RECENT DEVELOPMENTS IN TRANSONIC FLOW—PART II—EXPERIMENTAL

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ABSTRACT

Experimental investigation of transonic flow problems has always been an order of magnitude more difficult than the experimental investigation of problems in subsonic or supersonic flow. In the beginning, the generation of transonic flow in the wind tunnel was itself a problem which was later overcome by the development of ventilated wall wind tunnels. Though this development considerably improved the understanding of transonic flow to enable satisfactory development of aerospace vehicles at subcritical speeds, it was found that there were still many lacunae at supercritical speeds in terms of inability to properly correct for wall interference effects and inability to simulate flight Reynolds numbers. With the development of shock-free airfoil by Pearcey and the possibility of efficient supercritical flight speeds, the need to overcome these experimental difficulties become imminent. This paper gives a brief account of the phenomenal progress that has taken place in recent years in overcoming these experimental difficulties.

INTRODUCTION

In part I[1], the phenomenal progress that has taken place in recent years in overcoming many theoretical and computational difficulties that plagued transonic flow studies for quite some time was dealt with. Here, in part II, the experimental aspects of transonic flow investigation, which also posed many difficulties, will be discussed. Some of these difficulties were overcome in the early fifties enabling qualitative understanding of the transonic flow phenomena through experiments in transonic wind tunnels. There was, however, still considerable lacuna and limitations in these facilities because of which it was not possible to get accurate quantitative data from these tunnels. But, during the sixties, there was a lull in further developments, since, as discussed in part I[1], the possibility of sustained flight at supercritical speeds appeared remote. However, the experimental demonstration of shock-free airfoils by Pearcey[2] provided the impetus for renewed vigorous efforts not only on the theoretical and computational front as described in part I, but also on the experimental front as well. As a result of this renewed effort there has been phenomenal progress in the experimental field[3]. The main theme of this paper is to highlight the recent developments that have taken place. But to provide the necessary background a brief discussion of the status of this field prior to these developments and the need for overcoming many of the then prevailing limitations would be presented.

EARLIER STATUS OF THE FIELD

Because air is a compressible fluid, as the velocity of air is increased, not only is the pressure decreased, but the density also decreases. For this reason (without going into too much detail) if there is sufficient pressure difference across a flow channel, the velocity reaches sonic value at the minimum cross sectional area (called throat) and it would be subsonic upstream of it and supersonic downstream of it. That is why a supersonic nozzle has a convergent-divergent shape. A model kept in a conventional wind tunnel with solid walls forms a throat at that section: even for supersonic nozzle of Mach number close to unity. Thus test Mach numbers in the neighbourhood of one cannot be attained in solid-walled tunnels. This phenomenon, called choking, can however be prevented in an open jet tunnel (figure 1), but it provides slightly unsteady flow and the power requirements are higher. Apart from this, at subsonic speeds, the solid wall and open jet boundaries generate respectively higher and lower velocities over the model relative to that obtained in free flight. This interference due to boundaries is rather high at transonic speeds. At low supersonic speeds, the shock emanating from the

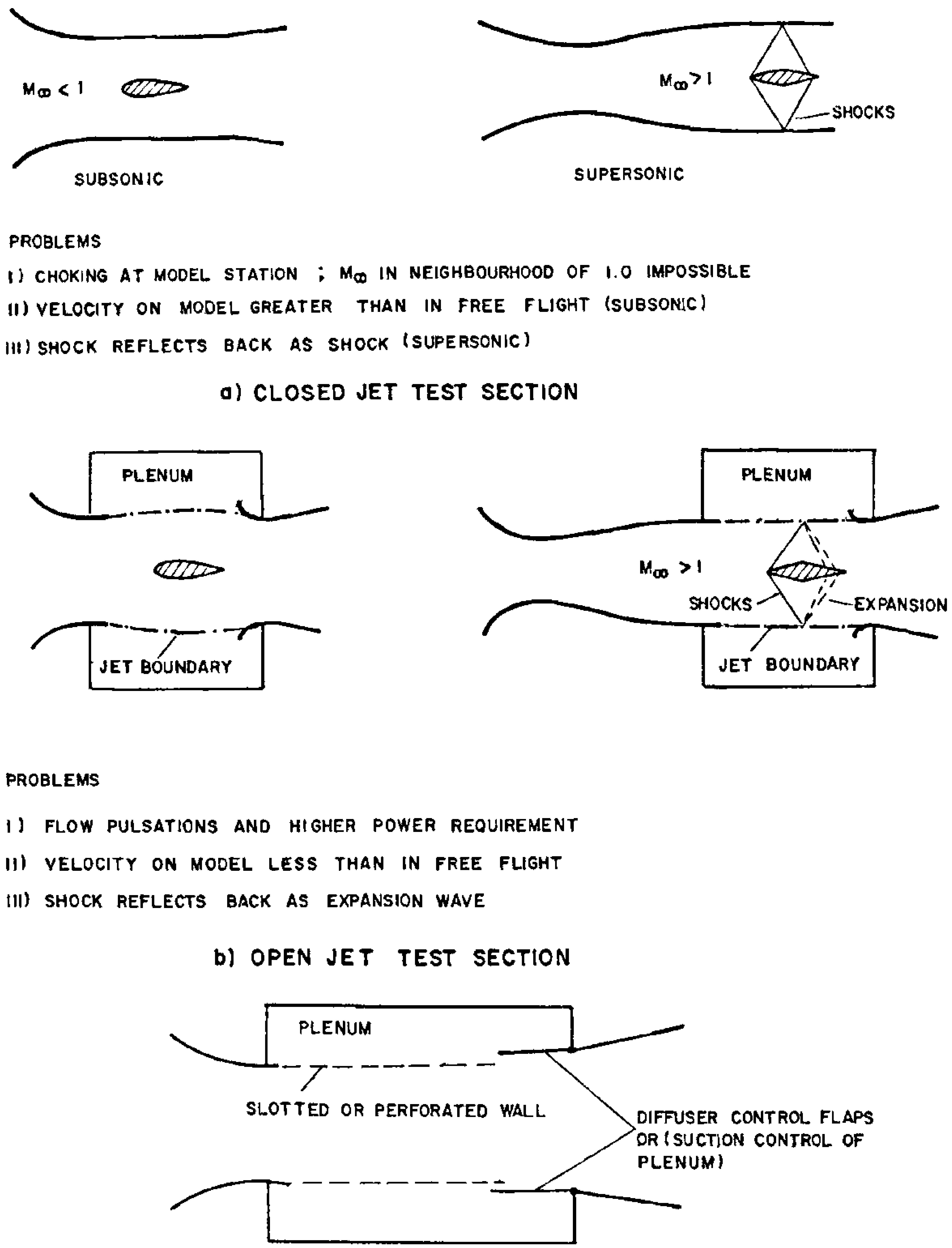


Figure 1. Development of ventilated transonic test section.

nose of the body reflects back from the boundary (as shock in case of solid wall and expansion wave in case of open boundary) on to the model, and invalidates the results. Thus, the interference produced by the solid wall boundary and open jet free boundary are opposite in character. Therefore, the development during 1950's of ventilated wall tunnels (slotted or perforated which is between closed and open jet test section) was the natural course (figure 1). This prevented choking, reduced interference and permitted transonic testing. For a more detailed account of such transonic wind tunnels, one may refer to the book by Goethert[4]. The development of these tunnels contributed enormously to a better understanding of transonic flow phenomena.

