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FACILITIES FOR AERODYNAMIC TESTING AT HYPERSONIC SPEEDS

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SUMMARY

An assessment is made of the usefulness and potential of existing European hypersonic facilities, on the basis of the planned U.S. space shuttle project and a hypothetical hypersonic transport aircraft. With respect to aerodynamic testing of space shuttle type of vehicles it is pointed out that a significant gap exists between M = 10 and M = 15.

At low-hypersonic Mach numbers the facilities in Europe will generally meet the minimum requirements for testing hypersonic transport models. European capabilities appear to be rather similar to those in the U.S. hypersonic wind tunnels, though the U.S. capabilities will be increased considerably in the near future.

It is further concluded that European facilities fall short in their performance of what is required, in the field of propulsion (including supersonic combustion tests) and also of hardware testing.

CONTENTS

1. INTRODUCTION

This LaWs paper will be mainly concerned with the requirements for aerodynamic and propulsion testing in wind tunnel facilities for developmental work in the hypersonic speed regime i.e. on vehicles flying at Mach numbers greater than about 5. Two types of vehicles are of current interest, namely the space shuttle and the hypersonic transport aircraft (H.S.T.).

The space shuttle is to enter the development phase by mid 1972. Outside the U.S. European participation in the project has been considered (Ref. 1) but is not very likely at the moment of writing (Ref. 2). The system under development will be propelled by rocket engines. In order to make regular space launches more economical, however, an airbreathing propulsion has to be used. It is generally expected that such a system will be developed after the rocket system under current development. This line of thought may eventually lead to the hypersonic transport system at the end of this century yielding less sonic boom problems at cruise as compared with the SST.

Advanced hypersonic missiles, either rocket or (sc)ramjet propelled are also to come, though little is known of the requirements.

The chances that Europe may participate with the U.S. in the development of one of these projects in the next few years are probably small. On the other hand much work of a more exploratory or fundamental nature, such as described for instance in references 3 and 4 respectively, has to be done before the development phase of a ramjet/scramjet propelled hypersonic transport can be initiated. In this field the present wind tunnel facilities in Europe with only limited performance, compared with full scale requirements are certainly of great value.

Before the development of a HST can be started the economic feasibility of the system has to be demonstrated. The payload being only a small percentage of the total weight, the aerodynamic, propulsive and structural characteristics should be known with a high degree of accuracy (ref. 3). To accomplish this, facilities must be available where aerodynamic testing at high enough Reynolds numbers is possible and where engine-airframe integration can be studied with representative intake and exhaust jet simulation, to mention just one aerodynamic problem. Also high performance long duration facilities are needed for hardware testing of the propulsion system and of the structure of the vehicle which will be subjected to severe aerodynamic heating.

In the next discussion on hypersonic facilities the requirements will be centered on the requirements for the development work on the space shuttle and the hypersonic transport taking two typical examples. *)
Hence a judgement can be made on the work that can be done in the European facilities that is of fundamental and of practical interest, and in which areas of research and development European hypersonic facilities show shortcomings.

In the past, several European facilities have been used for military project development (i.e. tactical and ballistic missiles). This type of activities will undoubtly remain in the future but the merits will not be discussed in the present paper, due to lack of detailed information. It can be remarked however that for such applications the usefulness and potential of present European facilities are quite satisfactory.

Typical trajectories for the various types of hypersonic vehicles are found in the altitude-velocity diagram of figure 1, which is a compilation of data found in the literature.

1.1 Basic studies

Before being able to write down realistic specifications for hypersonic vehicles such as semi-ballistic entry vehicles (Gemini, Apollo), lifting entry vehicles (Space shuttle) and the hypersonic transport many basic questions have to be answered first. As far as the fluid dynamic aspect is concerned, information is needed on topics such as boundary layer transition (location and occurrence), radiative heat transfer during re-entry, fuel injection in a scramjet engine, and ablation heat shield properties, mentioning only a few arbitrary examples.

Not only in the United States, which is the only Western nation that has developed manned hypersonic vehicles such as the Apollo, the X-15 and presently the space shuttle, but also in Europe a rather extensive hypersonic research program exists. An inventory of the European research and facilities can be found in references 5-7. This inventory is the result of the initiative to create

*) In a recent article(ref.84)military hypersonic cruise aircraft are foreseen in the late 1980's which would open a new corridor for weapons delivery. With speeds of Mach 5-12 at over 100.000 ft altitude it would have performances of a missile and flexibility and recallibility of an aircraft. The requirements will be comparable with those of the HST.

Eurohyp, a more or less informal body to bring people together working in the same field, to disseminate information on the hypersonic work in Europe and to create a better co-operation. A summary of the activities of Eurohyp since 1968 can be found in the introduction of reference 5.

It is concluded from the survey that a significant amount of work is being done in Europe and much of it will be applicable to the design of the real hypersonic vehicles that will be discussed in the subsequent sections.

1.2 The space shuttle

The proposed U.S. space shuttle will be a two-stage system, consisting of unmanned wingless solid propellant recoverable boosters and a manned orbiter. The orbiter will be a delta winged lifting vehicle with a length of 33.6 m and a span of 22.6 m. The wing load during re-entry is estimated to be somewhere between 200 and 250 kg/m^2. The landing weight will be about 75 tons. Orbiter propulsion is by three liquid rocket engines. The maximum acceleration during launch or re-entry is limited to 3 g (ref. 1, 8 and 9). Stage separation will occur at about Mach 7 at an altitude of about 60 km.

Experimental studies in Europe have been largely devoted to lifting bodies with lift/drag ratios higher than the present U.S. design. Quite a lot of work has been done in Germany (ref. 4, 10) not only at hypersonic speeds but also in the transonic and subsonic speed regime. In the U.K. high L/D configurations have been studied by the RAE. A good discussion on this subject may be found in reference 11.

For the proposed U.S. space shuttle, a cross range of 1100 naut. miles (ref. 1) will correspond with a hypersonic lift/drag ratio L/D = 1.3 (ref. 11). The aerodynamic coefficients of such a vehicle are considered to be represented accurately enough for the calculation of the flight trajectories by the data from reference 12.

These data give

α	C_L	L/D	C_N	C_x	Condition
(deg)	-	-	-	-	
25	0.3	1.3	0.37	0.08	Maximum C_L
55	0.7	0.6	1.36	0.10	Maximum L/D

? omwisselen

For a wing loading W/S = 200 kg/m^2 three equilibrium re-entry glide trajectories have been calculated, defined as re-entry with constant L/D and bank angle ϕ (ref. 13) and zero vertical acceleration. The pull-out phase which occurs at a flight altitude of about 80 km is not considered. Also viscous effects which affect the L/D ratio at high altitude have not been considered, following reference 13. The following three conditions have been calculated.

1. $C_{L_{max}}$, ϕ = 0 which minimizes the heat transfer rates to the vehicle
2. $(L/D)_{max}$, ϕ = 0 which gives the maximum range (zero cross range)
3. $(L/D)_{max}$, ϕ = 60°. This bank angle gives an acceleration in the horizontal plane of about 2.3 g, which is well below the maximum permissable value of 3. The cross range for this non-modulated trajectory has been calculated by a numerical integration in crude steps (13 steps for the complete trajectory) and was found to be about 925 naut. miles, which is not too far from the required maximum cross range. This simplified trajectory seems therefore a reasonable approximation of the real maximum cross range trajectory.

The results have been plotted in the altitude-velocity diagram of figure 2.

Finally it should be noted that the descend trajectory of the boosters is not indicated due to lack of data. When booster recovery is required however, the aerodynamic behaviour should certainly be studied.

1.3 The hypersonic transport

For quite some years the hypersonic transport concept has been studied. A review on the subject is for instance presented in reference 14. Early studies considered flight speeds up to Mach 15 but more recent studies by NASA are rather concentrated on the Mach 6-8 speed range. A typical example is the vehicle, presented in reference 15 powered with four hydrogen-fueled turboramjet engines designed to fly at a cruise Mach number of 6 at an altitude of about 30 km. The vehicle should carry a payload of about 23 ton which is about 9 °/o of the gross take-off weight. It has a length of 76.5 m and a span of 38 m.

The hypersonic transport will typically accelerate at q = .5 kg/cm^2 until the cruise Mach number and will than increase its altitude towards the level of optimal cruise (ref. 15) at close to maximal lift-over drag ratio. (Fig. 1 and 2). For a wing loading of 200 kg/m^2 (ref. 15) the lift coefficient is then C_L = 0.04. The airbreathing space shuttle which might come after the rocket powered launchers and orbiters will closely follow the same ascend trajectory as the HST (ref. 16).

At sustained hypersonic flight the vehicles will be propelled by ramjets or by scramjets (supersonic combustion ramjets), the former for low hypersonic Mach numbers ($M < 6$ to 7) and the latter for high hypersonic Mach numbers ($M > 6$ to 7). Figure 3 (ref. 14, 16) illustrates the superiority of the scramjet operation over the subsonic burning ramjet at high Mach numbers (H_2 fuel). Dual-mode scramjets which can operate with either subsonic or supersonic burning of fixed geometry are studied in the USA by AirResearch/NASA and in Europe by ONERA (ref. 17). ONERA has also performed an extensive wind tunnel testing program on hypersonic transport configurations (ref. 18). The other establishments in Europe who performed detailed studies on hypersonic lifting vehicles are the RAE and the DFVLR (ref. 4). Much work has been concerned with wave riders, based on simple flow fields such as wedge flows and cone flows. The significance of these shapes is seen not only for the hypersonic transport (refs. 19, 20, 21) but also for the space shuttle (ref. 11). It is advocated as distinctly European contribution to the design of lifting propulsive bodies (ref. 22).

1.4 Scramjets

The advantage of supersonic combustion is mainly due to the increased inlet performance relative to the subsonic burning mode (the flow remains supersonic, hence less static pressure rise and no normal shock losses), the low static temperature in the burner ($\sim 1000^\circ K$ versus well over $2000^\circ K$ and hence more sensitive heat addition due to less dissociation after combustion) and, last but not least, improved nozzle performance due to less freezing of chemical species in the expansion process (ref. 23). In case of a subsonic burning ramjet the inlet can be tested separately from the combustor and the combustor usually does not yield much performance problems since the subsonic combustor technology is well established. The increase in flight speed will only ease the burning rate problem due to the increased temperatures; however, cooling and material problems will show up.

The technology of supersonic combustion is however still rather new and much should still be done before good performance assessment is possible, particularly if the engine has to be run both at the subsonic and supersonic burning mode during the acceleration fase. A good match must be made between the three components: inlet, combustor and nozzle.

Two main problems exist for scramjet propelled aircraft namely, the performance assessment of the isolated engine and the engine integration into the airframe. At hypersonic speed the required engine frontal area (or free stream capture area) increases strongly with respect to the supersonic speed case. For example for Concorde the total inlet area is less than 1 °/o of the wing area whereas for a $M = 7$ airplane this ratio is of the order of 3 to 4 °/o. This increase is mainly due to the fact that the net thrust is only a small fraction of the engine gross thrust (the same situation as with high by-pass ratio fan engines at subsonic speed). This makes that inlet, supersonic combustor and nozzle performances are very critical since a 1 °/o gross thrust loss might give a 10 °/o increase in fuel consumption. Further, this large relative inlet area increases the engine-interference problems, both at the inlet side, but mainly at the nozzle side.

Studies (ref. 24) have shown that deflection of the large gross thrust vector can yield significant gains in lift with little loss in available thrust. In addition the pressures of an underexpanded nozzle flow of a HST configuration could provide favourable interference effects if the exhaust flow washed a large area of the wing lower surface (ref. 25). These engine forces will cause trimming problems of the aircraft.

Concluding: the airbreathing engines of a hypersonic vehicle form such an integral part of the aircraft that engine simulation (both inlets and exhaust) should always be performed and the installed engine performances must be carefully assessed on special test benches, which will look like hypersonic wind tunnels. The ONERA S4MA tunnel for instance is in fact a pebble bed heated wind tunnel with its test section nozzle replaced by a complete ramjet/scramjet engine (ref. 26).

2. FLOW PARAMETERS TO BE SIMULATED IN GROUND FACILITIES

In order to obtain information on the behaviour and performance in the design and development phase of the vehicles described above, wind tunnel testing under simulated environmental conditions is indispensable. This testing includes aerodynamics, propulsion systems and hardware. These three aspects may often be treated separately but also combined studies are needed, for instance airframe-engine integration (aerodynamics plus propulsion) and engine endurance and reliability testing (propulsion plus hardware).

2.1 Aerodynamic testing

The hypersonic flow regime can be divided into three regimes with different flow parameters of primary interest:

- the low hypersonic regime from Mach 5 up to say Mach 10 or 12, where duplication of the Mach number and the Reynolds number are of primary interest
- the hypervelocity regime where the flow velocity (or enthalpy) and the flow density are most important parameters, rather then Mach number and Reynolds number and where real gas effects may play an important role
- the low density regime at altitudes above 50 to 70 km where the mean free path between the molecules becomes comparable with characteristic body dimensions. The ratio of the mean free path and the laminar boundary layer thickness is proportional to $M/\sqrt{Re}$, which is the main parameter to be simulated. The low density effects become important above about $M/\sqrt{Re} = 0.01$ (ref. 4).

During the greater part of re-entry high temperature effects (hypervelocity regime) as well as low density effects are present. The scaling laws for these flight conditions are so different (convective versus radiative heating, non-equilibrium chemistry) that only partial simulation in ground facilities is possible (Ref. 27), when no full scale testing is done.

For the low hypersonic regime, with or without low density effects present, aerodynamic testing of scaled models is well possible.

In figure 4 the trajectories of figure 2 have been translated into a Mach number-Reynolds number diagram. Also the boundary of continuum flow $M/\sqrt{Re} = 0.01$ is indicated. In the next paragraphs its consequences for the flow parameters to be simulated for the space shuttle orbiter and the hypersonic transport are discussed in somewhat more detail.

2.1.1 The space shuttle

A good review of the aerodynamic problems related to the space shuttle vehicle is found in reference 28. Although data on the final North American Rockwell configuration are not presented, the data for the high cross-range orbiter (Ref. 28, p. 285 e.g.) show good agreement with the simplified trajectories presented in figure 4.

Ideally the flow conditions around the full scale vehicle should be duplicated around the model. This is done by duplicating the Mach number M, the Reynolds number Re and the wall temperature to free stream temperature ratio T_w/T_∞ if high temperature real gas effects are excluded for the moment.

2.1.1.A Mach number

Mach number duplication is necessary for shock shape duplication. It is known however, that at high enough Mach numbers the shock becomes very close to the under-surface of space shuttle-like bodies at representative angles of attack and is almost insensitive to further increase in flow Mach number. Also the situation at the leeward side where severe flow separations exist becomes insensitive to Mach number changes. It is therefore suggested that duplicating the flow Mach number at Mach 15 or 20 is not vital for the space shuttle (Ref. 29).

This suggestion is supported by looking at the slip flow boundary in figure 4. In the slip flow regime where the Mach numbers are above 15 to 20, the force data can be correlated on the rarefaction parameter $M/\sqrt{Re}$. For more details see section 2.1.1.C.

2.1.1.B Reynolds number

typ

Reynolds number duplication and temperature ratio T_w/T_∞ duplication is required for duplication of the boundary layer thickness and the ~~kind~~ of boundary layer (laminar or turbulent).

For a proper design of the thermal protection system knowledge of the location of the boundary layer transition region is essential. According to reference 30, there is yet no definite conclusion how the transition data, found in wind tunnels, should be interpreted for the full scale vehicle.

In reference 31 some new free flight transition data are presented and various transition criteria are discussed. For local Mach numbers above 5, the Reynolds number based on the conditions at the edge of the boundary layer and the wetted length Re_{x_t} which indicates the onset of transition was somewhere between 10^6 and 10^7 (data scatter). Below $M_e \approx 4$ values of Re_{x_t} between 10^4 and 5×10^6 are found with a data scatter of about two orders of magnitude. These lower Re_{x_t} values are obtained at high angles of attack on the lower side of the vehicle and are probably largely influenced by 3-dimensional effects. It should be remarked that Re_x is not the best correlating parameter but rather Re based on a boundary layer thickness, in combination with the local Mach number and Reynolds number per unit length (Ref. 31). In reference 31 it was found that for a $\alpha = 40^\circ$ re-entry onset of transition starts on the lower side of the vehicle somewhere between 70 and 80 km but reference 32 gives about 65 km.

Comparison of these data with figures 2 and 4 indicates that the somewhat vague boundary between continuum flow and slip flow coincides more or less with the boundary between full laminar flow and laminar plus transitional flow. Right and below this boundary the Mach number and the Reynolds number are the main flow parameters.

The required Reynolds numbers for correct boundary layer transition duplication in wind tunnels seems to be an unsolved problem, considering the accuracies required for the design of an optimum thermal protection system.

In this context the parameter T_w/T_∞ should also be mentioned. This is an important parameter not only for the skin friction coefficient or the heat transfer coefficient (see for instance reference 33), but also for the boundary layer transition point (see for instance reference 34).

