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From Wind Tunnel to Flight, the Role of the Laboratory in Aerospace Design

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Many costly experimental investigations in wind tunnels are made before the first flight of every aircraft prototype. In this lecture we aim at examining how, and to what extent, these investigations may lead to a determination of the aerodynamic characteristics, of the performance and of the flying qualities of the aircraft; this critical examination is based on a number of comparisons between wind-tunnel and flight results in a wide range of speeds extending from the low speeds of a V/STOL aircraft to the hypersonic speeds of a spacecraft at re-entry. After situating the flight corridors of typical airplanes and re-entry bodies (altitude, Reynolds number, Mach number, stagnation temperature), the various ground testing facilities allowing a more or less appropriate simulation of the flight conditions are surveyed.

Then, every speed range is considered: at low speed, the simulation of the transitional flight between vertical takeoff and cruise, as experienced by VTOL aircraft, makes it necessary to use sophisticated mountings in special wind tunnels; investigating high-lift devices of conventional airplanes must take place at Reynolds numbers sufficiently high to avoid too important corrections for the evaluation of the flight maximum lift; on the contrary, slender wing lift is well predicted from wind-tunnel tests on small models even in ground effect; in the transonic range, boundary-layer shock-wave interactions are dominant and well predicted in transonic wind-tunnels if the turbulent flow existing in flight along the airplane wetted surfaces is correctly simulated; on the contrary, cruise drag at subsonic and supersonic speeds is difficult to evaluate in wind tunnel at low Reynolds number since the corrections to be applied for friction are often not well known; in the supersonic range, wind-tunnel testing was developed considerably due to detailed research on SST's; it is now possible to satisfactorily predict the aerodynamic and kinetic heating characteristics from wind-tunnel tests, with the help of theoretical calculations; in the hypersonic range the methods of testing have been considerably improved during the last few years; it is now possible to predict, from wind-tunnel tests, the characteristics of drag, stability, and kinetic heating of ballistics bodies and re-entry spacecraft, with a good precision.

Finally, the need is shown for an ever closer cooperation between the research engineer, the manufacturer, and the flight test pilot so as to reach a better understanding of the wind-tunnel tests in the light of the flight tests, and therefore to enhance the practical usefulness of the laboratory.

IT is a great pleasure for me to pay homage to the pioneers of aviation, Wilbur and Orville Wright[1-3]; my youth was full of stories of this historical period. My uncle, René Quinton, although a biologist, had been in close contact with all those "magnificient men and their flying machines"; he founded the "Ligue Nationale Aérienne" (French National Air League) to encourage progress in aviation in 1908, just at the moment when Wilbur Wright arrived in France. The stay of Wilbur Wright in France, between June 1908 and February 1909, was extremely important not only for the Wright Brothers themselves, but also for the development of the newly born French Aviation.

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In this stimulating climate, designers and pilots soon broke record after record of range, speed and altitude.

Before the arrival of Wilbur in France, in 1908, the French were very suspicious of the claims made by the Wright Brothers. This attitude was mainly due to the lack of information about the work of Wrights.

I think that the Wright Brothers were also the first to have produced classified papers in aeronautics . . . but, alas, they were not the last!

Nevertheless, Wilbur's first flight at Camp d'Auvours, in August 1908 (Fig. 1) was a revelation for the whole world. Little by little, it was found that the real reason for the great advance of the Wright Brothers was their highly technical approach to aerodynamics and flight mechanics.

It was in rereading the story of their work, in their modest bicycle workshop in Dayton, that I found the subject of this lecture.

The Wright Brothers have been the first, in 1901, to use a wind tunnel to develop and improve their first airplane; in our days, this is still the right approach, and this is why I decided to entitle this lecture "From Wind Tunnel to Flight."

I realize now why this subject is so rarely discussed; this is because: *First,* the designers seldom make comparisons available, either because they lack time to make such comprehensive comparisons, or else because they do not wish to publicize them. *Secondly,* many flight measurements are very difficult to make; the drag of an airplane, for example, is generally deduced from the engine thrust measurement, which itself is seldom well known. *Thirdly,* the wind-tunnel model is rarely an exact copy of the airplane, and is often tested at too low a Reynolds number. Finally, there is a difference between the precision required in the wind-tunnel tests, for *research* purposes and for *development*. *Qualitative* results may be sufficient in *research,* since the main goal is often to compare several solutions; on the contrary, precise quantitative results are necessary in *development* work, the designer wanting to use them quickly for performance estimates of his project.

In this difficult situation, I shall nevertheless try to illustrate how tests in ground facilities may be extremely useful for predicting the characteristics of an airplane, or of a missile or even of a lifting re-entry body in its *development stage.** I shall use some unpublished results, kindly made available to me by many organizations,† and mainly by very helpful American and French colleagues.

First, let us examine the various *flight domains,* in the usual coordinates (altitude, Mach number), Fig. 2. We see that our aerodynamic interest lies inside the very thin atmospheric layer around the earth, up to about 300,000 ft; let us also consider the different types of vehicles capable of flight between zero speed and escape speed, like the Apollo capsule.

Kinetic heating is the major problem of high speed flight[5,6]: to indicate this, the equilibrium temperatures behind the front shock wave of a high speed vehicle are plotted; they reach already 150°C for "Concorde" cruising at Mach = 2.2, and more than 7000°C for the "Apollo" capsule during its reentry.

Finally, two limits of oxygen or nitrogen dissociation are drawn, to recall the importance of the real gas effects at hypervelocities.[7]

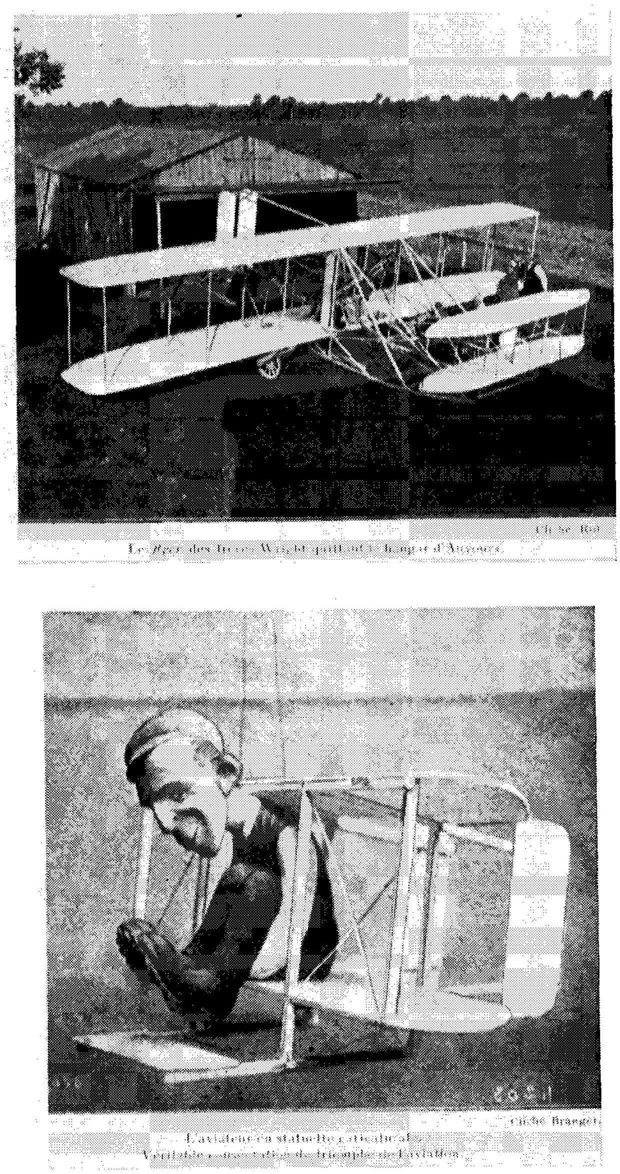


Fig. 1 The "FLYER" airplane of Wilbur Wright before his first flight in France (Camp d'Auvours, August 1908).

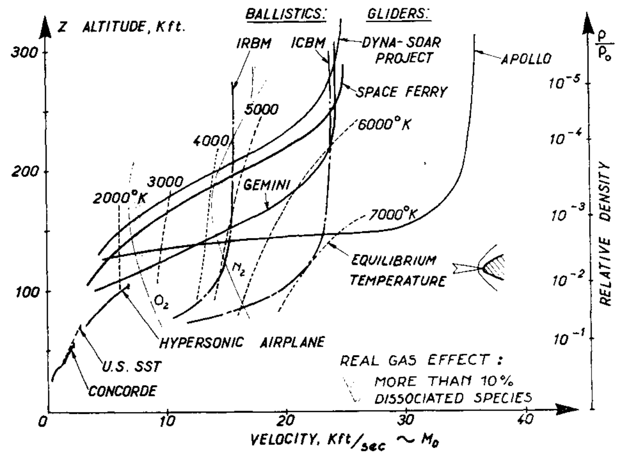


Fig. 2 Flight corridors for various vehicles, with indication of the equilibrium stagnation temperature and the onset of air dissociation encountered at high speed.

* A film has been prepared by the film section of ONERA to illustrate some wind-tunnel flight comparisons.[4]

† The author wishes to express his particular thanks to the following organizations for their advice and unpublished data: NASA Langley, Ames, and Edwards Research Centers; Arnold Engineering Development Center, Tullahoma (ARO, Inc. and U.S. Air Force); Royal Aircraft Establishment (Farnborough and Bedford); Aeronautical Research Institute of Sweden and the SAAB Company; Boeing, Douglas, Lockheed, and General Dynamics Companies in the U.S.A; Hawker Siddeley in the U.K.; Sud-Aviation, Nord-Aviation, A. M. Dassault, Bréguet, Matra, S.E.R.E.B. and the Flight Test Center (C.E.V.) in France.

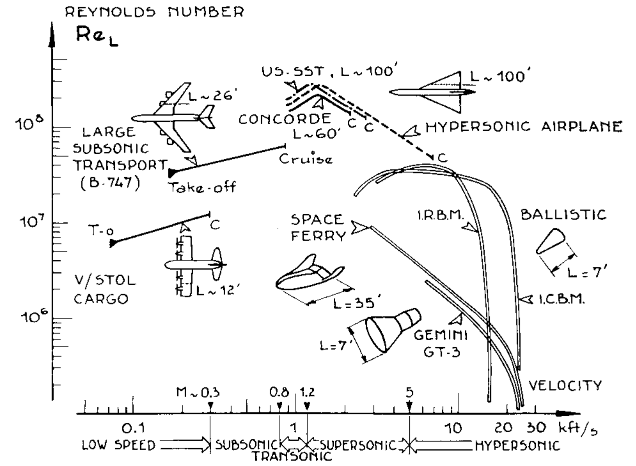


Fig. 3 Flight speed regimes and Reynolds numbers for various vehicles.

As is well known, the viscosity effects are often very important. That is why, in Fig. 3, I have plotted Reynolds number vs velocity, for some typical vehicles of interest. We shall see that, except in a few cases, it is impossible to duplicate the flight Reynolds numbers in ground facilities.

On the same chart, the speed domain is split into five parts: low speed, subsonic, transonic, supersonic, and hypersonic ranges, to facilitate our study.

1. Low Speed Regime

For ten years, low speed research has been mainly directed towards V/STOL airplanes. Much money has been spent on wind-tunnel studies and for the development of various research vehicles. Many of these V/STOL hardly have reached the exploratory flight stage; also, many have crashed.

Some reasons for such poor results are well known.[8,9] *First,* this type of airplane is more difficult to handle than a conventional aircraft, above all during vertical and transitional flight. *Another reason* lies in the difficulties found in predicting the flight behavior from tests on a small model in a wind tunnel: a good simulation of the lift engines or propellers is difficult, and the force measurement necessitates a very sophisticated instrumentation. *Last,* it is very difficult to predict, from wind-tunnel tests, the amount of control power required during the transition flight.[10]

Moreover, the model size must be small as compared to the tunnel size, to avoid too large wall effects because of the considerable amount of energy released by the lifting devices. The wall proximity can completely modify the flow around the model, especially when the lifting airflow hits the floor.[11]

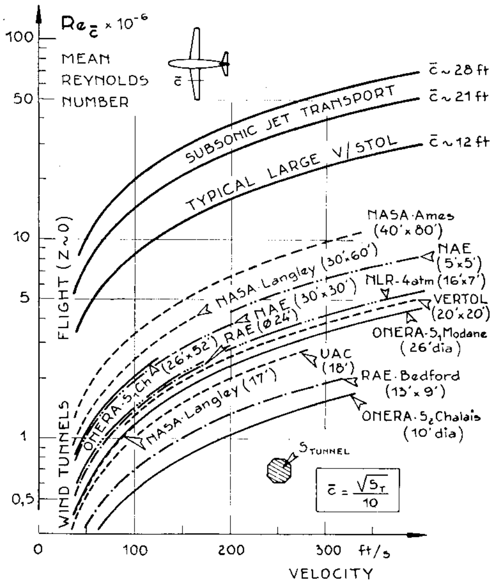


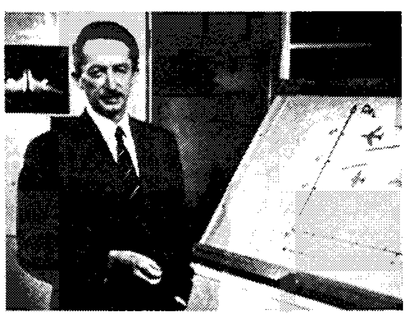
Fig. 4 Low speed test Reynolds numbers in large wind tunnels.

That is why the emphasis now seems to be on new large wind tunnels of at least 20 ft working section, specially equipped for V/STOL tests. Such wind tunnels are being designed and built in several countries.‡

In Fig. 4 the Reynolds numbers of the main existing tunnels are plotted as a function of the airstream velocity. These Reynolds numbers correspond to a model whose mean aerodynamic chord $\bar{c}$ is only 10% of the mean width of the tunnel. For the sake of comparison, the low speed Reynolds numbers found in flight are given for a typical V/STOL transport, and two conventional subsonic airplanes whose high-lift devices are to be studied. At 200 fps corresponding to the low speed flight of these airplanes, the mean chord Reynolds numbers are between 20 and 40 million, but in a conventional tunnel this Reynolds number is less than 2 million, and about 10 million in the biggest tunnel. We shall see how difficult it is

‡ In the United States, by Boeing-Vertol (20 × 20-ft, V_{max} = 270 knots), Lockheed-Georgia (26 × 30-ft, V = 100 knots), NASA-Langley (22 × 14.5 ft, V = 200 knots), and in Canada, by NRC (30 × 30-ft, V = 120 knots).

