DESIGN OF AERODYNAMIC TEST FACILITIES

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ABSTRACT

Very simple and basic theoretical considerations will determine the design features of an aerodynamic facility.

Desired Speed/Mach Number range, flight altitudes, Reynolds No etc. will call for different tunnel requirements with regard to model size, temperatures, pressures and energy supply.

For certain types of tests there will be special limitations which will be defined and discussed.

Restrictions can be of an economic as well as of a physical nature. High temperature simulation, stability and flutter belong to this category.

An attempt will be made to give a survey of expedient combinations of equipment and devices for covering this vast area of aerodynamic testing.

In the original planning of an aerodynamic facility it must be decided upon a long term stategy so that an optimum solution - with alternative development potentials - could be realized in stages.

In such basic facillties, the objectives can easily be changed and accommodated to comply with new strategies or R & D policies.

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INTRODUCTION

The difficulties in providing test facilities which could cover all aspects of aerodynamic testing have been presented in a graph, where flight altitude versus Mach Number is demonstrated with parameter curves for constant pressures and temperatures. This paper will only cover the subsonic, transonic and supersonic regimes. (Fig.1).

Each aerodynamic facility can be defined by or related to one or more characteristic features. In the subsonic flow the Reynolds Number is a dominating parameter, in the transonic and supersonic fields the Mach No is the important parameter.

Many tunnels are available today among others for aeronautical engineering, building aerodyanmics, meteorology, agriculture, automotive engineering etc. In order to have realistic conditions during testing, full scale Reynolds Numbers are generally desired.

In order to understand fundamental physical phenomena water tunnels have been used as an introduction to a problem area. A flow visualization technique, developed for this medium, partly explains the success of this tunnel. (Fig.2).

Subsonic facilities are vital instruments for testing difficult problems during the starting and landing of aircrafts under a variety of aggrevating conditions without risking lifes.

The transonic range is characterized by the simultaneous presence of subsonic and supersonic flows. By means of such a definition the range can in principle extend from $M = 0.3$ to rather high "supersonic" Mach Numbers. Subsonic tunnels are thus taking over below 100 m/s and supersonic tunnels are taking over, when cheap and simple aerodynamic means of establishing high transonic Mach Numbers are exhausted. Generally between $M = 1.2$ and 1.4.

In the transonic regime the experimental difficulties stem from the models' sensitivity to blockage and the extreme slopes of Mach lines and shocks, which can cause disturbances on the models by reflections via the transonic walls. This range is dominated by shock boundary layer interference problems, and test sections are still subject to subtle research and development works to improve the testing conditions. [1]

In the lower supersonic Mach No range the slope of the Mach lines ($\sin \alpha = 1/M$) are changing rapidly with M, but near $M = 4$ the characteristic lines are changing very little and so are the aerodynamic coefficients, which are flattening out at higher Mach Numbers.

In the subsequent hypersonic Mach No range ($\sqrt{M^2 - 1} \approx M$) thermodynamic phenomena like heat transfer problems, real gas effects, ablation processes will dominate the picture. Here the Mach No is no more a significant parameter.

If no preheating of the air is applied, condensation of the oxygen in the air will put a natural limit to the Mach No in the supersonic range (around $M = 4.2$).

The so called TWIN TUNNEL CONCEPT, however, can accept a supersonic tunnel leg to start with and later on prepare for a separate transonic tunnel leg with a 50% larger test section than in the supersonic tunnel. The two tunnels finally being operated on a common air supply system, whereby the capacity of the air supply system and the financing can be adjusted according to the order and time schedule, in which the facility is going to be realized.

The advantages when compared to older trisonic versions are:

- The transonic unit with the same test section size as in the trisonic tunnel can be designed for a much lower max pressure (4 atm). This pressure being determined by the maximum possible sting/balance forces.

- The weight of this transonic leg, especially when deprived of the heavy and expensive supersonic nozzle, also results in a reduced boundary layer at the entrance of the transonic test section.

- Only the separate variable supersonic tunnel leg, which now could be made much smaller than the conventional trisonic tunnel, must be designed for the high pressure, which is necessary for testing at high Mach numbers (without using ejectors).

- The supersonic nozzle system does not need a choke system or any transonic auxiliary systems.

- It will be ample of time for tests in the supersonic test section when operated on an air supply system planned for a 50% larger transonic tunnel (the test section area being 2.25 times larger).

- The control and data acquisition and reduction system could easily be shared.

In fact two tunnels could be run in parallel roughly at the same cost as one combined trisonic facility. The operational flexibility and the better financial options are obvious.

The charging of large amounts of air in pressure vessels over night or between runs in order to make short intermittent "blow down" tests is naturally attractive from both an installation and from an operation point of view. The continously operating facility is far more expensive and vulnerable.

With modern computer technique and instrumentation there is today little incentive in continuously operating tunnels. Already the installed power and the start up of large electric machines are creating special grid problems. Only in captive trajectory tests or in certain presssure measuring tests time may still be a critical factor.

SUBSONIC TUNNELS

Design Consideration and Quality Requirements

The ever increasing demand for high quality stems from the escalating effect that experimental and theoretical advancements mutually will have upon each other.

In all these domaines, but specially in the subsonic and transonic ranges, the Reynolds No is an important parameter. A lot of expensive efforts have been made in these two ranges to reach realistic levels e.g. by pressurisation (Fig.3) and by applying cryogenic techniques.

In engine development work correct Reynolds Numbers are equally important as realistic temperatures.

The high temperatures at higher Mach Numbers (Fig.4) will make the design of an engine test facility complicated and expensive.

During the planning of basic resources for expedient aeronautical facilities the optimum use of possible options and future development potentials must be carefully considered.

TYPICAL DESIGN FEATURES

Some design features in aeronautical test facilities covering the speed range up to $M = 4$ can be derived from fundamental theoretical considerations. They will also help to explain certain traits in the envelopes of wind tunnels. (Fig.5).

From Table 1 some characteristic changes of the typical design parameters for different wind tunnels are demonstrated.

Specially emphasized are the ever increasing demand (at increasing speed) for higher driving pressure ratios (i.e. for power). At higher supersonic Mach Numbers the more slender nozzles and the rapid change in the area ratio A_1/A_S (test section area/sonic throat area) are other typical features .

The consequence of this rapid change in area ratio (Fig.6) at higher Mach Numbers is that the settling chamber (from being 10 times larger than $A_1 \sim A_S$ near to $M = 1.0$) can be about the same size as the test section when operating near to $M = 4.0$. This is a good reason for making the hypersonic range starting up with axisymmetric nozzles below rather than above Mach No. 4.0.

Another reason being that at higher Mach Numbers ($M = 5$, $A_1/A_S = 25$) the two dimensional nozzles are sensitive to secondary flow phenomena, which are accentuated further due to sealing difficulties and the tiny slot-shaped minimum throat. In an axisymmetric nozzle, however, the boundary layer is spreading out uniformly in the nozzle. Another advantage is that the axisymmetric nozzle is much shorter than the two-dimensional version.

In the past many trisonic tunnels have taken care of the testing of models in the subsonic, transonic and supersonic ranges. A separate transonic test section, however, can much better match the size of a smaller supersonic test section with only one model ideally accommodated to the size of each test section.

The trisonic concept is a compromise offering poor testing conditions in the transonic regime and a too large test section in the supersonic regime.

During 100 years the tunnel design has bee improved thanks to developments in many areas. Diffusor half angles have been reduced to safe values ($\sim 2^\circ, 3$). Larger contraction ratios. better honeycombs and the introduction of expedient gauzes have improved flow quality and turbulence level.

Computer methods for the profiling of the contraction units have made design and manufacturing easy. Experimental results are positive. Thanks to mew traversing-measuring methods corner eddies have been discovered and remedied by using a uniform contraction (i.e. constant aspect ratios through the entire contraction unit).

By all these measures flow quality has been improved and typical figures are:

Total pressure variations	$\pm \Delta P_0$	within 0,2 percent of the dynamic pressure
Static pressure variations	$\pm \Delta P$	within 0,2 percent of the dynamic pressure
Angularity	$\pm\ 0.06^\circ$ $\pm\ 0.12^\circ$	in the vertical direction (1/1000) in the horisontal direction (2/1000)
Temperature	$1^\circ C$	
Turbulence: Longitudinal components		RMS/ u/U < 0,05 percent
Turbulence: Lateral components		RMS/ u/U < 0.1 percent

Computerized data acquisition systems, model manufacturing by NC machines, advanced strain-gauge balances and measuring techniques have made the testing faster, better and easier. As in all physical experiments an error analysis will tell, if the individual accuracies in a measuring sequence are in harmony with one another, so that excessivly expensive equipment not is purchased in vane. The most important development trend is, however,coupled to the tunnel concepts proper.

As a consequence of the "Prandtl School" open test section tunnels of the "Göttingen type" were in common use around 1927.

The closed test section was already used at an early stage and is still the dominant version for high quality aeronautical development.

During the last 30 years slotted walls have been used as a compromise between open and closed tunnels (compare Brown Bovery paper 1942 by Darrieux). The method must be supported by empirical evidence, and the optimum value of open area depends on the current type of test.

The recently (1983) invented "adapted wall" system [2] is the only which can produce experimental data according to the first principle (i.e. without using empirical corrections).

Calibration and Corrections

Reference pressures which could be affected by model attitude variations are placed upstream of the test section (in the downstream end of the contraction unit).

The drag of calibration rakes, traversing gears etc. should be so small that they do not contribute to a change of the natural free stream flow pattern. The same goes for attitude mechanisms, which very often are deteriorating a basically good quality free stream.

The türbulence level is measured by the classical sphere tests or by hot wire anemometry. Relevant frequencies are in this context higher than 5 Hz. It may therefore be important to have the resonance frequency of the sphere under control. The turbulence factor:

$$T.F. = 385.000/R_C = R_C(\text{in free air})/R_C(\text{in wind tunnel})$$

is important to know in order to relate future tests in the tunnel to the natural turbulence conditions.

A tunnel wall correction due to inferference e.g. between the boundary of the tunnel air stream and the down wash of a lifting airfoil has to be made.

The corrections according to Glauert are of the following type:

$$\Delta C_D = (\pm)\ SC_L^2/8A$$

$$\Delta\alpha = (\pm)\ 57.3 \cdot SC_L/8A \quad \text{where}$$

(±) is referring to closed and open trhoat tunnels respectively.

A cross sectional area of jet

S airfoil surface

C_L lift coefficient

ΔC_D drag correction

$\Delta\alpha$ angle of attack

An horisonal Buoyancy correction is made due to the boundary layer growth along the tunnel walls. That will cause a presssure drop along the test section, if no angular corrections of the walls are made. The buoyancy correction would be:

Model volume times dp/dx.

The level of the ground floor is generally horizontal so the B.L. corrections have to be made on the other surfaces. From a flow angularity point of view it is, however, not sufficient to let only the corner fillets in an octagonal section cope with this compensation.

Computerized (panel) methods for the calculation of any type of wings have improved the means to get correlations with refined results. The progress stems to a large extent from the improved accuracies in manufacturing of models by means of NC machines.

In the ambition to increase the Reynolds No ($\rho VL/\mu$), the speed tends to go up. With increasing speed (V) another ratio will be of considerable significance i.e. V/C, where C = speed of sound. This ratio is proportional to the inertia force/elastic force in the fluid, and as such the ratio serves as a criterion of similarity for high velocity flow.

The correction for compressibility in a pitot - static tube for a compressible fluid is $1/\sqrt{1+(M^2/4)}$ which amounts to 1% at about 100 m/s.

In the Glauert - Prandtl approach it could be shown that a body in a compressible flow is equivalent to a thicker body in an incompressible flow, and that the ratio between the two thicknesses would be equal to the $\sqrt{1-M^2}$. For an airfoil this also holds for the thickness, camber and angle of attack, so the lift coefficient of an airfoil at high Mach No would be

$$C_L(M) = C_L(0)/\sqrt{1-M^2}.$$

Practical/Economic Aspects

Smaller wind tunnels have mainly from an economic point of view open test sections. The performance would have been better with a closed test section and a conical diffusor.