However, the interference though reduced was still large enough and some interference corrections to model data were needed. An estimate of these corrections was obtained by using, the same linear theory that was used in subsonic tests in conventional tunnels but with appropriate boundary conditions to represent the ventilated walls. These boundary conditions are

$$\frac{\partial \phi}{\partial x} + K \frac{\partial^2 \phi}{\partial x \, \partial y} = 0 \text{ for slotted walls,} \tag{1}$$

$$\text{and, } \frac{\partial \phi}{\partial x} + \frac{1}{R} \frac{\partial \phi}{\partial y} = 0 \text{ for perforated walls,} \tag{2}$$

where ϕ is the perturbation potential, the gradient of which gives the perturbation velocity relative to free stream velocity, x is along the wall and y is normal to the wall, K and R are parameters which depend respectively on the slot geometry and open area ratio of the perforated walls.

A detailed review of this classical method of estimating the interference corrections was given by Garner *et al*[5] and Pindzola and Lo[6]. The various methods adopted in practice to determine the parameters K and R are summarised in ref. 7. References 8, 9 and 10 give additional information on this subject. Though these methods were reasonably acceptable for tests at subcritical speeds, they were not considered good enough to provide accurate data at supercritical speeds as would be explained in the next section.

Further the model Reynolds number that could be obtained was well below the flight Reynolds number, but were high enough to enable extrapolation of drag data to flight Reynolds number for subscritical speeds. But as will be seen in the next section, this Reynolds number capability was not sufficient to simulate proper flow at supercritical speeds.

NEED FOR FURTHER DEVELOPMENTS

In this section we shall bring out inadequacies of the then existing facilities and test techniques to meet the objective of obtaining accurate data at supercritical speeds so that one can have a better appreciation of the great progress that has taken place in the recent past to overcome these drawbacks.

a) *Limitations of interference corrections based on linear theory*

The parameters K and R used in classical linear theory of wall interference are affected by wall boundary layer[11,12], whose development depends on the pressure field induced by the model. Hence it is not possible to determine these parameters with any certainty. In view of this, the estimation of wall interference, though reasonably acceptable at subscritical speeds, was found to be inadequate at supercritical speeds. In fact, the results obtained on a supercritical airfoil[13] in a 2-D slotted tunnel of 8% open area ratio (a value considered satisfactory from linear theory) were so distorted that it could not be matched with interference free results for any M_∞ and α. These studies showed that the open area ratio of the walls had an enormous effects on the pressure distribution on the model (see figure 2) and an open area ratio in the range of 2–2.5% gave distortion-free results (not necessarily interference free). The experience with perforated walls[14] has also been similar.

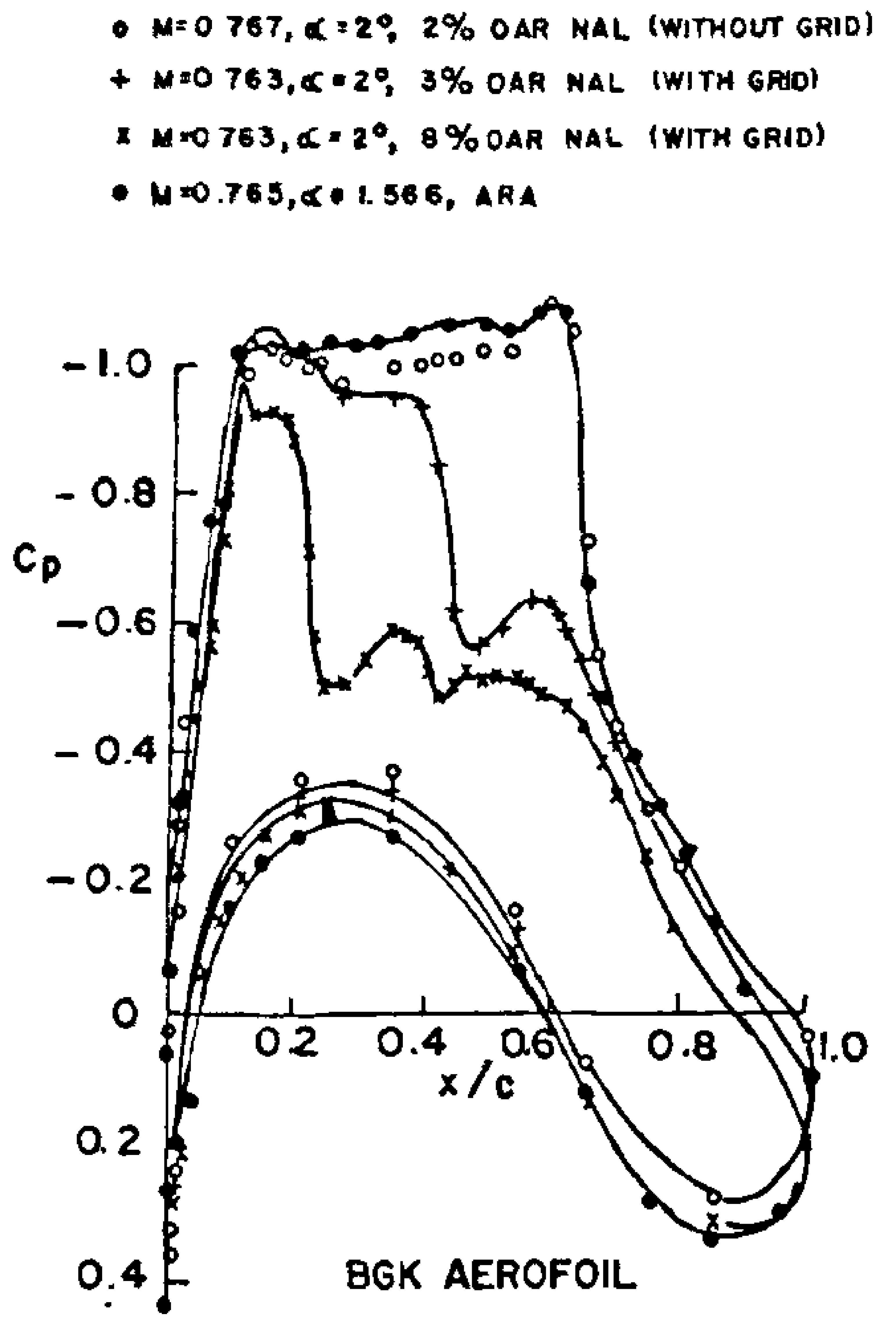


Figure 2. Effect of open area ration on C_p distribution.

Therefore it became essential that one had to find new means of overcoming this wall interference problem to generate accurate data, particularly at supercritical speeds.

b) *Effect of non-simulation of flight Reynolds numbers*

The effect of the non-simulation of flight Reynolds number on the qualitative flow over an airfoil at subcritical and supercritical conditions is shown in figure 3. In most cases, an artificial roughness (minimum required) is incorporated near the leading edge of the model to make the boundary layer just turbulent but not add excessive drag due to protuberance of the roughness. The displacement effect of the boundary layer is higher at the lower model Reynolds number compared to that at flight Reynolds number.