The wall temperature will depend on the method of cooling but may vary between 600 and 1700°K along the vehicle wall, which means that T_{w/T_∞} will vary between about 3 and 8. For wind tunnel testing the maximum Reynolds number is attained when the free stream temperature is as low as possible, the limitation being the condensation temperature of the gas which varies between 30 and 60°K for air and nitrogen, the value increasing with stagnation pressure (Ref. 35). A wall temperature of the model equal to room temperature may be a reasonable value to simulate an average of the non-uniform wall temperature of the full scale vehicle.

In conclusion the best way to deal with the transition problem seems to duplicate the Reynolds number and if possible the temperature ratio T_{w/T_∞} in the continuum regime indicated in figure 4. The boundary layer thickness will than be properly simulated and the transition region generally will be more upstream than on the full scale vehicle (see data of reference 31), which generally will not lead to too optimistic predictions for the thermal protection requirements. Mach number-Reynolds number duplication also eliminates the necessity for possible Reynolds number corrections for the phenomena on the leeward side of the vehicle where large regions of separated flow exist.

2.1.1.C Viscous interactions and low density effects

The space shuttle orbiter will experience peak heating and deceleration at 60-70 km altitude (Ref. 28, 36), where low density effects can certainly not be neglected. The slip flow regime which extends roughly between $M/\sqrt{Re}$ = 0.01 and 0.1 is indicated in figure 4. The low density regimes at higher altitudes such as the near free molecule and the free molecule regime are of less importance for the space shuttle re-entry from a practical point of view. From reference 29 the following remarks are quoted.

The most significant practical effects, as far as overall performance is concerned, occur on slender, high L/D vehicles. Not only are much larger viscous interaction induced forces generated on this type of vehicle, resulting in a large reduction in L/D (see reference 4 for instance), but the effects extend to relatively lower altitudes than the other low density effects.

In the case of the space shuttle a large percentage of the re-entry flight time is spent in manoeuvring in the high altitude regions dominated by rarefied flow effects. It has been established that rarefaction effects are likely to be significant over the whole of a slender vehicle, over localized regions such as on control surfaces, if the value of the viscous interaction paramter $M/\sqrt{Re_{\infty, L}}$ is greater than about 0.01. For a 20m long vehicle this corresponds to about 75 km altitude over the forward surfaces if they were at 40° incidence to about 55 km over the leeward surface at about -10° incidence (end of quotation).

In order to have an indication of the viscous interaction effects on the longitudinal range, calculations have been made for the highest re-entry trajectory of figure 2, where these effects will be most significant. For the sake of simplicity it is assumed that C_L remains unaffected. The effect on L/D is assumed to be represented by reference 37, figure 26 : at $M/\sqrt{Re}$ = 0.1 the L/D value is half the inviscid value of 0.6.

Starting from the vertical equilibrium condition at an altitude of 90 km the following longitudinal range is attained when the flight velocity has decreased to 1 km/sec : L/D = const. R = 6200 km and if L/D influenced by viscous effects R = 4600 km or about 75 percent of the inviscid value. This is the result of a rather crude calculation (numerical integration in eleven steps).

For vehicles with much higher L/D values for the same W/SC_L the influence of viscous interaction on the range performance is even much larger. In reference 38 calculations show for a L/D = 4 vehicle reductions in range of more than 50%. For a discussion on the relevance of such high L/D vehicles the reader is referred to references 11 and 39.

The examples mentioned show that knowledge of the aerodynamic behaviour at high altitudes is essential for the assessment of the vehicle performance. Analyses which do not take low density effects into account can give misleading results.

It is concluded from figure 4 that aerodynamic testing of the vehicle behaviour should be done for values of $M/\sqrt{Re}$ between 0.01 and 0.1 at Mach numbers above, say 15 (see also sub 2.1.1.A). Its significance should be considered against the background of the influence of high temperature real gas effects.

2.1.1.D High temperature real gas effects

At very high altitude, where the flow is completely free molecular and at low altitudes where the air is continuum, forces acting on a vehicle can be predicted theoretically with a good degree of certainty. Between these limits, however, the flow is a complex function of each type and most of our understanding has to be gained by experiments in wind tunnels (Ref. 29). Very unfortunately in this same area high temperature real gas effects complicate the picture considerably.

A discussion on the problems of aerodynamic testing in this hypervelocity regime where the flow velocity itself and the free stream density are the primary flow parameters can be found in reference 27. Proper scaling is difficult and often completely impossible : radiative heat transfer for instance is proportional to the nose radius of the vehicle and convective heat transfer is proportional to the inverse of the square root of the nose radius. Also non-equilibrium effects may become important, especially at higher altitudes : relaxation lengths of the order of several meters may occur on a space shuttle (Ref. 40).

The dissociation and ionisation of the air in the stagnation region and elsewhere around the vehicle could lead to adverse effects on lift, trim and heat transfer. Prediction of these effects is still uncertain ; they cannot be investigated in a wind tunnel of the usual blow down type, for the very process of expansion through a nozzle from stagnation conditions representative of high altitude re-entry causes thermodynamic non-equilibrium in the nozzle flow which is not like the flow around the vehicle in the real atmosphere.
An other reason is the incompatibility of the scaling laws forconvective and radiative heat transfer and for non-equilibrium chemistry (binary and tertiary collisions of recombining molecules for instance) (Ref. 27).

What can be done in this area experimentally is providing data for theoretical predictions such as chemical reaction rates and radiative heat transfer data obtained from facilities like shock tubes, expansion tubes and plasma facilities, eventually boosted by a magneto hydrodynamic device and the development of high performance facilities where flow non-equilibrium is largely avoided (see Ref. 41).

2.1.2 The hypersonic transport

In figure 2 the trajectory is given of a hypersonic transport accelerating at a constant dynamic pressure of $q = 0.5$ atm up to a non-specified cruise Mach number (Ref. 15). The corresponding flight conditions are plotted in the Mach number-Reynolds number diagram of figure 4. It is seen that the Reynolds numbers based on vehicle length for a given Mach number, are one to two orders of magnitude larger for the HST than for the space shuttle during re-entry. The consequences for the aerodynamic parameters to be simulated are discussed below.

2.1.2.A Mach number

The Mach number must be duplicated in wind tunnel tests to obtain the same shock shape as in real flight. This requirement should not be violated as was permitted for the high Mach number tests for the space shuttle. This may be illustrated by reminding to the engine intake region where the position of the shock waves from the external compression surface relative to the intake lip should be duplicated accurately.

2.1.2.B Reynolds number

From figure 4 it follows that a hypersonic transport designed for a cruise Mach number of 6 to 8 (see Ref. 15 for a typical example) will fly at Reynolds numbers well above 10^8, based on the total vehicle length. Although the significance of the transition data obtained in wind tunnels for the full scale vehicle is still not definitely settled (Ref. 30), the Reynolds numbers at which the HST will operate are so large that the boundary layer on the vehicle will be almost completely turbulent.

In reference 42 shock tunnel measurements are described on a HST-model over a Reynolds number range from about 0.5 million to 160 million. It is found that transitional boundary layer effects on the axial force coefficient begin to emerge at Re_L of about 2 million. These effects predominate for about a decade in Reynolds number until the turbulent boundary layer exerts the major influence at Reynolds numbers of about 20 million.

From this figure it is concluded that a Reynolds number of about $Re_L = 20$ million is a minimum requirement for wind tunnel tests for the development of hypersonic transport aircraft where absolute performance data should be obtained. Reliable extrapolation to the much higher full scale Reynolds numbers seems feasable in that case (Ref. 42).

The value $Re_L = 20$ million is in fact still open to discussion. A value of 50 million as was presented in the first provisional version of the present paper is probably on the safe side but a value of 5 million as suggested in reference 29 is apparently too low when the drag data of reference 42 are considered. Up to what value of Re_L testing is necessary will also depend on the required accuracy of the data which have to be extrapolated to full scale Re_L values.

When the model is tested at Reynolds numbers below say 20 million, the transition region moves too far downstream. In that case the transition might be moved upstream again by artificial trips, but at hypersonic velocities the trips must be so large that even the flow outside the boundary layer is disturbed, causing an additive interference drag (Ref. 29, 43, 44). Artificial boundary layer transition is therefore not an attractive method at hypersonic speeds.

For cases where the boundary layer itself is an important parameter such as for engine intakes preceded by a compression ramp and for shockwave-boundary layer interaction as for instance occurs near flap hinges, the consequences of testing at lower Reynolds numbers than the full scale values should be considered with great care. The turbulent boundary layer thickness being inversely proportional with the one-fifth power of the Reynolds number, testing at $Re_L = 2 \times 10^7$ gives a boundary layer thickness which is about 60 % larger than testing at the real value of $Re_L = 2 \times 10^8$ at Mach 8. Isolated testing of partial models in the correct flow environment may yield usable results in these cases.

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The average wall temperature/free stream temperature ratio in the wind tunnel will have a value of the same order as for the full scale vehicle when the tunnel is operated near its condensation limit and the model is at room temperature (see also 2.1.1.B).

2.1.2.C Low density and real gas effects

Only near the leading edges viscous interaction effects may occur. Testing at too low Reynolds numbers may lead to undue conclusions such as concerning aerodynamic heating. Partial model testing may be useful in this area. At speeds up to, say Mach 8 the high temperature real gas effects on the aerodynamic behaviour are still small. For refined measurements, however, they may be taken into account.
As an example reference 45 gives for the lower side of a flat plate flying at 30 km altitude at Mach 8 at an angle of incidence of 20° a pressure coefficient which is about 1 percent lower than in a wind tunnel where the free stream temperature is 55°K.

High temperature real gas effects are however very important as far as the airbreathing propulsion is concerned (see section 2.2).

2.1.3 Conclusions for aerodynamic testing

For space shuttle type of vehicles the following is concluded for the flow parameters to be simulated for wind tunnel testing.

The Mach number should be duplicated up to about 15 to 20. Above this Mach number the viscous parameter $M/\sqrt{Re}$ is of primary importance for the aerodynamic behaviour, when $M/\sqrt{Re} > 0.01$. When $M/\sqrt{Re}$ is smaller than this value, the flow behaves as a continuum and for the space shuttle this coincides approximately with the onset of boundary layer transition. Interpretation of wind tunnel transition data for full scale flight behaviour still being in discussion, the best to do is testing at duplicating Mach number and Reynolds number when $M/\sqrt{Re} < 0.01$. This also gives the correct boundary layer thickness.

A model temperature equal to room temperature will provide a reasonable T_w/T_∞ value in many cases. In practice, however T_w is non-uniform.

High temperature real gas effects such as occur during a great part of the re-entry may be studied experimentally by partial simulation only.

For hypersonic transports development testing in wind tunnels the following conclusions are made.

Mach number duplication is essential to duplicate the shock shape. The boundary layer being predominantly turbulent, in many cases wind tunnel testing at Reynolds numbers above, say 20 million allows probably good prediction of the vehicle performance by a correction of the skin friction to the full scale Reynolds number value.

For cases where the boundary layer itself is an important parameter such as for engine intakes and shockwave-boundary layer interaction, Reynolds number duplication and/or partial model testing will be necessary.

A model temperature equal to room temperature will provide a reasonable T_w/T_∞ -value in many cases. High temperature real gas effects on the aerodynamic performance are rather small up to say Mach 8.

2.2 Propulsion testing

Since the ramjet and scramjet do not contain devices to increase the total pressure of the internal flow (turbomachinery), the engine flow simulation at hypersonic airplane models is in principle easier to achieve than at the lower speed regimes. If the internal flow is not heated, either in a subsonic or supersonic stream, the nozzle total pressure will not be matched with the scaled nozzle geometry and the stream tube leaving the nozzle will be too small. Hence, the interference with the outer flow is wrongly matched. Two methods are available to obtain the simulated nozzle flow field and pressure distribution, namely by adding large quantities of additional gas such that $\dot{m}\sqrt{RT}$ is simulated for the nozzle, or burning a fuel within the internal flow. In the latter case it is required to use air as the tunnel fluid and the total temperature of the air should be duplicated if the same fuel is used as for the full scale flight. In that case the scaling law for the burning rate process is approximately equal to the scaling law for the Reynolds number ($p.l$ = constant). The high total temperature required for duplication however, is in conflict with the lowest possible stagnation temperature for maximal Reynolds number and wall-free stream temperature ratio. Until recently few tests have been performed on HST models with simulation of the engine flow (Ref. 24 and 25).

In a scramjet the heat release within the supersonic flow is either of a 2-dimensional or 3-dimensional nature. As yet a good understanding of heat release in multidimensional supersonic flows has not been attained. A very strong unknown interplay exists between the chemical kinetics, mixing, fuel jet penetration, shock waves and duct area. Local heat release in supersonic flow will cause thermal compression, however shock waves should be avoided. In supersonic combustion tests the entrance Mach number, pressure level, temperature and chemical composition are of primary importance. For propulsion testing the total engine mass flow and flow duration give additional requirements to the test facilities.

Figure 5 gives in the flight Mach number-altitude plot the required stagnation temperatures, pressure and mass flow per unit capture area of the inlet for complete environment duplication. For large hypersonic Mach numbers these conditions are hard to achieve in the laboratory as is also the case for hypersonic wind tunnels. Therefore in the next discussion emphasis will be focussed on the parameters which are of primary importance for engine and combustion tests.

2.2.A Mach number in the combustor

It is evident that Mach number duplication in the combustor is essential due to the strong interaction between the flow field (compression) and local heat release. In the following it will be always assumed that the Mach number is duplicated.

2.2.B Temperature

For low static temperature at the supersonic combustor entrance the overall reaction rate will be limited by the chemical kinetics, whereas at sufficient high static temperature the turbulent mixing between the fuel and air will be the rate limiting factor. Hence static temperature duplication in the combustor is of primary importance at low hypersonic Mach numbers, since if the flow is decelerated from hypersonic speed to supersonic speed in the combustor, the static temperature will be between the ambient static temperature and stagnation temperature depending upon the Mach number ratio. For various performance reasons, a rough rule of thumb is that the combustor entrance Mach number is about one-third of the flight Mach number (Ref. 23). Figure 6 gives the typical static temperatures versus Mach number for the NASA Hypersonic Research Engine (HRE) burning hydrogen (Ref. 46). Figure 7 shows the importance of the initial temperature for hydrogen as fuel. Since hydrogen is the most probable fuel for scramjets for performance and cooling reasons, the combustor entrance static temperature should be above about 1000°K. Over about 1500°K mixing will be the dominant factor. These conditions occur at $M \gtrsim 10$ to 11. This means that for supersonic combustion tests the stagnation temperature should be duplicated up to M = 10 à 12, hence T_o = 4000°K to 5000°K. Other fuels might also be used such as the metalized fuels (for example Trimetylaluminium, Triethylaluminium, Trimethylborane) or hydrocarbons (Ref. 23). The first group will yield spontaneous ignition even at atmospheric temperatures, whereas the latter group will also need preheated air for fast ignition.

2.2.C Pressure

The static pressure level in the supersonic combustion chamber depends primarily on the flight altitude, the inlet process efficiency (K_D) and inlet Mach number ratio. The actual value will be at about 1 kg/cm^2 (say 1/5 to 5 kg/cm^2). Figure 8 gives some typical inlet values as will be encountered in flight, respectively total pressure ratio, and static pressure levels (Ref. 48). Particularly in this pressure regime and for temperatures between 1000 and 1500°K the ignition delay time for hydrogen is a strong irregular function of the pressure, making use of appropriate scaling laws for pressure unsuitable (Fig. 9). For higher temperatures the induction time is inversely proportional to the pressure level ($\tau_{i.t.} p = f(T)$).

Once the chemical reaction is started the characteristic reaction time for hydrogen is proportional to $p^{-1.65}$. For hydrocarbon the reaction time is proportional to $p^{-1,8}$ (sometimes also taken as p^{-2}). (See also Ref. 49).

Therefore for good understanding of the combustor phenomena the pressure level should be duplicated as well as the geometry. Scaling laws can only be used if the overall chemical kinetics behaviour can be described by simple rules and if variable induction times do not exist.

Ideally the hypersonic engine (scramjet) should be placed in the freejet of a hypersonic facility, duplicating the stagnation condition (temperature and pressure) and free stream Mach number in which the complete system can be tested (inlet, combustor and nozzle). This may be difficult to achieve in a hypersonic wind tunnel due to the high required stagnation levels, particularly if the duplicated Mach number approaches the value 8 (see section 3.2).

One means of omitting the high stagnation pressure level is to utilize a direct connection setup at which the flow is expanded only to the required supersonic speed in the combustor (See fig. 10 of ONERA from Ref. 50). This will reduce the required tunnel reservoir pressure by the ratio as indicated in figure 8 (typically a factor 2 to 5, depending inlet geometry). The inlet performance and combustor extrance flow field can be determined in separate wind tunnel tests at the full scale Reynolds number (but reduced temperature) at the representative relative boundary layers thickness.