Speed could in this way have been increased by 40 to 100% with the same power.

An asymmetric location of an open tunnel without return ducting can cause severe deterioration of the flow pattern. Large outdoor or indoor obstacles can have the same bad influence on the velocity distribution.

To avoid dust and paper to be moved around in the laboratory a closed circuit with corner vanes is a natural though expensive improvement.

For smaller and medium sized low speed tunnels a vertical i.e. a so called up and over type of tunnel is preferred to the horizontal or race track tunnel:

Better accessibility (from both sides)
Symmetrical flow (with regard to roll)
Occupies a small floor or ground area

For atmospheric High Reynolds No tunnels in which temperatures are determined by the ambient temperature and possible cooling arrangements (max 50°C) the optimum choice must be made between size, pressure and speed for obtaining high Reynolds No at a reasonable cost.

The wind tunnel cost (C) when geometry, speed (V) and pressure (p) are constant is proportional to the linear scale raised to the 1.7 power:

$C_L/C_S = (D_L/D_S)^{1.7}$ where

D = A typical linear dimension

L = Large tunnel

S = Small tunnel

From Fig.7 it is clear that the most expensive way of achieving high Reynolds No is to increase the size partly because that will rapidly increase the area and the power is proportional to $(D_L/D_S)^2$.

An increase of speed and a moderate increase of pressure will, however, be very cost effective. [Though the power is proportional to $(V_L/V_S)^3$]. Fig.8.

The cost effectiveness with regard to a pressure increase depends upon the type of design layout which will be chosen.

From a structure stability and from weight economy point of view it is a great advantage for subsonic wind tunnels to chose sections, which are very near to circular (Fig.3).

Quasi circular [1] and Quasi Square test sections are thus attractive alternatives. They can - specially at moderate sizes - preferably be of the up and over type and give good flow conditions in corner vanes, transition parts and at the test section proper.

The most economic way to reach high Reynolds No is by applying moderate pressures in the tunnel (2 - 2.5 atmospheres). Here again the quasi circular sections will be the optimum design from a structure point of view.

A rapid pressurization and access to the test section is essential. An ordinary "diver" technique has been suggested for making changes on models between tests. The new technique will allow model service between runs without depressurization of the entire tunnel or the test section. That can be done by entrance arrangements similar to pressure sluices in hyperbaric chambers. Thanks to moderate pressure differences, short exposure times at elevated pressures and low frequencies between overpressure exposures, the physical strains are very low, when compared to the conditions normally prevailing for divers. (2 atmospheres equivalent to 10 m diving depth).

With only one permanent test section it is possible to work with different test equipment mounted on one or, if necessary, several exchangeable turntables. In this way bulky exchangeable test sections could be omitted.

With regard to pressure the cost differences due to an increase in pressure from 1 to 2 atmospheres is of rather small influence, since stiffness and buckling criteria already at atmospheric conditions will determine the structure, which thus can stand moderate overpressure.

A doubling of the pressure from atmospheric would approximately result in a 10% cost increase and that certainly pays off. A further pressure increase means that deformations of slots, tunnel operation, handling etc. rapidly will grow more and more difficult.

TRANSONIC TESTING

In the subsonic regime starting and landing conditions for aircrafts are of a major concern. Low speed and transonic manoeuvrability is also important fields of investigation.

Compressibility effects start at speeds long before Mach No 1.0 locally appears on the wing. On thick wings M_c could be 0.6 but on air intake lips it can occur already at $M_c = 0.4$.

When in a tunnel with solid walls Mach No. 1.0 is approached, a model blockage of 1% will already create choked conditions in the test section at $M_c = 0.89$. At $M_c = 0.95$ a model blockage of 0,2% will choke the test section. This high sensitivity to blockage (Fig.6) is reduced by the introduction of ventilated walls - slotted on the subsonic side and perforated (or combined with slots) on the supersonic side. In the transonic range above M = 1.00 it is desirable to have a variable porosity, so that the open area increases from 2% near to M = 1.00 to 6.5% at M = 1.2.

One of the difficulties to establish M = 1.2 or higher Mach numbers by means of suction through a progressively increasing porosity upstream the testing area is generally that the transition length has been chosen too short. The result has therefore been an over expansion with no possibility to get an extinction of the compressing waves forming at the end of the transition part. An expensive solution to this problem is to introduce a short flexible nozzle upstream the transonic test section.

It could be cheaper to make the transition part longer and combine it with a device, which could allow the Mach No to increase stepwise (in 2 or more steps) so that higher Mach Numbers than M = 1.2 can be obtained, without loosing too much air by ventilation through the transonic walls. That procedure will make it easier to overbridge the Mach No range 1.2 - - 1.4, which otherwise will be leapfrogged in a transonic test section, where Mach No are obtained by plenum suction only.

On the supersonic side of the transonic range there is another limitation caused by the shockwave from the model which after reflecting on the ventilated walls can impinge on the model or the sting. In order to improve the conditions for overlapping tests in the transonic and supersonic test sections (and assuming that the same models can be used in the two test sections) the height of the transonic test section should preferably be 50% larger than the supersonic test section.

The total head applied in transonic test sections is, due to possible stresses on models and stings limited to 400 kPa. In supersonic tunnels, however, the maximum pressure is generally determined by the starting conditions at the maximum Mach No. (Say near 10 atm. at M = 4.0).

From a cost and manufacturing point of view there are several advantages in splitting up a facility into two separate tunnel legs (Fig.9) instead of using one single trisonic tunnel.

The so called Twin Tunnel System Facility is attractive with regard to utilisation, accessibility and performance.

For aerodynamic testing in the subsonic and transonic domains, the temperature is generally of a minor importance. This is not true, however, if very high Reynolds No must be considered.

Reynolds No is proportional to speed, pressure, linear dimension and viscosity.

As a result the Reynolds No goes up with the temperature $T^{-1.4}$. The required power (E), however, is reduced according to $E \sim Re \cdot T^{1.9}$. Despite high complexity and costs the cryogenic technique has already been applied [3] and seriously considered in other projects [4] for obtaining full scale Reynolds numbers.

The very need for such Reynolds numbers is still under dispute despite on heavy argument, which originated from discrepancies between wind tunnel tests and and test flight experiences with a C141 aircraft. Though the Reynolds No interest should be focused to the subsonic regime (starting and landing conditions, high lift devices etc.) the large and exotic cryogenic facility projects are today primarily transonic facilities.

The cost estimation of a cryogenic high Reynolds No tunnel alternative is complex and depends very much on the technical solution and the local cost of liquid nitrogen. [5]

SUPERSONIC TUNNELS

The aerodynamic testing in the supersonic range is primarily devoted to research and development of aircrafts, missiles and power plants.

In the early days each Mach No required a separate nozzle, In 1940 - 50th several methods were developed for making possible a continuous variation of the Mach No by means of variable Mach No nozzles. (Fig.10).

Most of these devices are of the mechanical type. The simplest and at the same time most accurate version Fig.10E) is the single-jack system, which lately has been provided with a Boundary Layer compensation Jack, which can be used separately for trimming purposes. The slotted or ventilated nozzle, however, is subject to an artificial expansion in an axisymmetric geometrically fixed nozzle by aerodynamic means only. (Fig.10F).

The variable Mach No in combination with rapid advances in measuring and computer techniques increased tremendously the speed of collecting data.

During the last 30 years, the testing techniques, strain-gauge balances, pressure measuring techniques and model manufacturing have improved the overall quality and accuracy of the experimental results.

It is imperative to dry the air so that no water condensation shocks can take place during the expansion in the nozzles. In some ejector tunnels preheating (Fig.11) of the air has been used instead of using large drier beds of Silicagel or activated aluminum. In blow down facilities compact driers are installed downstream the compressors. (Sometimes combined with a pre-cooler for a rough separation of water).

In a unique blow down facility at VOLVO-FLYGMOTOR AB in Trollhättan, Sweden, (Fig.12), water is used to chase the air out of a huge (11.000 m^3) rock chamber (height 4 m). The rock chamber is 90 m below sea level. During the summer, when the water is about 20^{o}C the partial water vapour pressure is slightly too high but in winter acceptable (~ 0,4g/kg air). The conclusion is, that a rock chamber of the same kind should rather be about 240 to 320m down in order to avoid condensation problems (water vapour partial pressure being the same as above).

In supersoic tunnels the noise level is usually high. Even if the noise level in the settling chamber can be reduced by cascading arrangements in the valve diffusor, it is not possibe to get rid of the noise generated in the boundary layers of the supersonic nozzle. At higher Mach No this noise source is dominating.

An important design feature for the few flexible wall tunnels, which can change Mach No during testing, is the sealing of the flexible nozzles towards the sidewalls.

It is very important, that the seals are properly designed and mounted in order to get a good Mach No distribution.

In many tunnels it can take 20 minutes (between two runs) to change from one Mach No to another. In the single screw semiflexible wall nozzles (Fig.10E) the entire Mach No range can be covered within 10 seconds. The variation of the Mach No can be very useful, when e.g. studying hysteresis effects in air intakes and during flutter tests.

The slender nozzle types generally have good starting load characteristics. With a variable Mach No nozzle thè start can take place at a low Mach No, so the high model loads at high Mach No are avoided (Fig.13).

With an angle of attack mechanism, which is combined with a roll mechanism any combinations of lift and side forces can be arranged e.g. for 6 component balance or pressure measuring models.

The angle of attack mechanism can also be used for calibration devices (rakes, angularity probes, pneumatic balances etc.).

The sting can be exchanged by a hollow cylinder with a variable nozzle for varying and measuring mass flows through air intakes.

Assuming that there is no blockage (Fig. 14) the expected starting conditions for any of the trisonic tunnels would be as per Fig.15.

In order to keep the models within the test rhombus the Mach No range, planform and attitude angles must be known. A typical model length to test section height (L/H) versus Mach No curve is shown in Fig.16.

Thermodynamic Tests

In supersonic tunnels which specially are devoted to thermodynamic tests (turbojet engines, ramjets etc.) several superimposed requirements must be considered:

- The static temperature (T_S) goes down linearly with an increasing flight altitude (H_S) up to 11 km. Thereafter the temperature T_S is assumed constant for another 10 km.

- In order to obtain the equivalent static temperatures in the test section of the wind tunnel the Total temperature T_0 must be established in the settling chamber and accomodated to the Mach No which is required in the test section:

$$T_0/T_S = [1 + M^2 (\kappa - 1)/2] \qquad \text{(fig.4 and 17)}$$

- The pressure variation (P_S) with flight altitude (H in km) is changing as in Fig.18.

- The isentropic pressure ratio (P_0/P_S) over the supersonic nozzle will give necessary total head in the settling chamber in order to reach the wanted pressure (P_S) at the flight altitude (H). (Fig.19).

$$P_0/P_S = (T_0/T_S)^{\kappa/\kappa-1} = [1 + M_0^2 (\kappa - 1)/2]^{\kappa/\kappa-1}$$

In the supersonic range the aerodynamic conditions for testing are very neat and simple. For research and development on powerplants, when combustion processes are involved, there are many difficulties which all can be referred to the Reynolds Number under true flight conditions:

Operation of Flame holders
Heat transfer problems
Mixing processes
Evaporation
Ignition
Combustion
Extinction
Cooling etc.

In all these tests the correct simulation of tempeartures and flight altitudes is of a primary importance.

A simple method to obtain correct temperature simultation is by a combustion heater installed upstream or in the settling chamber. By oxygen injection upstream the combustion zone it is possible to have the same oxygen content after the combustion as in normal air. This is naturally important, if the test object further downstream will need the oxygen for its operation in the test section.

It is also possible to control the total temperature by mixing cold air with air passing a preheater of a matrix type. Heat exchangers or direct electric heating elements are other possibilities.