Philippe Poisson-Quinton



Philippe Poisson-Quinton, born in 1919 in France, was educated at the Sorbonne and at Ecole Superieure d'Aeronautique in Paris and Institut Aerotechnique in St-Cyr. He started to do research work at ONERA in 1946. Now he is Chief, Applied Aerodynamics Division, and in this position he is in close contact with the French Technical Services and Aerospace Manufacturers who consult ONERA for their development projects.

He contributed to the first research programs on boundary-layer and circulation control for lift increase on conventional airplanes and later to jet-lift and rotor studies as applied to VTOL aircraft.

He took part in the aerodynamic development of many transonic and supersonic French airplanes and later to the first investigations aiming at a definition of the SST Concorde. More recently, he directed basic research on the variable-geometry airplane and contributed to the Airbus and the hypersonic glider projects in ONERA wind-tunnels.

For five years, Poisson-Quinton has been lecturer of Applied Aerodynamics at the Test Pilots School of the French Flight Test Center of Istres.

Philippe Poisson-Quinton is Associate Fellow of the AIAA, member of the American Helicopter Society, and member of AGARD's Flight Mechanics Panel. He was awarded the French Médaille de l'Aéronautique.

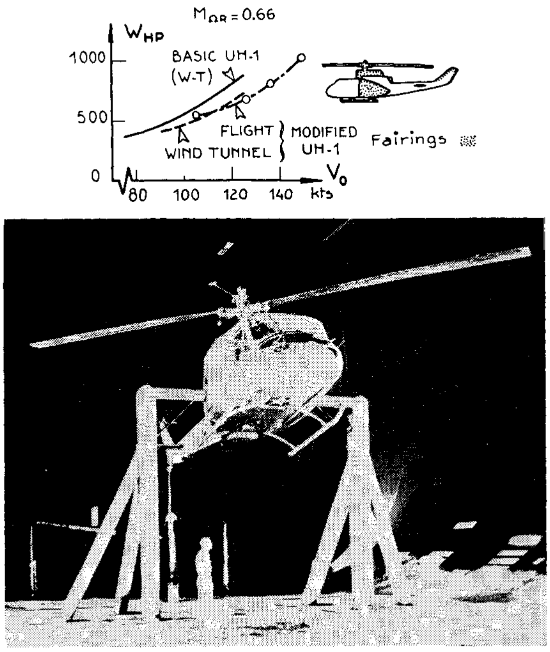


Fig. 5 Improvement of the Bell UH-1 helicopter performance: full scale tests in the Ames 40- × 80-ft wind tunnel (NASA) and flight data.

to predict some aerodynamic characteristics in flight from low Reynolds number wind-tunnel results.

The so-called "full scale tunnels," like those of Ames and Langley at NASA, or Chalais-Meudon at ONERA, are used both for real airplane tests and for free flight tests on dynamically similar models[12,13]; the S_1 Modane sonic tunnel (D = 26 ft) is also used for special full scale tests such as release of canopies and the cabin environment before bail out, rain simulation upstream of a cockpit, de-icing systems efficiency, turbo-ramjet performance, etc.[4,68]

The last point to mention here is the choice of the maximum velocity required in these low speed tunnels. It seems important to be able to simulate the transonic flow around the leading edge of a wing at high angle of attack; this corresponds to a Mach number of about 0.3, or 330 fps. Such conditions are encountered during flight tests of high-lift devices at altitude and may be present on high speed airplanes in climb. Also, the receding blade of a helicopter rotor works near maximum lift at a Mach number of 0.3 or 0.4. These conditions have to be simulated in the two-dimensional tests of the blade sections.[14] Naturally, the best solution is to separate the Reynolds number and Mach number effects by the use of a variable density tunnel.

Now, I should like to review briefly some V/STOL problems and the low speed characteristics of fast airplanes.

V/STOL Problems

In the case of *helicopters*, calculations and past flight experience are mainly used for the performance predictions of a new machine, because small scale models are not really appropriate for rotor studies in wind tunnels, except for fuselage interactions in a rotor slipstream. However, full scale tests have taken place recently for improving poorly designed fuselages. Figure 5 shows a good agreement between the results obtained in the Ames wind tunnel and in flight on the Bell UH-1.[15] These tests resulted in an improved fuselage shape, thus reducing the drag and substantially increasing the speed. It seems that large scale tests, both in wind tunnel and in flight, with a very sophisticated instrumentation on the blades, should lead to improved helicopter performance in the near future.[16]

Turning now to jet lift VTOL, we have, in France, some experience with the Dassault "Balzac"[17] and the "Mirage 3-V." The wind-tunnel tests with a small model are particularly difficult since although the intake and exhaust flowfields are simulated, all force components on the airframe have to be measured.[18,19] After many successive improvements, wind-tunnel tests, by the Dassault Company, of a very sophisticated, one-thirteenth scale model of the "Mirage 3-V,"

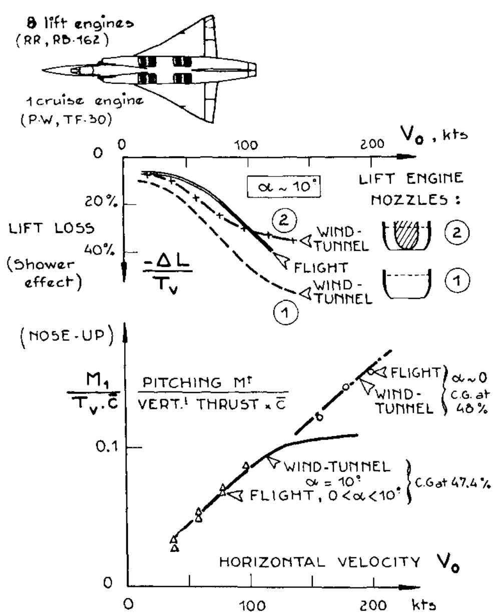


Fig. 6 Jet lift VTOL Mirage 3-V. Wind-tunnel/flight comparison of lift loss and nose up pitching moment during transition.

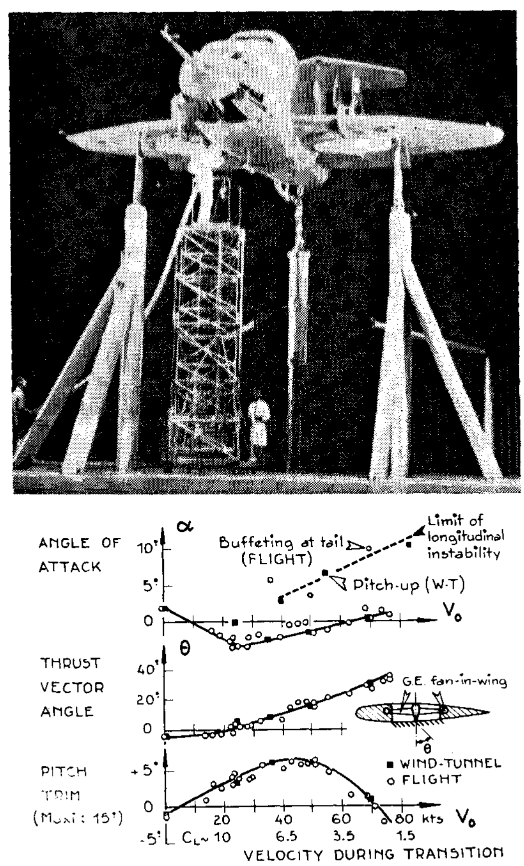


Fig. 7 Fan-in-wing VTOL Ryan XV-5A: aerodynamic characteristics of transition of the prototype tested in the Ames 40- × 80-ft wind tunnel before flight tests.

(Fig. 6), were in good agreement with flight results obtained during transition between hovering and cruise. The lift loss and the increase of the nose-up pitching moment with forward speed are two typical characteristics of this type of VTOL.[20] This experimentation in wind tunnel has shown also that it is very important to simulate exactly the nozzle shape of the lift engines, and that it is possible to improve these transition characteristics with a small modification of the jets' configuration.

Another example of good agreement between wind-tunnel and flight tests is given on Fig. 7 for the Ryan XV-5A, fan-in-wing VTOL. Here the prototype was tested full scale in the Ames 40- × 80-ft wind tunnel before its first flight.[21] Such a study was very useful, not only for the flight-test center, but also for the fan-in-wing manufacturer.

In the case of the tilt-wing XC-142 A, Fig. 8, this propeller-VTOL was too large to be tested full scale. Several models were used but the comparison of results[21,22] was somewhat disappointing because—as we know—the models were never exactly similar, and also because the same parameters were not measured on every model.

However, the wind tunnel gave a good indication of the relation between the velocity during transition and the wing angle of attack, or the thrust required (Fig. 8); also shown are the wing tilt incidence angles given by the tests of a dynamic model mounted on the horizontally moving carriage of the Princeton University track. This semi-free flight facility[12] is primarily used for dynamic longitudinal stability research on VTOL in, and out of, the ground effect.

It is difficult to predict, in a wind tunnel, a most important parameter of VTOL performance, namely, the rate of descent. This limitation is due to the wing stall and flow separation, and hence is very sensitive to Reynolds number. On small models the descent boundary given by the tufts visualization of the separation is often very pessimistic. The same trend was found with the free flight of a 11% scale model in the 30- × 60-ft Langley wind tunnel[8]: a 5° difference on the maximum flyable slope.

In flight, the pilot's judgment is required to appreciate the descent limitations. At first, a light buffet due to local flow separations on the wing, and then a strong deterioration of the handling qualities are felt. These two boundaries of the descent rate are shown in Fig. 9 in terms of the transition velocity.

For the tests in the Ames 40- × 80-ft wind tunnel of a large model (0.6 scale) of the XC-142 A, the descent limit was determined by the onset of a complete flow separation on the wing (tuft visualizations) and by the sudden increase of horizontal tail vibrations. This boundary was about the same as the "light buffeting" limit in flight, that is to say, a

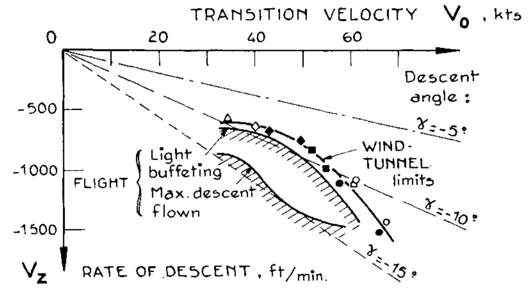


Fig. 9 Descent limitations during approach of the LTV XC-142 A: wind-tunnel prediction and flight results.

descent slope prediction 5° less then the "maximum descent flown" ($\gamma = -15°$).

Considering another propeller slipstream configuration, I should like to present some comparative results on the Bréguet 941. This STOL airplane resulted from much basic research in wind tunnels and flight, on the experimental Bréguet 940 ten years ago.[23]

The polar curves given by the Bréguet wind-tunnel on a one-ninth scale model are very close to flight results, except near the stall, as shown in Fig. 10, but the very large $C_{L\,\max}$ increase with the propeller slipstream on the deflected flaps is about the same in wind tunnel and in flight. Incidentally, some recent results from NASA on a large model of a similar configuration show that a leading edge device (slat) is very efficient as regards improvement of $C_{L\,\max}$ when the airfoil section has no camber (unlike the Bréguet 941).

Again, a high rate of descent is mandatory to perform a very short landing. The slope of descent in flight, given in Fig. 11, is about $-7°$ during a "standard" approach,[24] with a very large safety margin (at 9° below the incidence of stall).

It is interesting to mention here the spectacular increase of the descent slope, from $-7°$ to $-11°$ proven recently in flight, and obtained by use of what we call "propeller transparency," that is to say, a zero equivalent thrust on the external propellers. This improvement of the descent capability due to a larger drag for the same lift was well predicted in the Bréguet wind tunnel.

To deflect the propeller slipstream, blown flaps are also very efficient, as verified in flight on the NASA modified Lockheed

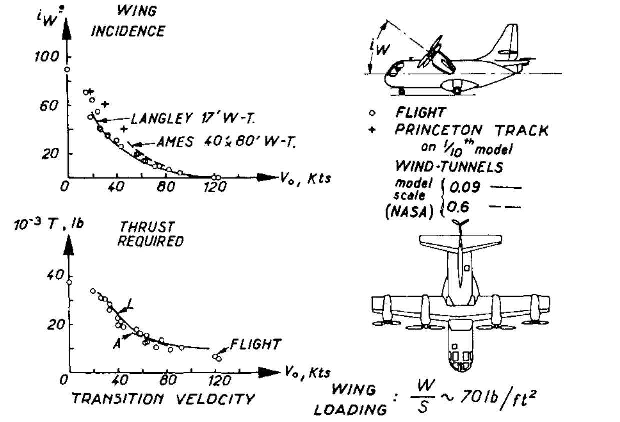


Fig. 8 Tilt wing VTOL LTV XC-142 A: wing tilt angle and thrust required during level flight transition and from wind-tunnel predictions.

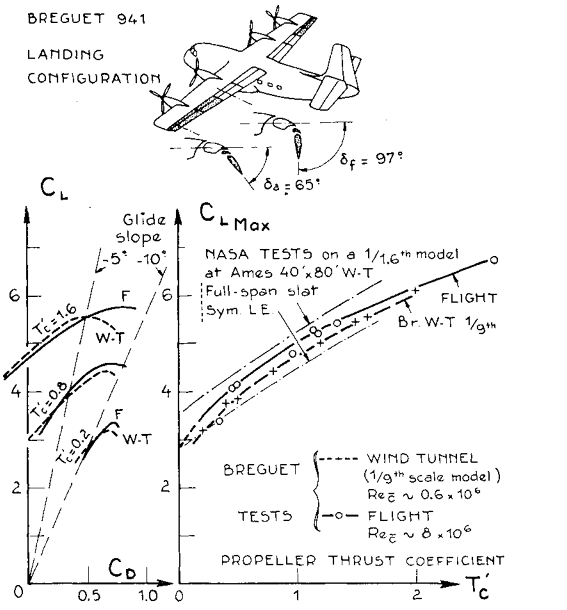
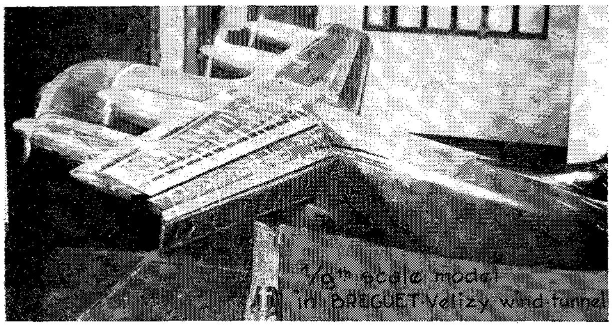


Fig. 10 STOL Bréguet 941: influence of the propeller slipstream on the polar curves and the $C_{L\,\max}$ in wind tunnel and flight.