2.2.D. Mass flow

The required mass flow for propulsion testing depends primarily on the required net thrust to overcome drag, to accelerate and to climb. The first term is of primary importance and depends on the flight L/D ratio of the vehicle. For a cruise vehicle this value will be between 4 and 6. For a hypothetical 200 ton HST with four engines the net thrust should be over 10 tons. Flying for example at $M = 7$ where the thrust coefficient is about $C_T = 0.6$,

$$\left(C_T = \frac{F_N}{q\ A_{capture}}\right)$$

for a hydrogen fueled engine, the required mass flow per engine is about 200 kg/sec. This value increases for other fuels. Typically, using kerosine the required mass flow will roughly double the value using H_2 at $M = 7$. It can be computed that the HRE of NASA-Lewis consumes about 5 kg air per second at $M = 7$ and 100 000 ft altitude.

A mass flow rate of 200 kg/sec requires a combustor entrance of 0.4 m^2 at $M_{entrance} = 2.3$. These conditions are rather typical for medium size unheated supersonic wind tunnels, but are exhaustive for hypersonic wind tunnels, which are necessarily fitted with a heating system. Only the Tripltee tunnels at NASA Langley and at AEDC (under design) fulfil these conditions up to $M \approx 7$ (Ref. 51 and 52).

2.3. Hardware testing

2.3.A. Ablation testing

For the structure of hypersonic vehicles one of the most significant paramters is the aerodynamic heat load to which it will be subjected. The magnitude of the heat load and the exposure time are rather different for the space shuttle and for the hypersonic transport and hence the structural concepts which deal with these heat loads. The feasibility of these concepts will have to be verified by hardware testing under simulated flight conditions.

For the space shuttle the heat transfer rates are one order of magnitude larger than for the HST (Ref. 53), but the total heat transferred to the vehicle (per unit wetted surface and per flight cycle) will be much smaller than for the HST.

For the space shuttle a passive heat protection system has been considered for early versions consisting of ablative material on an aluminium substructure, which requires refurbishment after each flight (Ref. 8). At the moment of writing this paper, three different insulation systems are foreseen: a low-weight elastromer on the upper surface (up to 340°C), a new ceramic material on the lower surface (up to 1370°c) and a new oxidation-inhibited, reinforced carbon material for the wing leading edges and the nose cap where temperatures up to 1650°C are anticipated (Ref. 54).

A discussion on the testing of ablative materials which is also useful to understand the problems of testing of non-ablative protective coatings can be found in reference 55.

For ablation studies near the nose region of a re-entry body the major requirement is to simulate the stagnation enthalpy and the pitot pressure on the model (Ref. 27). For the shuttle lower re-entry trajectory of figure 2 the following conditons are found at the vehicle stagnation point (equilibrium flow assumed) (Ref. 56).

Altitude H (kft)	300	250	200	150
Velocity u (kft/sec)	25.8	25.0	22.6	12.1
Stagn.enthalpy h_s/R (°K)	10.8×10^4	10.2×10^4	$8,8 \times 10^4$	2.4 a 10^4
Pitot pressure p_s (atm)	1.39×10^{-3}	1.97×10^{-2}	1.20×10^{-1}	2.3×10^{-1}
$\dot{q}\sqrt{r_n}$ $\left(\frac{BTU-in}{ft^2-sec}\right)$	73	262	533	201

$\dot{q}$ is the convective heat transfer rate at the stagnation point and r_n is the nose radius in inches. For nose radii up to 1 foot the radiative heat transfer is more than one order of magnitude less than the convective heat transfer (Ref. 57).

It is found that the required values of stagnation enthalpy can be generated in arc heater facilities but for $h_{s/R} = 10^5$ °K the reservoir pressure should not exceed 5 atm (Ref. 58, state of the art 1961). The total pressure in a wind tunnel, required for flow duplication, is however of the order of 10^3 to 10^6 atm.

The solution is to duplicate the stagnation enthalpy and to test the model at fairly low Mach numbers, typically Mach 2-5 (Ref. 27, 55).

2.3.B Structure testing

For the hypersonic transport a rather simple passive thermal protection system as for the space shuttle will not be employed, but the walls are to be cooled by the hydrogen fuel. This cooling may be accomplished directly, using wall materials with good thermal conducticity, or indirectly by blowing pre-cooled sheets of air over the outer wall surface i.e. by slot cooling (Ref. 53). These active cooling systems are much more vulnerable to failures and extensive testing will be necessary on reliability, thermal fatigue, effect of transients, etc. The same arguments are valid for the testing of the HST propulsion system. A discussion on the development and hardware testing of airbreathing engines for large hypersonic vehicles can be found in reference 51.

The facility requirements for development testing of full or large-scale airbreathing propulsion systems and associated airframe can be appreciated by considering the requirements as described in section 2.2. The same requirements for complete flow duplication (p_t, T_t, M) are valid, and result in the use of large tripltee tunnels. The required running time of these tunnels will largely influence the design of the tunnel heating systems. It is generally agreed that a least a few minutes running time is required, however, aeropropulsion people quote figures as high as 15 minutes.

It should be noted that such a large facility allows also Mach number-Reynolds number duplication of the HST up to Mach 7 when the tunnel is operated at equilibrium condensation condition (see fig. 11), $p_o L_m$ being of the order of 750 atm.m.

3 PRACTICAL AND PRINCIPAL LIMITATIONS FOR GROUND TEST FACILITIES

3.1 Hypersonic wind tunnels for aerodynamic testing

For the bulk of aerodynamic tests at hypersonic speeds in ground test facilities the primary aim is to duplicate the Mach number and the Reynolds number of the full scale flight condition (see section 2.1). In order to achieve this with the least amount of energy the tunnel is operated at the lowest possible temperature. This temperature follows from the requirement that the static temperature in the test section flow should not be below the condensation temperature (Ref. 35). In figure 11, which will be discussed later, the stagnation temperatures, necessary to avoid equilibrium condensation are indicated as a function of Mach number and stagnation pressure for a perfect gas. For a real gas the minimum stagnation temperatures are lower than indicated in figure 11 (dashed lines).

Calculations for a real gas (with Refs. 35, 45 and 59) show that at Mach 18 at minimum stagnation temperature the degree of dissociation of the gas in the stagnation region of a blunt body model in the test section is about 1 $^o/o$ for stagnation pressures between 10 and 1000 atm. It is concluded that in a wind tunnel operated at minimum stagnation temperature, high temperature real gas effects are restricted to molecular vibration only, when the test section Mach number is below Mach 18. The discussion on aerodynamic facilities in this section 3.1 will be confined to facilities operating under conditions where molecular vibration is the only high temperature real gas effect to be taken into account. Its effect may, however, still be considerable: the pitot pressure may be 65 $^o/o$ of the ideal gas value for instance (Ref. 45, fig. 20).

Methods to correct the wind tunnel data for the full scale vehicle real gas effects such as dissociation and ionization will not be discussed here. They may be provided by theoretical analyses supported by experimental data of a more fundamental character such as radiative heat transfer measurements and chemical reaction rate data and partial flow simulation.

The "pure" aerodynamic phenomena are duplicated when the Mach number and Reynolds number around the vehicle are duplicated and the correct wall temperature-free stream temperature ratio T_w/T_∞ exists. For practical reasons the non-uniform wall temperature distribution along the full scale vehicle is often approximated by a uniform wall temperature of the wind tunnel model (room temperature) Non-uniform increase of the wall temperature during wind tunnel tests should not be overlooked. At flight conditions where $M/\sqrt{Re}$ is larger than about 0.01, i.e. during high altitude re-entry, the parameter $M/\sqrt{Re}$ should be duplicated, rather than M and Re separately.

In order to find the stagnation pressures needed to generate the necessary Reynolds numbers, indicated in figure 4, calculations have been performed for a wind tunnel model with a standard length of 1 meter, operated at the equilibrium condensation temperature for air (Ref. 35). This will give the highest Reynolds number for a give stagnation pressure and Mach number. No real gas effects have been taken into account. The results have been plotted in figure 11.

Real gas effects however, may have considerable influence as is indicated in the example below, calculated with aid of reference 59.

real gas				equivalent perfect gas	
p_o atm	T_o ^oK	Re/m at m^{-1}	$M_{cond.}$ -	p_o atm	T_o ^oK
488	1500	2.2×10^6	12.77	379	1671
1072	1500	6.5×10^6	12.56	1085	1736
2308	1500	18.7×10^6	12.48	3241	1846
4920	1500	50.7×10^6	12.80	10960	2084

The real gas effects are in fact twofold, namely high pressure effects (van der Waals effects) and high temperature effects. For given free stream conditions at hypersonic Mach numbers the stagnation temperature is lower than according to the perfect gas calculations and the stagnation pressure can be either higher or lower.

In reference 60 the combination of both effects has been considered and is presented in graphical form, two figures being reproduced as figures 12a and 12b of the present paper, valid for nitrogen, which closely resembles air. These graphs can be used in combination with figure 11, which is valid for a perfect gas, to calculate the real required tunnel stagnation conditions.

As was already pointed out, above $M/\sqrt{Re} = 0.01$ the viscous interaction paramter $M/\sqrt{Re}$ should be duplicated rather than Mach number and Reynolds number. From figure 11 it follows that this procedure allows testing at lower stagnation temperatures and pressures than if Re and M both had to be duplicated along the complete re-entry trajectory of a space shuttle. This is a very important consideration from a facility engineering point of view.

It can be discussed whether M-Re duplication for the lower shuttle re-entry trajectory should be pursued up to Mach 19 where $M/\sqrt{Re} = 0.01$, because of the necessary high values for $p_o L$ and T_o (facility engineering and high-temperature real gas effects). For a model length $L_m = 1$ meter figures 11 and 12 give as required real gas stagnation conditons about $p_o = 2600$ atm and $T_o = 2900^oK$ at the point considered.

A minimum tunnel requirement may be that M-Re duplication must be possible up to such a value that interpolation to the $M/\sqrt{Re} > 0.01$ data is possible with acceptable and reasonable accuracy. This value may be different for tests where for instance boundary layer transition is important (heat transfer for example) or for force tests where M and $M/\sqrt{Re}$ are more important. In the following sections the practical and principal limitations will be discussed, which are important for validation of the performance of hypersonic facilities for aerodynamic testing.

3.1.A Stresses in the sting support

Due to the aerodynamic forces stresses will develop in the model and the sting which supports it. For a given model and sting support these stresses are a function of the angle of incidence and the dynamic pressure only and a weak function of the flow Mach number.
The stress in the sting with diameter d_s at the model base is calculated for a static load assuming that the normal force N (perpendicular to the model axis) acts on a point at a distance 2/3 L_m from the nose.

For the space shuttle a minimum sting diameter $d_s = 0.1$ L may be employed (about equal to the base diameter of the vehicle) and for the HST a value $d_s = 0.04$ L is a realistic value(Ref. 61). The bending stresses due to aerodynamic forces in the sting are then calculated from the moment

$$M = (1 - \frac{2}{3})LN = \frac{1}{3}L^3 \quad C_N q . \frac{S}{L^2}$$

where L is the model length, S the reference area for the normal force coefficient C_N and q is the dynamic pressure. For a solid circular sting follows

$$\sigma = 3.3(\frac{L}{d_s})^3 \frac{S}{L^2} \quad C_N q$$

A value $\sigma = 5000$ kg/cm^2 at nominal static aerodynamic load is considered as the structural limit: starting and stopping loads and a safety factor are not included. Also the strength of the model itself and the allowable elastic deformation of model and support are not considered.

When a maximum normal force coefficient C_N is assumed for the space shuttle $C_N = 1.5$ (see chapter 1.2), which is the modified Newtonion pressure coefficient for an angle of incidence of 65 degrees, and for the HST a maximum $C_N = 0.25$ corresponding with an angle of attack of 16° for a typical configuration at Mach 6 (Ref. 62) the following maximum dynamic pressures q_m are calculated for a stress in the sting $\sigma_m = 5000$ kg/cm^2 from

$$q_m = 0.3 \ \sigma_m \ (\frac{d_s}{L})^3 \ \frac{L^2}{S} \quad \frac{1}{C_N}$$

This gives a maximum dynamic pressure $q_m = 1.5$ kg/cm^2 for the space shuttle and $q_m = 2.2$ kg/cm^2 for the HST. This may limit the performance for facilities with high stagnation pressure capabilities at low hypersonic Mach numbers.

For instance at Mach 8 a maximum stagnation pressure of 325 atm is allowed for space shuttle testing if q is limited to 1.5 kg/cm^2 and to 480 atm for HST-testing at $C_N = 0.25$. In reference 42, however, a HST-model has been tested at a stagnation pressure of 1300 atm. In that case however C_N was about 0.085 and the sting diameter d_s was about 0.05 L_m. This will give a stress in the sting of 2400 kg/cm^2 which is well below the limit set at 5000 kg/cm^2. For $d_s = 0.04$ L_m the stress would have been 4600 kg/cm^2, which indicates that the limits for q_m of 1.5 kg/cm^2 for the space shuttle and 2.2 kg/cm^2 for the HST are not exact boundaries. It will be largely dependent on model-sting geometry and safety factor which has to include starting and stopping loads which are also facility dependent and the maximum α at which the HST is to be tested.

An other criterion may be derived from the assumption that the space shuttle model should be tested at values of dynamic pressure and angle of incidence where the normal force is four times the weight of the vehicle along the full scale trajectory (3-g is maximum design acceleration) and for the HST a normal force of two times the weight of the vehicle is attained, corresponding with a 2-g turn.

For a full-scale model and flow duplication the stresses in the sting would then be calculated from

$$\sigma_f = 3.3\left(\frac{L}{d_s}\right)^3 \frac{S}{L^2}\ C_N\ q_f$$

where $C_N . q_f = 4\frac{W}{S} = 800$ kg/m^2 for the space shuttle and $C_N\ q_f = 2\frac{W}{S} = 500$ kg/m^2 for the HST. However, due to the fact that the tunnel can operate at a much lower temperature, the Reynolds number can be increased by a factor of about 4 (was 3 in the draft version of the present paper) when M-Re duplication is employed at tunnel equilibrium condensation conditions instead of flow duplication.

Introducing a scale factor $B = L_f/L_m$, the ratio between the full scale length and the model length, one finds for M-Re duplication of the N = 4W and N = 2W conditions that $\sigma_m/B = \frac{1}{4}\sigma_f$ = 22 kg/cm^2 for the space shuttle and 111 kg/cm^2 for the HST (σ is proportional with q or p_o).

For a maximum stress σ = 5000 kg/cm^2 the maximum scale factor B or minimum model length L_m is found as

Space shuttle	B_{max} = 228	$L_{m_{min}}$ = 0.15 m
Hypersonic transport	" 45	" 1.70 m

When $Re_L = 2 \times 10^7$ is required and not full scale Re_L duplication then the model size can be decreased proportionally, keeping the stagnation conditions constant.

Both sting load criteria can be worked out and the results are found in figure 13. The p_o = constant curves for M-Re simulation were calculated from figure 11. The sting load limits are indicated in figure 13 as C_N = 1.5 (0.25), which is equivalent with q_m = 1.5 (2.2) kg/cm^2 and as N = 4W (2W) valid for the space shuttle (HST).

Although these sting load criteria are not to be used as exact figures it follows from figure 13 that both criteria indicate approximately the same minimum model size (within a factor 2) and that the minimum model size for M-Re duplication of the HST is one order of magnitude larger than for M-Re duplication of the space shuttle and testing of a HST-model at $Re_L = 2 \times 10^7$.

3.1.B Maximum stagnation pressure

In figure 11 the stagnation pressures are presented which generate the required Reynolds numbers over a model with a standard length of 1 meter. The wind tunnel is operated at the equilibrium condensation limit. The curves are valid for a perfect gas.

The results of figure 4 and 11 are used to calculate the required model length L_m for M-Re duplication for various stagnation pressures. The results are plotted in figure 13. Also the minimum model lengths as determined by the tolerable sting loads as was calculated in section 3.1.A are indicated.

From a practical point of view a value of 5000 atm should be considered as an upper limit for the stagnation pressure p_o which can be contained in the reservoir of a blow down wind tunnel: the highest design value of present facilities is 60,000 psi or 4200 atm (Ref. 36, 63). This limit is also indicated in figure 13.

For force testing of wind tunnel models with an internal balance, fairly small models can be used. In reference 64 for instance force tests are reported on a space shuttle configuration in a gun tunnel with a length of 0.10 meter. For more detailed measurements, however, such as pressure and heat transfer distribution on the model surface, larger models are required. For a well instrumented model, such as used for development work, a model length of more than 0.3 to 0.5 m seems to be a sensible requirement. It also makes aerodynamic loading of the sting and the model less critical.

From figure 11 it follows that for a HST configuration with a model length L_m = 0.50 m , $Re_L = 2 \times 10^7$ can be obtained up to Mach 8 with a reservoir pressure p_o = 200 atm. Recent HST studies do not indicate higher design Mach numbers (see also section 13).