In most cases it is necessary to install suction devices downstream the test section for simulating different flight altitudes. For this purpose ejectors (driven by air, hot water or steam) or vacuum spheres, rock vaccum chambers, vacuum pumps or similar can be used.

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CONCLUDING REMARKS

Most of the wind tunnels of today have been in operation for more than 30 years. Therefore the planning today of wind tunnels for the next period of 30 years should consider:

That the highest tunnel quality should be required from the start of a project because wind tunnel equipment cannot easily be changed afterwards. Ancillary equipment, however, can more easily be improved or exchanged at a later occasion.

Wind tunnel resources for research and development will improve the know how and also contribute to the general education in the aeronautical field.

This way of building up a high competence level is profitable not only if the interest is focused on national industrial activities but also when purchasing advanced equipment abroad.

REFERENCES

[1] Agrell, N., Mattsson, R., Nyberg, S-E. — Investigation of the Transonic Drag Characteristics for Non-Slender Wing-Body Combinations and their equivivalent axisymmetric Bodies at Zero Lift
ICAS PROCEEDINGS, September, 1978

[2] Whitfield, J.D., Jacocks, J.L., Dietz, W.E., Pate. S.R. — Demonstration of the Adaptive-Wall Concept Applied to an Automotive Wind Tunnel
SAE Technical Paper Series No 820373

[3] Kilgore, Röbert A. — Development of the Crypgenic Tunnel Concept and Application to the U.S. National Transonic Facility
AGARDOGRAPH No 240, November, 1979

[4] Hartzuiker, J.P. — The European Transonic Wind Tunnel ETW.
The Royal Aeronautical Society, January, 1984

[5] Nelander, Curt — A Quasi - Continuous Transonic Wind Tunnel for Cryogenic Operation
ROLLAB MEMORANDUM RM096, November, 1983

Mach Number Range	Pressure Ratio for Operation	Nozzle Slenderness Ratio L_N/H	Test Section Size L/H	Area Ratios A_0/A_1	A_1/A_S
Subsonic 0 - 0.3	~ 1	Contraction Unit $L/H_0 \sim 1.2$	2.25	9	(5.8 - 2.0)
Transonic 0.3 - 1.2 (1.4)	1.1 - 1.5	Transition or nozzle 1 - 2.8 (4.0)	2.25	12	1.6 - 1.03 (1.12)
Supersonic 1.3 - 3.8 (4.2)	1.5 - 9 (15)	5 - 7.2	0.6 - 4.0	12 - 1	1.06 - 9.0 (12.8)
Hypersonic 4 ⇒	10 ⇒	4 ⇒	4.0 ⇒	1 ⇒ ↓	12.8 ⇒

INDICI

O = Settling Chamber
S = Sonic Throat
1 = Test Section
A = Atmospheric

Table 1 The Variation of Characteristic Design Parameters in the Trisonic Mach No Range

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14-16 May 1985 , CAIRO

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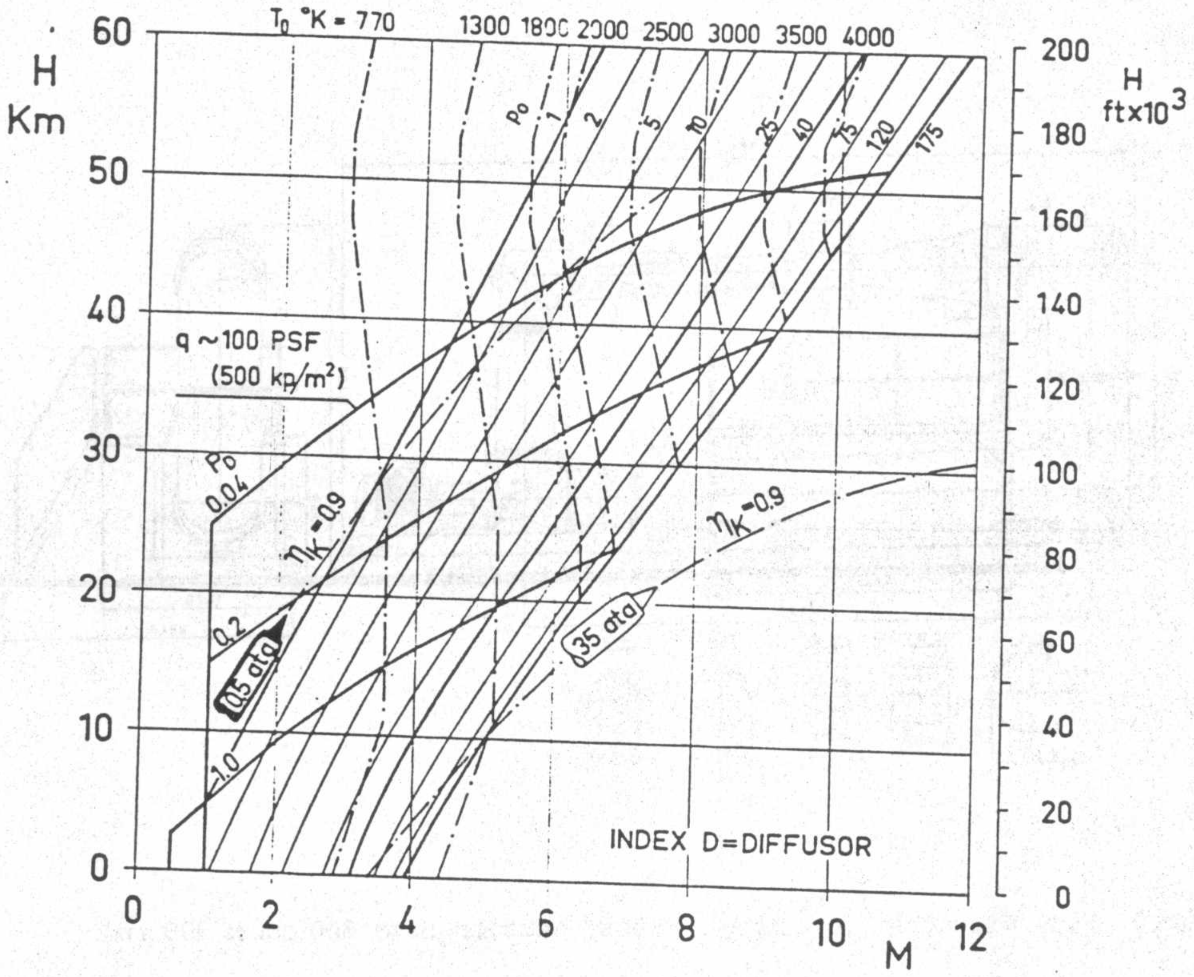

Fig.1 Flight Altitude versus Mach No (M) at Constant Temperature (T_0) and Total Pressure (P_0)

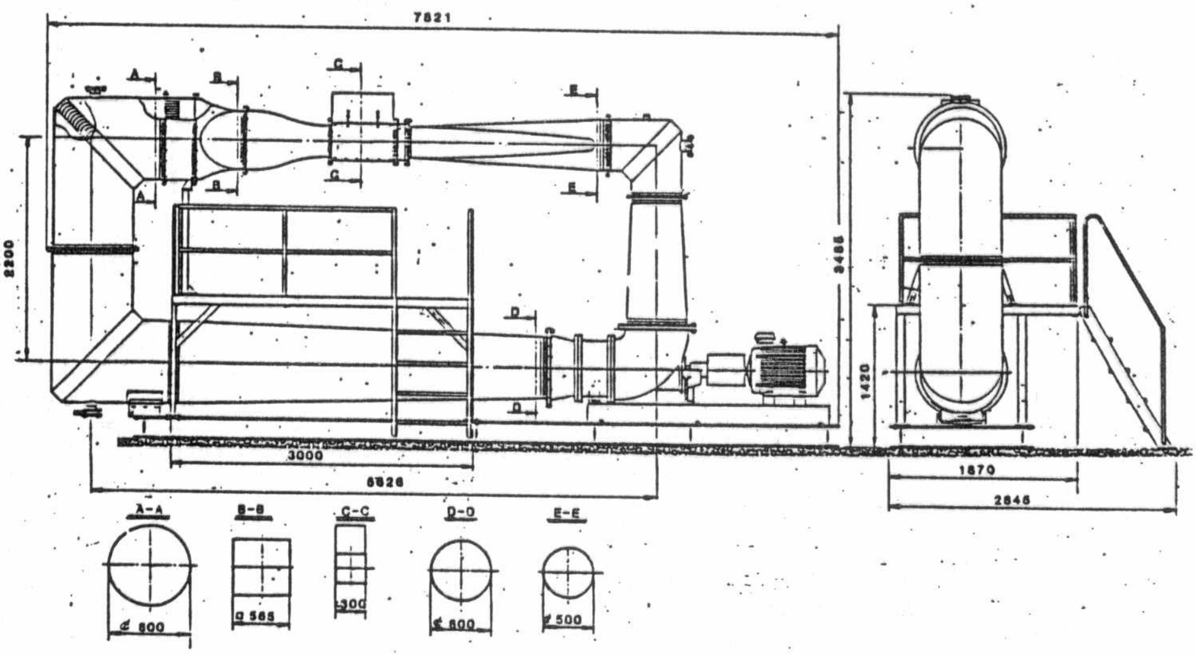

Fig.2 Flow Vizualisation Tunnel (Test Section Size 300 mm x 300 mm)

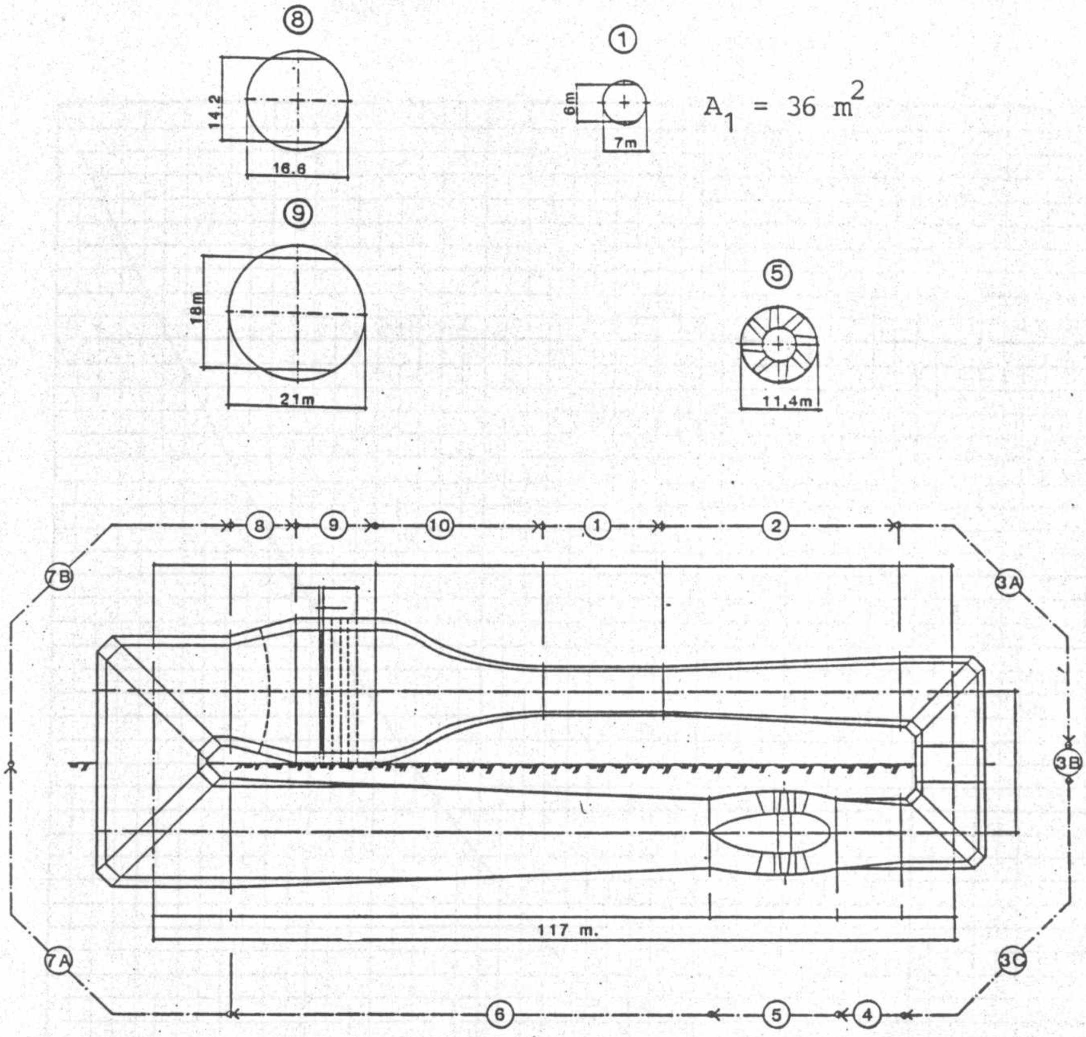

Fig.3 A quasi-circular Test Section Wind Tunnel (7m x 6m) suitable for pressurization according to S.O. Ridder