At subcritical speeds, the qualitiative flow pattern does not however alter much and methods have been developed to extrapolate the results obtained at model Reynolds number to flight conditions. On the other hand, as can be seen from the figure, at supercritical speeds at which shocks occur on the surface of

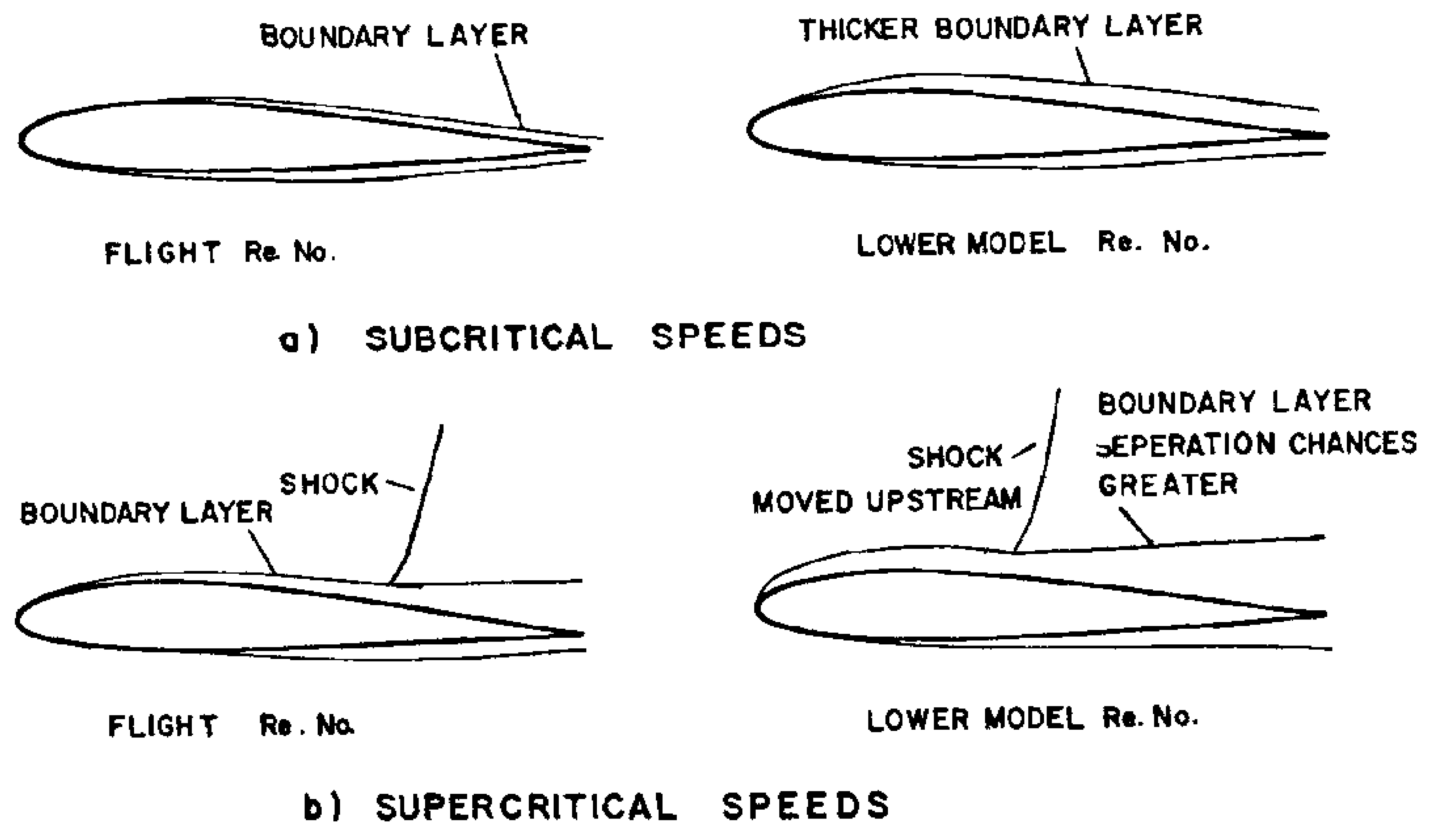


Figure 3. Qualitative effect of non-simulation of flight Reynolds number on flow pattern.

the body—because of shockwave boundary layer interaction—the qualitative flow pattern itself changes. The shock occurs considerably farther upstream on the model compared to that under flight conditions. Therefore, there will be a large effect on the quantitative data as seen from figure 4 (from ref. 15) which shows the comparison of the pressure distribution on the wing of C. 141 in flight compared with model data. No methods exist for applying these large corrections. Therefore it is necessary to simulate in wind tunnels, high Reynolds number close to the flight values.

The various developments that have taken place in recent times to overcome the wall interference problem and obtain flight Reynolds number in Wind tunnels are described in the next section.

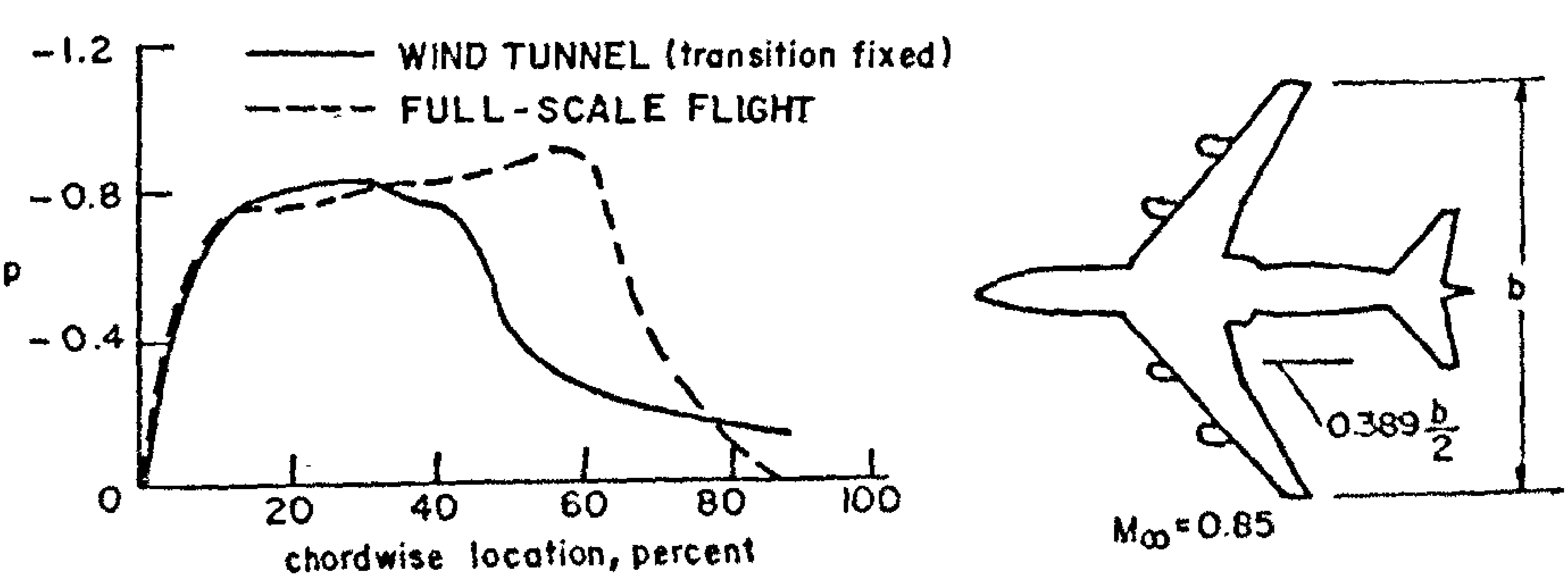
RECENT DEVELOPMENTS

Estimation of ventilated wall interference:

A recent development to overcome the uncertainty in the wall boundary condition of the linear theory is to make pressure measurements close to the wall surface and use this data as the wall boundary condition for assessing the interference effects. At present the linear theory continues to be used, the justification being that the wall is far enough from the model and perturbations at the wall are small enough for linear theory to be applicable. In this method, the model is represented by singularities whose strength depend on the force or pressure data taken on the model. Then, in effect, the distribution of wall singularities is obtained to satisfy the measured wall pressure data. The induced interference flow field at model location, due to these singularities at the wall, is then determined. Based on this, corrections to free stream velocity, angle of attack etc are determined. The question of proper and adequate singularity representation for the model, particularly for three-dimensional models of the aircraft type, is presently receiving attention. An overview of these methods is given in ref. 16. Extending this approach using nonlinear transonic theory is also being attempted[17].

Adaptive wind tunnels:

Another approach to the tunnel wall interference problem is to eliminate it by adapting the walls of the tunnel under each model flow condition such that the stream lines close to the wall correspond to unconfined interference flow over the model. The general idea of using adaptive (using flexible) walls for reducing interference was thought of[18, 19] in the early 1940s at the National Physical Laboratory, U.K. But the shaping of walls was carried out to match with the stream lines based on linear theory using model representation by singularities. Because of limitations and uncertainties of linear theory and proper model representation, this approach did not gain much acceptance at that



time. However, the situation has now completely changed and this approach is now considered very promising and much progress has in fact taken place. There are two main reasons for this change in attitude. The first is the principle put forward by Ferri and Baronti[20] and Sears[21] (figure 5), independently, that by comparing the measured flow variables at the walls of adaptive wind tunnels with the computed flow variables over the exterior contour of the walls and iteratively modifying the shape of the walls until the two match, one would be contouring the walls to match with the streamlines of the infinite interference free flow field over the model. It is rather important to note that, in this approach, *no information on the flow over the model (not even the forces acting)* is needed. The second reason is the development of nonlinear potential flow methods used to compute transonic flow over the exterior contour using the now available fast powerful digital computers. This is a good example in which computation and experiment have been blended to push the frontiers of the field. The only limitation of this approach is that the shock on the model should not reach the wall. There are two reasons for this limitation. First, if the shocks reach the wall, then the irrotational potential flow method used for computing the external flow would not be strictly speaking valid. Perhaps the more important point is that it would be very difficult to incorporate in the adaptive wall teachnique, sudden changes in the flow direction across the shock: in general the shock could be oblique.

Broadly speaking, there are two approaches to adaptively modify the effective stream lines near the wall. One of them is to make the solid walls flexible so that they can be contoured as required. The other approach is to have ventilated wall test section whose plenum is divided into many segments and alter the stream lines near the wall by controlling the flow field in the vicinity of walls through suction or blowing from the various segments. Ref. 22 which deals with the use of computers in adaptive wall wind tunnels gives a good summary of the various adaptive wind tunnels that are existing in the world. In the case of adaptive wind tunnels using ventilated walls with segmented suction or blowing, two flow measurements near the wall would have to be carried out, in place of the wall pressure and displacement data obtained in case of solid flexible walls. These flow measurements could be either the magnitude of the velocity and its direction at one surface near the wall surrounding the model or the x component of the velocity along two surface surrounding the model or some other equivalent measurements. Apart from details, the principle is the

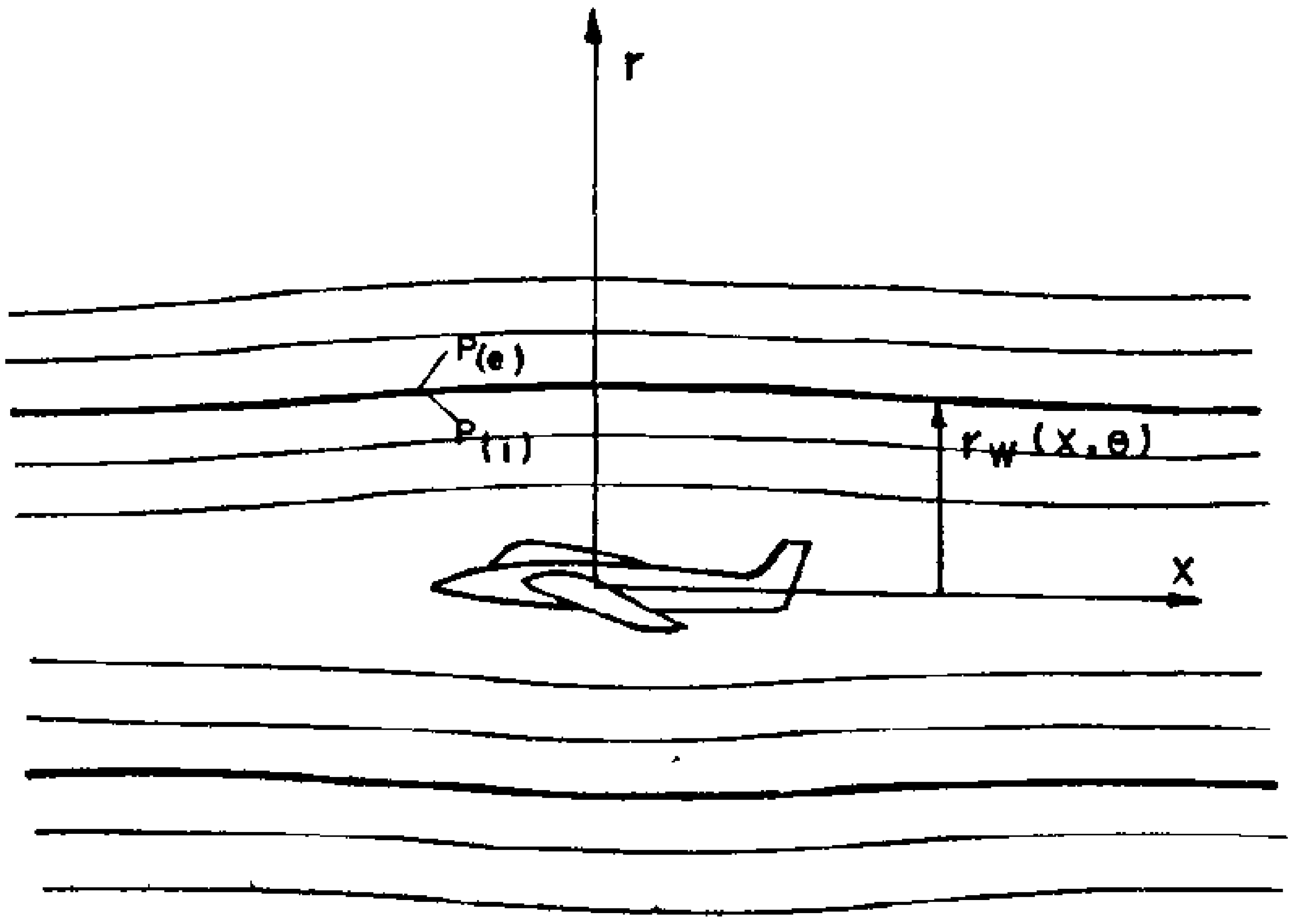


Figure 5. Principle of adative wall technique.