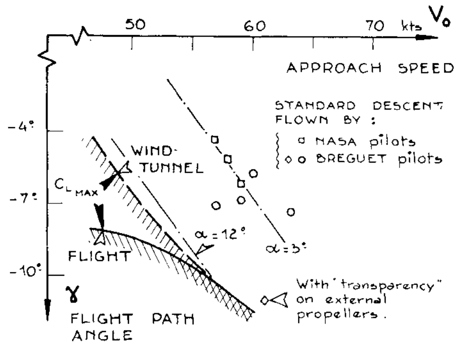


Fig. 11 Descent capability during the Bréguet 941 approach.

C-130, and more recently, on an experimental four propeller Japanese seaplane, the UF-XS. Here, both propeller thrust coefficient C'_T, and blowing momentum coefficient must be carefully analysed in wind tunnel to estimate the high-lift performance. In the case of UF-XS, the wind-tunnel tests on a small model at very low Reynolds number,[25] under predicted the C_L in STOL configuration by about 10%. In fact, takeoff and landing were performed at less than 40 knots, i.e., with C_L in excess of 6.

Low Speed Characteristics of High Speed Airplanes

The study of the low speed characteristics of conventional *high speed airplanes* is still a very fundamental subject. Here the main problem is to predict, with a small model, the maximum *usable* lift coefficient, with *acceptable* stability and control in pitch, yaw and roll. In general, the usable $C_{L\max}$ is limited: 1) by the wing stall, on large aspect-ratio airplane; 2) by some stability troubles, mainly in pitch and roll, experienced with moderate aspect-ratio swept wings; 3) by ground clearance and visibility on slender-wing airplane.

In every case, it is mandatory to study the wind-tunnel model up to a combination of incidence and yaw angles much larger than normally encountered in flight. First, for the large aspect-ratio airplanes, it is difficult to predict maximum lift from wind-tunnel tests, when only small models are available, due to the Reynolds number effect and the difficulty of duplicating the high-lift devices on a small scale. Model testing always gives a maximum lift lower than flight test results. This is shown in Fig. 12 for three turbo-propeller driven airplanes and for the "Caravelle" jet transport.

We have tried a rough evaluation of the Reynolds number effect based on large number of results obtained both in wind tunnel and in flight. In each case, we have plotted the respective values of the $C_{L\,\max}$ at various flap settings, for every airplane and model, as shown on Fig. 13a, for several transport configurations from three firms. Then, the ratios of the $C_{L\,\max}$ obtained in wind tunnel to that in flight are plotted vs Reynolds number in Fig. 13b, assuming that the mean Reynolds number of a large transport airplane is about 12 million during landing. We can see that most of the experimental values fall near a mean curve, which indicates an underprediction of about 20% when a model is tested at a Reynolds number of about 1 million, a value typical of a conventional wind tunnel. Flight $C_{L\,\max}$ values given here were obtained during stall tests with negligible angular velocity, but the $C_{L\,\max}$ increases slightly with the pitch velocity during rapid maneuvers, as proven both in-flight and in wind-tunnel tests.[26] We must not forget that, beyond the $C_{L\,\max}$ incidence, serious trouble may quickly develop, particularly in some cases of swept wing, *T*-tail airplanes. Generally, a violent pitch-up is followed by a stable longitudinal attitude at a very large angle of attack. This is known as "deep-stall"[9,27,28] and it leads to disaster, unless the horizontal tail is powerful enough to bring back the airplane into a normal level flight. These dangerous flight configurations can be easily predicted by means of wind-tunnel tests, as long as these tests are carried out very carefully at large angles of attack, up to 50°, far beyond the stall. Recent examples of deep stall accidents show how large the responsibility of the wind-tunnel engineer is, and how important it is that he should be in close contact with the designer during the development phase of a new prototype.

Another dangerous situation, experienced mainly by fighter aircraft, is the entry into a spin after a dynamic stall during a sharp turn on a sudden pull out. Here again the laboratory is very useful in the evaluation of the spin tendency and in determination of the best recovery maneuvers. In fact, the French Air Ministry requires that the spin characteristics be determined in the vertical spin tunnel of the Fluid Mechanics Institute of Lille before the first flight of any prototype.[30] For example, in the case of the supersonic delta fighter "Mirage 3," the wind-tunnel tests had shown that the only way to recover from the spin was to move the control column

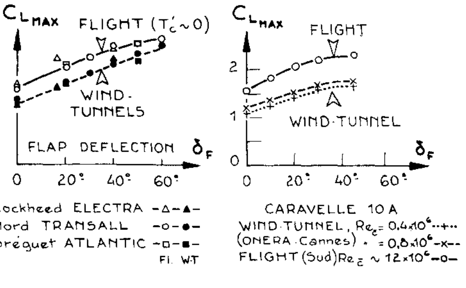


Fig. 12 Maximum lift coefficient vs conventional flap deflection in wind tunnel and in flight for turbo-propeller and jet aircraft.

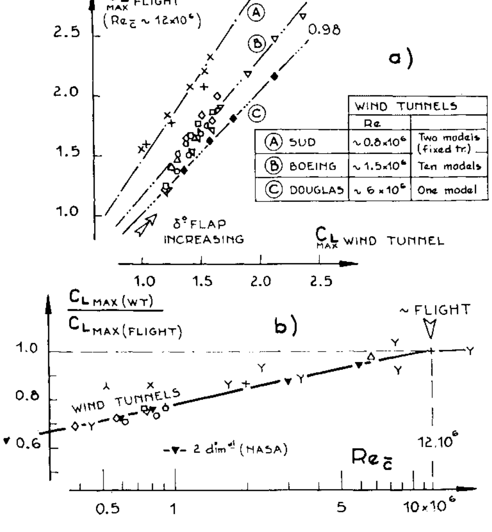


Fig. 13 Wind-tunnel/flight correlation of maximum lift for several high aspect-ratio transport aircraft.

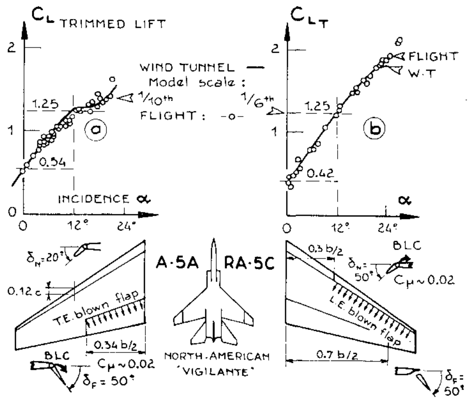


Fig. 14 High-lift performance with boundary-layer blowing on the two versions of the carrier based NAA "Vigilante."

towards the center of the spin. Flight tests showed that this prediction was correct.[4,30]

High maximum lift and good stability and control characteristics during landing approach are mandatory for carrier-based strike aircraft, and that is why much wind-tunnel research was undertaken to improve the usable $C_{L\,max}$ of the swept, low aspect-ratio wings of these airplanes, mainly by means of boundary-layer control (blowing slots near the leading edge and on the trailing edge flaps). Flight test results have confirmed the large improvements predicted in wind tunnels, on the $C_{L\,max}$ and maneuverability at low speed, when using boundary-layer control (BLC) (the U.S. "Phantom"[31] and "Vigilante,"[32] the "Buccaneer" in England, and the experimental "Etendard" in France,[33] for example). The case of the North American Aviation "Vigilante" is most interesting because two configurations of BLC were applied: 1) blowing on the trailing edge flap for the A-5A, Fig. 14a; 2) blowing on the drooped leading edge for the RA-SC, Fig. 14b.

In both cases the lift curves measured in flight were in good agreement with the wind-tunnel measurements at the same momentum coefficient C_μ, but the $C_{L\,max}$ was larger in flight; the leading edge blowing was very effective in avoiding the leading edge stall, i.e., the pitch-up, and also in improving roll control during carrier approach.[32,33]

Turning now to slender wings used for supersonic flight, their usable lift is limited either by some stability troubles or by the ground clearance at takeoff and landing, or even by visibility during the approach.

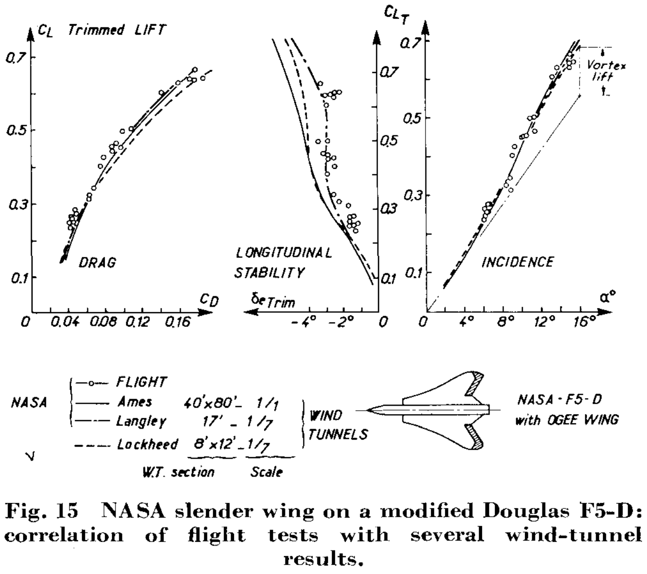


Fig. 15 NASA slender wing on a modified Douglas F5-D: correlation of flight tests with several wind-tunnel results.

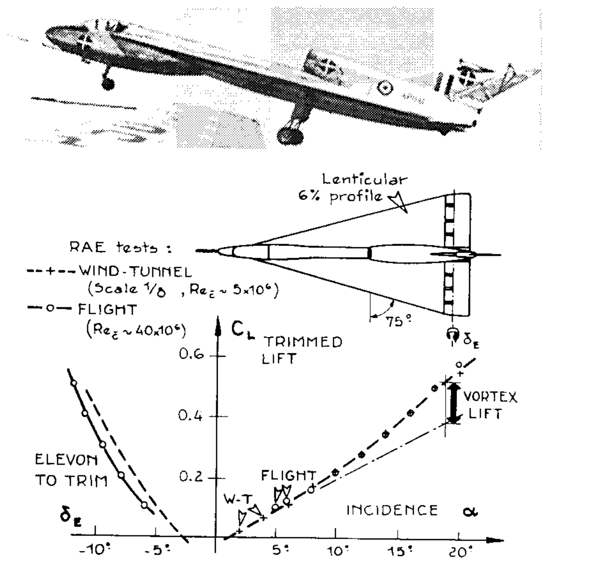


Fig. 16 Vortex lift correlation on the Handley-Page 115 research airplane.

For this type of thin wings, whose leading edge is quite sharp, a laminar separation initiates the well known vortex sheet. A comparison of the vortex visualization in flight, on a F-5D modified by NASA, and in the ONERA hydrodynamic tunnel at very low Reynolds number,[4,34] shows not only the same position for the vortex, but also bursting on the upper surface of the wing at about the same incidence. The wind-tunnel-flight comparisons on the modified F-5D[35] have demonstrated that, on this type of thin wing, the vortex lift increase is practically independent of the Reynolds number, as shown by the (C_L, α) curves obtained in several tunnels, Fig. 15. The correlation is not as good for the longitudinal stability, but the reduction of the static margin appears in every case at about the same CL; the wind-tunnel polar curves are in good agreement with flight measurements. The vortex lift contribution is even larger on the 75° delta research airplane tested by the RAE, as shown on Fig. 16. The wind-tunnel flight correlations on the H.P. 115 were satisfactory, even for the lateral stability derivatives.[36]

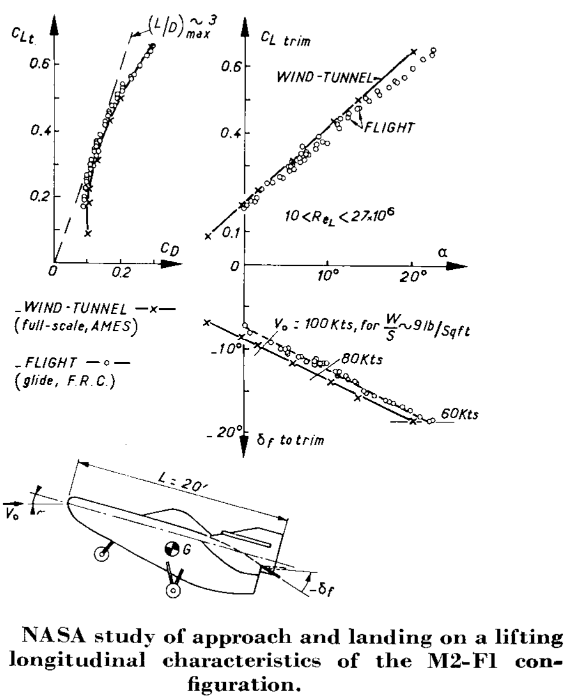


Fig. 17 NASA study of approach and landing on a lifting body; longitudinal characteristics of the M2-Fl configuration.

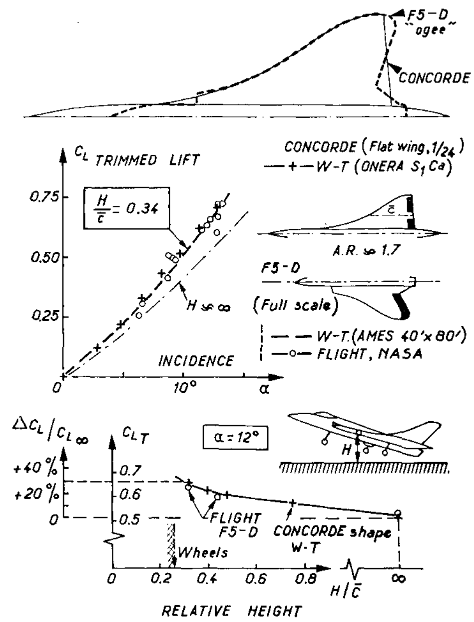


Fig. 18 Favorable ground effect with a slender wing: comparison between a "Concorde" model and the modified F5-D having about the same platform.