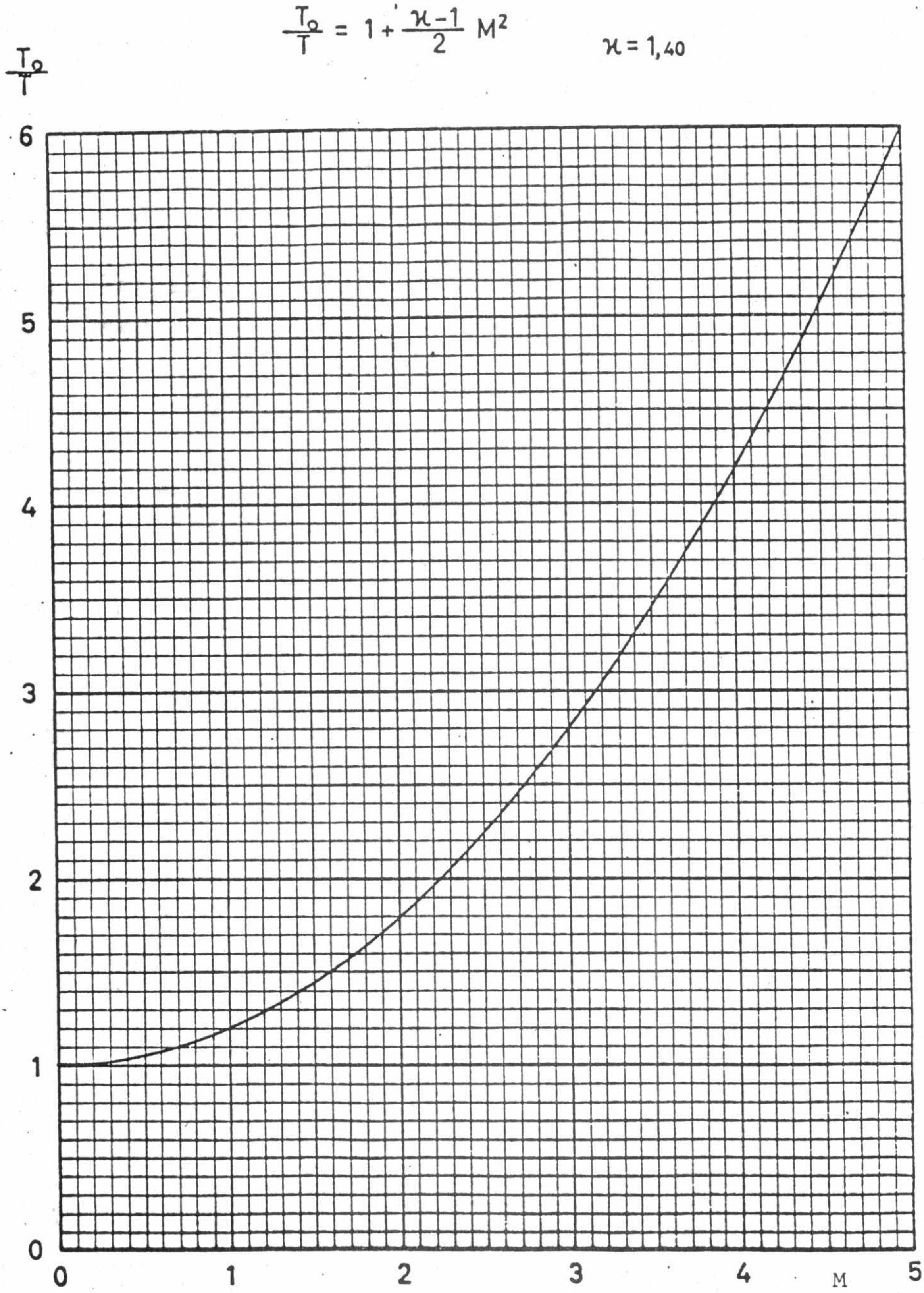

Fig.4 Total/Static Temperature Ratio (T_0/T) versus Mach No (M)

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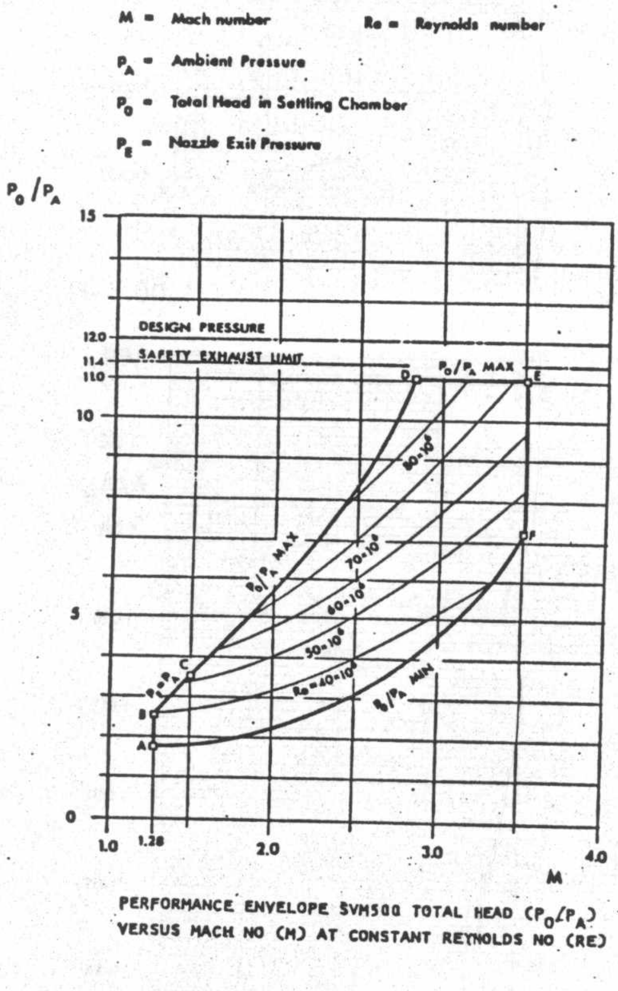

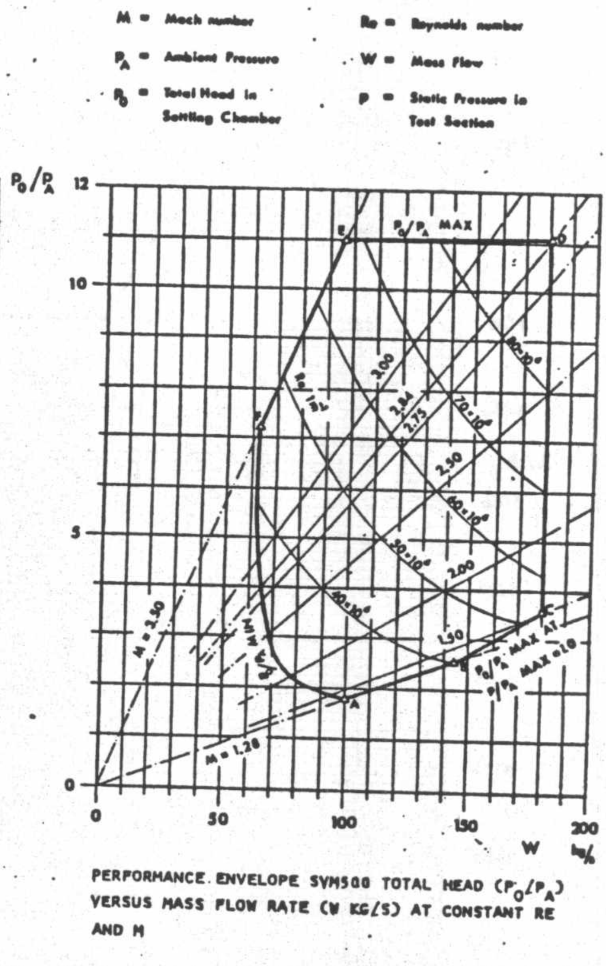

Fig.5 Typical Supersonic Performance of a Trisonic Tunnel (Test Section Size 500 mm x 500 mm) .

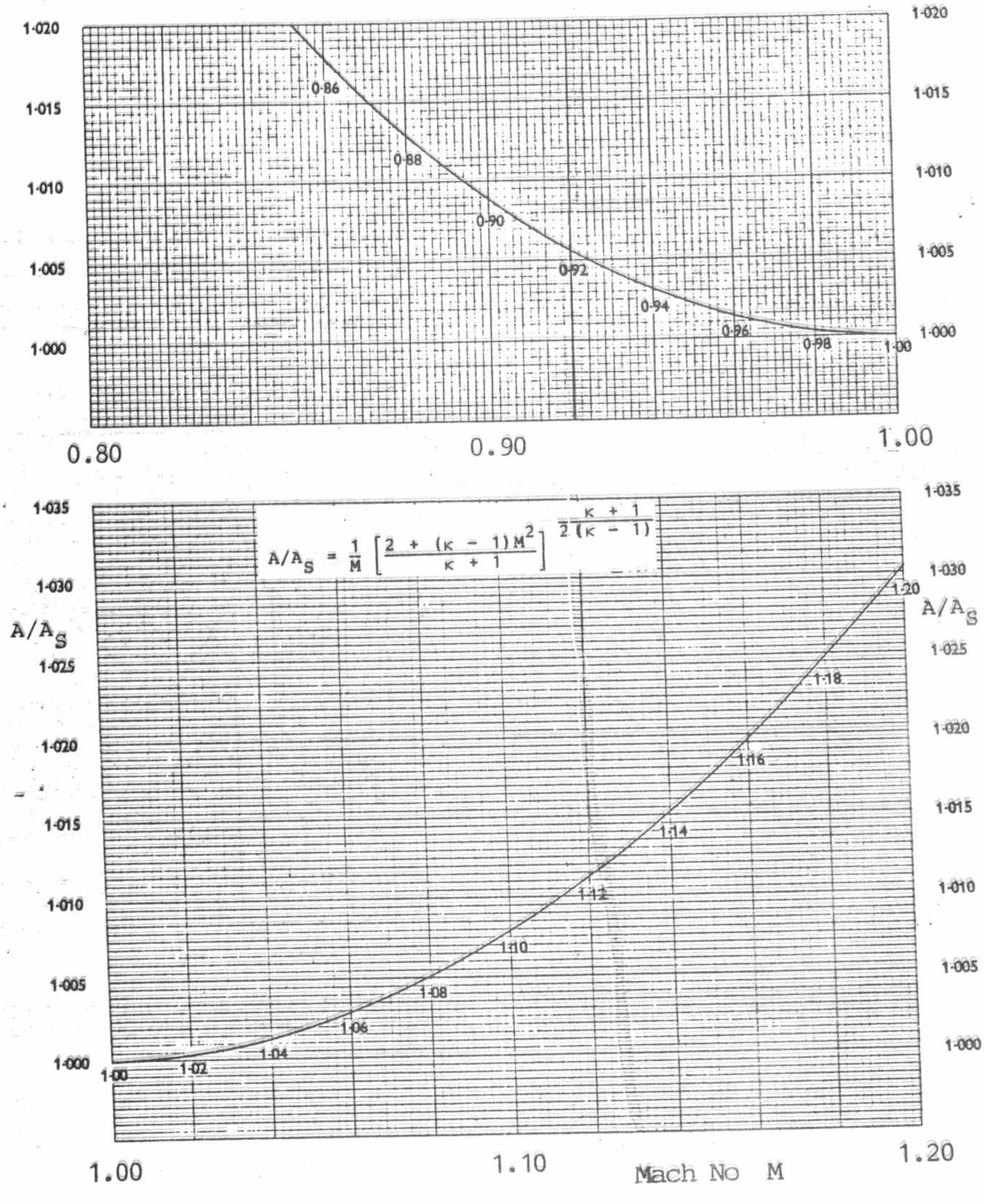

Fig.6 Area Ratio (A/A_S) versus Mach No (M)