For M-Re duplication of the space shuttle lower trajectory conditions the requirements are more demanding than for a HST model at Mach 8 and $Re_L = 2 \times 10^7$ as can be seen from figures 11 and 13. This is due to the fact that the Mach number range of interest is much higher than for the HST. If for instance M-Re duplication is necessary up to Mach 15, a stagnation pressure p_o = 2000 atm is required for a model length of 0.50 m.

It is concluded from figure 13 that stagnation pressures larger than p_o = 2000 atm and models smaller than about L_m = 0.3 to 0.5 m are not very interesting for development testing of the space shuttle or HST- configurations when the "pure" aerodynamic phenomena along the whole trajectory should be simulated, including boundary layer transition.

For M-Re duplication of the HST up to Mach 8 only large models with $L_m \sim 2$ meter and $p_o \sim 1000$ atm can keep model and sting loads within acceptable values when testing up to $C_N = 0.25$ ($\alpha \simeq 16^o$) is required.

3.1.C Facility power

It is found that already for a small hypersonic wind tunnel for development work with a test section of 0.50 m and a stagnation pressure of 1000 atm operating at the condensation limit, very large energies are contained in the flow. For example at Mach 5 the total energy flux trough the test section would be about 600 MWatt and at Mach 10 about 50 MWatt. This lower figure for a higher Mach number is due to the fact that although the velocity is about 50 percent larger, the throat aera is only 5 °/o of the value at Mach 5.
From these illustrative figures it is clear that only blow down facilities are to be considered. Such a facility is charged between the runs with a limited power (compressors, storage heaters, capacitor bombs). The accumulated energy is released during the running time which is only a small fraction of the time interval between the successive runs.

The released power P can be written as

$$P = \frac{1}{2}\,\rho\,uA.u^2 = \frac{1}{2}\,Re_L M^2 a^2 \mu\,\frac{A}{L.}$$

For a given Mach number and Reynolds number and a given free stream temperature, which determines the local speed of sound a and viscosity μ , it is found that the released power increases proportional with a linear dimension of the facility. From this point of view a small facility, working at a high stagnation pressure is attractive. Also facility and model costs will be generally lower than for large wind tunnels working at the same Reynolds number. The minimum size will be determined by considerations, discussed elsewhere in this chapter 3.1.

3.1.D Throat erosion and cooling

The feasability of a high pressure facility however, is not only limited by the strength of the pressure reservoir but also by the limit of throat melting. This becomes a problem at high Mach numbers when high reservoir temperatures are required to avoid condensation of the test gas.

The heat transfer to the wall of the nozzle throat is higer than anywhere in the facility. Its value is given as (Ref. 65, p. 192) :

$$Q = 0.0014\,\rho^* u^* C_p (T_o - T_w)$$

where ρ^* and u^* are the density and flow velocity in the throat. This equation can be written to :

$$Q = 0.56\;p_o\;\;T_o\left(1 - \frac{T_w}{T_o}\right)\;\frac{Watt}{cm^2}$$

where p_o is the stagnation pressure in atm and T_o the stagnation temperature in ^oK.

For water cooled nozzle throats limits for the tolerable heat load are given in reference 58. A practical upper limit is 5 kW/cm^2. For a given p_o this determines the maximum Mach number for condensation free flow in a facility with a water cooled nozzle. It follows for a wall temperature T_w = 600°K (Ref. 65).

p_o (atm) :	100	500	1000	2000	5000
$T_{o_{max}}$ (^oK) :	9100	1225	860	720	645

When these figures are compared with figure 11, it follows that for a wind tunnel with a water cooled nozzle throat only limited possibilities exist when M-Re duplication is to be realized.

Instead of water cooling also film or transpiration cooling of the nozzle throat may be employed such as in the NASA Ames 3.5 ft tunnel (helium cooling) (no reference known with detailed information) and the Northrop Mach 10 hypersonic facility (Ref. 66). Less than 10 percent of the tunnel weight flow is injected upstream of the throat. For a more analytical approach to the problem, see references 67 to 69.
Much higher heat fluxes than 5 kW/cm^2 can be tolerated when running times are employed which are so short that the surface temperature rise is acceptable. In reference 41 the heat flux required to melt a tungsten throat within 1 millisecond is presented as a function of reservoir pressure and temperature (melting temperature is 3700°K) and oxygen-free nitrogen must be used as a test gas to prevent throat erosion.

The permissable pressure is inversely proportional with the square root of the running time (Ref. 70) and from reference 41, figure 6, the following maximum reservoir pressures are permitted for a running time of 100 milliseconds:

p_o (atm):	100	500	1000	2000	5000
$T_{o_{max}}$ (^oK):	10000	5750	4700	4000	3700

It is concluded that throat heating does not limit the M-Re duplication capabilities as required in figure 11, when the maximum Mach number is limited to Mach 18 (see section 3.1) and for running times shorter than 100 millisec. For longer running times and other materials the Mach number limit may be lower.

3.1.E Real gas effects

Almost all hypersonic facilities are based on the blow-down principle where the gas is rapidly expanded and accelerated in a converging-diverging nozzle from the stagnation or reservoir condition. This may happen so rapidly that the various degrees of freedom cannot accommodate rapidly enough and flow non-equilibrium occurs in the nozzle. A certain amount of the available enthalpy "freezes" and cannot be transformed into kinetic energy of the test section flow. Non-equilibrium effects make the flow diagnosis and definition of the test section flow conditions much more complicated and should be avoided, if possible (Ref. 71).

In reference 41 a value for the entropy $S/R \leqslant 32$ is selected as the criterion for equilibrium flow to be present. Reference 71 prefers $S/R \leqslant 31$. Reference 59 gives for these entropies (real gas effects included) :

p_o = 200 atm	S/R = 31	T_o = 4700^oK
	32	5260^oK
p_o = 1000 atm	S/R = 31	6080^oK
	32	6870^oK

As was pointed out in the general remarks of section 3.1 high temperature real gas effects in the wind tunnel can be restricted to molecular vibration only, when the test section Mach number is below Mach 18 and the tunnel is operated at its equilibrium condensation temperature.

For tunnels operating at these conditions, comparison of the figures quoted above with figure 11 would indicate that no-equilibrium nozzle flow is completely avoided. In fact however, reference 41 sets $S/R = 32$ as the boundary for chemical non-equilibrium. Vibrational non-equilibrium however, appears to be present at much lower entropy values than $S/R=31$ or 32 as is concluded from the data, presented in reference 72.

For a throat diameter of 0.25 inch = 6.35 mm and a 5 degree half angle nozzle with a parabolic throat contour, which is a realistic case for the present discussion, it was found (Ref. 72) that for a stagnation pressure p_o = 4000 psi = 280 atm the frozen enthalpy is about 3.7 % of the stagnation enthalpy for T_o = 2000^oK and 4.1 % for T_o = 3000^oK . These stagnation conditions correspond with entropy values $S/R = 25.6$ and $S/R = 27.6$ respectively.

It is concluded that vibrational non-equilibrium effects in the nozzle flow as well as around the model (Ref. 72) are in many cases not negligible. They can be minimized by a using slender nozzles. For a fixed value of T_o and of $p_o L_m$ and thus also p_ox throat diameter (if L_m is proportional to the test section diameter), the slenderness of the throat and the nozzle is the only parameter that affects the vibrational non-equilibrium phenomena (Ref. 72).

3.1.F Running time

Finally, some lower limits for the running time should be considered.

Firstly, the running time must be long enough to permit steady flow to be established in the test section and around the model. For facilities with short running times, say less than 100 milliseconds, the flow is started impulsively in general by breaking a diaphragm near the nozzle throat. The starting process has been described in the literature, see for instance references 13 and 14.

A praticial definition of the start time is the time during which the steady flow through the nozzle throat is not transformed into a steady flow in the test section but is rather passing through unsteady expansion waves and/or shock waves which exist in the nozzle during the starting process. As follows from consideration of the data presented in reference 73 and 74, this time is found by constructing the u-a characteristic for the steady-state nozzle flow in a wave diagram along the nozzle, the singularity at the throat being discussed in reference 73 and the particle trajectory of the steady state flow. The starting time is then equal to the ordinate in the wave diagram (or x-t diagram) in the throat where x = 0, of the particle trajectory which arrives at the same time in the test section as the u-a characteristic from the origin of the wave diagram (Ref. 74). This applies for an initial pressure below the free stream static pressure that will be generated when the tunnel has started. For a conical nozzle with a throat radius r_t, a nozzle half-angle tangent β and a sound speed a_o at reservoir conditions the nozzle starting times t_s have been calculated and also the time t_b between the breaking of the throat diaphragm and the establishment of steady flow in the test section.

M :	5	10	15	20
$t_s \, \beta \, \frac{a_o}{r_t}$:	1.35	2.65	4.14	5.85
$t_b \, \beta \, \frac{a_o}{r_t}$:	3.54	13.30	32.10	62.06

For instance a Mach 15 conical nozzle with a half-angle of 5 degrees at $T_o = 2000^{\circ}K$ and an exit diameter of 100 cm would give $t_s = 0.5$ millisec and $t_b = 3{,}9$ millisec ($r_t = 0.95$ cm with high temperature real gas effects included). A contoured nozzle, which is about twice as long as a conical nozzle with the same maximum half-angle and exit diameter (without boundary layer effects) will have starting times which are roughly two times longer than the values from the table above.

For the flow establishment around the model stabilization of separated flows is the governing factor. For laminar separated flows it has been found that about 30 body lengths of flow were required for the pressure in the base region of a sphere to stabilize (body length equal to sphere diameter) and for the heat transfer a factor of two longer time was needed to reach equilibrium. For shock induced separation the flow establishment times are much shorter for the cases of interest (Ref. 75). On the leeward side of the space shuttle large regions of laminar separated flow may occur during re-entry at large angles of incidence. The diameter of the fuselage being of the order of 0.15 of the length and assuming that 60 diameters are required for flow establishment (a somewhat arbitrary value) the flow establishment time t_e is 9 L/u, where u is the free stream velocity. For a local sonic speed of 150 m/sec (near condensation) it is found that t_e=60 L/M millisec, where L is the body length in meters. This figure should be considered as an order of magnitude and will depend on the body geometry and flow conditions such as Reynolds number etc.

For a contoured nozzle with a test section diameter of 1 meter and a maximum wall angle of 5 degrees and a model length of 1 meter the following data on starting times are found

M :	5	10	15	
T_o :	500	1000	2000	^{o}K
r_t :	10.o	2.23	0.95	cm
t_s :	6.9	2.1	1.0	millisec
t_b :	18.0	10.7	7.6	"
t_e :	12.0	6.0	4.0	"
t_s+t_e:	18.9	8.1	5.0	"

The quantity $t_s + t_e$ is to be considered as a maximum for the non-useful duration of flow from the reservoir. In practice this time will be shorter because t_s and t_e will partially overlap each other.
Shock tunnels are in fact the only facilities which have been used for configuration testing which have running times that are so short that the question of flow establishment time arises. The throat size of such a facility must be compatible with the preceding shock tube diameter. A throat diameter of 20 cm as required for a Mach 5 nozzle with an exit diameter of 1 meter is certainly not realistic for present or even future high-pressure shock tunnel technology. In fact the nozzle of the Cornell shock tunnel which has been used for HST-configuration testing (Ref. 42) has a 0.61 m diameter nozzle when operating between Mach 5.5 and 8.2 and a 1.22 m nozzle for Mach numbers between 10 and 17.

The running time is 2 to 13 milliseconds, the high value being for the lowest stagnation temperatures (Ref. 76).

The flow establishment times being proportional with the linear dimension of the tunnel and the model it is concluded from this discussion, that possibilities of shock tunnel testing of models with a length larger than 0.5 meter are marginal at low hypersonic Mach numbers from considerations of the required running time.

Another minimum testing time criterion follows from the requirement that force measurements on complete vehicle models should be made. The consequences for the required running times are discussed in reference 71. In impulse facilities such as shock tunnels and hot shot tunnels the force data are obtained with acceleration compensated balances. These are designed on the premise that the test model being evaluated, vibrates as a rigid body (Ref. 71). Slender bodies and/or large models tend to generate vibrations within the model, yielding imperfect inertia compensation. When these bending resonance frequencies are high enough, they can be filtered out. For shock tunnels a minimum frequency of roughly 1000 Hz can be tolerated and this limits the model scale to something of the order of 45 cm for a model slenderness ratio of 10. For the same model geometry this frequency is inversely proportional with the model length. When a balance system, based on the premises mentioned in reference 71 is employed, facilities with test times of the order of 50 milliseconds or more should be used for development work where model lengths of more than 0.5 meter are tested and shock tunnels fall short of this requirement.

It should be noted, however that at Cornell Aero Labs force testing was done with a HST-model with a length of 0.66 meter (Ref. 42). Interference of the vibration modes with the inertial compensation may have been avoided in this case by placing several accelerometers inside the model and then excite the model on a shaker to establish the compensation required in analogue circuits (Ref. 29).

Free flight testing techniques (Ref. 77 for instance) are probably not attractive for the development testing described in this paper because of the relatively large aerodynamic loads and the required integrity of the light weight model during flight and of the facility after impact of the model. Also the cost of complicated throw-away models should be considered.

In summary it is concluded that the running times of present shock tunnels which are of the order of 10 milliseconds are marginal for development testing of models longer than, say 0.5 meter but testing is not impossible. Test times of the order of 50 milliseconds require much less attention as far as flow starting times and inertia compensation for force measurements is concerned. Also the feasibility of probe traverses (Ref. 78), scannivalves (Ref. 79) and variation of the angle of incidence during the run (Ref. 43) is greatly increased, which will increase the efficiency of each run. Also dynamic testing is facilitated when the running time is 50 milliseconds or more.

3.1.G Conclusions

The practical and principal limitations of hypersonic wind tunnel facilities for Mach number-Reynolds number duplication can be summarized as follows.

1. Acceptable sting loads require for M-Re duplication of hypersonic transport aircraft models with a length of the order of 2 meters. For testing up to Mach 8 reservoir pressures of the order of 1000 atm are then required. Whether the construction and use of such a facility is justified is open to discussion

2. For the case of M-Re duplication of a space shuttle and for boundary layer transition close to the leading edges of a hypersonic transport ($Re_L = 2 \times 10^7$) flying not faster than Mach 10 the requirements of tunnel size/stagnation pressure are largely overlapping (Fig. 13). If one facility should do both jobs the design should lie within the following limits:
   a. Model length $L_m > 0.3$ to 0.5 m to prevent excessive sting loads and to allow ample instrumentation
   b. Stagnation pressure $p_o < 5000$ atm from structural considerations for the tunnel
   c. Stagnation pressures $p_o L_m > 500$ atm.m (better is $p_o L_m > 1000$ atm.m) in order to duplicate the Reynolds number up to large enough Mach numbers
   d. Up to Mach 18 at equilibrium condensation conditions high temperature real gas effects are restricted to vibrational excitation of the molecules only, and up to that Mach number the required Reynolds number can be generated without severe throat erosion problems for running times shorter than 100 milliseconds for a tungsten throat and oxygen-free nitrogen as test gas
   e. Running times of less than 10 milliseconds are marginal for development testing on models longer than 0.5 m from the point of view of flow establishment times. A running time of more than 50 milliseconds offers more flexibility in this respect and in measurement techniques.

3.2 Facilities for combustion and propulsion testing (including hardware testing).

Many of the arguments which are discussed in section 3.1 hold also for the facilities for scramjet combustion and propulsion tests. These facilities must generate the correct environment in the supersonic combustion chamber with regard to Mach number, pressure and temperature. For a given flight Mach number simulation, these facilities will run at higher stagnation temperatures and lower stagnation pressures than wind tunnels, which makes that no direct limitations exist with respect to stress levels, either in the tunnel reservoir or in the combustor model. Limitations will show up with respect to tunnel power and heating system, throat cooling and erosion, running time and the composition of the air.

3.2.A Facility power and air heating

For mass flows of the order of 200 kg/sec as quoted in section 2.2 a power is required of the order of 10^3 MWatts during the run. This power is equal to a large power station and hence prohibitive in many circumstances. Therefore short running facilities with accumulation energy storage are developed. The compression heating systems (shock tunnels and gun tunnels) are the cheapest facilities in this respect. The shock tunnel might give too short running times ($<$ 10 msec), but gives almost unlimited stagnation temperature duplication (Ref. 48).
Calculations show that a gun tunnel with preheated barrel may generate about 3000°K during 50 msec. with mass flows in the 10-50 kg per second range. Also hot shots (arc heating) show this performance (Ref. 48). In the gun tunnel the temperature drop due to cooling may be compensated by increasing stagnation pressure during the run (see Ref. 78 for a pressure record); in the hot shot this cooling effect is much more troublesome (see also 3.2.C).
Regeneration heating systems (such as the pebble bed) are limited by the maximal solid material temperatures attainable, typically less than 2500°K. The running times may be however several

seconds to a minute or longer and therefore show good promise for hardware testing (see also 2.3).