same viz making the wall streamlines to match with the unconfined infinite free stream flow.

The flexible wall tunnel is perhaps ideally suited for 2-D testing and the ventilated wall test section with segmented suction is probably simpler for 3-D testing. But as given in ref. 22, quite a variety of adaptive wall wind tunnels are in existence. Table 1 taken from ref. 22 gives a list of the various tunnels and their general characteristics. The use of thick-walled rubber tube supported and deformed by a set of 64 jacks as an adaptive 3-D test section (figure 6) is one example of the many ingenious techniques that are being tried. It should be mentioned that due to 3-D refljef, the perturbation velocities near the wall are much less in 3-D testing compared to 2-D testing. Therefore, for using adaptive wall technique for 3-D testing, a need for improving the resolution of instruments which measure the flow conditions near the wall is being felt.

It may be mentioned that the principle of adaptive wall wind tunnel is strictly speaking valid only when the test section length is infinite, but finite length test sections do not introduce too much error. Also, effect of finite length of the test section can be taken into account.

The use of adaptive wall wind tunnel, apart from reducing or nearly eliminating interference, also permits much bigger models to be tested. Thus the Reynolds number capability of the tunnel also gets considerably increased. A spin-off from this aeronautical development is that this principle has been used in an automotive wind tunnel, where good results

Table 1 *Adaptive wall wind tunnels*

2-D Test sections:

		Test section dimensions [cm] and characteristics			
		Cross-Sect.	L.	Walls	No. of Controls
AEDC	1 foot	30.5×30.5	95	perforated	global + local
CALSPAN	1 foot	30.5×25.4	142	perforated	10 + 8 Plen. Ch. Comp.
NASA Ames I	13×25 cm	13 ×25	74	slotted	10 + 10 Plen. Ch. Comp.
NASA Ames II	2 feet	61 ×61	153	slotted	16 + 16 Plen. Ch. Comp.
NASA Langley	30 cm	36 ×36	144	flexible	19 + 19 Jacks
ONERA S4 LCh	18 cm	18 ×18	75	flexible	10 + 10 Jacks
CERT (ONERA) T2	40 cm	37 ×39	132	flexible	16 + 16 Jacks
SOUTHAMPTON Univ.	6×12 in	15.2×30.5	107	flexible	15 + 15 Jacks
SOUTHAMPTON Univ.	6 in	15.2×15.2	112	flexible	19 + 19 Jacks
T.U. Berlin	15 cm	15 ×15	69	flexible	8 + 8 Jacks

3-D Test sections:

	Test section dimensions [cm] and characteristics			
	Cross-Sect.	L.	Walls	No. of Controls
AEDC/CALSPAN	30.5×30.5	95	4 perforated	64 wall segments
NASA Ames I	13 ×25	74	2 slott., 2 sol.	44 Plen. Ch. Comp.
SOUTHAMPTON Univ.	15.2×15.2	107	2 flex., 2 sol.	40 Jacks
T.U. Berlin	15 ×18	83	8 flexible	78 Jacks
Sverdrup	30.5×61	245	·12 flex. slats	204 Jacks
Wright Aer. Lab.	24 ×24	130	2 rodwalls, 2 sol.	180 Jacks
DFVLR Göttingen	80 diam.	240	rubber tube	64 Jacks

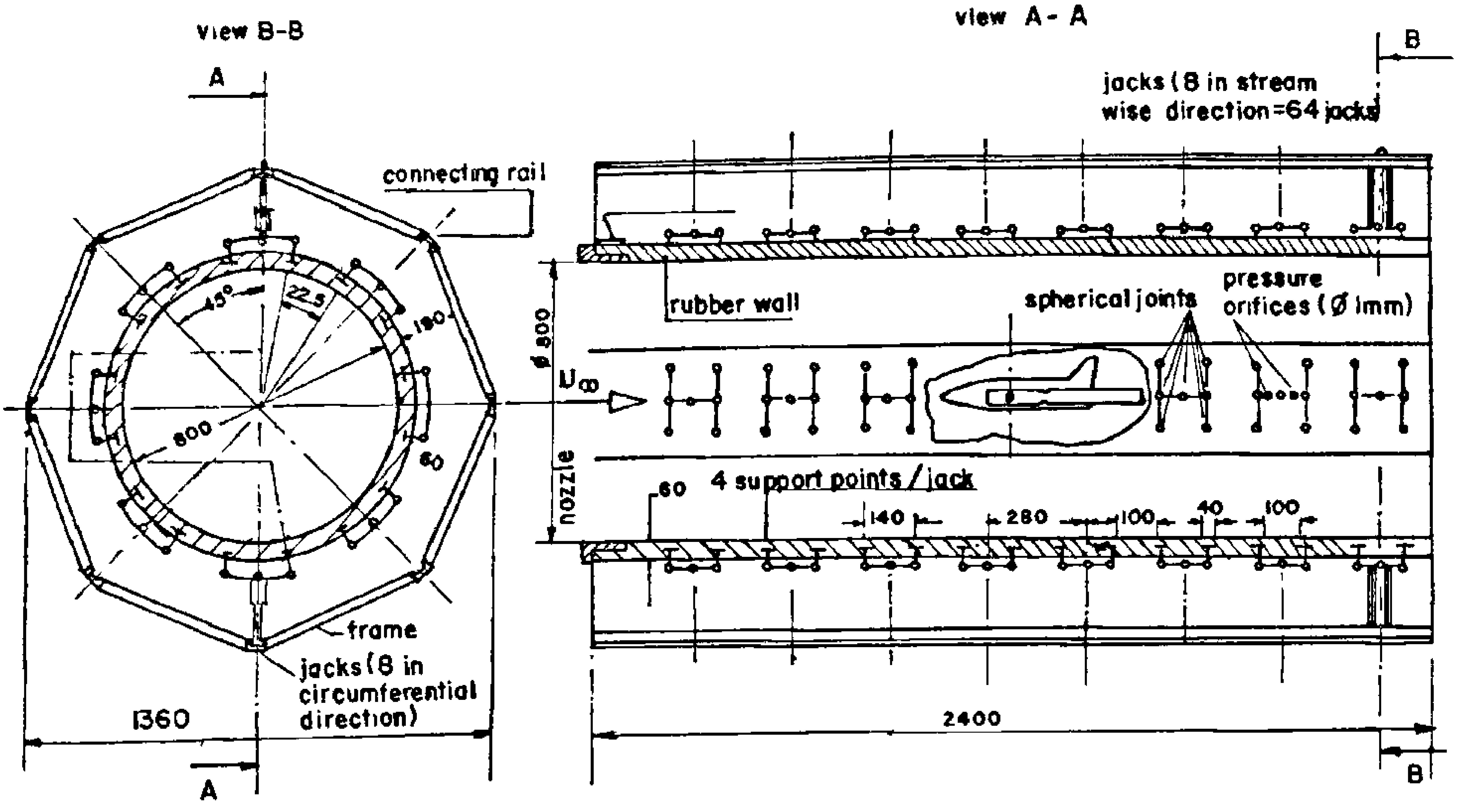


Figure 6. Sketch of adaptive rubber tube test section.