Re = C x P x V x L

. is referred to the tunnel diameter (L = D)

A typical chord for a half model arrow wing is 0.2 x D

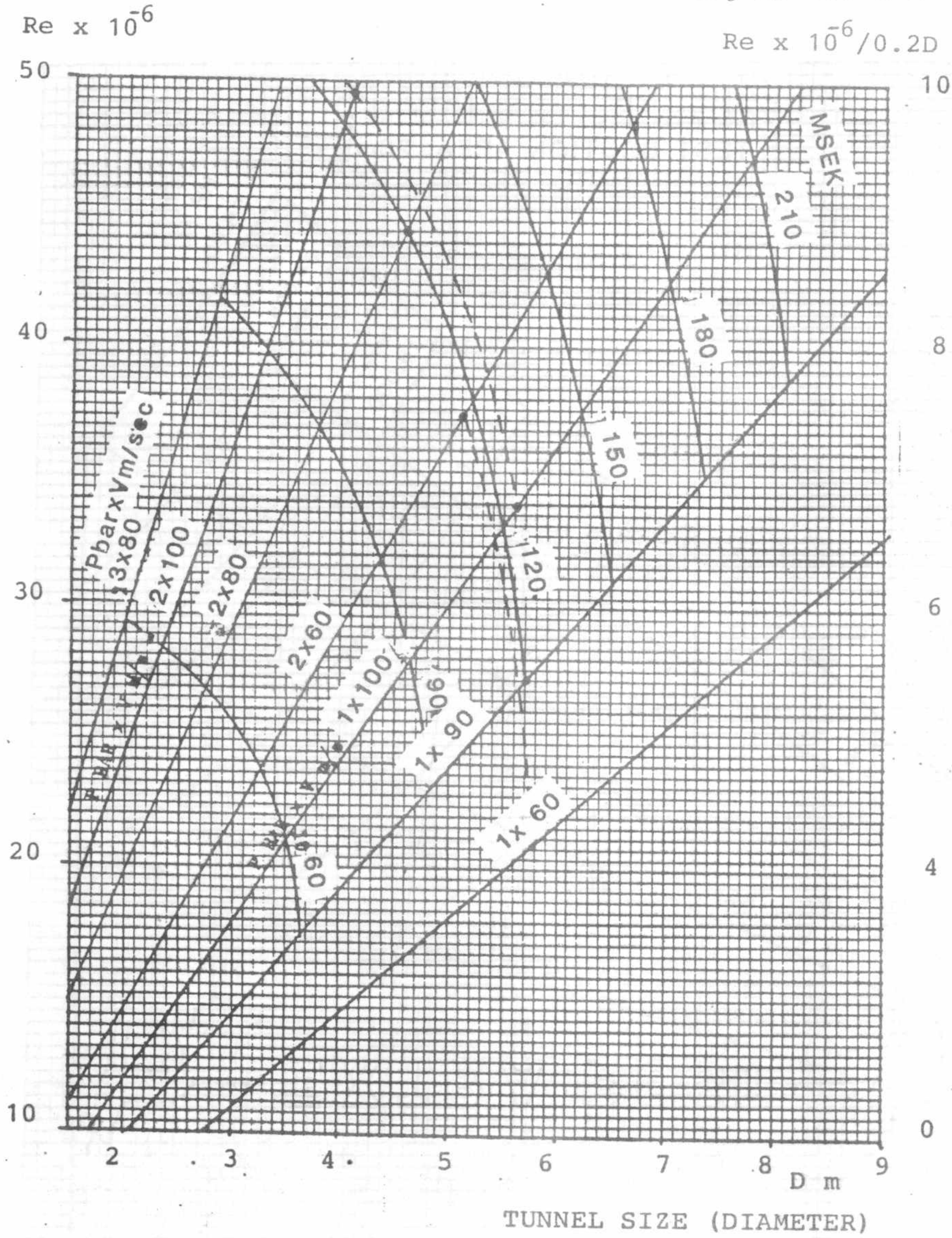

Fig.7 Reynolds No (Re) versus Tunnel Diameter (D) at Constant Pressure times Speed (P x V) and Cost (C)

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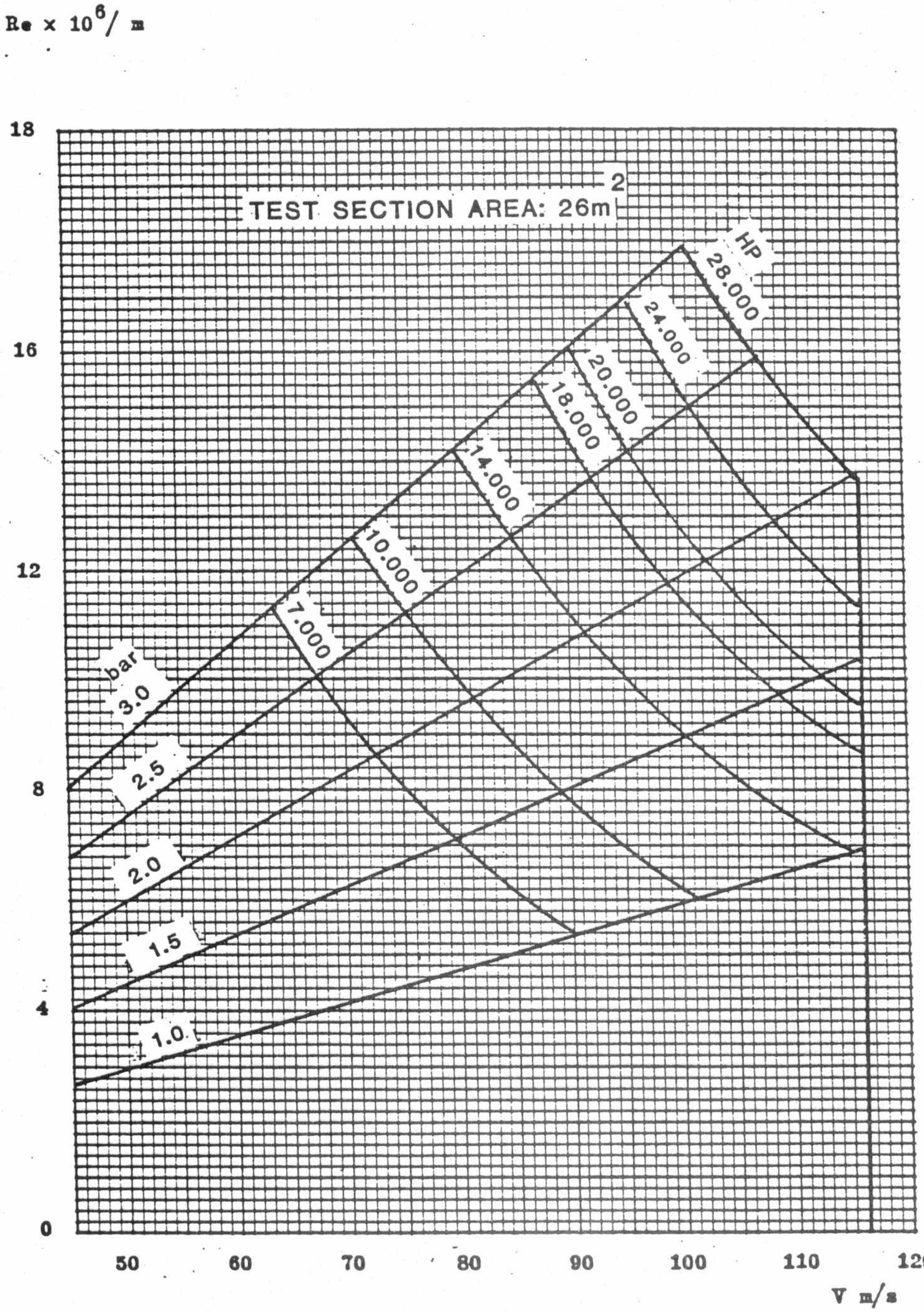

Fig.8 Reynolds No (Re) per m versus Speed (V) at Constant Pressure (P) and Power (E) for a 26 m^2 Subsonic Tunnel