An attractive means to increase the stagnation temperature of the supersonic burning test facility is to burn upstream of the nozzle a hydro-carbon, hydrogen or nitrogen containing fuel (NH_3, N_2H_4) and add additional oxygen (Ref. 80, 81). The air is then called vitiated air. The attainable temperature in combination with the regenerative heating system might be in the 3000°K range.

The only means to obtain higher temperatures for long periods of time, $\geq$ 10 sec, is to use arc heating. However continuous arc heating is limited to only moderate pressures (a few hundred atm) (Ref. 55), decreasing with increasing temperatures and hence flight Mach number, conflicting with the requirement that the tunnel stagnation pressure must increase with Mach number to allow flow duplication. The flight altitude for which arc heater facilities can provide flow duplication above say Mach 8 to 10 is therefore too far above the HST real flight altitude to allow realistic hardware testing in free jet test sections of the blow down type. However, continuous arc facilities are very useful for ablation testing, since pitot pressure and wall static pressure duplication is required rather than Mach number.

3.2.B Throat cooling

The heat transfer equation of section 3.1.D is used to compute the temperature-pressure limitation of continuous propulsion and hardware test facilities. This line is represented in figure 5 (upper boundary). If the connected pipe supersonic combustion testing method is used (Fig. 10) the lower line in figure 5 is obtained taking into account the total pressure recovery of figure 8. These lines probably will coincide with the arc heating capacities of the former section. Hence it can be concluded that for hardware testing (long run times) the laboratory facilities are limited to approximately free flight Mach numbers of 8 for free jet facilities, and to M = 10 for connected pipe testing for a q = 0.5 kg/cm^2 flight condition (Fig. 2).

3.2.C Flow duration

The least required flow duration is of primary consideration for facility type, heating system desing and throat cooling requirements. Two kinds of tests can be distinguished, namely: first, fundamental flow field and combustion tests and second, hardware tests. The first kind can be performed in short duration facilities in which the certainty of flow establishment is the main criterion apart from data collection time considerations. Reference 48 reveals that a few milliseconds seem to be sufficient, though reference 49 indicates that tests in a hot shot tunnel, having a 200 msec runtime, yields difficulties interpreting the results, mainly due to temperature variation. Test times of at least 10 msec seem preferable. Hardware tests should be of representative duration, hence of the order of several minutes (see also section 2.3).

3.2.D Air contamination

From power requirements point of view the vitiated air system is very attractive, since direct heating occurs by burning. However, almost all fuels for the vitiation system contain hydrogen, sothat free radicals such as OH will be present at the entrance of the combustor. These free radicals will substantially shorten the ignition delay times in the combustor, and hence will yield unrepresentative results. Furthermore the thrust as produced by the nozzle will be typcially 10°/o less than for clean heated air (Ref. 80). Many facilities use the vitiated cycle as a topping cycle for regenerative heated air (Ref. 81). Caution must be exercised to translate results from vitiated facilities to flight conditions. For duration tests and mixing tests vitiated air systems will be useful.

3.2.E Conclusions

For long duration combustion and propulsion tests (> 1 sec) complete scramjet performance assessment is limited to $M \simeq 8$, due to throat cooling capabilities. The same is true for structural testing in the real flow environment. For combustor and nozzle tests the flight Mach number duplication is limited to about 10 in the laboratory. For higher Mach number free flight testing is the only means. For scramjet hardware test a vitiated air system gives an economical solution with respect to power requirements.
For supersonic combustion and aerodynamic performance tests of scramjets short duration facilities can be used with sufficient running time. Facilities with a combination of preheating and compression heating such as gun tunnels with preheated barrel or tunnels such as the ONERA R4Ch will probably show good prospects. Also shock tunnels such as Sheffield University are very useful though the latter might be short of flow duration. For ablation studies continuous arc heaters are the best choice.

4 REMARKS ON EUROPEAN FACILITIES FOR HYPERSONIC TESTING

This section will present a brief evaluation of the performance of the major hypersonic facilities in Europe. In these facilities as well as in the smaller facilities which are located mainly at colleges and universities much work has been done of a more fundamental or exploratory character, as was mentioned in section 1.1 (see also Ref. 5, 6 and 7). In this section, however, the usefulness of the available major facilities will be considered against the background of the requirements for testing related to the development of the space shuttle and the hypersonic transport as described in the previous sections.

4.1 Aerodynamic testing

In reference 7 information is presented on all hypersonic facilities existing in Europe. Their usefulness for development testing at high enough Reynolds numbers as described in the preceding chapters can be appreciated if their maximum p_oL_m performance (stagnation pressure x model length) is plotted as a function of the Mach number at which the facility can be operated. Figure 11 where the required p_oL_m values are plotted is then the background against which the available p_oL_m can be projected.

During re-entry the space shuttle will fly at angles of attack of 25° to 60°. In order to avoid blockage of the wind tunnel flow, the model length L_m should not exceed half the core diameter (Ref. 29). For the present evaluation $L_m = 0.5\ D_m$ is assumed for the space shuttle with D_m equal to the nominal test section diameter, which is listed in reference 7.

For the HST which is much more slender than the space shuttle and which is tested only at small angles of attack up to say 10° or 15° (Ref. 42, 62) the model length L_m can be much larger without blockage. For the present case $L_m = D_m$ is assumed for the HST.

In reference 7 only the maximum stagnation pressure p_o of each facility is given. Having no detailed information on the dependence of p_o on the Mach number it is assumed that all facilities operate at their maximum p_o over the full Mach number range with the restriction that the maximum dynamic pressure q_m is not exceeded. This is assumed to be $q_m = 1.5\ kg/cm^2$ for space shuttle testing and 2.2. kg/cm^2 for HST testing (section 3.1.A).

This assumption on p_o can be criticized, but if the true p_o-M data had been plotted in figures 14 and 15 this had to be done for all facilities to obtain a fair basis of comparison. These data are presently not available. It is suggested that these should be included in the next edition of reference 7 and/or it should be stated at what Mach numbers the Reynolds numbers, mentioned in reference 7 are attained. Some remarks dealing with the p_o assumptions are given in the next sections.

From the facilities listed in reference 7 the "major" facilities should be selected. The following criteria have been used:

Test section diameter D_m >0.25 m. This means that not only facilities for models longer than say 0.3 m are included as was required for development testing (see section 3.1.G), but also smaller facilities which are suitable for more basic or exploratory studies.

p_oL_m > 10 atm.m. Lower values allow adequate M-Re simulation below Mach 5, which is outside the hypersonic regime.

Running time longer than 5 milliseconds, so that most shock tunnels may also be included.

An exception is made for the few arc heater facilities and low density tunnels where no restriction is made on p_oL_m. The arc heater facilities can in fact be used as low density facilities, but also high temperature phenomena such as occur during re-entry can be studied.

The relevant characteristics of the remaining facilities from reference 7, together with some additional information from reference 4, 29 and 30 can be found in table 1. The resulting p_oL_m performance is plotted in figure 14 and 15.

It should be remarked that most facilities operate only at specific Mach numbers rather than infinitly variable Mach numbers and the lines in figures 14 and 15 are to be considered as facility potential performance, which can often be used by simply employing different throat blocks. Also operation at lower values of p_oL_m than indicated is of course possible. For the HST, facilities which operate only at Mach numbers above 15 and/or which have maximum Re_L-capability below 10^6 have not been plotted in figure 15. For shock tunnels the maximum model length is assumed to be 0.5 meter to avoid too large difficulties with flow establishment times (see section 3.1.F).

From the data on tunnel performance the following conclusions may be drawn:

4.1.A Hypersonic testing of space shuttle configurations

From figure 14 it is concluded that the estimated performance of the large shock tunnel at TH Aachen covers the required M-Re performance for space shuttle re-entry above Mach 7.5. It is not known whether the running time is long enough, while it is given in reference 7 as 1-10 millisec. Also the conical nozzle is a disadvantage because this entails axial pressure gradients which makes corrections necessary.

In the low hypersonic regime between Mach 5 and Mach 9 several facilities are available which have Reynolds number performance close to the lower trajectory. Some remarks are made below.

The Imperial College gun tunnel is probably still the only gun tunnel with a test section larger than 0.25 m presently available in Europe since work at Bristol and RARDE (both in the U.K.) has come to an end (Ref. 30). Although the model length is rather limited ($L_m \sim 0.15$ m) the Reynolds numbers are rather high, which combined with a good parallel flow quality, makes the facility a useful tool in the European hypersonic tunnel inventory.
In a recent note data came available of a gun tunnel at the Institute de Méchanique des Fluids at Marseille. Though this tunnel has good performances the flow in the test section is of the source type and the running time is only limited.

The ONERA S4MA blow down tunnel has an attractive Reynolds number potential capacility. The facility can also be used as a combustion tunnel for ramjet/scramjet testing.

The smaller shock tunnel of TH Aachen shows no superior performance while the disadvantages are the same as for the large shock tunnel.

The blow down facilities of FFA (Sweden), ARA (U.K.) and of CEAT and the ONERA R3Ch (France) have about the same performance. Model size and tunnel costs will have to be considered.

From an economic point of view the Ludwieg tube facilities of the DFVLR and the ONERA R4Ch slow piston tunnel are attractive but their present performance is rather low.

It is doubted whether the DFVLR arc facility PK2 can work at a stagnation pressure of 100 atm down to a Mach number of 6 as indicated in figure 14. The performance envelopes shown in reference 4 show rather a constant maximum Re_L between M = 5 and 15. It is therefore supposed that the PK2 facility is not suitable for M-Re duplication and is to be used exclusively as a low density and/or high enthalpy facility.

Between about Mach 10 and 15 a gap exists where no M-Re duplication on space shuttle wind tunnel models can be realized.

Above Mach 15 the simulation of the Mach number is less urgent (see section 2.1.1.A) but several facilities for testing between Mach 15 and 20 are available in Europe namely the VKI long shot and the hot shot tunnels of ONERA. The Sud shock tunnel has the same performance as the hot shots but with a considerable shorter running time. The Reynolds number performance of the long shot duplicates Re_L at Mach 15 for the lower trajectory and is twice the hot shot tunnel Re_L performance. The model length L_m is of the order of 0.30 m for all M > 15 facilities. A disadvantage is the divergence of the test section flow caused by the conical nozzle except for the ONERA Arc 2 tunnel, which has a contoured nozzle.

It should be noted that testing at lower Mach numbers with the present long shot and hot shot facilities has only limited possibilities. This is because for both facility types the test gas flows out of a rather small reservoir with a constant volume, causing the stagnation pressure to drop during the run. This drop is proportional with the throat size and hence lower Mach numbers bring about a faster pressure drop (and temperature drop) during the run. This disadvantage is not present in gun tunnels and slow piston tunnels.

For low density testing several facilities are available as is shown in table 1. It should be noted that the inviscid core diameter of the test section flow is considerably smaller than the nominal nozzle exit diameter for low density facilities due to the thick boundary layers. Therefore only the largest facilities should be considered for testing complex models. These are the DFVLR PK2 tunnel with a 60 cm nozzle diameter and the new RAE low density tunnel with a 76 cm nozzle. The PK2 facility has a uniform core diameter of 10 cm at Re/cm = 5000 at Mach 15 (?) (Ref. 4), giving a viscous interaction parameter $M/\sqrt{Re} = 0.7$ for a model length of 10 cm which is probably too small for development testing.

It is concluded that up to Mach 9 several facilities exist or are planned in Europe which offer good possibilities for space shuttle tests at duplicating Reynolds numbers. A facility with a larger test section diameter of about 1 meter would offer a Re_L simulation capacility covering the lower trajectory requirement as indicated in figure 14 without being limited by the q=1.5 kg/cm^2 boundary set for excessive sting loads. A model length of 0.5 m could then be accommodated, which allows more detailed measurements than presently available. Also extension of the M-Re duplication capability to higher Mach numbers to say Mach 12 is advisable to close the gap between the present Imerial College Mach 9 gun tunnel and the VKI Mach 15-20 long shot facility.

It should be noted that the lower flight altitude boundary given in this paper (Fig. 2) may shift to larger altitudes when vehicles are operated at higher lift coefficients than the present shuttle to alleviate aerodynamic heating (Ref. 11) with a corresponding decrease in maximum Reynolds number requirement. For low density research and development a facility which is large enough to accept large and complex models is presently not available in Europe. Such a tunnel should cover values of $M/\sqrt{Re}$ between 0.1 and 0.5. Provisions should be made in the tunnel pumping system for accepting not only nozzle flows but also exhaust gases from motors, reaction jets or mass injection gas (Ref. 29).

4.1.B Hypersonic testing of HST configuration

For testing of hypersonic configurations up to Mach 8 at ReL = 2×10^7 European facility performance is adequate as can be concluded from figure 15. Reynolds number duplications requires test section diameters one order of magnitude larger then presently available.
The HST is a more slender configuration than the space shuttle and force measurements will be more sensitive to axial pressure gradients. Conical nozzles as employed for instance in the shock tunnels

of TH Aachen are therefore unsuitable for HST tests, which require great accuracies to determine the economic feasability of a HST system. It is questionable whether the available facilities are suitable for HST development testing where the aerodynamic behaviour of the complete airframe-engine system should be properly simulated. This requires proper engine flow simulation and a correct boundary layer thickness (at the engine intake for instance). Then higher Reynolds numbers are required which can only be realized in large facilities to avoid excessive sting loads. This requires a facility like the large Tripltee tunnels described in section 4.2.

4.1.C Comparison with U.S. facilities

In reference 82 data on hypersonic facilities in the United States are presented, exclusing those of the AEDC. The older AEDC facility data can be found in reference 65. These data have been plotted in figure 16 from which a performance envelope could be drawn. For the lower Mach numbers no dynamic pressure limits are set and only facilities with contoured nozzles have been selected. It should be noted that p_o times the test section diameter D_m is plotted rather than p_o times the model length L_m as was done in figures 14 and 15.

In the same figure the performance envelope of the European hypersonic facilities with contoured and conical nozzles has been plotted. The S4MA facility which is presently used exclusively for propulsion testing is indicated separately.

It is found that up to about Mach 10 the M-Re duplication performance of the European facilities and the selected U.S. facilities is not very different. However, when the large Tripltee facilities mentioned in section 2.3, fitted with contoured nozzles are added, the picture is drastically changed. The NASA Langley 8 ft structures tunnel attains $p_o D_m$ = 670 atm.m at Mach 7.5 which is close to the HST Reynolds number duplication requirement and one order of magnitude larger than the performance of European facilities.

Between Mach 10 and Mach 15 there is a wide gap between U.S. and European facility Reynolds number performance, which has been widened by the new NOL 5 foot blow down facility which is to be operational by late 1972 (Ref. 63). Between Mach 15 and 29 this facility with contoured nozzles has even a higher Re_L capability than the VKI *long shot*.

When only facilities with contoured nozzles should be considered it should be noted that above Mach 15 no facilities are available in Europe, except for the Imperial College gun tunnel fitted with a Mach 18 nozzle and the ONERA Arc 2 tunnel.

4.2 Combustion, propulsion and hardware testing

Table 2 gives a review of the facilities as used in Europe for scramjet tests and supersonic burning studies. A comparison is made with the largest facilities in the U.S. at NASA.

The main capacity in Europe, the S4MA tunnel of ONERA and the main facility in the U.S. are projected in figure 5 also. It shows that research and testing in the field of supersonic combustion and scramjets is as yet only possible at the very lower end of the practical applicability region for this propulsion means. Hence, there is a need for better facilities for this field of research and development if the hypersonic flight with the economical air breathing propulsion units is going to come.

In Europe good work of fundamental nature in the field of supersonic burning is done at the University of Sheffield in the high enthalpy shock tunnel and at the DFVLR at Porz Wahn in the pebble heated facilities.

For heat shield ablation testing the arc heated facilities of the DFVLR might be used in principle, their stagnation temperatures being of the order of 5000°K (air, nitrogen) to 10000°K (argon) In order to duplicate the pitot pressure of the full scale vehicle during re-entry the facility will have to be operated at low Mach numbers, between say Mach 2 and 5 (see section 2.3).

The PK2 facility which is the largest facility available has a power supply of 1000 KW and operates at reservoir pressures between 0.1 and 100 atm (Ref. 4). This is well within the range of the facilities for ablation testing listed in reference 55.

On the other hand at the sixth meeting of the IaWs working group it was stated that ablation tests cannot be performed adequately in Europe. It seems therefore open to discussion whether the PK1 facility is large enough for ablation testing on a development scale.

For hardware testing connected with the hypersonic transport and its propulsion system no facility is available in Europe with sufficiently long running times, which should be more than several minutes.

However considerable effort is put in the U.S. in the design and development of Tripltee tunnels as might be concluded from the next survey.