have been obtained on automobile models having tunnel blockages of the order of 30% (ref. 23).

Simulation of flight Reynolds number—Cryogenic tunnel

The enormous gap in Reynolds number between that needed and that which existed until recently is indicated in figure 7 (from ref. 3). The Reynolds number is $\rho VL/\mu$ or $\rho MaL/\mu$, where ρ is the density, V is the velocity, L is the characteristic length, μ the viscosity of the medium, M is the Mach number and a is the speed of sound. Mach number simulation being even more primary, only the remaining parameters affect the Reynolds number.

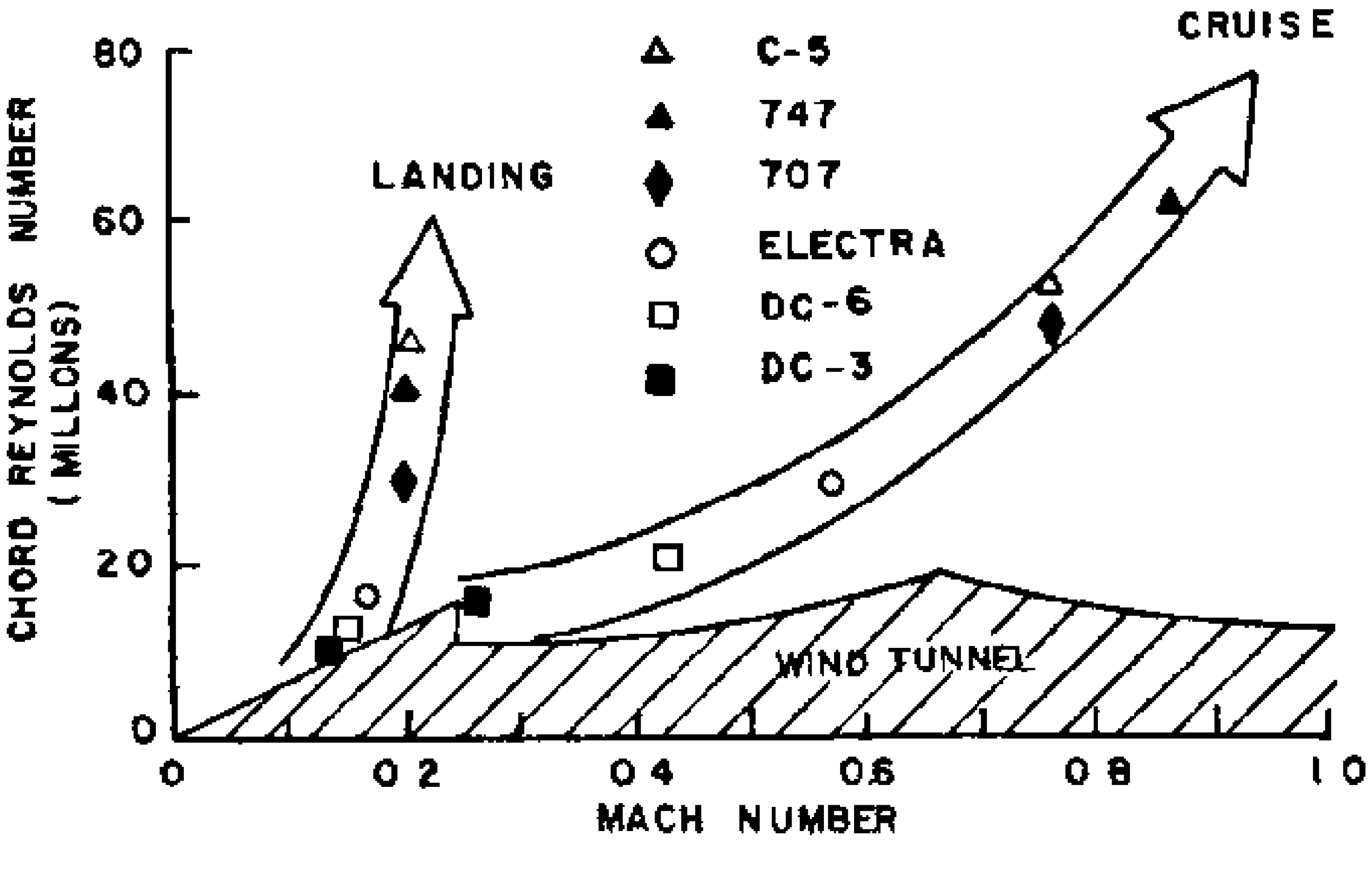


Figure 7. Reynolds number GAP (NTF excluded).

Since the cost of tunnel varies as L^3, increasing the size of the tunnel beyond a certain limit makes it prohibitively expensive; also inconvenient for operation. Therefore, the one important parameter by which the Reynolds number can be increased is to increase the density of the fluid medium. There are essentially two approaches to increase the density ρ; one is to increase the pressure P, the other is to decrease the temperature T ($\rho \propto p/T$). A brief review of the many studies that were made during 1970s using both these approaches has been given by Murthy[24].

The development[24] of Ludweg tube, the Evans clean tunnel, hydraulic driven tunnel etc which were all different approaches to the essential idea of operating at high pressures proved to be not too successful mainly for the following reason. Ambient temperature being the same, and Mach number being the same, increasing ρ by increasing p, increases the dynamic head proportionately. Hence model loads also increase. The strength of the model and excessive aeroelastic deflection of model limits on the maximum operating pressure to about 5–10 atmospheres. So extension of Reynolds number capability by this approach is limits the maximum operating pressure to these facilities was the short duration of test time—much less than a second.

On the other hand, increasing the density by reducing the temperature, reduces the velocity V also since V is proportional to $T^{0.5}$ (figure 8). Therefore the dynamic head which is proportional to ρV^2 does not change. Therefore the high Reynolds number can be obtained without increasing the model loads. Also the power needed to drive the tunnel, proportional to ρV^3, actually reduces somewhat. It should be noted that though V decreases with temperature, ρV increases with temperature as $T^{-0.5}$. Hence Reynolds number, which depends on ρV, increases with decrease of temperature. It may be noted that reducing the temperature reduces μ also. Hence it has a beneficial effect in further increasing the Reynolds number. On the whole, as can be seen from figure 8, the use of cryogenic temperature can increase the Reynolds number capability by an order of magnitude.