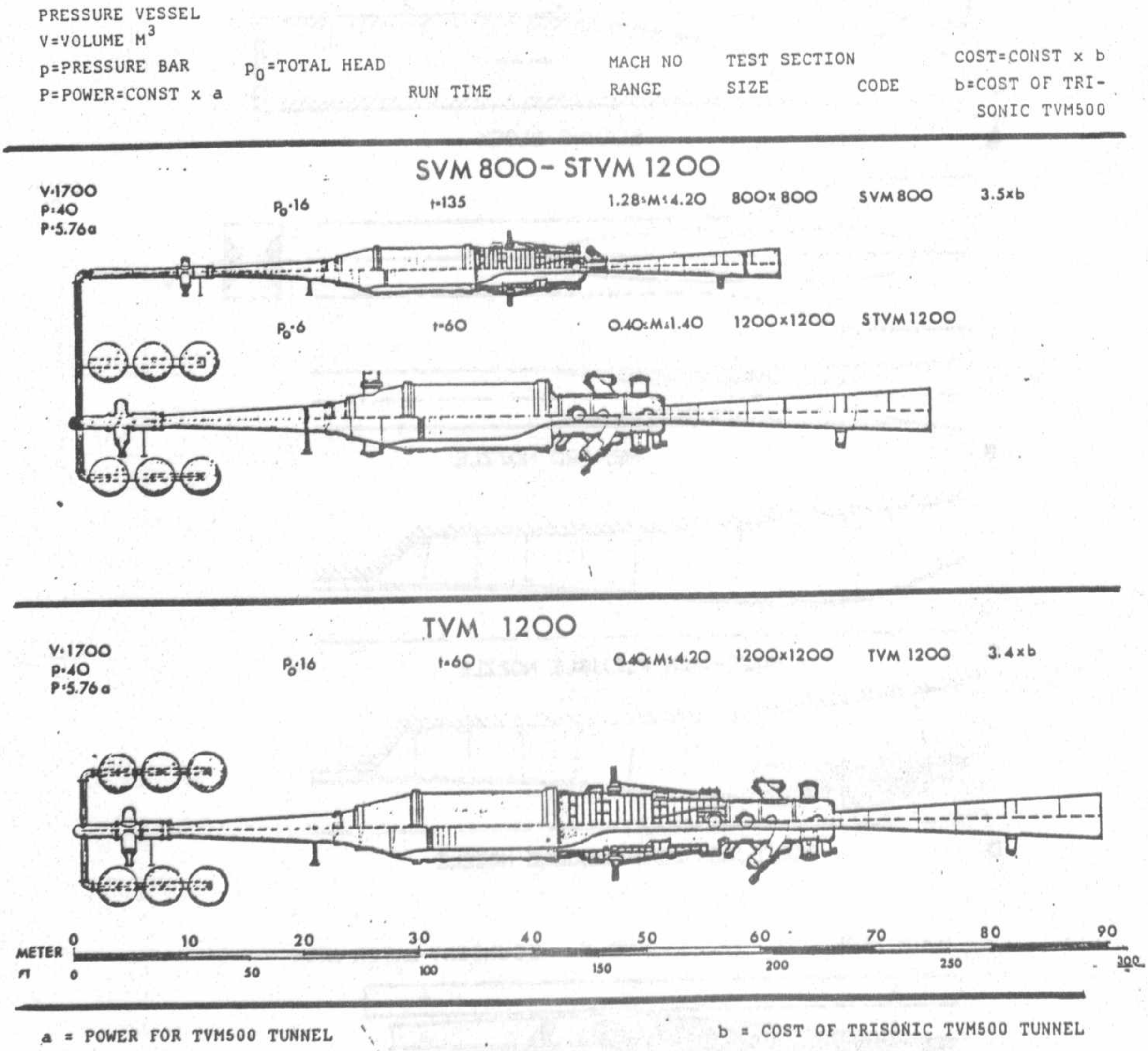

Fig.9 The ROLLAB Twin Tunnel System

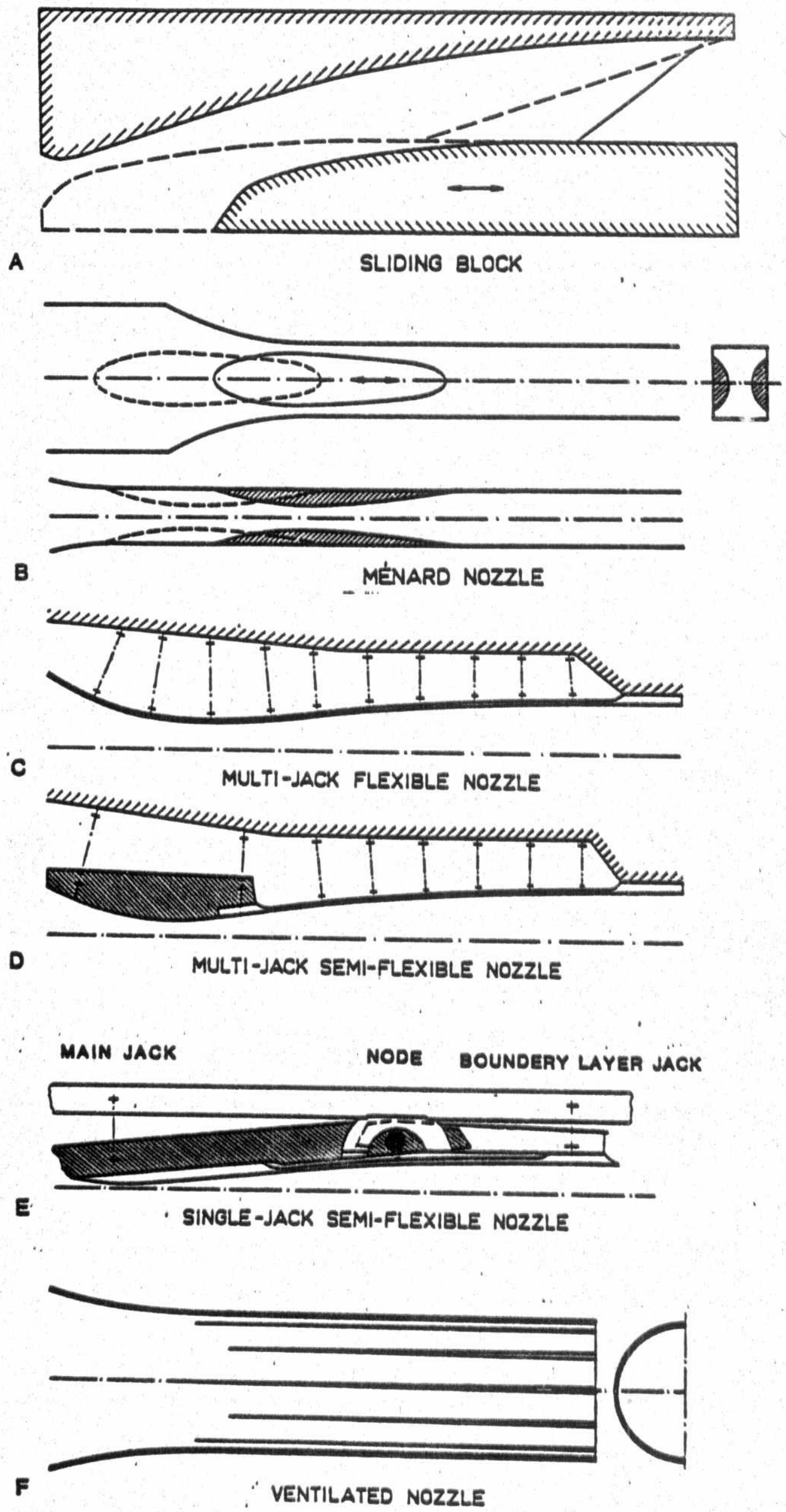

Fig.10 Variable Mach No Nozzles design concepts

6

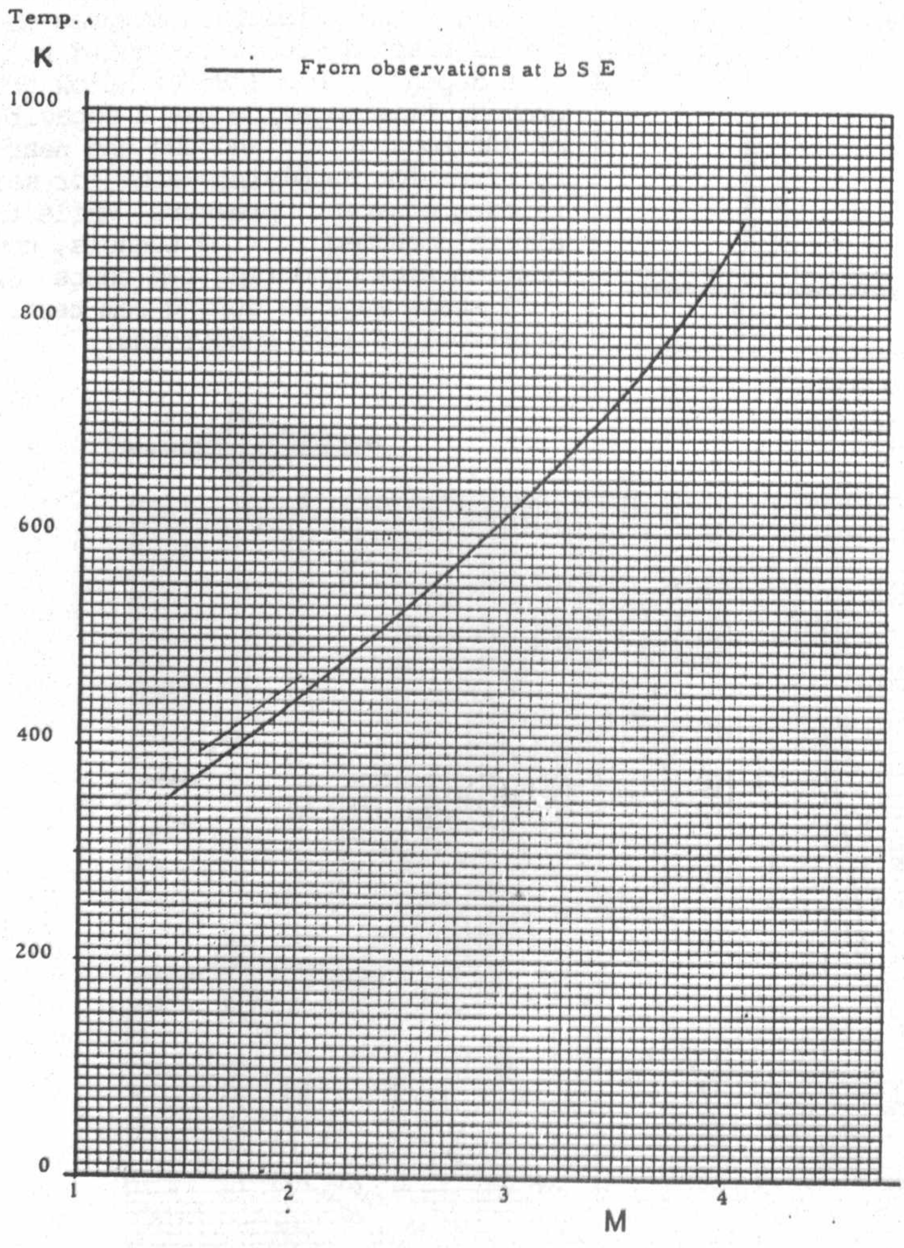

Fig.11 Minimum Total Temperature (T_0) to avoid water condensation shocks at a given Mach No (M)

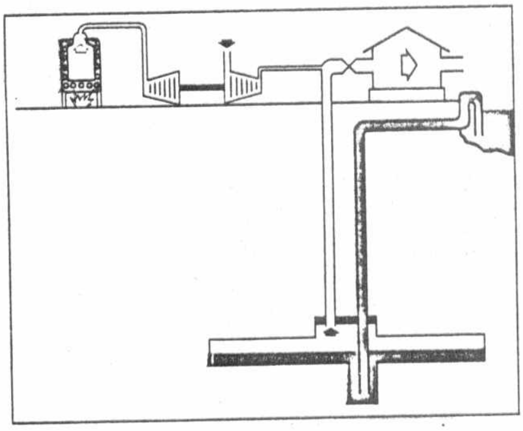

Fig.12 A unique air storage system

At Volvo - Flygmotor AB, Trollhättan Sweden, a magazine for compressed air is blasted out of the solid rock at a depth of 90 m (295ft) below the ground. This magazine has a capacity of 130 tons of air and use the nearby river for supplying water for maintaining constant pressure while the air is consumed in wind tunnels, combustion rigs or during ram tests on full scale engines during ram tests at supersonic flight conditions.