We have to mention here also the case of lifting bodies, being tested now by NASA at low speed in preparation of the future manned re-entries from space. In this case, the leading edge is very blunt to reduce the hypersonic heating rates and no longer any leading edge laminar separation initiates the vortex lift, except on small models at low Reynolds numbers. This is why the aerodynamic characteristics of various configurations (M2, HL-10, SV-5A) are determined from full scale tests in the NASA 40- × 80-ft tunnel before the flights at Edwards.[37] So far, very good correlations have been obtained, as shown for the first M2-F1 vehicle, Fig. 17.

The Ground Effect

Finally, *the ground effect* is a very important phenomenon which must be carefully analyzed in wind tunnels for all types of airplanes.

This is a *favorable* effect in the case of slender wings (and a fortunate one with respect to the Concorde). Here again, the flight tests carried out by NASA with the modified F5-D[35] confirmed our Concorde predictions obtained with a fixed ground board in a wind tunnel, Fig. 18. A trimmed lift increase of about 30% was obtained with the airplane very near the ground.

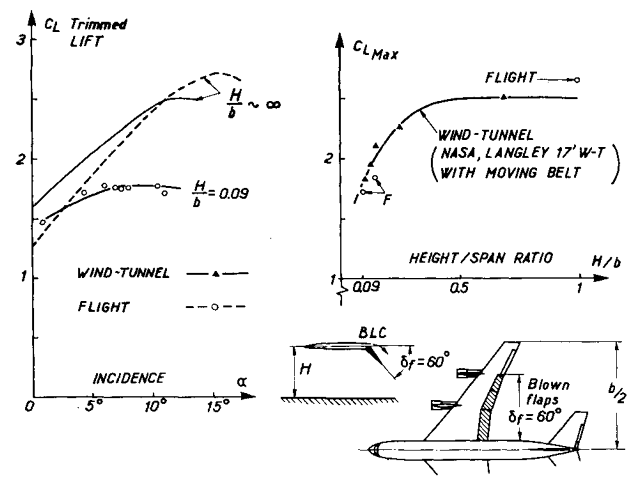


Fig. 19 Unfavorable ground effect on the Boeing 367-80 with blown flaps.

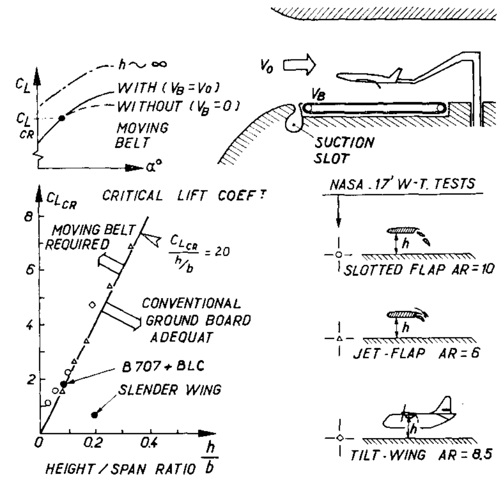


Fig. 20 Ground effect tests with a moving belt in the 17-ft tunnel of the NASA Langley Research Center.

The ground effect becomes *unfavorable* on large aspect-ratio wings with powerful high-lift devices. This is the case for the Boeing 707 prototype (367–80) equipped with BLC blowing over the flaps[38]; Fig. 19 shows that, when the wing is very near the ground, the maximum lift is reduced by more than 30%. Wind-tunnel and flight tests are also in good agreement on the drag reduction (increase of effective aspect ratio) and on the nose-down pitching moment (large change in downwash at the tail near the ground).

These wind-tunnel tests were conducted by NASA in the Langley 17-ft tunnel, equipped with a moving belt to obtain a ground velocity equal to the air flow velocity[8,20,39]; thus was avoided the formation of a thick boundary layer, which would easily separate from the ground board due to impinging flow deflected downward by the blown flaps.

By a systematic investigation of this effect, Turner, at Langley,[8] has determined the condition in which the moving belt is needed. Figure 20 summarizes this research with three configurations equipped with powerful lift devices on large aspect-ratio wings. A moving belt is needed when the ratio of the lift coefficient to the relative height above the ground is larger than about twenty.

It was found also that the moving belt is useless in the cases of jet lift and some fan lift VTOL configurations because the lifting units affect only a small part of the wing span.

More important ground effects occur during hovering of many VTOL configurations. The aerodynamic suck-down and the hot gas recirculation predicted by model static tests are generally in good agreement with full scale results.[8–10]

2. Subsonic Regime

I shall speak briefly about the conventional subsonic tests, that is to say, below the critical Mach number and at moderate angles of attack, where the boundary-layer separation problems are not too critical.

Generally, there is good agreement between flight test and wind-tunnel results for lift, stability derivatives and control efficiencies (see, for example, the comparisons for both steady and dynamic tests, Figs. 35, 37). In every case, at the low Reynolds number of the wind-tunnel tests, it is mandatory to trip artificially the boundary-layer transition on every surface to avoid some parasite laminar separation.

However, in general, the wind tunnel cannot give directly a good drag prediction for the following reasons: first, all details of an airplane configuration cannot be reproduced with the model; second, wind-tunnel model mounting systems

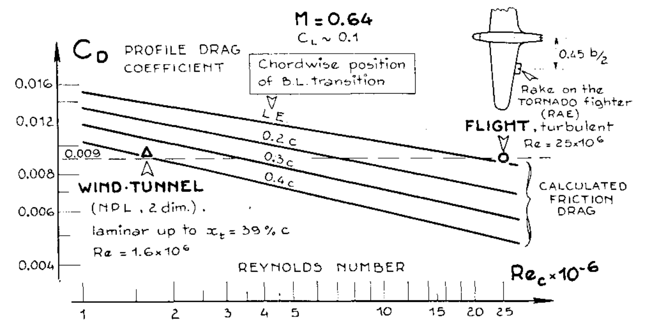


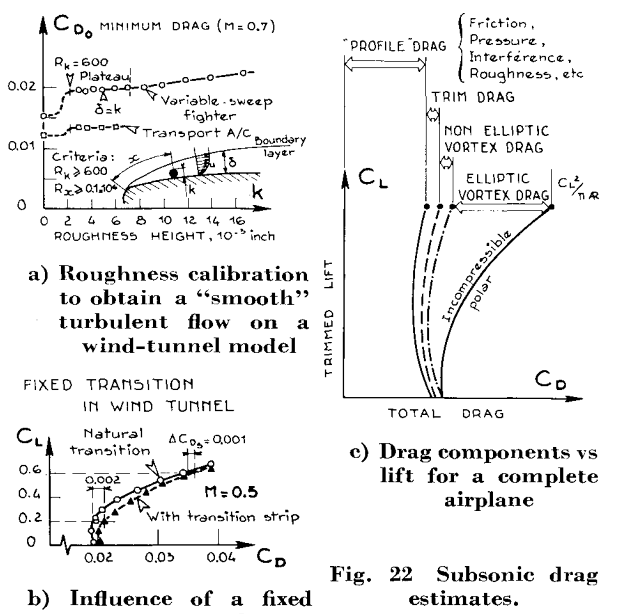
Fig. 21 Effects of Reynolds number and transition location on the profile drag measured by the wake method in wind tunnel and in flight.

often produce, or eliminate, some possible drag sources (and also some parasite moments); third, it is very difficult to estimate which drag components are Reynolds number dependent and should be corrected in order to predict the full scale values.

Frictional Drag

In order to estimate the friction drag, a parameter necessary for the calculation of the full scale minimum drag, we have to know exactly the condition of the boundary layer (laminar or turbulent) on all wetted surfaces of the model. A very basic example, shown in Fig. 21, illustrates the importance of the transition location on the model. Here are plotted the minimum profile drag values, obtained by the wake method, in wind tunnel on a two-dimensional profile, and in flight at the mid-span of a fighter aircraft.[41] In the latter case, at high Reynolds number ($Re = 25 \times 10^6$) and with a conventional wing skin, the boundary layer is turbulent just after the leading edge, and the drag is well predicted by the friction calculation based on the profile pressure distribution and transition at the leading edge. At the low Reynolds number ($Re = 1.6 \times 10^6$) of the wind-tunnel test, the boundary layer is laminar as far as 40% of the chord, and the measured drag has, fortuitously, about the same value as in flight, but, had the transition been fixed near the leading edge, the wind-tunnel drag would have been more than 50% larger than in flight.

In a free transition test on a complete model, it is difficult to determine exactly what parts remain laminar and thus to



a) Roughness calibration to obtain a "smooth" turbulent flow on a wind-tunnel model

b) Influence of a fixed transition on the induced drag

c) Drag components vs lift for a complete airplane

Fig. 22 Subsonic drag estimates.

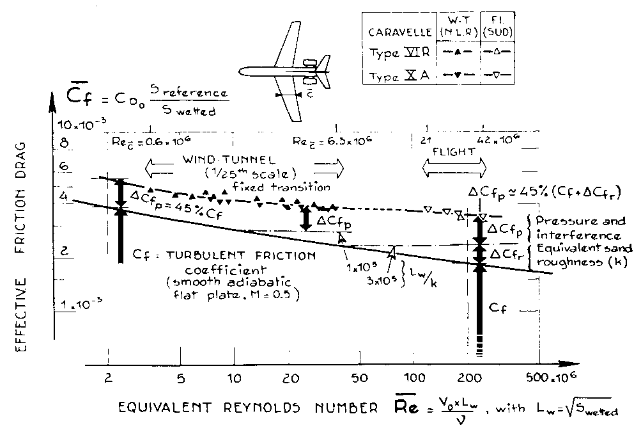


Fig. 23 Wind-tunnel correlation of minimum drag flight for two types of "Caravelle": effective friction drag vs equivalent Reynolds number.

calculate a friction correction for every surface. This is why artificially tripping the transition at low Reynolds number is the only practical method of extrapolating the turbulent friction up to the flight Reynolds number; the size and the location of the strips (Fig. 22a) must be chosen after a careful experimental analysis[42–46] to insure a turbulent flow without, or with a well known, parasite drag. This problem, as we shall see, is even more difficult for high speed tests.

Fixing the transition permits us also to avoid large errors in the induced drag estimates because the boundary layer, at first laminar, becomes turbulent on the upper surface when the incidence increases,[47–49] giving a pessimistic induced drag factor dC_D/dC_L^2 (Fig. 22b). Even with a fixed transition on the model, there remains a Reynolds number effect because, generally, the usual induced drag factor takes account of all the parasite drags that grow with angle of attack (Fig. 22c). Finally, the "suction effect" recovery on slender wings depends also on the normalized leading edge Reynolds number: $Re_{l.e.} = V_n\, r/\nu$, if $Re_{l.e.} < 20{,}000$, as shown by Henderson.[49]

When enough wind-tunnel and flight test data at various Reynolds numbers are available, it becomes possible to attempt some minimum drag correlation. In Fig. 23 we have plotted the effective friction drag vs an equivalent Reynolds number, taking account of the total wetted surface area of two different types of "Caravelle," to compare the experimental values with the theoretical friction drag of the flat plate. If we assume that the pressure and interference drags depend upon the boundary-layer development, then

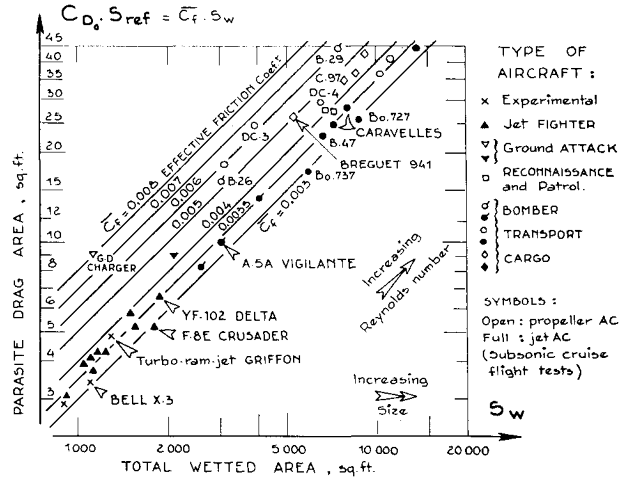


Fig. 24 Subsonic minimum drag of various aircraft from flight measurements.

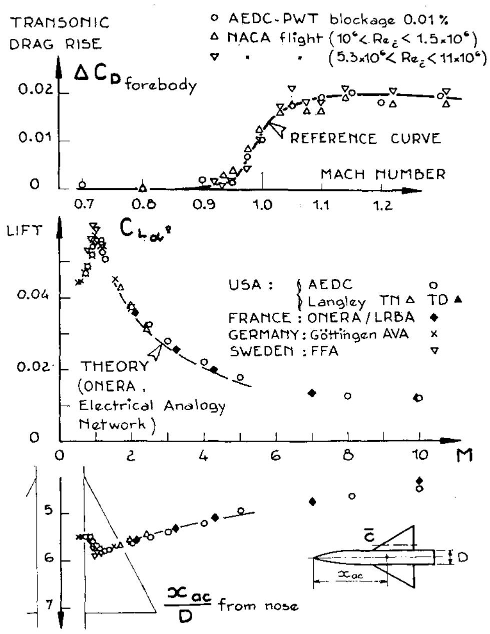


Fig. 25 AGARD "B" calibration model for high speed wind tunnels: transonic drag divergence, and lift and stability at supersonic and hypersonic speeds.

they must be proportional to the theoretical friction drag[47] and, for that, we have to consider a new set of theoretical friction curves taking account of some equivalent surface roughness[50] for both the model and the full scale airplane; nevertheless, the total effective friction drag is about 50% larger than the flat plate value.[47,54]

A much less precise correlation of the effective friction drag coefficient of various types of airplanes[54] is given in Fig. 24, in terms of parasite drag area and total wetted area, to show that most of the recent high speed airplanes have an effective friction drag coefficient equal to or lower than $\bar{C}_F = 0.0035$. Of course, the number is much larger for a cargo-type aircraft.

In fact, until now, for drag predictions, most designers rely on *analytical methods*, a mixture of friction drag calculations and of very personal formulae, taking past experience into account.

Parasite Drag

It is possible to estimate some parasite drag terms directly from flight measurements on airplanes of the same family,[51–54] but the wind tunnel remains very useful to compare a new project with a previous configurations well known in flight. These methods are commonly used by the transport aircraft firms.