Tripltee facilities with a test section diameter of about 1 meter are the NASA Lewis facility which recently became operational (temperature duplication up to Mach 7) and the Aerodynamic and Propulsion Test Unit (APTU), presently under construction at AEDC (Ref. 52). The NASA Lewis Hypersonic Propulsion Research Facility has a nozzle exit diameter of 1.06 meters and operates at a maximum stagnation pressure of 80 atm (Ref. 81) and a stagnation temperature of 2300°K which can be boosted to 2670°K by viatiation. The running time is 2 to 3 minutes.

In the APTU clean air flow duplication is possible up to Mach 6 (clean air) and the maximum stagnation pressure is 210 atm, which allows flow duplication at Mach 6 down to 17 km. The NASA Lewis facility was designed for testing ramjet type engines which operate at high altitudes. The APTU will provide the higher pressures and massflows needed for testing low-level and run-in missile engines (Ref. 52).

Nominally these facilities will produce the desired test flows at less than 1 m diameters. Free jet testing of missile engines can therefore be conducted only at small angles-of-attack. Succesful development of the ramjet and air-augmented rocket engines has long awaited availability of these facilities. However, these facilities are not large enough to test the small research and missile engines at high angle-of-attack conditions or any of the larger engines which will be needed for aircraft applications.

Large Tripltee facilities are the NASA Langley 8-Foot High Temperature Structures Tunnel which has been considered for testing structural test engines to be adopted for the HST (Ref. 51). In this blow-down Mach 7.5 tunnel stagnation pressures up to 280 atm and temperatures up to 2500°K can be generated (Ref. 65).

At AEDC a large Tripltee facility has been designed of the same category i.e. a nozzle with an exit diameter of 10.2 ft (3.2. meter), a stagnation pressure of 240 atm and a stagnation temperature of 2400°K. At these stagnation conditions the flow at Mach 7.7 at an altitude of about 25 km is duplicated and the running time is then about two minutes (Ref. 52). Running times of 30 minutes are possible when the flow is duplicated at Mach 7 and 43 km altitude or Mach 4 at 30 km altitude.

The performance of these U.S. Tripltee facilities corresponds with the requirements already discussed in section 3.2.

5. CONCLUSIONS

1. During the past two decades hypersonic research has been substantial in Europe. Present activities and facilities are reflected in the Eurohyp inventory. Up to now the European facilities have been used primarily for research of a more fundamental and exploratory character. Much of this work will be applicable to the design of the space shuttle and the hypersonic transport.

2. For the space shuttle aerodynamic development testing, M-Re duplication is necessary up to about Mach 15 to include boundary layer transition effects. For the lower re-entry trajectory of the present space shuttle orbiter design this corresponds with $Re_L = 3 \times 10^7$ at Mach 6 and 6×10^6 at Mach 15.
At high flight altitudes and velocities where the boundary layer is fully laminar, high altitude phenomena occur in the form of viscous interactions. For $M/\sqrt{Re} > 0.01$ this viscous interaction parameter should be duplicated rather than M and Re separately.
High temperature real gas effects cannot be duplicated in sub-scale testing due to conflicting scaling laws. Partial simulation and experiments of a more basic nature should provide the required information.

3. For the hypersonic transport attention is focussed to Mach 6-8 cruise conditions. The Reynolds number Re_L for a 75 m long vehicle is then of the order of 2×10^8, giving a boundary layer which is almost completely turbulent. $Re_L = 2 \times 10^7$ is considered as a minimum requirement for HST development testing where absolute performance data should be obtained. Reliable extrapolation to the full scale Re_L seems then feasible.
The boundary layer thickness at $Re_L = 2 \times 10^7$ is however about 60 °/o larger than at $Re_L = 2 \times 10^8$, the consequences of which should be considered with care.

4. For scramjet propulsion testing the Mach number in the combustor must be duplicated.
For hydrogen fueled scramjets the combustor entrance static temperature should be above 1000°K and up to about 1500°K reaction rates are dominant over mixing in the combustion process. This corresponds with a flight Mach number of 10 to 12 or a stagnation temperature of 4000°K to 5000°K. Up to these temperatures T_o duplication is essential.
For good understanding of the combustion phenomena the pressure level should be duplicated as well as the geometry. Scaling laws can only be used if the overall chemical kinetics behaviour can be described by simple rules and if variable induction times do not exist.

5. Hardware testing of ablative materials is generally performed in arc heated facilities where the stagnation enthalpy and the pressure on the vehicle are the primary simulation parameters.
For hardware testing of the HST airframe and propulsion systen large true temperature tunnels (Tripltee facilities) are required with running times of at least several minutes. Typical characteristics are $p_o = 250$ atm, $T_o = 2500$°K and a test section diameter of the order of 3 meters.

6. For aerodynamic testing of space shuttle and HST configurations the following limitations were found :

   a. Sting loads. For space shuttle testing the dynamic pressure should not exceed about 1.5 atm when testing at maximum normal force coefficient. For HST models the dynamic pressure limit will be determined by the angle of attack range. At $C_N = 0.25$ the dynamic pressure should not exceed about 2 atm (cruise condition is $C_N \sim 0.04$). Starting and stopping loads have not been considered.

b. The tunnel reservoir pressure should not exceed 5000 atm for structural reasons.

c. Up to Mach 18 throat heat transfer does not limit the Reynolds number capacity when the tunnel is operated at minimum temperature for condensation free flow and for running times shorter than 100 milliseconds (tungsten throat, nitrogen test gas). For longer running times and other materials the Mach number limit may be lower.

d. Real gas effects can be restricted to molecular vibration only, if the flow Mach number is below 18 when the tunnel is operated at equilibrium condensation conditions.

e. A running time of the order of 10 milliseconds as is current shock tunnel practice is marginal for testing of models longer than say 0.5 meter due to tunnel starting and flow establishment times.

7. For propulsion testing the following factors affect the facility performance. For long duration (> 1 sec) test complete scramjet performance assessment, including intake performance, is limited to about Mach 8 due to throat cooling capabilities. For combustor and nozzle tests the flight Mach number is limited to about 10 in the laboratory. For scramjet hardware testing a vitiated air system gives an economical solution with respect to power requirements.
For combusting and aerodynamic performance tests of scramjets short duration facilities can be used. Gun tunnels with preheated barrel or shock tunnels will show good prospects, though the latter may be short of flow duration.

8. The following conclusions are made on the European facility performance for aerodynamic development testing:

a. For testing of space shuttle configurations at duplicating Reynolds numbers and of HST configurations at $Re_L = 2 \times 10^7$ several facilities are available in Europe up to Mach 9. Between Mach 9 and Mach 15 no facilities exist with high enough Reynolds numbers. Between Mach 15 and 20 some facilities with high enough Reynolds numbers are present but they have conical nozzles which give no parallel test section flow. Low density phenomena can be simulated up to $M/\sqrt{Re}$ values of more than 1.

b. The model sizes which can be accommodated are rather small and sometimes even too small for development work. Space shuttle models with a length of about 0.3 m can be accommodated and in the low density facilities the model length will be of the order of 0.1 m at high $M/\sqrt{Re}$ values.
For HST configurations a model length of about 0.5 m can be accepted and this is again rather small for development work.

c. For HST development testing at duplicating Reynolds numbers no facility is available in Europe.

d. Three extensions of the present European testing capability can be considered :

1. A facility with a contoured nozzle with an exit diameter of about 1 meter and operating up to Mach 12 or 15 and a maximum stagnation pressure of the order of 1000 atm.

2. A low density facility which can accommodate fairly large and complex models in which also jet pluming phenomena (reaction jets for instance) can be studied. The facility should cover values of $M/\sqrt{Re}$ between 0.1 and 0.5 and the inviscid core diameter should be of the order of at least 0.5 m.

3. A large facility with a test section diameter of about 3 m, and stagnation conditions of 250 atm operating up to Mach 7 or 8 for HST development testing.

9. On the combustion testing capability in Europe it is concluded that this is presently only possible up to corresponding flight Mach numbers of about 6 which is at the very lower end of the applicability region of scramjets. The work in the Sheffield University shock tunnel at high stagnation temperatures but rather short running times and the available longer duration facilities (> 1 sec) might be complemented by a 100 millisecond facility with a stagnation temperature capability of 2500 - 3000°K and mass flows between 10 and 100 kg/sec.

10. For hardware testing the following is concluded.
For ablation tests on a reasonable scale the facility performance of the largest available arc heated facilities is marginal and possibly not adequate.
For hardware testing connected with the hypersonic transport a large Tripltee facility with a running time of several minutes or longer is required. No such a facility at present exists in Europe.

ACKNOWLEDGEMENT

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Table 1 Hypersonic wind tunnels in Europe (Ref.38)

Test section > 0.25 m diameter or height
Running time > 5 millisec
Test gas: air or nitrogen
Stagn. pressure x test section diameter ⩾ 10 atm-m.

Facility	Test section/ diameter m	M	$p_{o\,max}$ atm	$T_{o\,max}$ °K	Running time sec.	Contoured nozzle -
BLOW DOWN TUNNELS (running time more than 2 seconds)						
ONERA R2Ch	0.33	5,6,7	80	650	35	yes
" R3Ch	0.33	5-10	170	1100	10-35	yes
" S4MA	0.69	6	40	1850	60	
" S4MA	0.7 - 0.9	7-12	150	1850	10-60	planned
CEAT Poitiers	0.63	7,8.2	100	1000	40	yes
DFVLR Hyp H2	0.60	6-11.2	60	1400	120	
" Üs A_H	0.30 x 0.30	- 6.3	40	570	30-60	
CRA	0.35	6-8	100	800	30	
"	0.35	10-12	100	1400	40	
FFA Hyp 500	0.50	7.15	120	800	180	yes
RAE Bedford 3' x 4'	0.92 x 1.22	5	12	420	cont.	yes
ARA Bedford	0.30	6,7,8	200	850	60	
BAC 18"	0.46 x 0.46	6	20-34	460	380	
TUBE TUNNELS					millisec	
DFVLR Göttingen	0.5	5-7	40	400-600	300	
	0.5	9-12	150	750-1100	300	
SLOW PISTON TUNNELS						
ONERA R4Ch	0.325	10-15	200	1700	200	
HOT SHOT TUNNELS						
ONERA ARC1	0.50	15-20	2000	5000	100	
" ARC2	0.70	15-20	1500	7000	100	yes
CRA Hot shot	0.60	10-20	-	8000	40-70	
LONG SHOT TUNNELS						
VKI	0.61	15-20	4000	2600	10-40	no
GUN TUNNELS						
Imperial College No. 2	0.31	9	700	1500	20	yes
" " No. 2	0.46	18	700	1500	20	yes
I.M.F. Marseille	0.35	9-10	400	~1500	5-10	no
SHOCK TUNNELS						
Sud C_2	1.20	18	1000	4500	12-16	
TH Aachen	0.50*	6-15	200	7000	1-10	
" "	2.00*	6-25	2000	8000	1-10	no
RAE Farnborough 15"	0.38 x 0.38	7-15	250	4500	10	yes, to be uprated
ARC HEATERS						
DFVLR PK1	0.30	5-20	10	5000	cont.	
" PK2	0.60	5-20	10(100)	6000	cont.	
" Göttingen	0.25	4-10	0-4	8000	cont.	
LOW DENSITY TUNNELS						
ONERA R5	0.35	7-10	~1	1100	50	
CNRS SR 3	0.36	18,20	4	1800	cont.	

*) See next page.

Facility	Test section/ diameter m	M	$p_{o\,max}$ atm	$T_{o\,max}$ ^oK	Running time	Contoured nozzle -
LOW DENSITY TUNNELS						
DFVLR Göttingen VK1	0.25	7,25	3;500	3000	8 hr	
" " VK2	0.40	10,15,20	30	1200	cont.	
RAE LDT	0.76	6,10	50	1400 (Air) 2300 (N_2)	cont.	yes, to be uprated

*) A maximum throat diameter of 5 cm - which is current shock tunnel practice (Ref. 76) - has been assumed, giving a smaller test section at the lower Mach numbers. No real gas effects are taken into account for calculation of this test section diameter.

Table 2 Facilities in Europe for hypersonic engine tests and U.S. Tripltee facilities

Institute, name	$T_{o\,max}$ ^oK	p_o atm	mass flow kg/sec	combustion test section area cm^2	kind of heating	run time sec	Ref.
ONERA S4MA *)	1850	15	5 at T_o=1850^{o}K	~130	pebble bed	10-60	26
	1850	150	35	-	"		
R4Ch	1700	3.7 (after throttling)	2.5 at T_o=1700^{o}K	80	slow piston compression in preheated tube	0.2	17
Univ. of Sheffield	2000-6000	500		80	shock tunnel	0.004	48
DFVLR P.W.	1800		0.15	25	pebble bed		83
P.W.?	1800	60	1		"		83
NASA Langley 8ft HTST	2500	280		47000 (free jet)			65
NASA Lewis TTT	2300(2670)	80	100	8000 (free jet)	inductive heating (with vitiation)	120-180	81
AEDC APTU	1700	210			pebble bed		52
AEDC TTT (design)	2400	240	2300	80000	"	30-3600	52

*) Condition for a particular case

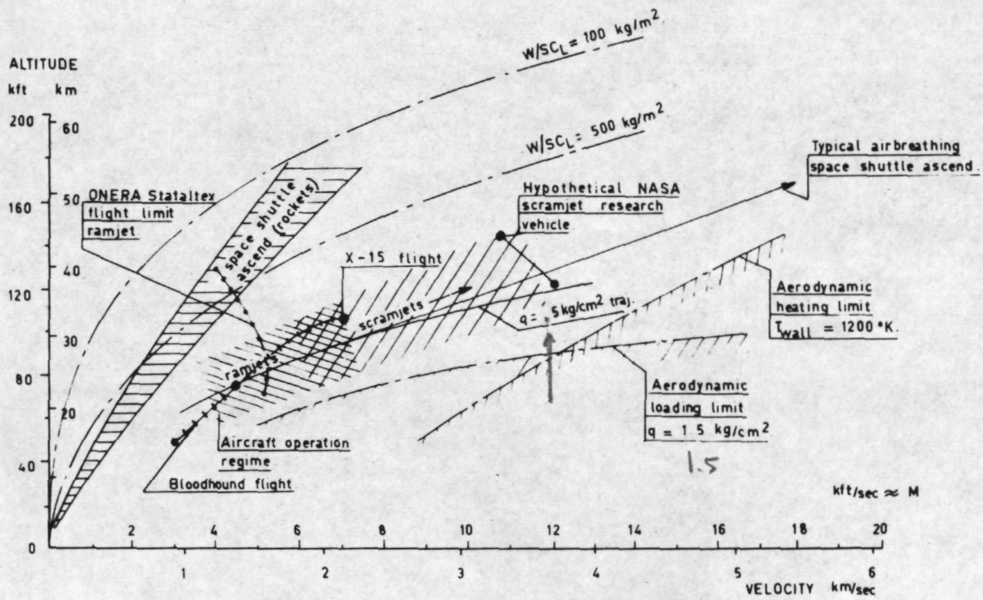


FIG. 1 TRAJECTORIES OF VARIOUS FLYING AND HYPOTHETICAL VEHICLES WITH AIRBREATHING ENGINES.

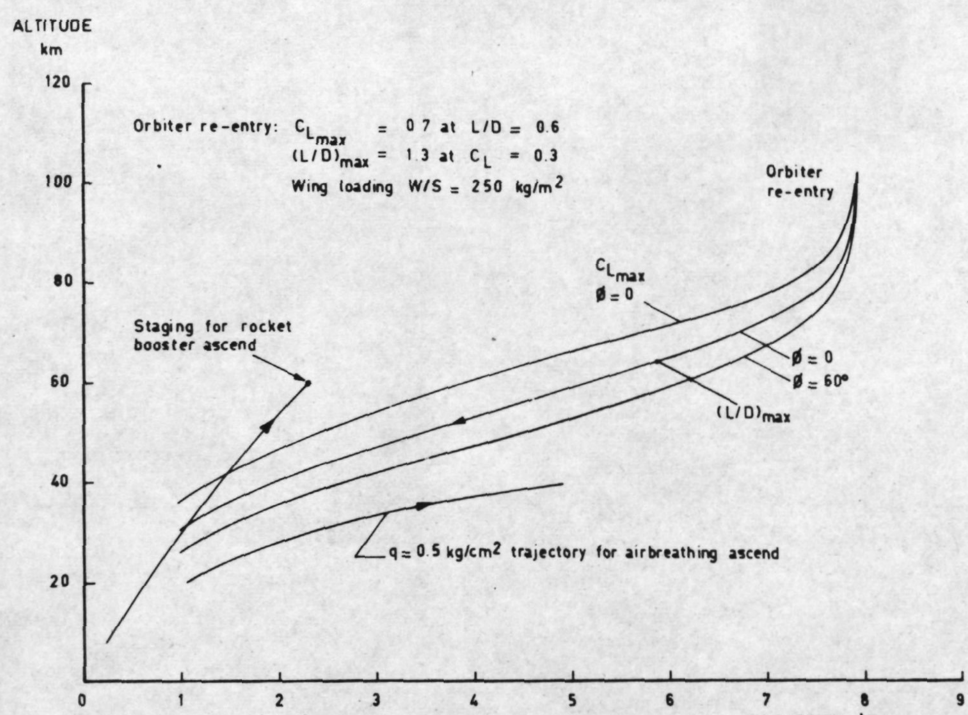


FIG. 2 SPACE SHUTTLE TRAJECTORIES FOR ASCEND AND DESCEND.