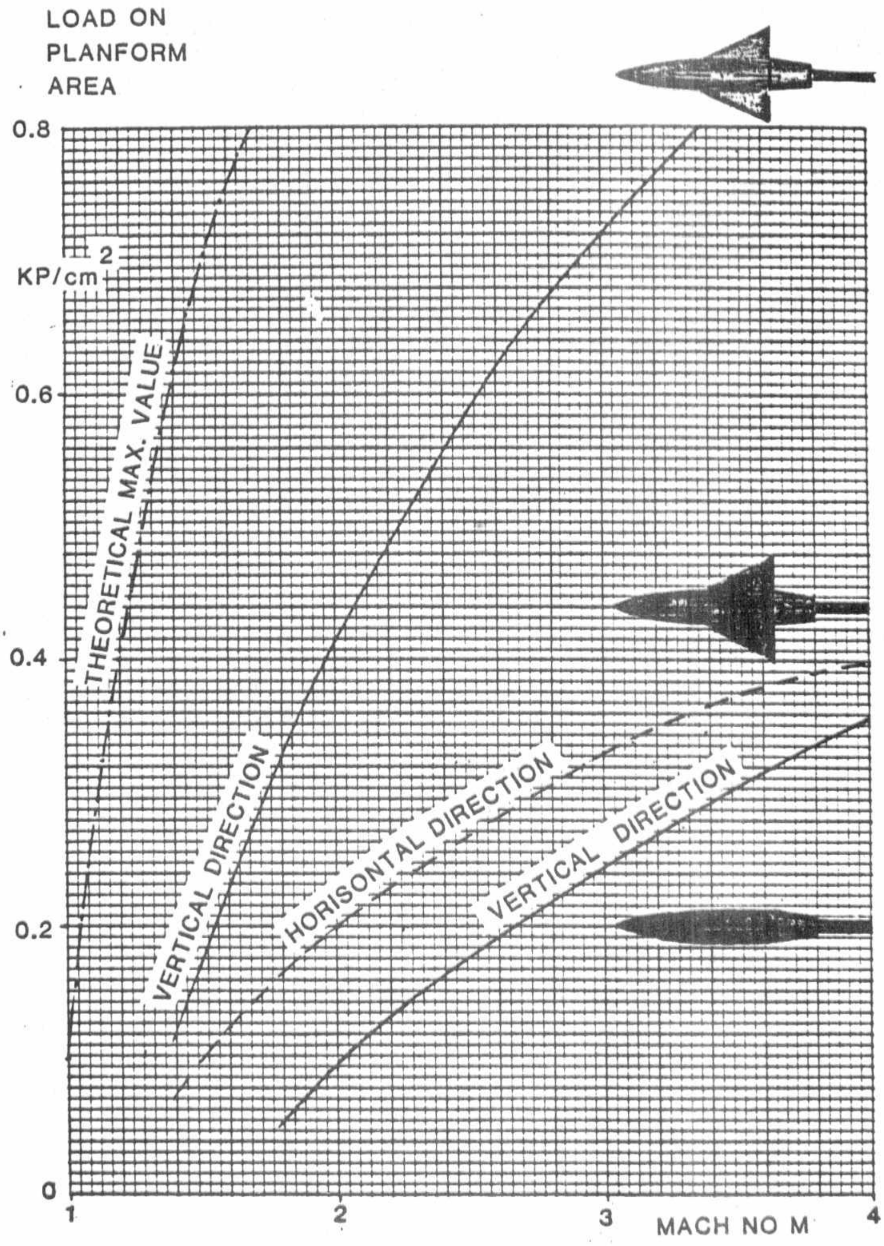

Fig.13 Starting Loads versus Mach No

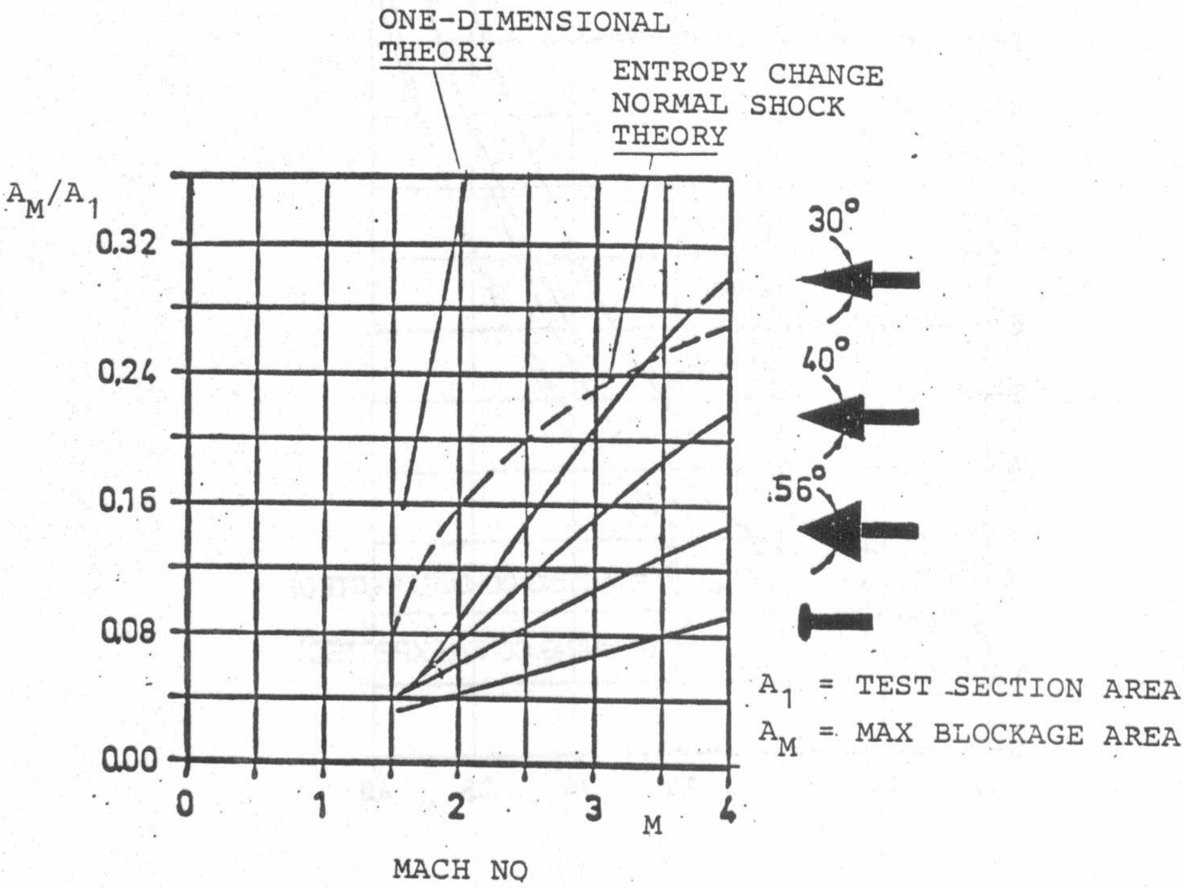

Fig. 14 Maximum Model Blockage (A_M/A_1)

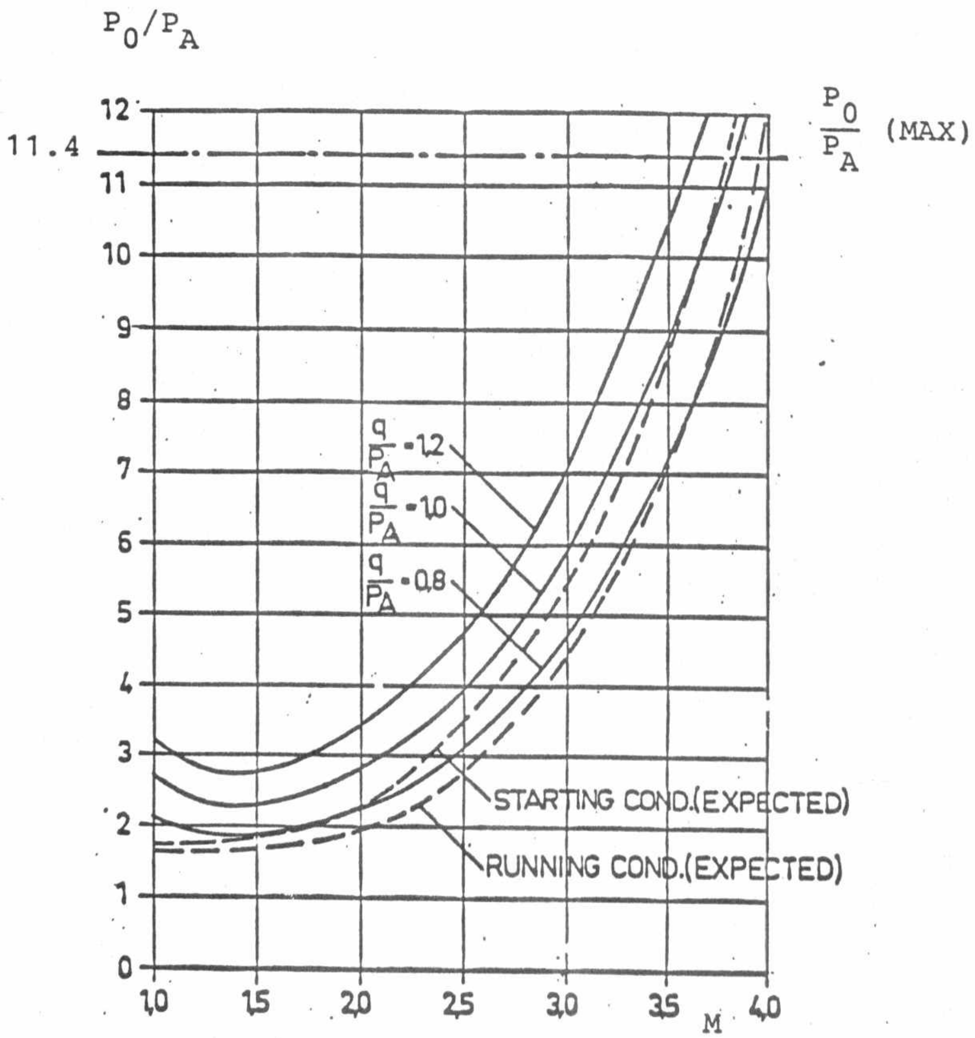

Fig.15 Expected Operating Conditions in TVM500

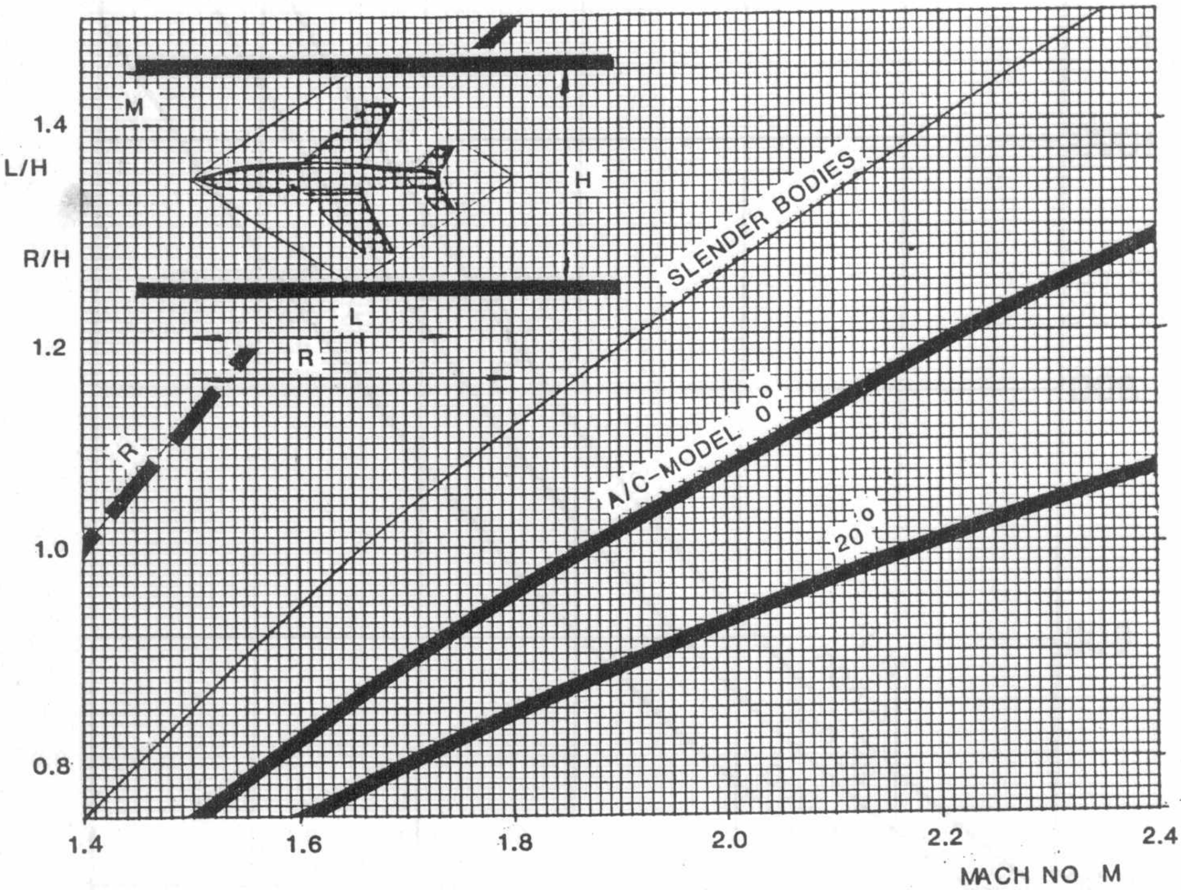

Fig.16 Maximum Model Length (L/H) as determined by Test Rhombus for a typical Configuration