Let us also note that the designer must be very cautious about many small sources of parasite drag on his full scale airplane: for example, between the prototype and the first production airplane of the executive "Jet-Falcon," Dassault designers were able to reduce the cruise drag by 6%, thanks to operation "clean-up." They eliminated all parasite drag sources, such as windshields, protruding aerials, etc., and they filled up all gaps, slots, and discontinuities. To estimate such a gain with a small model in a wind tunnel is, unfortunately, impossible.

3. Transonic Regime

Now, I should like to say a few words about the transonic domain well explored by the military airplanes and, soon, to be traversed by the SST. But let us remember that, between 1945 and 1955, we were, all of us, in about the same situation as at the time of the Wright Brothers, when every step forward cost the life of a pilot.

In fact, in those days, flight testing was *ahead* of the laboratory, where we did not know how to simulate transonic flow in wind tunnels. At the same time, it was possible to fly with the first jet fighters, in these very dangerous regimes, with little scientific background. It is only when slotted and perforated test sections were developed that detailed investigations of transonic regime became possible.[55] At about that time Dr. von Kármán founded AGARD, which became an organization for true international cooperation. One of the first proposals of AGARD was a "calibration model" for the evaluation of the wind tunnels of the member nations.

These comparisons[56] have been most valuable to all, and permitted to develop the test section geometry for correct transonic measurements, and also to check the validity of supersonic measurements (Fig. 25).

It was also the time when the aerodynamicist, just like a magician, was able to explain to the pilots the reasons for such transonic troubles as "buffeting," "pitch-up," "wing-dropping," or "aileron reversal efficiency." The best thing for the aerodynamicist, as for a surgeon, was to study a typically "sick" airplane, meeting all these troubles in the same flight. Figure 26 shows these unusual longitudinal characteristics up to Mach 1 and a very good agreement between wind-tunnel tests on a small model and flight results.

The pitch-up problem, for some swept wing configurations, grows worse when shock waves cause boundary-layer separation at the wing tips.[57] This trouble is very well predicted

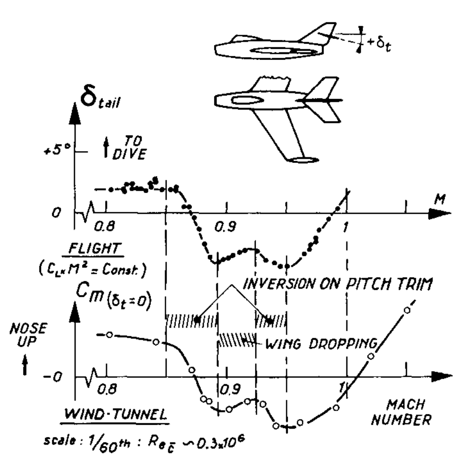


Fig. 26 Flight/wind-tunnel comparison of transonic longitudinal characteristics of an experimental "Mystère II" fighter tested by ONERA.

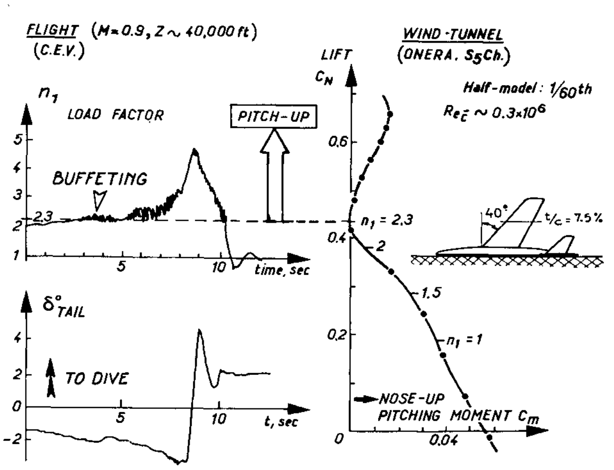


Fig. 27 Record of transonic pitch-up of "Mystère IV" in flight at $M = 0.9$, and comparison with wind-tunnel prediction from the longitudinal stability curve shape.

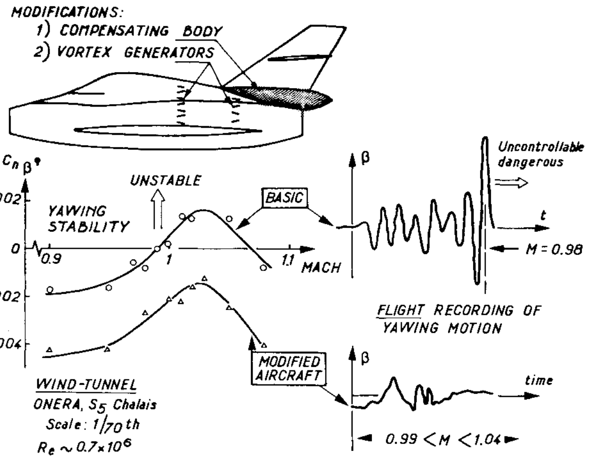


Fig. 28 Transonic yawing instability of the experimental delta Nord "Gerfaut" cured by after-body fixes studied in wind tunnel.

by a typical kink on the longitudinal stability curve, as shown in Fig. 27.

In some cases, a strong shock wave located at the rear fuselage, near Mach = 1, is sufficient to give a large boundary-layer separation on the fin and this explains the very dangerous yawing instability that appeared during the first transonic flight of an experimental delta (Fig. 28); here again, it was possible to duplicate this trouble in a small transonic wind tunnel, exactly at the same Mach number, and to find a most unusual, but very efficient solution, to cross the sonic barrier.

Nowadays, high speed airplanes are slender enough to experience no trouble of this kind in the transonic range. In fact, we are more concerned about improvements and optimization of subsonic jet transports. In particular, we try to increase the limit Mach number, when drag divergence and buffeting appear successively. Then, the problem is "how to work out an economical solution in spite of the shock-waves?"

Shock-Boundary-Layer Interaction

A wind-tunnel study of this "lower transonic range" is not so easy, because of the unfavorable Reynolds number

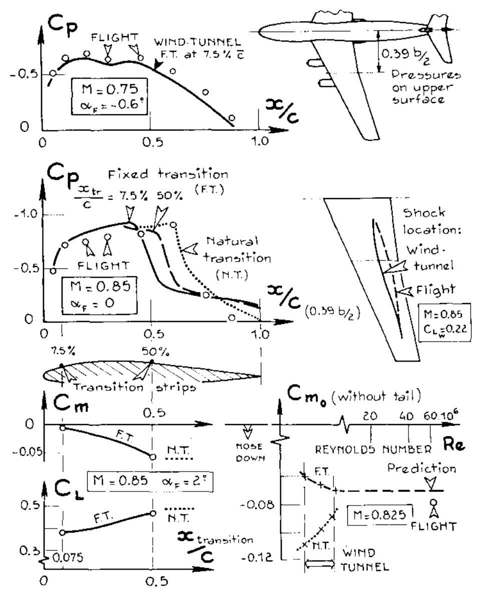


Fig. 29 Wind-tunnel/flight comparisons on the Lockheed C-141 jet cargo: detrimental effect of transition strips on the shock location and the pitching moment in supercritical regime.

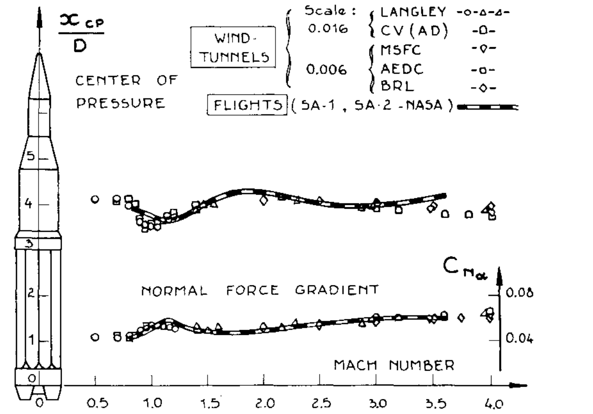


Fig. 30 Normal force and center of pressure for the Saturn IC booster at transonic and supersonic speeds from wind tunnel (fixed transition) and NASA flight tests.

effect, often leading to premature separations as soon as the first shock waves appear. Here again, tripping the boundary-layer transition is generally helpful, but in some cases it is necessary to try several strip locations on the wing model when the shock-wave location is very sensitive to boundary-layer thickening due to the roughness. A very good example of this difficulty is found in the wind-tunnel/flight comparisons of the cargo-jet C-141,[58,59] where a free transition on the model gives a pressure distribution much more similar to flight at the supercritical Mach numbers, as shown on Fig. 29b. In this case, the wind-tunnel prediction of the pitching moment (i.e., the tail load), with conventional roughness near the leading edge, was erroneous because of a shock location much nearer to the leading edge than in flight, particularly on the upper surface in the mid-span region, where the flow is quasi-two-dimensional. This problem is not yet well understood and needs a more detailed analysis not only for the typical "roof top" chordwise pressure distribution, but also for some more modern "peaky" types, where a supercritical regime exists near the leading edge at cruise conditions.[60]

The transonic tests of missiles or rockets at low Reynolds numbers are also difficult because of a frequent shock-wave-boundary-layer interaction in the neighborhood of the base, near Mach = 1. Here again, an artificial turbulent flow, given by roughness, is useful for good predictions of the lift and the stability derivatives, as shown in Fig. 30 for the Saturn IC booster, tested in many facilities at a small scale and then in flight by NASA.

Transonic flow may occur locally at the leading edge, in spite of a low Mach number, when the angle of attack is high, as on the receding blade of a helicopter rotor, or on a modern

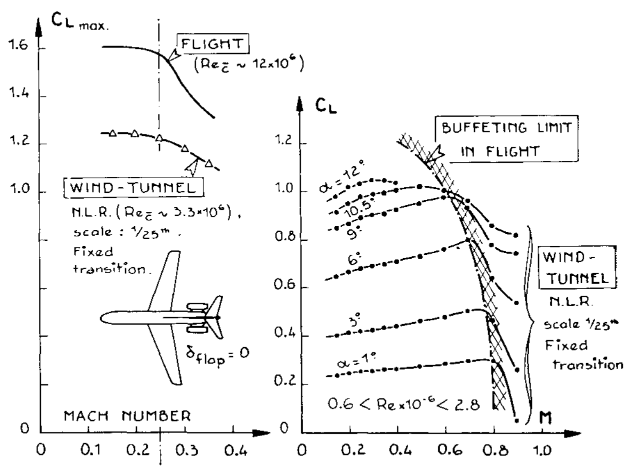


Fig. 31 Mach number effect on the $C_{L\,\max}$ and the buffeting onset on the "Caravelle."

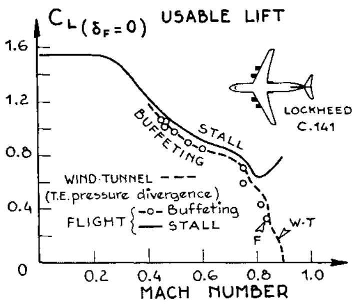


Fig. 32 Buffeting limit vs Mach number predicted by trailing edge pressure divergence on a model of the Lockheed C-141.

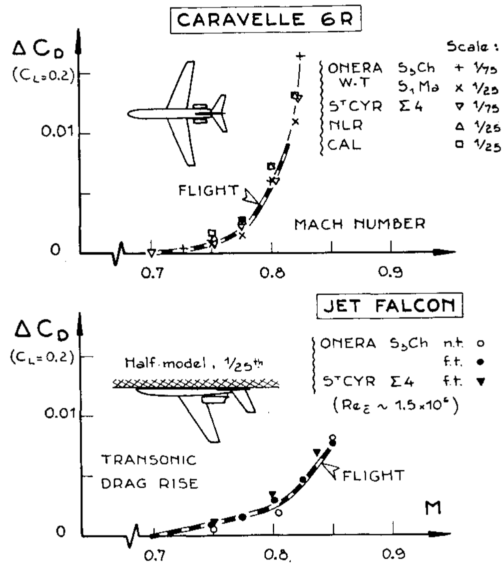


Fig. 34 Wind-tunnel/flight comparisons of the transonic drag rise for "Caravelle" and "Jet-Falcon."

transport wing during climb-out. That is why the $C_{L\max}$ performances must be studied including the Mach number effect if we want to predict the flight envelope.

In the case of "Caravelle," Fig. 31a, the boundary-layer separation behind the shock near the wing leading edge gives a loss of the maximum lift as soon as the flight Mach number reaches Mach = 0.25, as predicted in wind tunnel. These curves remind us how large the Reynolds number effect is on the $C_{L\max}$ for a small model.

A very important flight limitation is the buffeting onset at high speed, due to unsteady boundary-layer separation behind shock waves. For high aspect-ratio airplanes, the buffeting detection in wind tunnel is quite easy. A very good prediction of the buffeting envelope vs Mach number is given by the locus of the transonic lift loss for given incidences, as shown in Fig. 31b, in the case of "Caravelle." The buffeting onset is also well predicted in wind tunnel by the trailing edge pressure divergence at the upper surface,[62] and Fig. 32 shows that this limit is very near the stall for the swept wing of the C-141. This pressure analysis is often used for swept wing fighter configurations, as well as measurements of unsteady bending moments at the wing root, or the analysis of some discontinuities of longitudinal stability curves (C_L, C_m, without tail).

Drag Rise Mach Number

A correct determination of the "limit" Mach number, corresponding to the drag divergence, has been always one of the main goals of the transonic wind-tunnel tests. The first flight correlations were given by NACA, thanks to the famous family of experimental airplanes (X-1, D-558-II, X-2, X-3, X-5, etc.,[63,64] for example). The next step was to improve the limit Mach number, and the wind tunnel was an essential research tool to develop many new aerodynamic concepts, such as the area rule[65] and the conical camber,[66] later applied on many production airplanes.

A precise drag divergence Mach number is fundamental when designing a modern jet transport, if we want to predict its maximum economical cruising speed and its payload range capability.[47] Here, wind-tunnel and flight tests often give different (C_D, M) curve shapes. These discrepancies are mainly due to: 1) a poor transonic tunnel test section (not checked by a well known calibration model); 2) too large a model; 3) parasite laminar separations at low Reynolds numbers, or ineffective transition strips; and 4) interferences due to model support.

The latter case is illustrated by the Fig. 33, where three types of model support are seen to give different drag rise shapes. The most often used technique is the sting balance because of its ability to measure the six components on the model, but, when the rear fuselage has no jet nozzle, as on most transports, the conical rear, modified into a boat tail shape, and the sting itself, can strongly change the real drag and the moments. In every case, these parasite interferences must be carefully analyzed on each model to obtain the "interference free" results at transonic as well as at supersonic speeds.