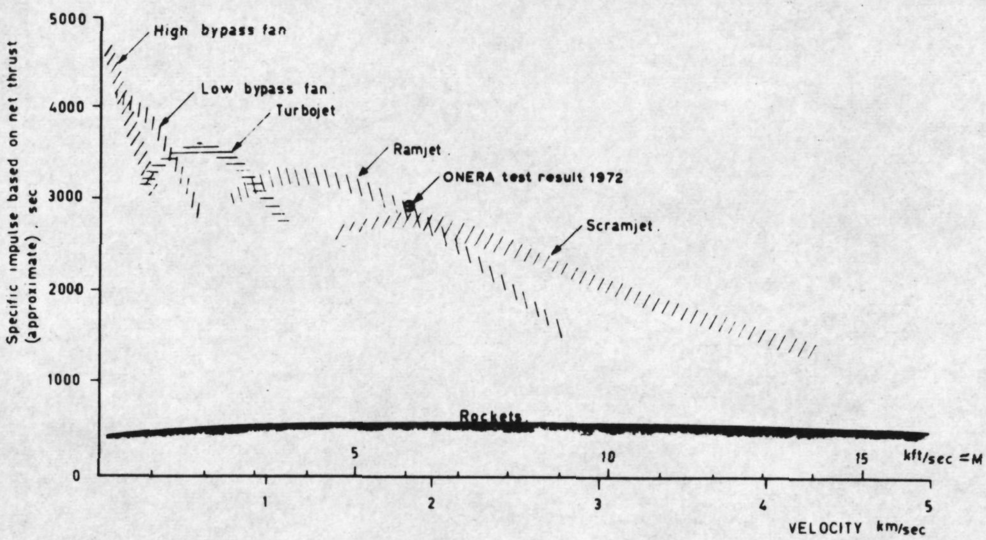


FIG. 3 SPECIFIC IMPULSE VS. FLIGHT SPEED FOR VARIOUS ENGINES.

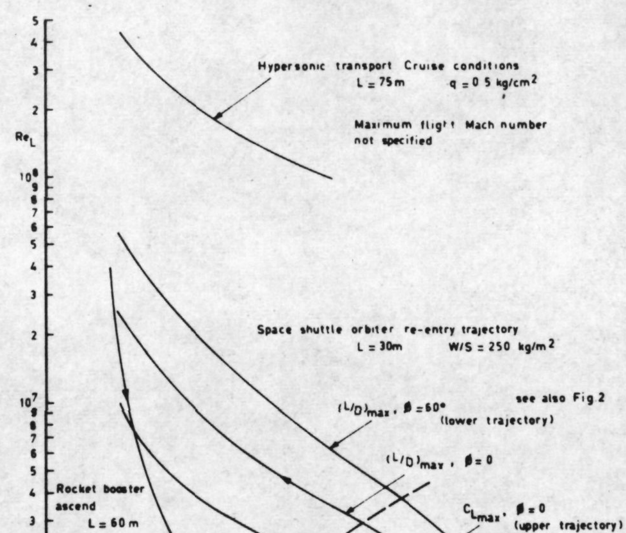


FIG. 4 REYNOLDS NUMBER – MACH NUMBER CHART FOR THE SPACE SHUTTLE AND A HYPERSONIC TRANSPORT.

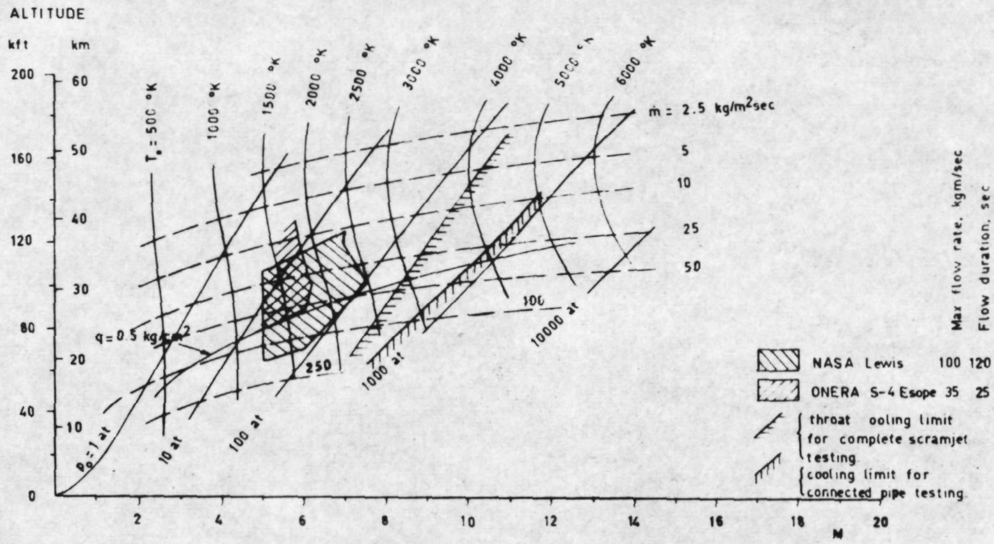


FIG. 5 STAGNATION CONDITIONS FOR FLOW DUPLICATION AND REQUIRED MASS FLOW PER UNIT CAPTURE AREA FOR GROUND TEST PROPULSION FACILITIES.

FIG. 6 COMBUSTOR ENTRANCE TEMPERATURE FOR A RANGE OF FLIGHT MACH NUMBER.

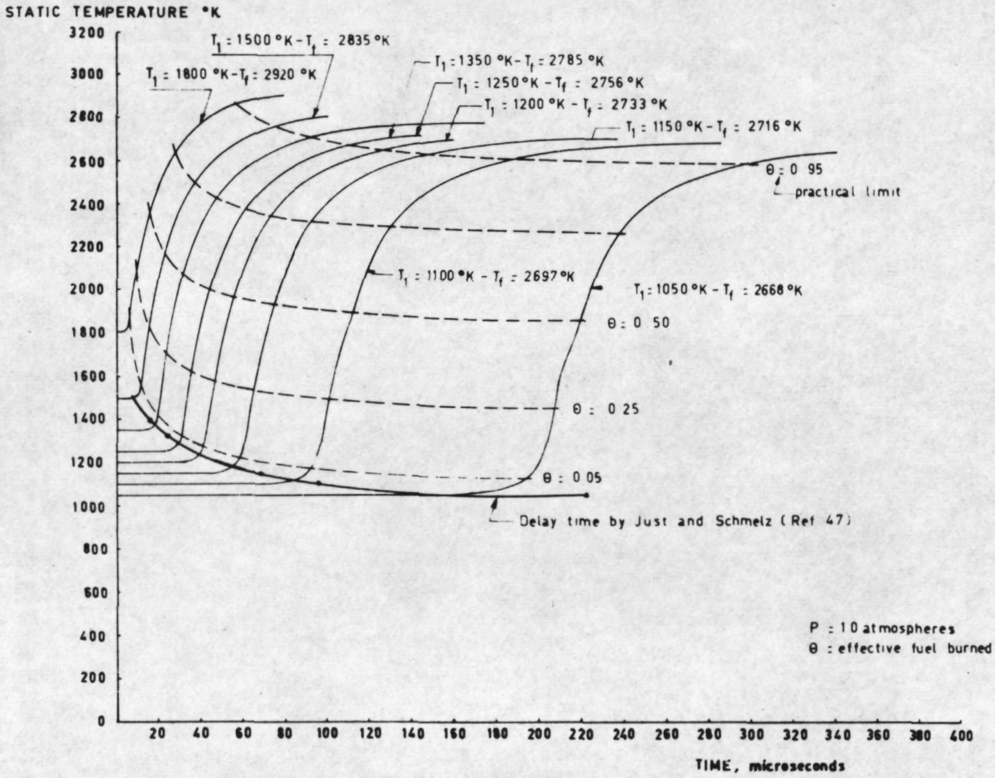


FIG. 7 COMBUSTION TEMPERATURE VARIATION WITH TIME FOR PREMIXED STOICHIOMETRIC HYDROGEN.

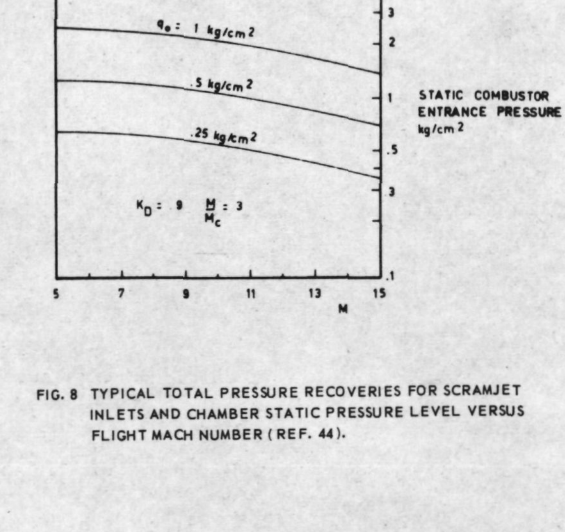


FIG. 8 TYPICAL TOTAL PRESSURE RECOVERIES FOR SCRAMJET INLETS AND CHAMBER STATIC PRESSURE LEVEL VERSUS FLIGHT MACH NUMBER (REF. 44).

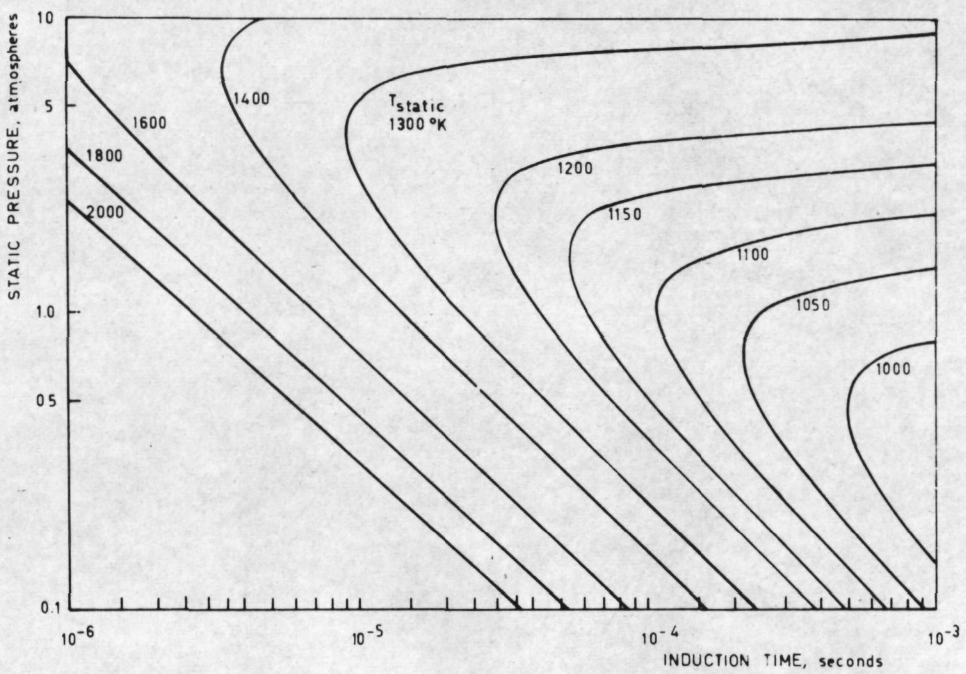


FIG. 9 STOICHIOMETRIC HYDROGEN– AIR INDUCTION TIME VERSUS STATIC PRESSURE AND TEMPERATURE. OH CONCENTRATION AT 10^{-5} MOLES/LITER.

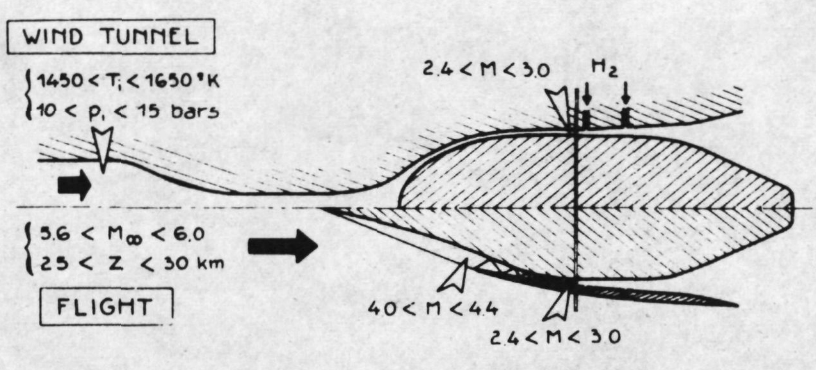
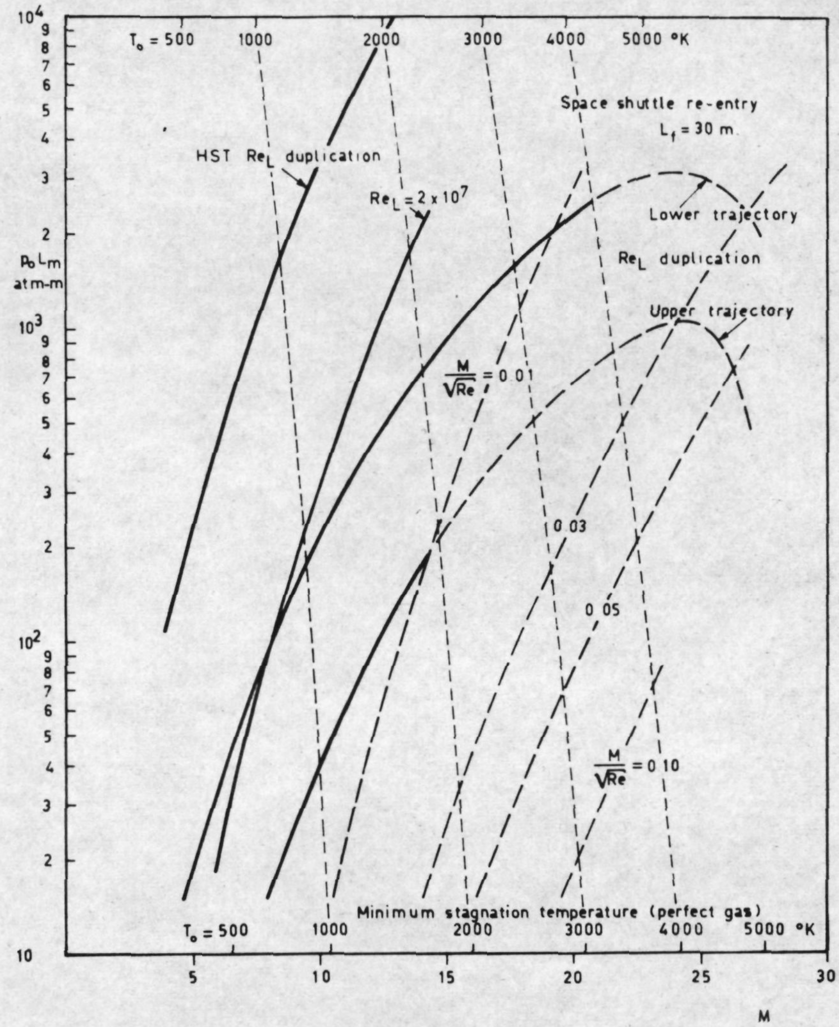


FIG. 10 SCRAMJET SIMULATION WITH CONNECTED PIPE (ONERA – ESOPE).



PERFECT GAS
EQUILIBRIUM CONDENSATION LIMIT
MODEL LENGTH L_m = 1 METER
SEE ALSO FIG. 4

FIG. 11 WIND TUNNEL STAGNATION PRESSURE TIMES MODEL LENGTH, REQUIRED FOR REYNOLDS NUMBER DUPLICATION.

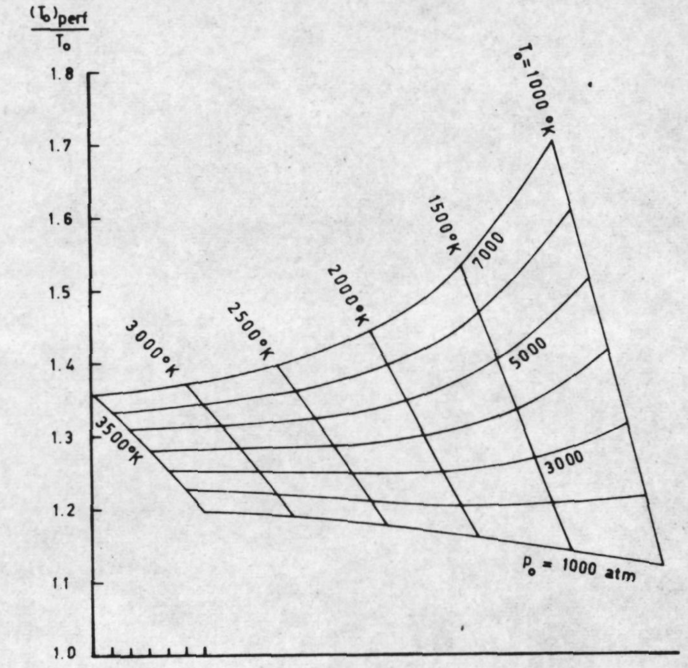


FIG. 12a EQUIVALENT PERFECT CORRECTION FACTORS FROM MEASURED SUPPLY CONDITIONS (FROM REF. 60).