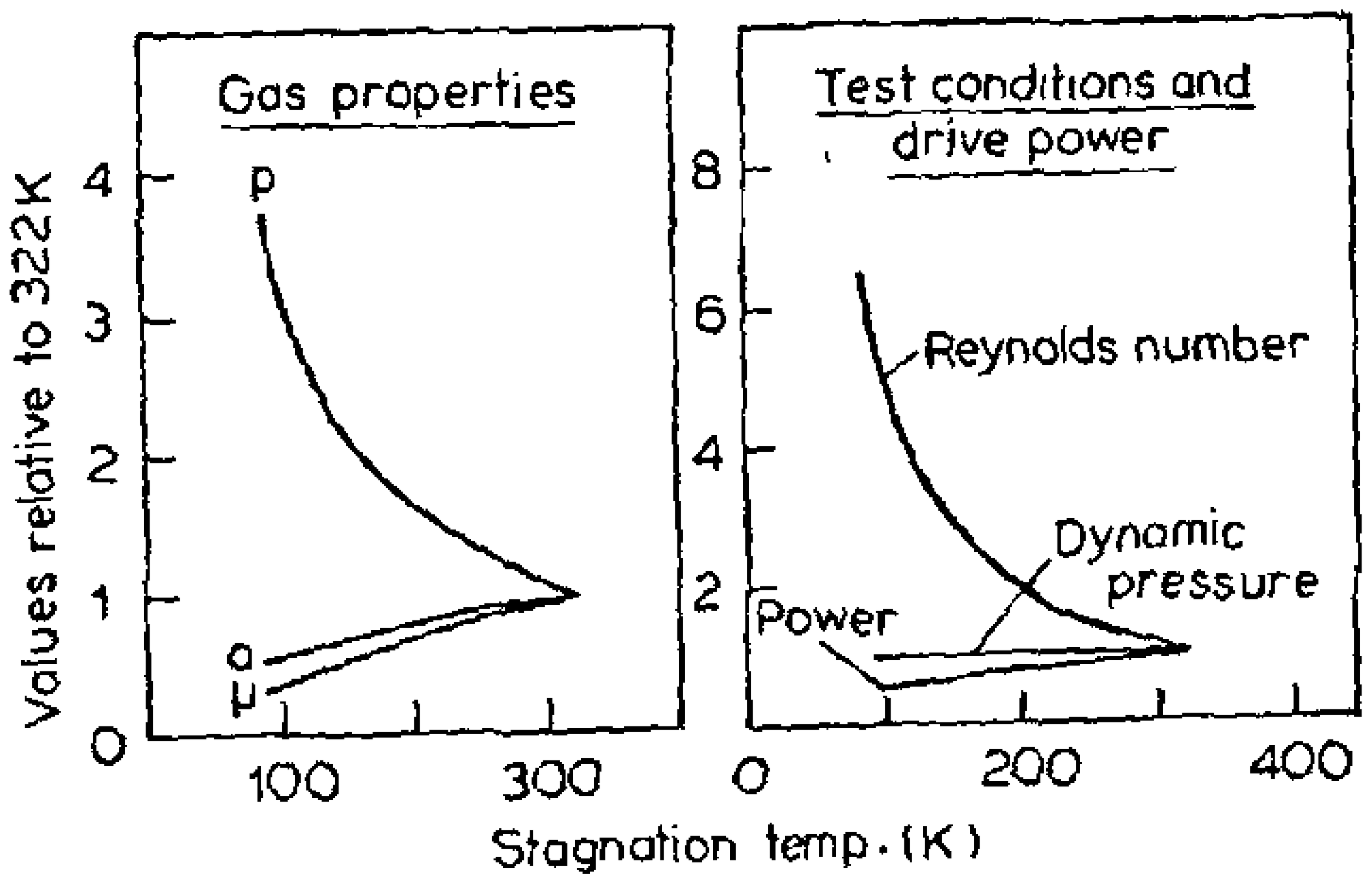


Figure 8. Effect of temperature reduction $M_\infty = 1$ (constant stagnation pressure and tunnel size).

This concept of obtaining high Reynolds number by reducing the gas temperature was successfully demonstrated[25, 26] through the use of cryogenic nitrogen as the working medium, in the pilot low speed and transonic tunnels at NASA Langly Research Centre. As a result, the development of a 2.5 meter cryogenic