However, it is often possible to predict the drag rise Mach number with a good accuracy, as shown in Fig. 34 where we

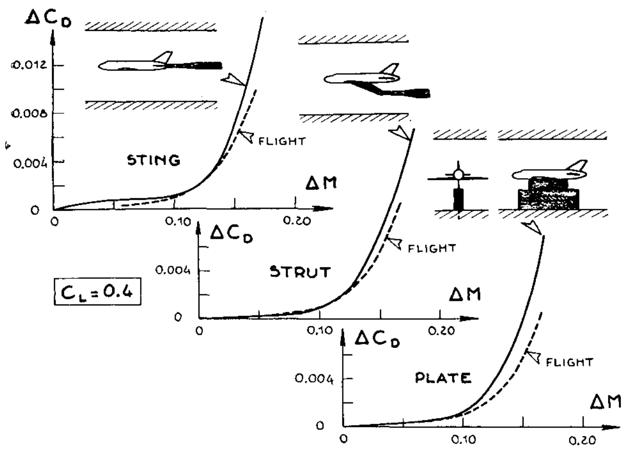


Fig. 33 Influence of the wind-tunnel model support systems on the drag rise (Boeing Co.).

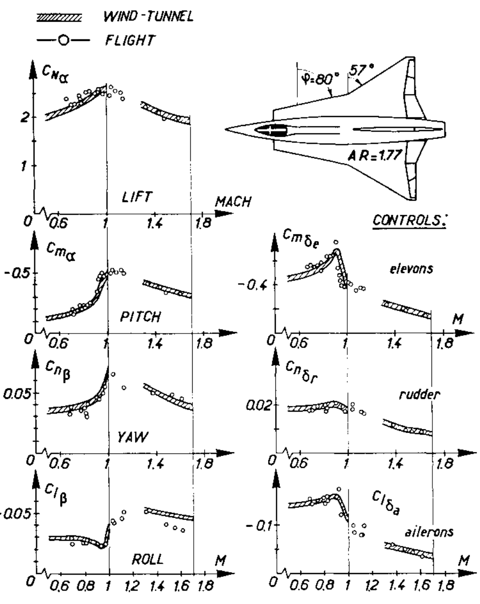


Fig. 35 Aerodynamic derivatives of the SAAB "Draken" between M = 0.6 and M = 1.7 from FFA wind tunnels and SAAB flight measurements.

compare, in the case of the Sud "Caravelle" and of the Dassault "Jet-Falcon," the points given by several facilities with the flight test curves. Nevertheless, it is always safer to check the basic configuration in several wind tunnels having different types of transonic walls and different test section sizes

4. Supersonic Regime

During the last few years, many aerodynamic investigations were carried out in support of several SST developments. The supersonic transport must be highly optimized to become successful, and, therefore, the wind-tunnel measurements must be much more precise than, for example, were required in the past for military aircraft.

Stability Derivatives

Wind tunnels usually give static stability and control derivatives in good agreement with flight results. An example of good correlation is given in Fig. 35 for the Swedish supersonic fighter "Draken," tested in the FFA tunnels and then by the SAAB in flight. An equally good agreement was obtained by NASA on the X-15 in a very large domain of supersonic Mach numbers, Fig. 44.

In the case of the huge B-70 supersonic bomber,[9,69] it is interesting to compare the wind-tunnel predictions with flight results on the role of the folding wing tips deflected at $-25°$ to cross $M = 1$, and then at $-65°$ for the supersonic acceleration ($M = 1.3$ to 3). Figure 36a shows that the longitudinal static margin reductions for increasing tip deflection were well predicted, but the actual airplane was always less stable. Thus, the elevon deflections for trim were more positive in flight. Finally, the longitudinal maneuverability (elevon deflection per "g") was about the same as predicted (Fig. 36b), because of lower $Cm_{\delta e}$ values in flight. Because of the wing tip deflections, the yawing stability remained about the same between $M = 0.6$ and $M = 2.4$ (Fig. 36c), and the dihedral effect ($C_{l\beta}$) continued to decrease until it was negligible at supersonic cruise, Fig. 36d, as predicted by the tunnel tests. On the contrary, the adverse yawing moment due to aileron deflection reported in flight was not predicted in the laboratory.

For the unsteady derivatives, we have fewer comparisons, and that is why we have started a joint program (ONERA and the French Flight Center) on a "Mirage 3B" especially instrumented for variable stability purposes.[70]

a) Static margin

b) Longitudinal maneuverability

c) Yawing stability

d) Induced roll

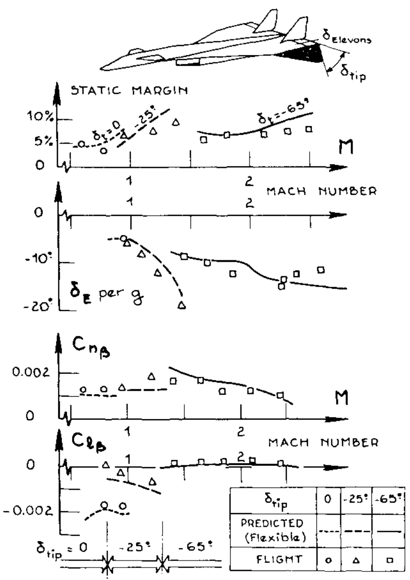


Fig. 36 Effect of wing tip folding on the supersonic bomber B-70; comparisons between wind-tunnel predictions taking account of the aeroelasticity and flight results.

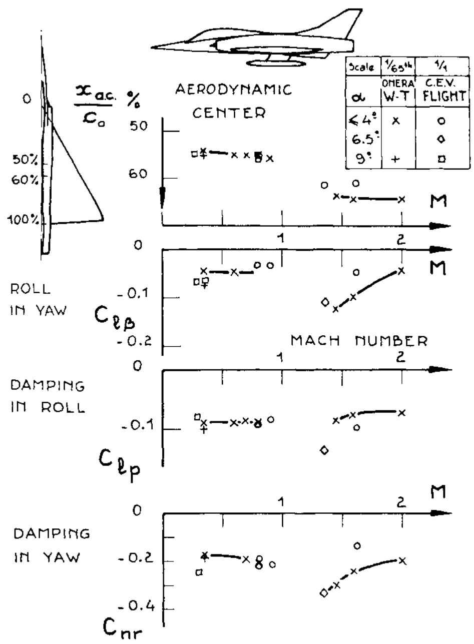


Fig. 37 Dynamic tests on a "Mirage 3B;" wind-tunnel/flight correlation at subsonic and supersonic speeds.

So far, we have only a few results from wind tunnels on a small model tested with a forced oscillation method,[71] and from some flights on the conventional configuration of "Mirage 3B."[4] Some examples of correlation given in Fig. 37 show that it is difficult to obtain the unsteady derivatives.[72]

Drag Prediction

As regards the wind-tunnel drag prediction, the problem is even more difficult than in subsonic flow, because we have to isolate yet another term, the wave drag. Thus, from the measured minimum drag, we have to subtract a mean friction drag, computed from the local Reynolds numbers of each part of the model and taking account of the laminar regions (boundary-layer visualization), and of some parasite roughness drag (systematic tests with various grit sizes).[42-45]

Precise supersonic drag estimates are obviously vital in a case such as "Concorde." That is why Sud-BAC have tried very hard to cross check the measurements on a reference

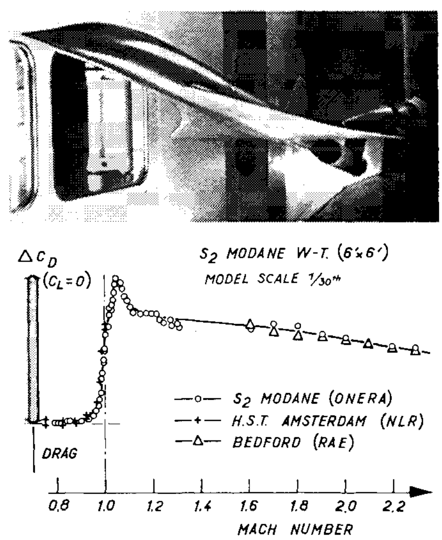


Fig. 38 Comparison of wind-tunnel measurements of the minimum drag vs Mach number of the first "Concorde" project model (1964).

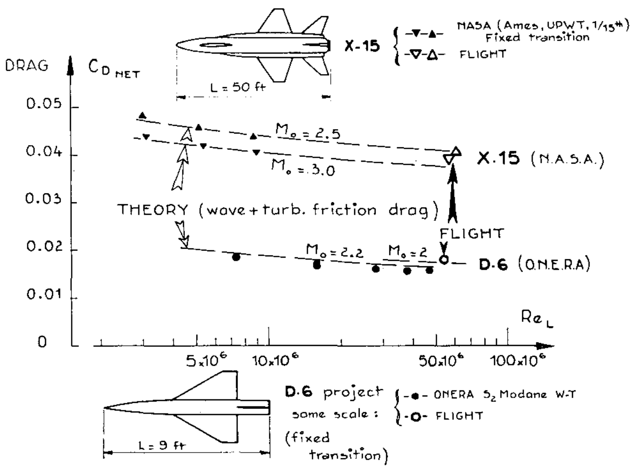


Fig. 39 Correlation of theoretical estimates and wind-tunnel and flight measurements of the minimum drag of the X-15 research airplane and the D-6 missile.

model with about the same transition strips, obtained in various facilities, Fig. 38.

When the configurations are not too complex, it is possible to calculate quite accurately both the wave and friction drags to compare with the minimum drag (base drag deduced) measured in wind tunnel and in flight. In Fig. 39, such good correlations for the X-15 tested by NASA,[74] and for an ONERA experimental missile, are given vs the Reynolds number. It is interesting to note that the D-6 missile itself was tested in the large variable density tunnel of Modane before being launched at Mach 2 with a two stage booster.[4,77]

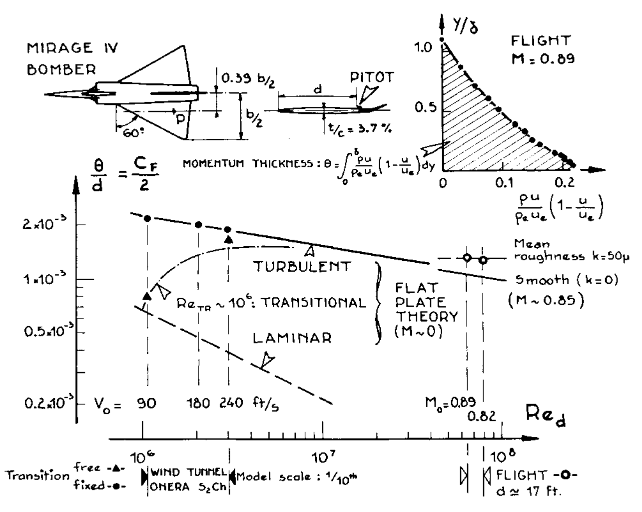
For these two configurations, the flight measurements were made after rocket burn-out, i.e., in a much more favorable case than for an airplane whose turbojet is still operating (net thrust taking account of the inlet loss and of the after-burner characteristics). In these tests, however, the drag evaluation necessitated a careful measurement of all the base drags, which are very large at low supersonic speeds, particularly on the X-15. It is important to note that the base drag predictions from wind-tunnel tests of the X-15 were systematically too low by about 10%, mainly because of sting interference.[74]

Returning to the friction drag prediction, this term, in the case of the SST, accounts for almost one-third of the cruise drag[78] and is therefore extremely important. Here, the main problem is the validity of a friction drag calculation based on some conventional turbulent flat plate approximation, at Reynolds and Mach numbers much higher than flown in the past.

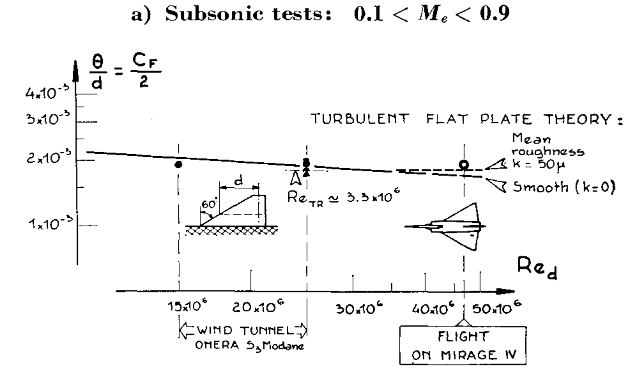
Much basic experimental research was recently carried out on flat plates in large American and British supersonic wind tunnels to obtain friction coefficients at very high Reynolds numbers.[79] The results are within the boundaries given by the various approximate turbulent theories, but the question remains whether these calculations are valid for a slender wing.

In an attempt to find an answer, our colleagues of NASA have prepared a flight program of boundary-layer measurements at various locations on the upper surface of the experimental North American B-70 bomber. In France, we at ONERA, in cooperation with the French Air Ministry and the Dassault Company,[4,80] have tried similar tests on a "Mirage 4" bomber. The friction drag coefficients in flight were deduced from the boundary-layer measurements made with a rake (26 pitot tubes, 1 static, 5 stagnation temperature probes, 1 wall temperature) mounted on the wing upper surface 17 ft behind the leading edge. The subsonic results obtained in flight at Mach number of about 0.85 ($Re_d \cong 70 \times 10^6$) are compared with a calculated turbulent friction coefficient (adiabatic flat plate[81]) in Fig. 40a. The agreement is excellent if we assume an equivalent sand roughness[50] on the wing surface ($k \cong \frac{2}{1000}$-in.). We have plotted on the same figure some boundary-layer measurements made by a pitot traverse, at a similar location on the wing of a $\frac{1}{10}$ scale model tested at low speed. With a natural transition, the values agree with a transitional boundary-layer calculation, whereas a fixed transition, by means of roughness along the leading edge, gives experimental results in accordance with the turbulent calculation on a smooth plate.

A supersonic flight with a local Mach number $M = 2.15$ at the rake location ($Re_d \cong 47 \times 10^6$) has given a friction coeffi-



a) Subsonic tests: $0.1 < M_e < 0.9$



b) Supersonic speeds and comparisons with theoretical predictions ($M_e \sim 2.15$)

Fig. 40 Mean friction drag coefficients obtained by boundary-layer measurements on the delta wing of the "Mirage 4" bomber in flight and on wind-tunnel models.

