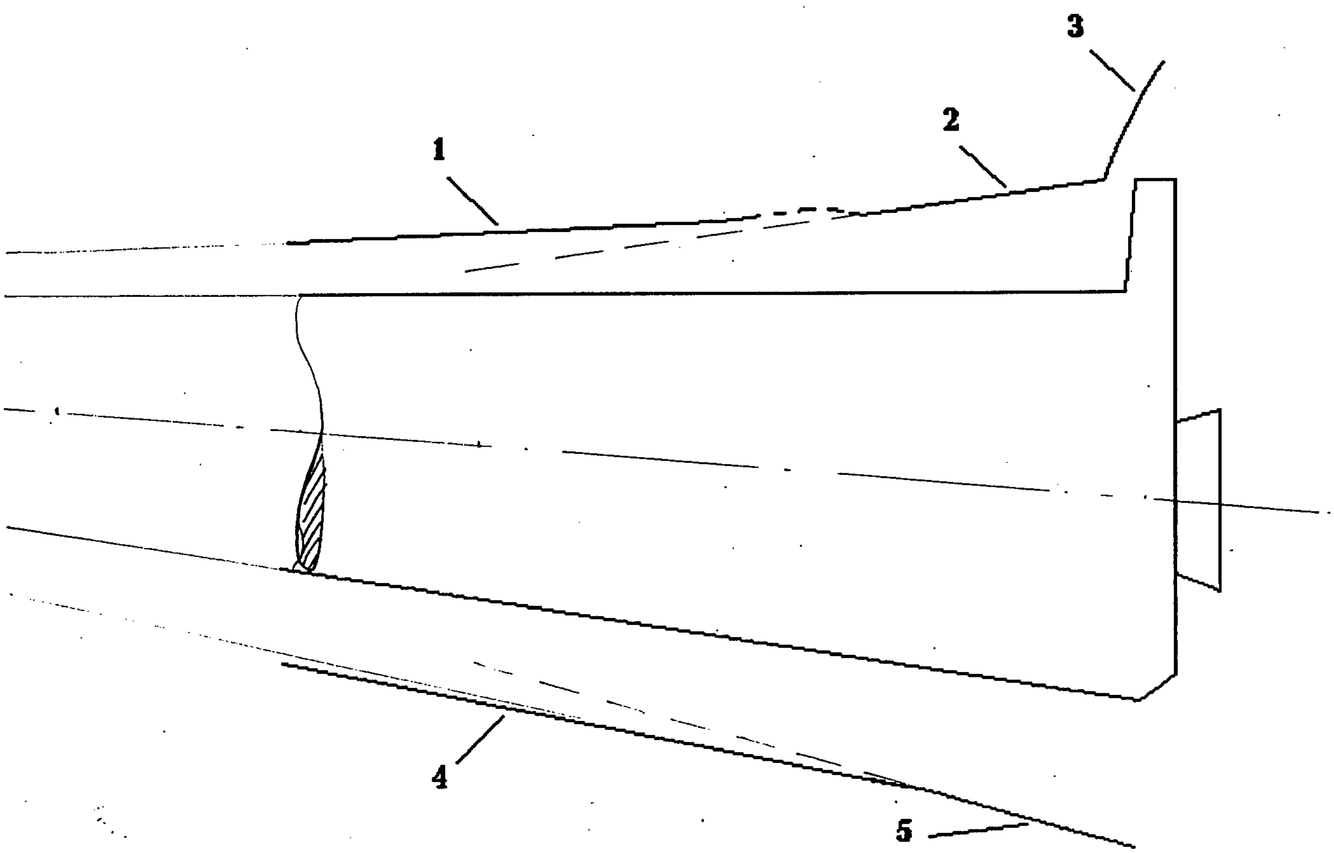

Fig. 11

Flow patterns over 5°-cone model with flap at different t_w

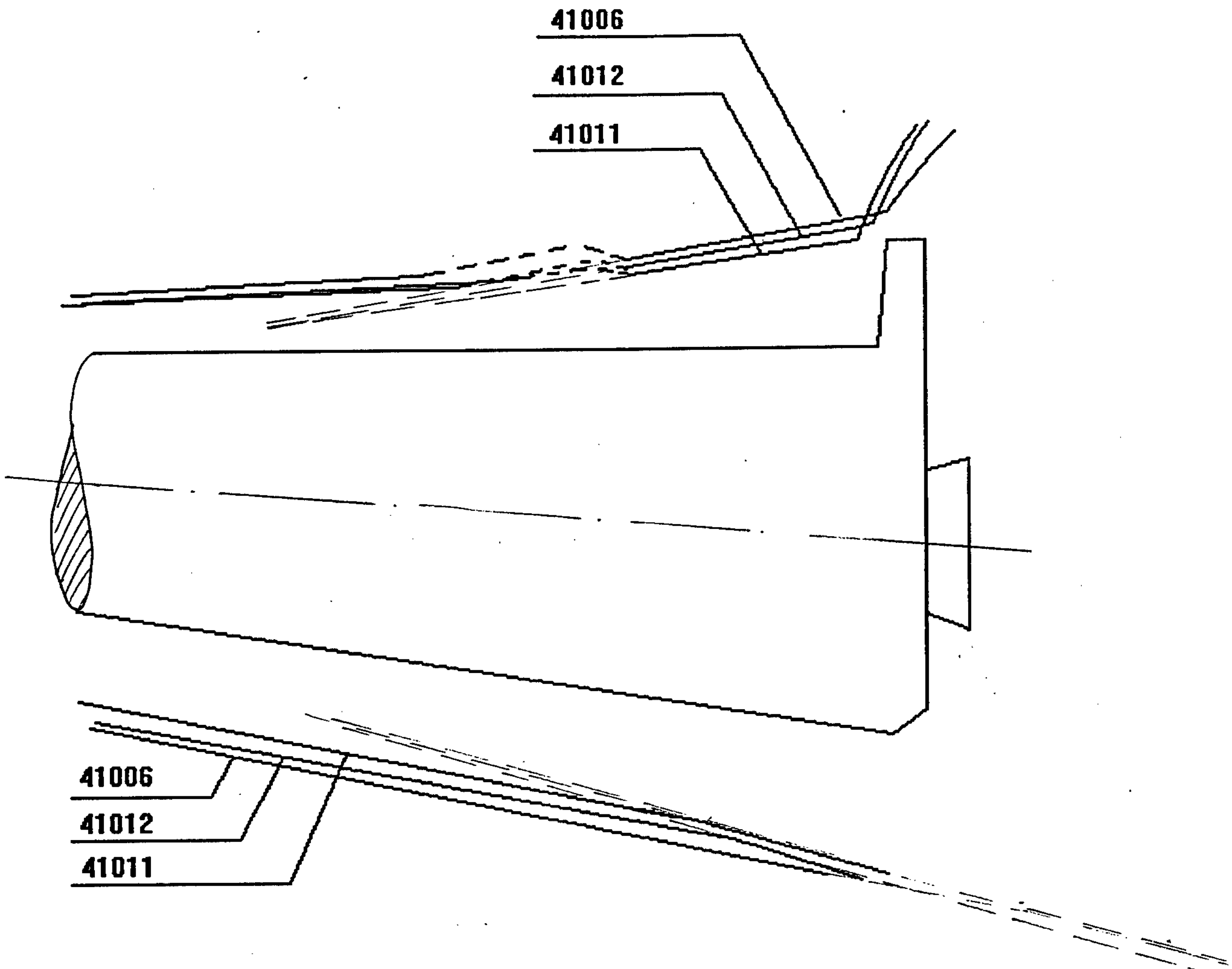