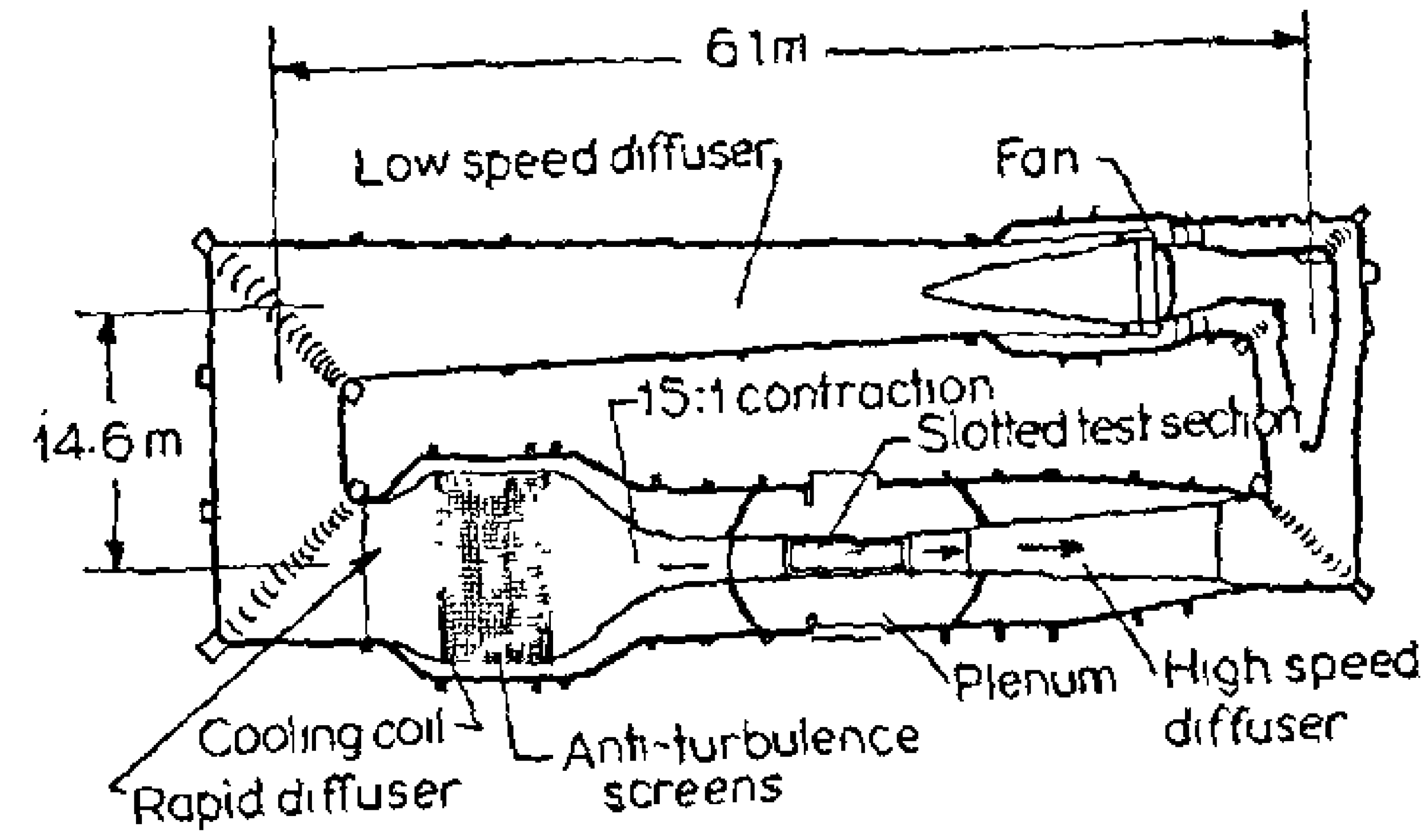


Figure 9. NTF tunnel circuit.

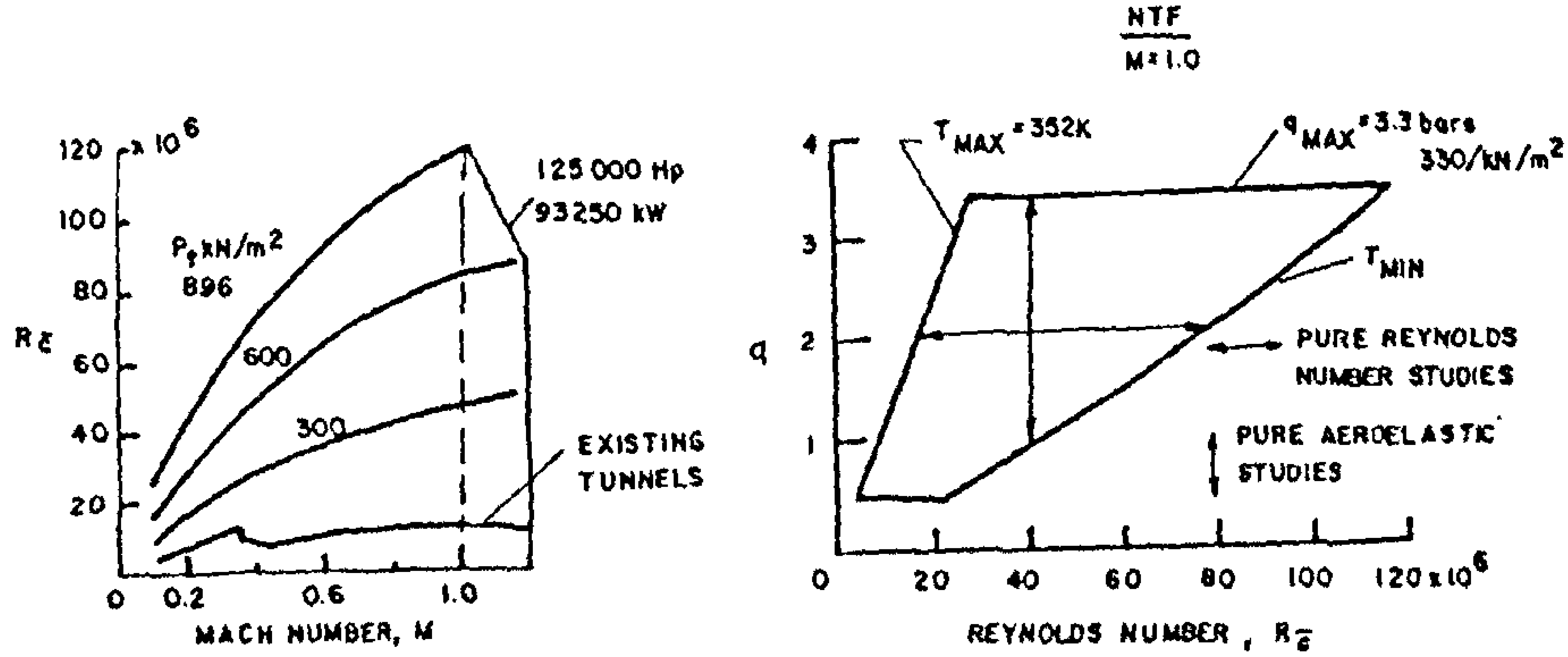


Figure 10. Operating range of NTF.

Table 2 *N.T.F. Facility*

Test section: 2.5 m × 2.5 m (Slotted walls)
Test gas: Nitrogen and air
Max. pressure: 130 psi
Min. temperature: Minus 320°F (−195°)
Max. Horse Power: 126000
Speed range: 85 to 850 mph
Max. Mach number: 1.2
Max. Reynolds number: 120×10^6

tunnel, as a national transonic facility was taken up in United States and has just been commissioned. Brief particulars of this facility are given in table 2. Figure 9 shows the general layout of this tunnel. This tunnel with the capability to vary the temperature and pressure independently provides an unique opportunity to study the effects of Reynolds number and dynamic pressure (aero elastic effects) independently (figure 10). It can be seen that full scale Reynolds number can be achieved.

Though the cryogenic concept is simple in principle, practical utilization of it in the national transonic tunnel required solution to many technical problems, such as good internal insulation for the tunnel, the access to the model without having to purge the tunnel circuit of nitrogen medium; design of models and instrumentation to operate at cryogenic temperatures etc. Details on these aspects are contained in references 27 and 28. The successful development of this facility is indeed a technical marvel. This facility would be of immense value not only to generate data on specific projects but also in furthering the frontiers of knowledge in high Reynolds number transonic flow. It may be mentioned that there are plans to build a similar, though slightly smaller, cryogenic transonic wind tunnel in Europe also.

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ANNOUNCEMENTS

NATIONAL DOWN'S SYNDROME CONGRESS

This Congress will be held at the Institute of Genetics Hospital for Genetic Diseases, Osmania University, Hyderabad, during 8–10 September 1987. Abstracts of papers are invited on Educational, Psychosocial and Biomedical aspects. The last date for receipt of abstracts is 30th May 1987.

Further particulars may be had from: P. Usha Rani, Clinical Psychologist and Organizing Secretary, National Down's Syndrome Congress, Institute of Genetics, Hospital for Genetic Diseases, Osmania University, Begumpet, Hyderabad 500 016.

TECHNICAL INFORMATION ON NICKEL AND ITS APPLICATIONS

Nickel Development Institute, Canada and Zinc Development Association, London have joined hands to provide technical information on nickel and its applications to industries in India and neighbouring countries from now on. The needful information will be serviced by the Indian Lead Zinc Information Centre, the Indian Branch of Zinc Development Association, London. For details please contact: Indian Lead Zinc Information Centre, B-6/7, Shopping Centre, Safdarjung Enclave, New Delhi 110 029.