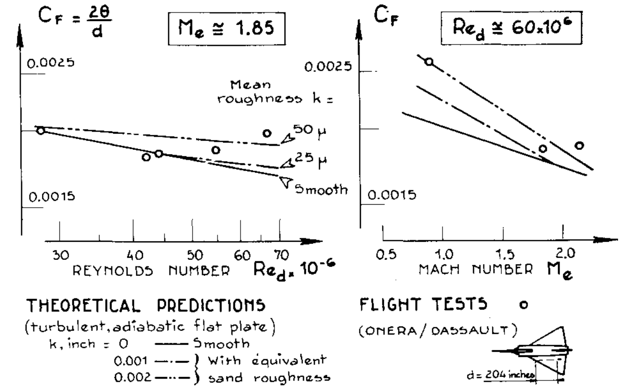


Fig. 41 Reynolds and Mach number effects on the mean friction drag coefficient measured on the "Mirage 4" wing in flight and predicted by usual rough flat plate calculations.

cient larger than the "smooth" calculation, Fig. 40b, but here again the hypothesis of a mean equivalent roughness gives a better prediction. The supersonic wind-tunnel tests, made on a half-wing of the D-6 project (very similar to that of "Mirage 4"), are in good agreement with the theoretical prediction on a smooth turbulent flat plate.

Finally, it was possible to study the Reynolds number effect during flights at various altitudes at about the same Mach number ($M \cong 1.85$, Fig. 41a) and the Mach number effect at about constant Reynolds number ($Re_d \cong 60 \times 10^6$, Fig. 41b).

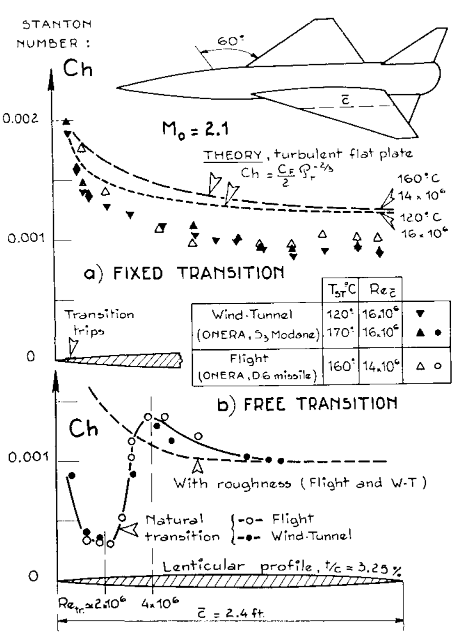
Concluding the review of this very preliminary research, it seems that for a thin and slender wing at low incidence, a conventional compressible turbulent flat plate calculation is quite adequate for the friction drag calculations if we assume that the wing skin presents a small equivalent roughness at large Reynolds numbers; thus, we must have a very smooth skin surface on the SST to avoid a large parasite drag ($Re_{\bar{c}} \cong 130 \times 10^6$ for "Concorde" in supersonic cruise).

Kinetic Heating

Friction means heating, and this problem becomes very important with the long duration of an SST flight, counted in hours instead of minutes for a fighter. Thus, it is mandatory to know accurately the heat-transfer coefficients to be used by the designer to calculate the heating history of the airframe during supersonic flight. As for the friction drag, we have tried, at ONERA, to check, by comparison with flight, the validity of the results on heat transfer obtained both in the supersonic heated flow of a blow-down wind tunnel (S3 Modane) and by the usual Reynolds analogy calculation.

The D-6 program[4,77] was initiated to compare directly the Stanton numbers measured on similar delta wings, equipped with the same instrumentation (40 thermocouples on the stainless steel skin of calibrated thickness), at about the same stagnation temperature, and for about the same Mach and Reynolds numbers ($M \cong 2$, $Re_{\bar{c}} \cong 15 \times 10^6$), on a half-wing mounted at the tunnel wall and on the third stage of an experimental missile flying at constant altitude ($Z \cong 30{,}000$ ft).

Figure 42a shows that wind-tunnel and flight experiments are in good agreement for the Stanton number along the mean



	T_{ST} °C	$Re_{\bar{c}}$	
Wind-Tunnel (ONERA, S3 Modane)	120°	16.10⁶	▼
	170°	16.10⁶	▲ ●
Flight (ONERA, D6 missile)	160°	14.10⁶	△ ○

Fig. 42 Kinetic heating measurements on the delta wing of the ONERA project D-6 at $M \approx 2$: a) Heat-transfer coefficient in turbulent flow predicted by conventional calculations and measured in flight and in wind tunnel on the same model; b) Boundary-layer transition on the smooth wing as detected by the sharp increase of heat transfer.

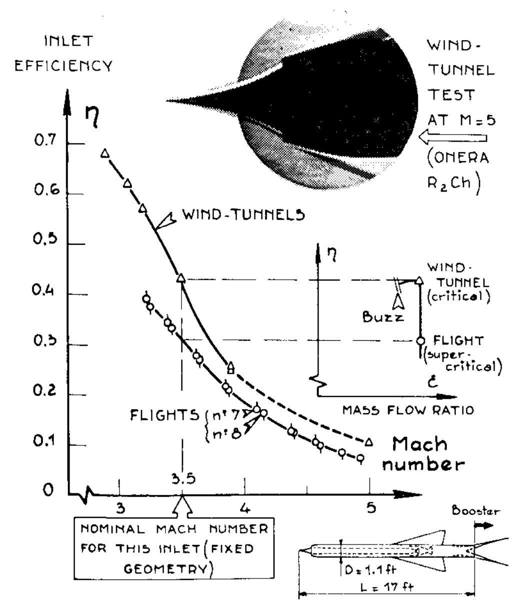


Fig. 43 Flight tests on "Stataltex" ramjet (ONERA 1962) up to $M \approx 5$: inlet pressure recovery in flight and in wind-tunnel tests.

chord of the delta wing. The values measured are lower, mainly near the trailing edge, than those predicted by the conventional Reynolds analogy connecting turbulent friction and heating. This pessimistic prediction of the theory was also observed on the X-15.[73]

The heat-transfer measurements, in the case of a free transition, give a very precise location of the transitional regime in the boundary layer, the Stanton number at the beginning of the free turbulent flow ($Re_{tr} \cong 4 \times 10^6$) being larger than in the case of a fixed transition at the leading edge. Here again, excellent agreement was obtained between flight and wind tunnel (Fig. 42b).

A good prediction of the total drag and of kinetic heating is important for an SST project, but the aerodynamicist also has a large responsibility with respect to propulsion efficiency, since, typically, the air intakes contribute 60% and the exhaust nozzle 30% of the nacelle net thrust, as in the case of "Concorde."[82] These problems have required the development of new sophisticated mounting systems in many wind tunnels to optimize separately the inlet and exhaust on models with variable geometries, for the supersonic cruise and then for the very important subsonic requirements.

Ramjet Propulsion

The best propulsion system to fly from Mach 3 up to hypersonic speeds seems to be the ramjet and much research is devoted to each component of this configuration (inlet, combustion chamber, nozzle); however, the complete test in a wind tunnel of a ramjet with a full simulation of flight conditions, i.e., at the real air stagnation temperature and pressure, is increasingly difficult as the Mach number rises and that is why the few ramjet tests made in the past were performed with free flight models. Five years ago, ONERA had undertaken such a flight program with the "Stataltex," a ramjet mounted on top of a one stage rocket,[83] to study the technological problems of a ramjet (subsonic combustion, behaviour of the hot structures, etc. . .) during its acceleration period from Mach 3.5 to 5 at various altitudes, between 40,000 and 100,000 ft.

To simplify the mechanical problems on these small experimental models, an entirely fixed geometry was chosen, even for the external compression inlet (designed here for Mach 3.5), and the fuel-air ratio was programmed constant during the accleration period; the inlet was operated at supercritical con-

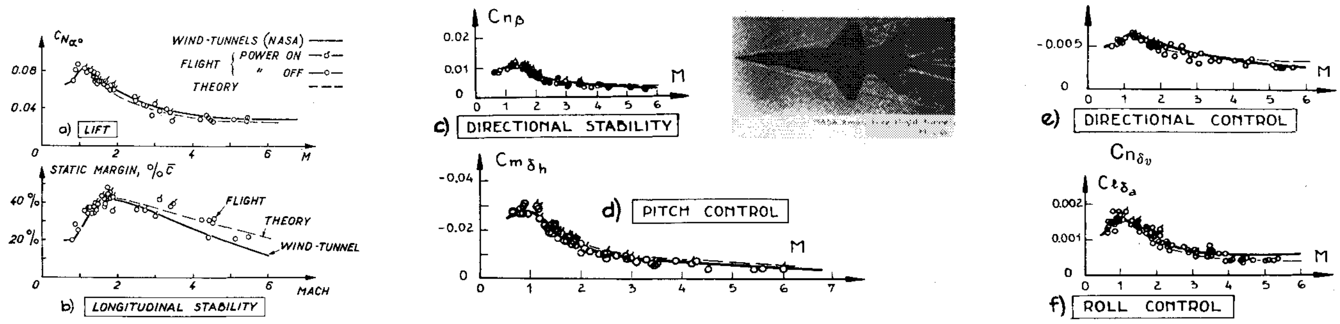


Fig. 44 Correlations of stability and control derivatives of the experimental rocket airplane X-15 between $M \approx 0.8$ and 6.

dition during the entire flight, to insure freedom from intake buzz; that is why the inlet efficiencies given in Fig. 43 from two flight tests were lower than those predicted using wind-tunnel data at critical regimes for each Mach number. Nevertheless, both tunnel and flight results show the same large loss in efficiency for the inlet operating between Mach 3 and 5.

X-15 Correlations

The best way to introduce the hypersonic regime is to mention the remarkable results obtained by NASA with the experimental rocket airplane X-15.[73–76] On the whole, wind-tunnel results, flight measurements, and theoretical predictions are in very good agreement between Mach 0.8 and 6.

Figures 44a to 44e show this unique correlation for stability and control derivatives.

Figure 45 gives a typical Reynolds number variation and the maximum temperature recorded on the wing leading edge, for flights up to $M = 6$, and the corresponding minimum drag. For this very truncated configuration, the total base drag represents more than 50% of the total minimum drag at Mach numbers lower than two, but decreases sharply with increasing Mach number.[74] Wind-tunnel measurements on the various base surfaces of X-15 models always gave a base drag lower than in flight, mainly due to support interaction. That is why the power-off L/D, predicted from wind-tunnel tests, was optimistic at low Mach numbers. Finally, the net minimum drag variation with the Mach number was well estimated by theoretical calculations, summing up the friction drag (with Reynolds and Mach number effects) and the wave drag (predicted by both supersonic and hypersonic theories).

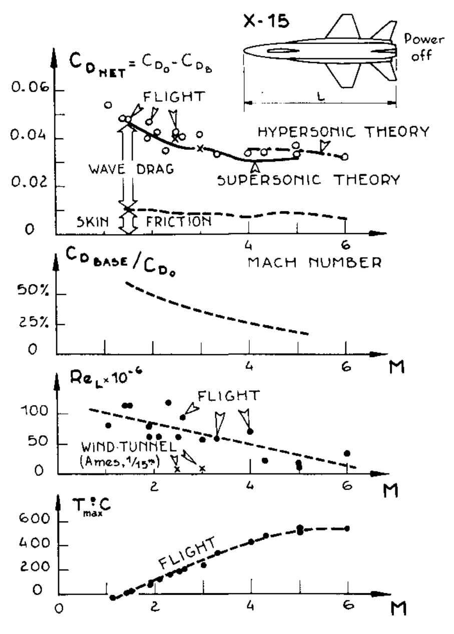
Figure 46a shows a good agreement for the pressure coefficients near the mid-semi-span, as measured in wind tunnel and in flight at $M = 4.7$ and $\alpha = 10°$. In these conditions, the fuselage bow shock crosses the wing near the mid-span lower surface and this effect was not included in the theoretical calculation given here.

Figure 46b gives the heating history on two points of the X-15 during a typical flight up to $M = 5.5$. Both theoretical calculations and wind-tunnel tests have overpredicted the kinetic heating and that is why, until now, NASA computes the heat rates of each new flight by a modified theory taking account of previous experiments.

To conclude, the X-15 program, to be continued in future years with a ramjet up to Mach 8, is most valuable for the theoreticians, the experimental aerodynamicists, and the designers who are in charge of hypersonic airplane development of the next decade.

5. Hypersonic Regime

Our last domain is the largest in Mach number range, from about five to values corresponding to re-entries from near or deep space, i.e., Mach numbers of 28 to 40. The very high enthalpies encountered at these flight speeds in our atmosphere produce the so-called "real gas effects" around the vehicle, with some dissociation, ionization and radiation of oxygen and nitrogen components, That is why new parameters, including time, have to be introduced[84] to study these phenomena, both theoretically and experimentally in new ground facilities, such as shock tubes, plasma jets, ballistic ranges, counterflow firing tunnels, etc.[85]

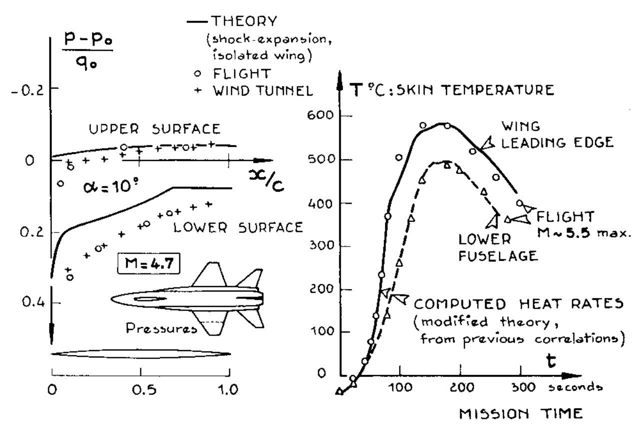


c) Minimum net drag compared with theoretical calculations

b) Base to total drag ratio

a) Typical conditions during power-off deceleration from $M \approx$ to 6 $M \approx 1$

Fig. 45 Supersonic flight of the X-15.



a) Pressure coefficient on the wing mid-span

b) Heating history during an accelerated flight from the B-52 launch up to $M \approx 5.5$

Fig. 46 Typical X-15 measurements at high supersonic speeds.