Fig. 12

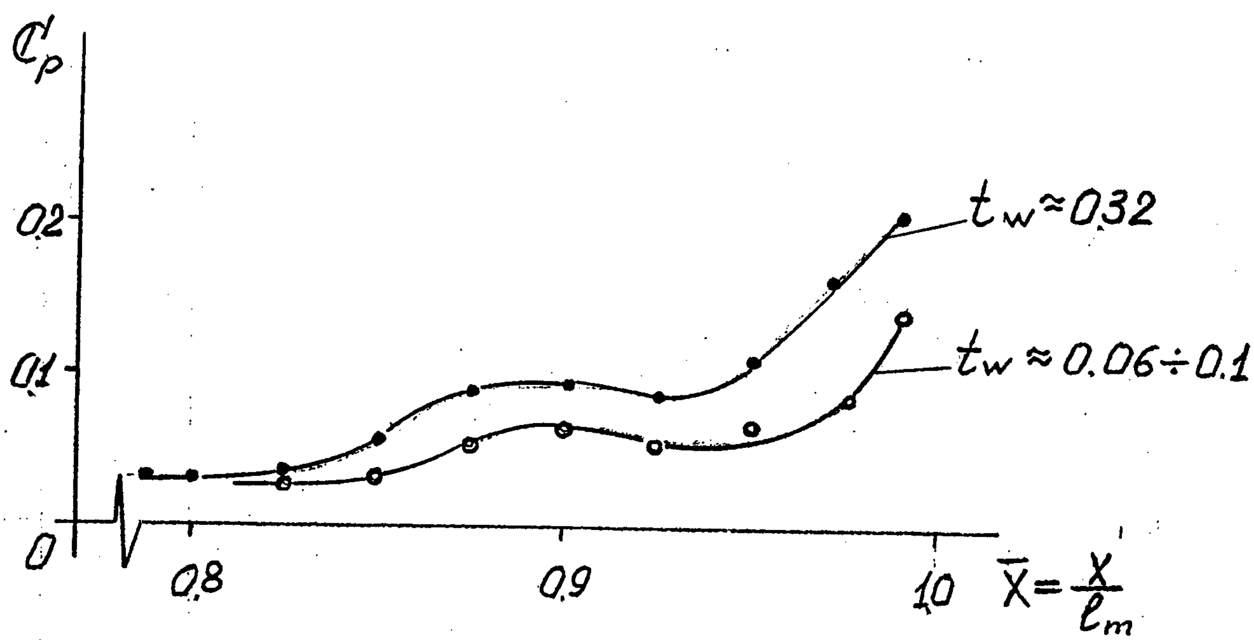

Fig. 13 Side model pressure distributions upstream of flap at $t_w \approx 0.3$ and $t_w \approx 0{,}06 - 0.1$

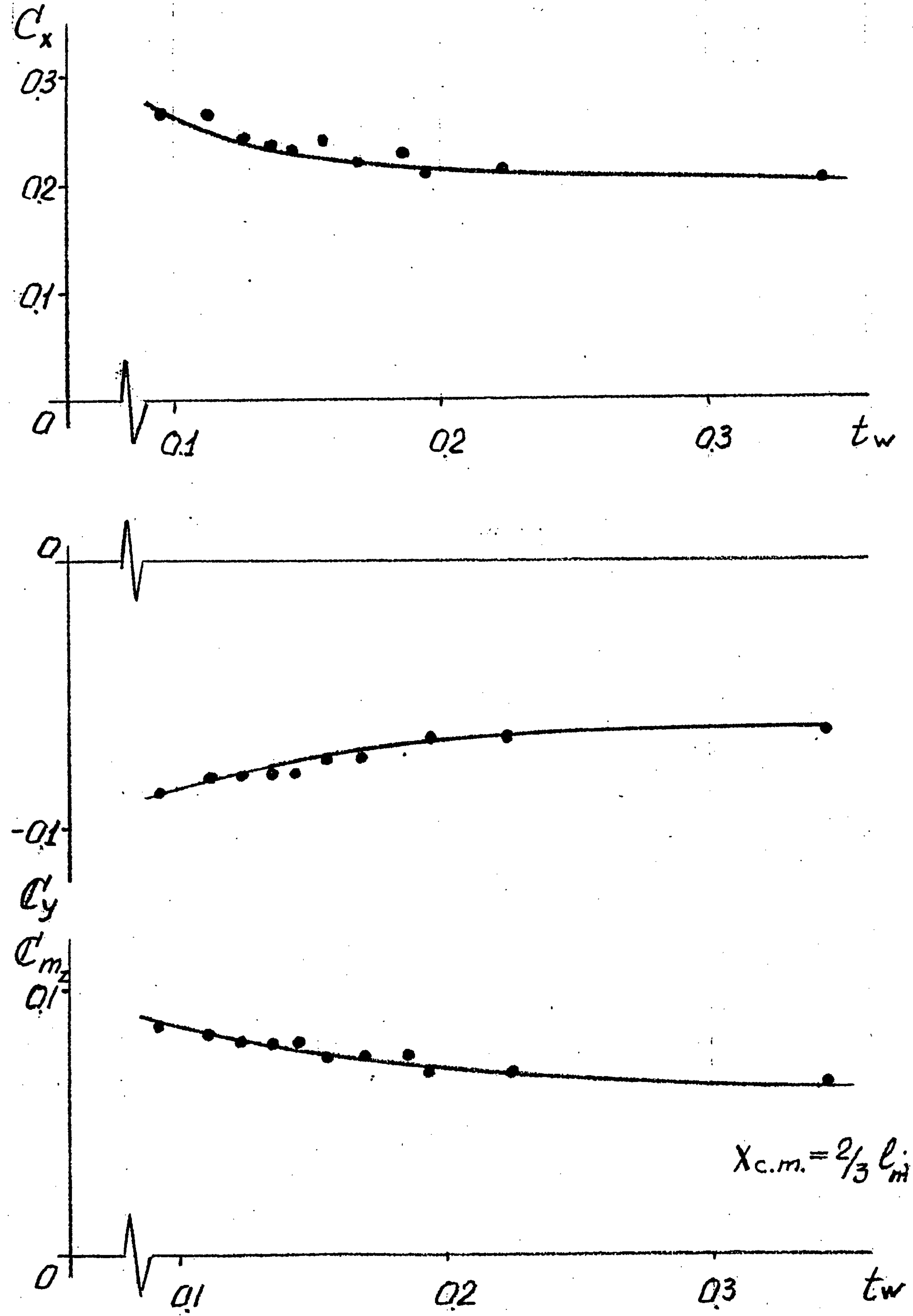

Fig. 14 Aerodynamic characteristics of 5°-cone model with flap as a function of t_w