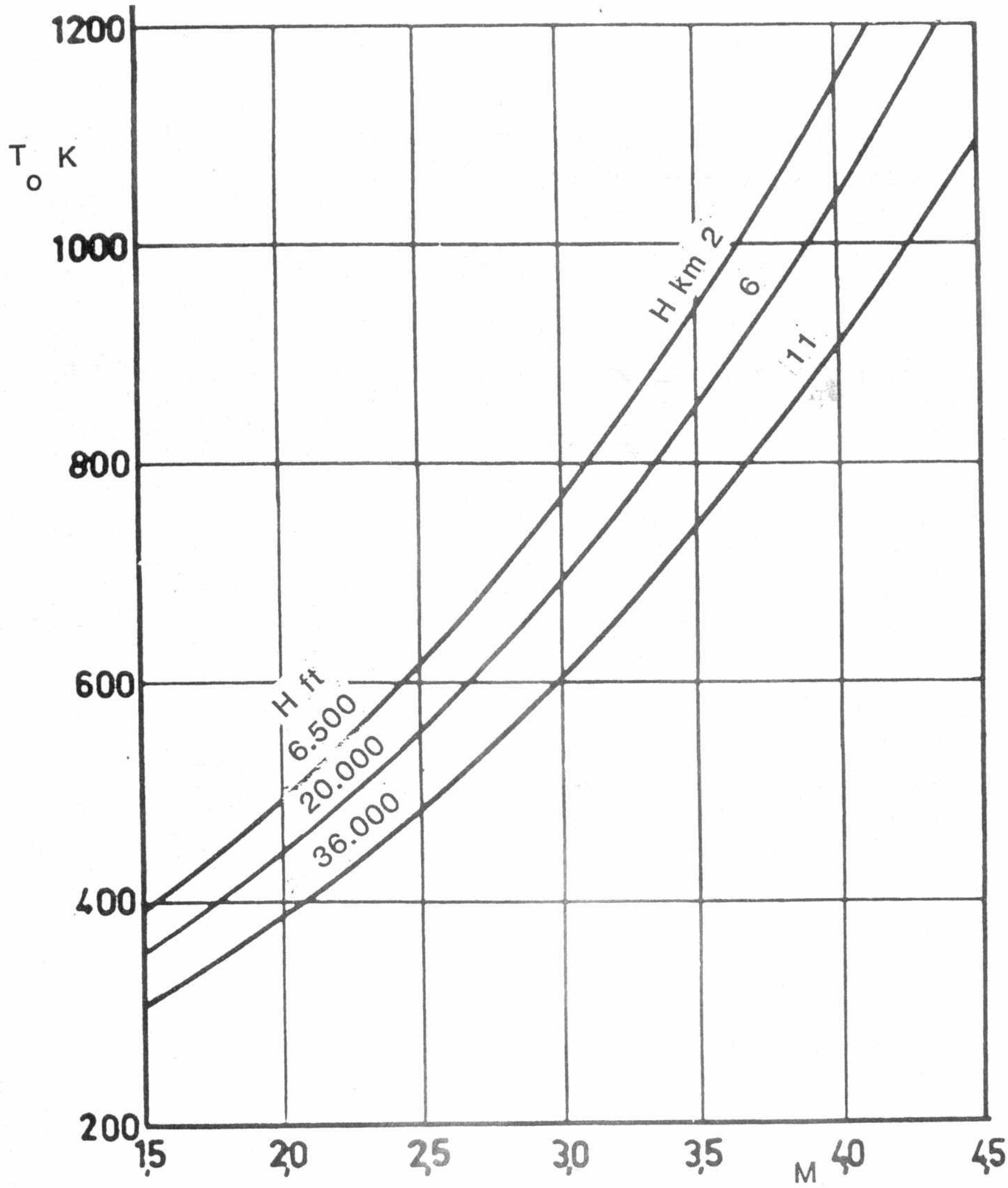

Fig.17 Total Temperature (T_0) versus Mach No (M)

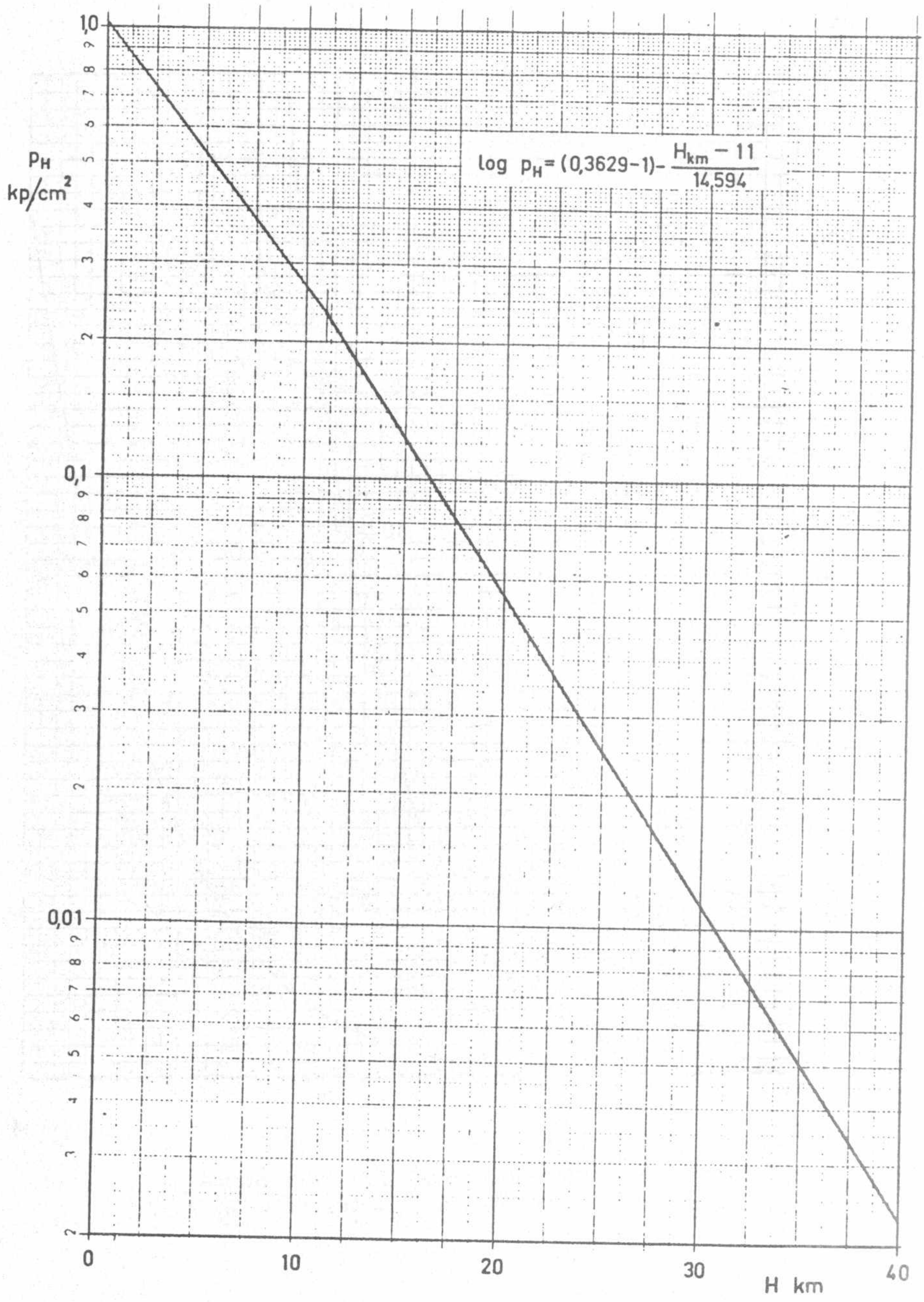

Fig.18 Pressure (P_H) decrease versus flight altitude (H km)

$$\frac{P_0}{P_s} = \left(1 + \frac{\varkappa - 1}{2} M^2\right)^{\frac{\varkappa}{\varkappa - 1}}$$

$\varkappa = 1{,}40$

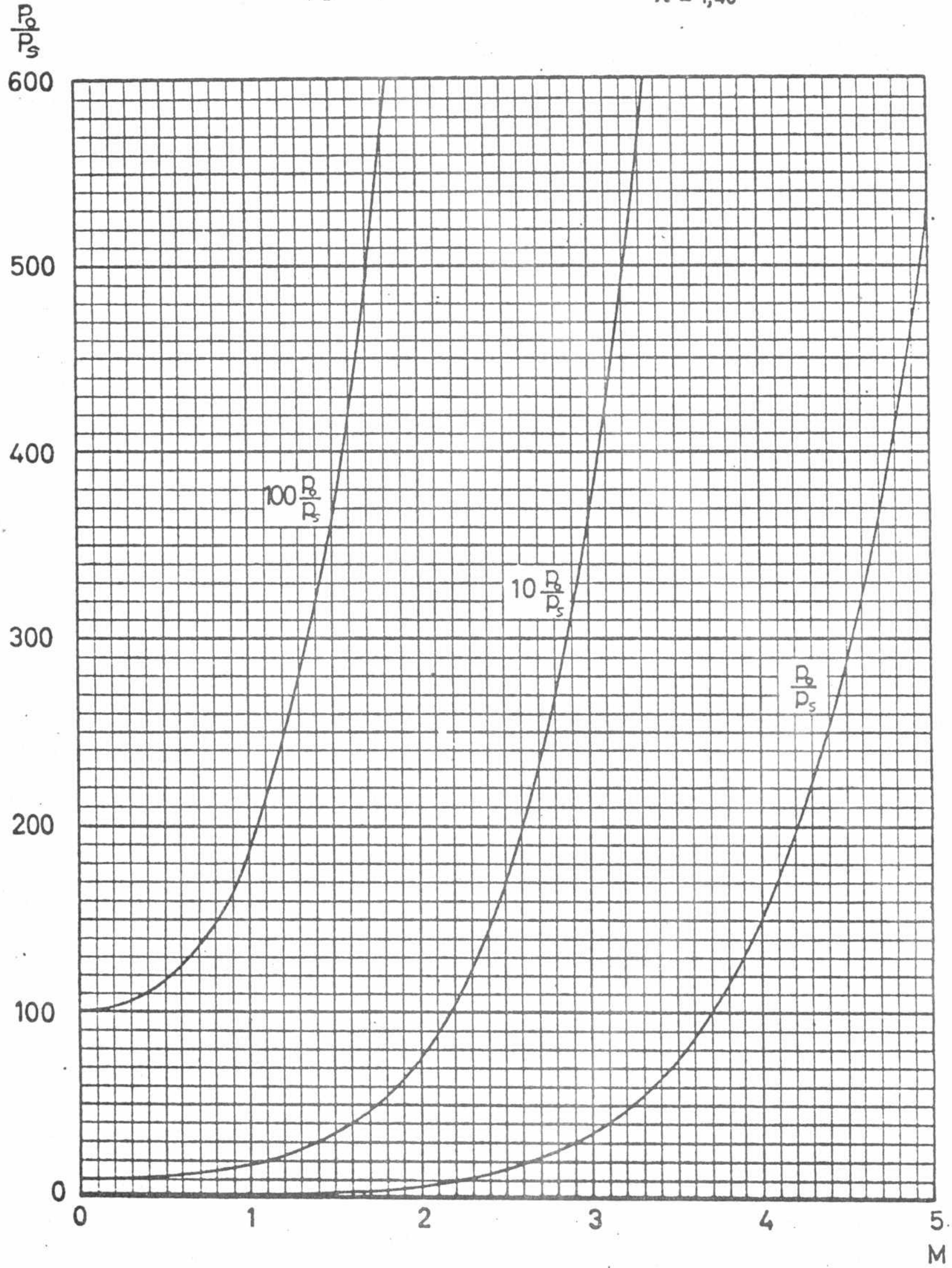

Fig.19 Total Head (P_0) over Static Pressure (P_S) versus Mach No (M)