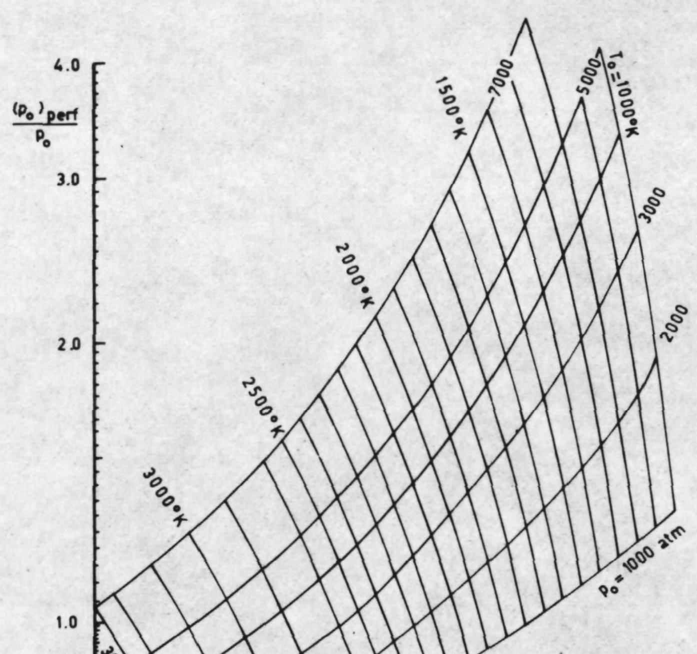


FIG. 12b EQUIVALENT PERFECT CORRECTION FACTORS FROM MEASURED SUPPLY CONDITIONS (FROM REF. 60).

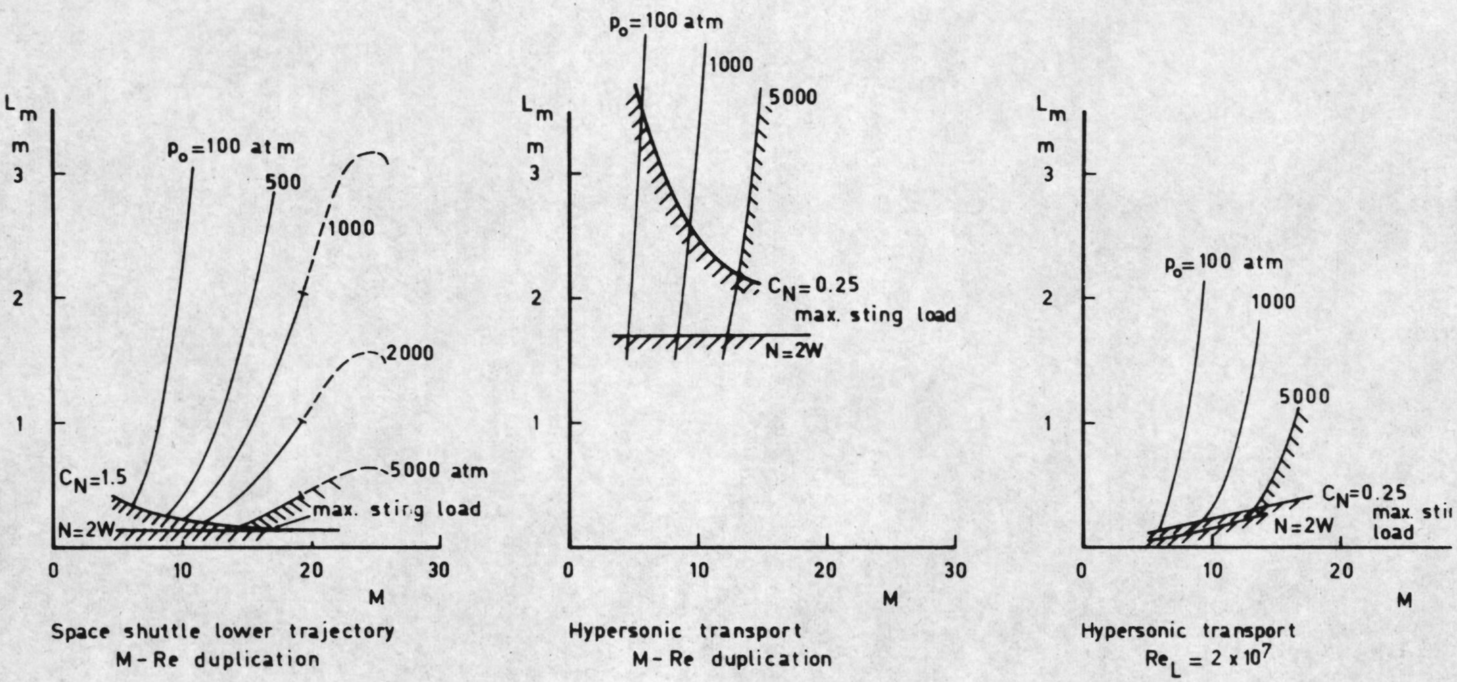
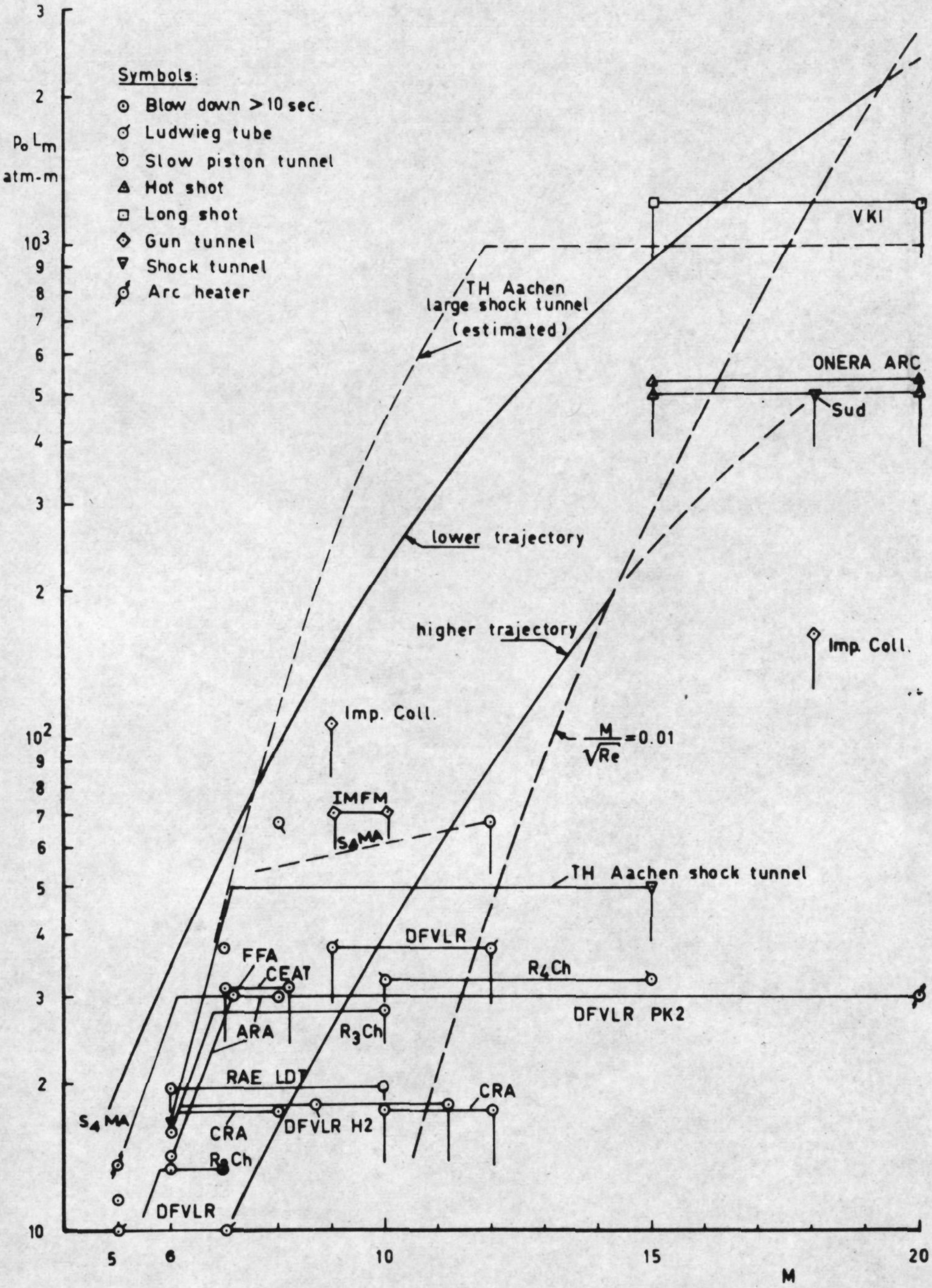
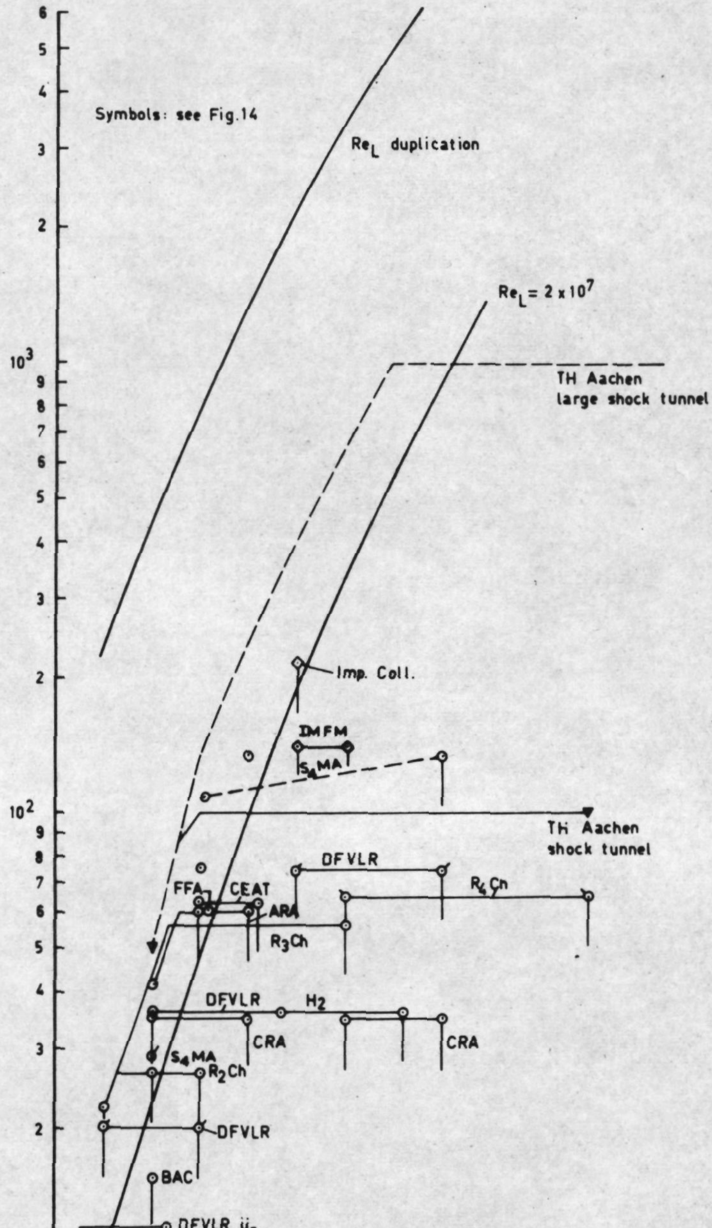


FIG. 13 MODEL LENGTH L_m REQUIRED FOR MACH NUMBER – REYNOLDS NUMBER DUPLICATION, CALCULATED FOR A PERFECT GAS (SEE FIG. 11).



MODEL LENGTH L_m = ½ TEST SECTION DIAMETER D_m.
DYNAMIC PRESSURE $q \leq 1.5\ kg/cm^2$. TRAJECTORY DATA FROM FIG. 11

FIG. 14 MACH NUMBER – REYNOLDS NUMBER SIMULATION CAPABILITY OF EUROPEAN WIND TUNNELS – SPACE SHUTTLE TESTING.



MODEL LENGTH L_m = TEST SECTION DIAMETER D_m.
DYNAMIC PRESSURE $q \leq 2.2\ kg/cm^2$.
TRAJECTORY DATA FROM FIG. 11.

FIG. 15 MACH NUMBER – REYNOLDS NUMBER SIMULATION CAPABILITY OF EUROPEAN WIND TUNNELS – HYPERSONIC TRANSPORT TESTING.

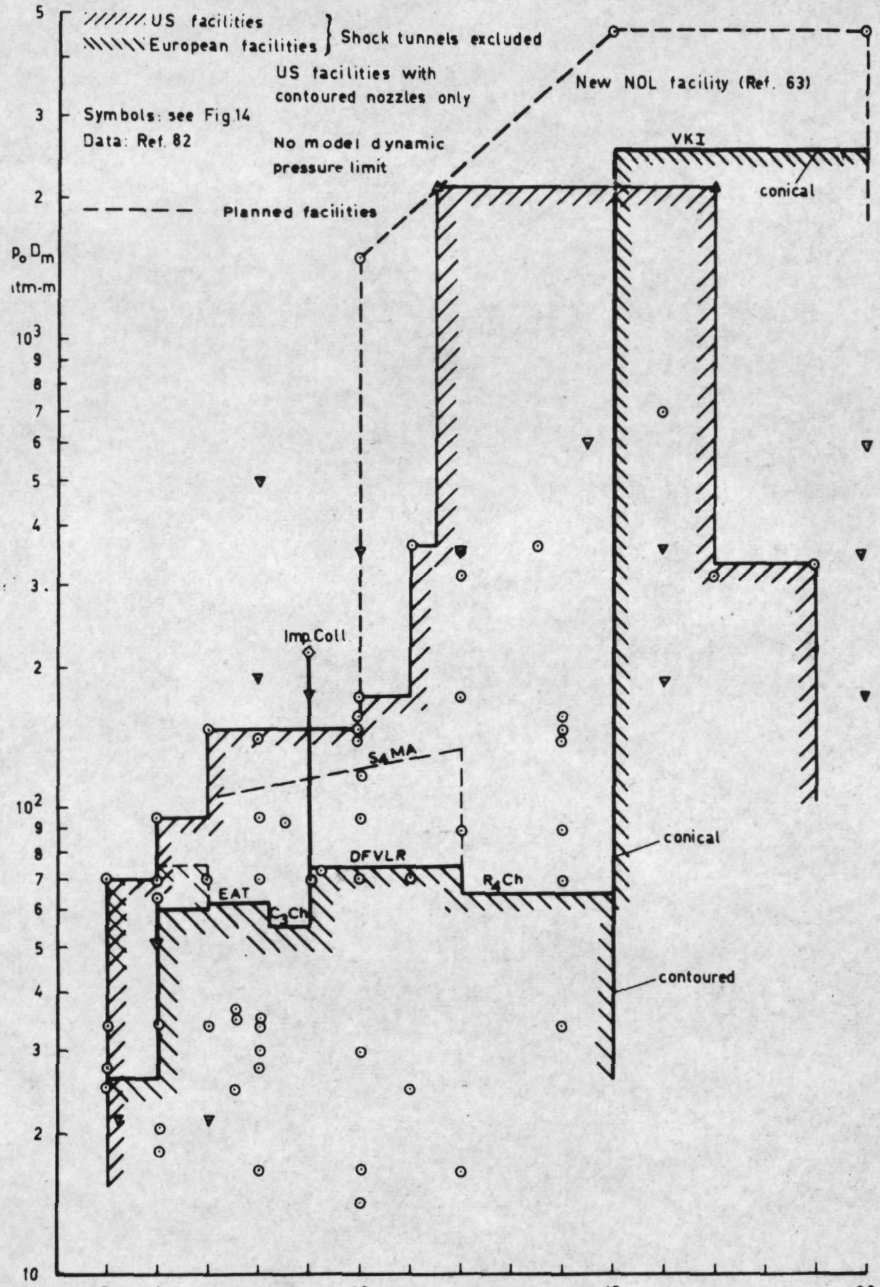


FIG. 16 PERFORMANCE OF US HYPERSONIC FACILITIES.