However, the real gas effects, although fundamental in heating, radio transmission, or detection problems, are not necessarily essential in the aerodynamic studies of a vehicle. Here we shall discuss the hypersonic regime under conditions in which the Mach, Reynolds number simulation is still adequate.

Figure 47 presents the flight trajectories of typical hypersonic vehicles in the earth's atmosphere, in terms of Mach and Reynolds numbers, superimposed on the maximum performance reached by the major present day facilities. Although, at a first glance, it would seem that a large part of these flight trajectories can be conveniently simulated in various large size facilities, several difficulties develop. For example, the highest Reynolds numbers are achieved in ballistic ranges, where the instrumentation is very limited and models may experience ablation. The helium tunnels can reach very high Mach numbers without heating problems, but they are used mainly for basic research where the nature and the temperature of the gas are not essential. On the other hand, the shock-tunnel tests are primarily limited by the very short run times available. Actually, the conventional hypersonic tunnels and the hotshot tunnels are the ones mostly used for the aerodynamic tests of complete models with enough run time to obtain a complete polar curve (in the ONERA hotshot tunnel, data are taken at 25 incidences during 25 msec, thanks to a special model mount); however, there are two limitations of the Reynolds number simulation at the extremes of the Reynolds number range, because it is difficult: 1) to duplicate the very high values encountered during the last part of the trajectory, and 2) to reproduce the very low Reynolds numbers corresponding to the beginning of a re-entry.

In this latter case, special low density wind tunnels must be used for a proper simulation of the viscous interaction parameter,[86] whose effect is to increase the drag and to decrease the lift. In the case of a lifting body re-entry, this causes a reduction of the L/D and of the range from re-entry to landing. Such large effects of low density are evidenced, for example, by the maximum L/D values obtained on a blunted 9° cone[87] tested in several low pressure facilities with high values of the viscous interaction parameter. Figure 48 shows that the loss of $(L/D)_{max}$ is large for altitudes above 200,000 ft. The same effects are apparent from tests of the HL-10 lifting body models[88] and in the analysis of the beginning of the "Gemini" re-entry.[89,90]

Conversely, tests carried out in high Mach-moderate Reynolds number facilities are to be interpreted to give a correct flight prediction at higher Reynolds numbers. For the sake of calibration of hypersonic facilities, a standard model was adopted by AGARD; this typical re-entry body, called HB-2, proved to be useful for comparisons of force and pressure measurements in a very large Mach-Reynolds number domain. Figure 49 compares minimum drag measured in several facilities. The C_D,M curves show that the base drag becomes negligible in hypersonic flow; the discrepancy above $M = 14$, due to a strong viscous interaction, is explained when the drag is plotted vs this parameter, as shown in the lower figure (the AEDC values,[91] and the ONERA results at $M = 16.5$ were obtained in hotshot tunnels).

Unfortunately the HB-2 flight test results are not yet available, but we have obtained in France some interesting wind-tunnel/flight comparisons on the ballistic missile "Thesa," developed by Sud-Aviation and launched by SEREB.[92] This ballistic shape was tested in many French facilities, including a range, from Mach 0.8 to 17, and we have also the flight results obtained by telemetry during four re-entries, among them three at about Mach 14. Figure 50 shows very good correlations of the total drag, the lift gradient, and the aerodynamic

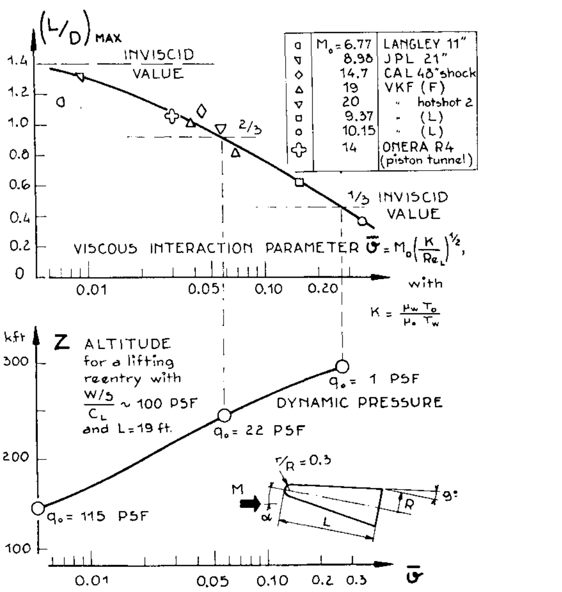


Fig. 48 Loss of lift/drag ratio of a 9° blunt cone due to the viscous interaction effects at high altitude and large Mach numbers.

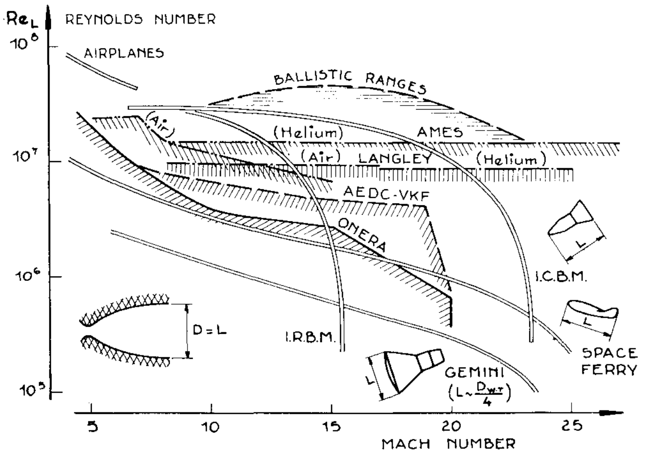


Fig. 47 Typical trajectories of hypersonic vehicles and performance envelopes of the major ground test facilities in terms of Reynolds and Mach numbers.

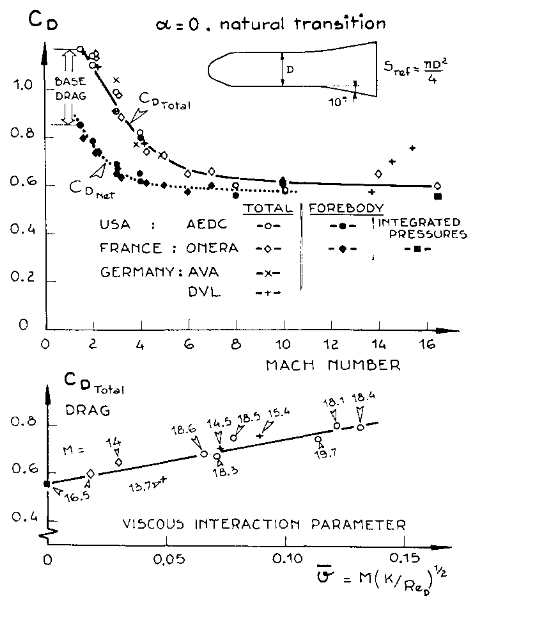
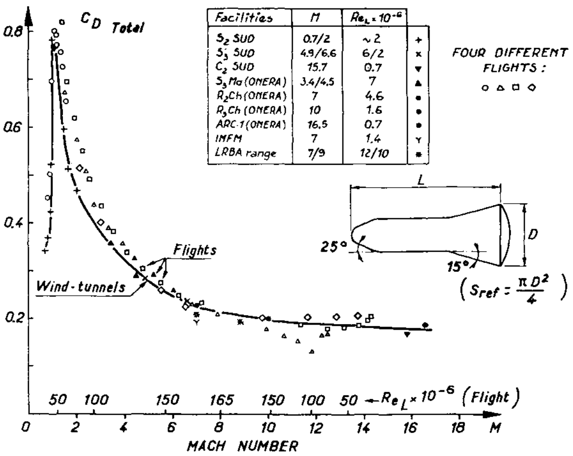
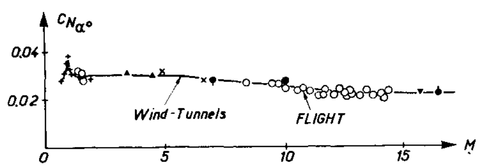


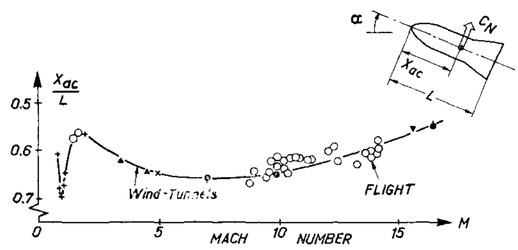
Fig. 49 AGARD HB-2 calibration model for hypersonic tunnels; minimum drag vs Mach number and viscous interaction parameter.



a) Total drag



b) Lift gradient



c) Aerodynamic center

Fig. 50 Wind-tunnel/flight comparisons on the French ballistic missile "Thesa."

center; notice that normal-force gradients for Mach numbers below about 8 cannot be easily derived when the body oscillations are damped out, whereas above Mach 14, initial oscillations are too large for a precise computation. Nevertheless, there are enough values to confirm the large variations of the aerodynamic center location with decreasing Mach number, which had been predicted by wind-tunnel tests.

As regards the tunnel determination of the aerodynamic center of these flare stabilized bodies, it is important to recall how difficult it is to obtain an attached flow on the flare (a laminar separation results in large errors in the stability derivatives) due to the relatively low Reynolds number achieved in most of the present hypersonic facilities. This situation can hardly be improved, because it is increasingly more difficult to trigger transition as the Mach number increases[93,94]; beyond about Mach 7, large roughness sizes must be used on the model nose, with the danger of deeply disturbing the boundary-layer profiles and even the external flow.[95] So far, no adequate solution has been proposed, and it is still highly desirable to carry out tests at Reynolds numbers sufficiently high to insure a natural transition upstream of a flare (or a control surface) to duplicate the usual flight cases. Typically, the only way to obtain pressure and heat-transfer measurements with a naturally turbulent boundary layer was to test an experimental ballistic shape in flight. For this purpose the "Berenice" program was undertaken some years ago by ONERA and Sud-Aviation with a four stage rocket missile, the last two stages being fired during the re-entry to attain about Mach 12.[4,96] The telemetered flight measurements were then compared with the predictions obtained at Mach 10 in the conventional hypersonic blowdown tunnel R_3 of Chalais. Figure 51 shows a pressure correlation along the ballistic shape at zero angle of attack; attached turbulent flow in flight and separated laminar flow in wind tunnel give a very different recompression on the flare, but there is quite good agreement between the theory and the experimental results, except on the flare where the boundary-layer effects are dominant.

Moreover, the heat-transfer coefficients, Fig. 52, obtained on "Berenice" in flight are very interesting because they show that, at about $M = 10$, the boundary-layer transition moves upstream and reaches the nose at re-entry, when the Reynolds number based on body length reaches about 20×10^6. We thus have obtained, for a point on the nose cone, a complete history through the laminar, transitional, and turbulent regimes, well confirmed by the theoretical predictions. The heat-transfer measurements in wind tunnel were made at a Reynolds number of only about 3×10^6, i.e., with a fully laminar flow on the entire shape. Two methods were used for these tests: Thermocouples welded on the very well calibrated skin, and thermosensitive coating (four color changes[95]) on two models of different sizes, molded with a highly insulating plastic material (silastene). Results from both methods are in good agreement with those from flight and with the laminar theoretical prediction. On the subject of missiles and rockets, the aerodynamicist has had, during the last ten years,

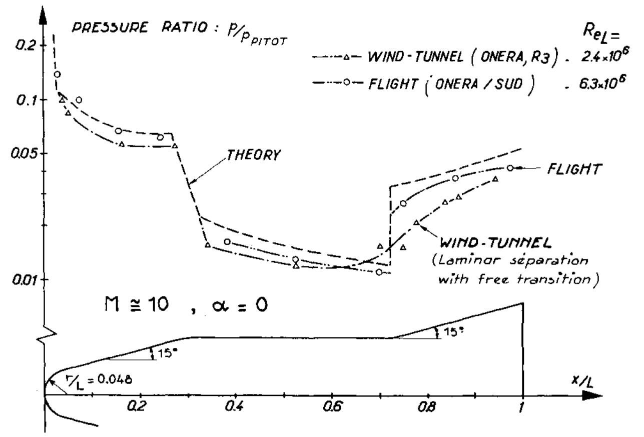


Fig. 51 Pressures on the ballistic head "Berenice" at $M \approx 10$; comparisons between wind-tunnel, flight, and theoretical prediction.

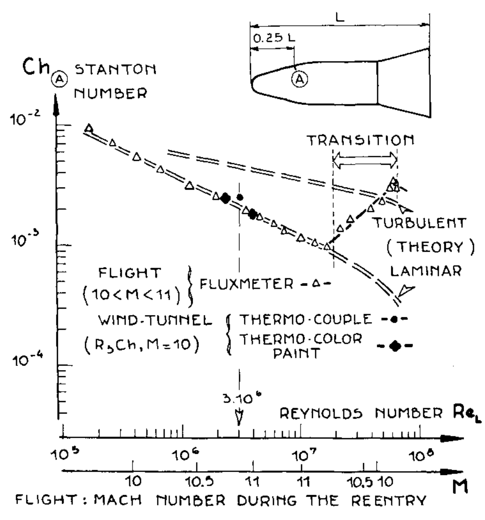


Fig. 52 Heat-transfer coefficients measured in flight on the "Berenice" nose cone vs Reynolds number during re-entry; comparison with wind-tunnel tests and calculations.

to solve numerous unexpected and difficult problems, often after the occurrence of spectacular and expensive failures of configurations in flight.

Although it is difficult to obtain data for *quantitative* comparisons between laboratory and flight, in this domain, we can say however that the wind-tunnel predictions were adequate if there was *no more failure* in the next flight. In general, their solution requires special and unusual experiments in the wind tunnel, of which some examples will be cited:

1) In the case of multi-stage launch vehicles, three aerodynamic problems may occur in succession: a) at subsonic or transonic speeds, a very hot recirculation flow, induced by the nozzle jet and impinging on the base of the stage, can cause the destruction of control elements; b) at the moment of separation of a spent stage and ignition of the next stage, the jet can trigger a serious boundary-layer separation, causing severe aerodynamic instability, or even the destruction of the vehicle; c) at higher altitude, jet pluming can also lead to the flow separation on the stabilizing flare, and thus to a loss of stability. It has been possible to adequately simulate these conditions in wind tunnel, using compressed air jets on the models, and to find solutions.

2) In the case of the initial motion of a body ejected in the near-field of a vehicle, a free flight test on a dynamic model in the wind tunnel is often the best technique to study its trajectory (analysed with the help of high speed camera). This technique is used, for example, to study the jettisoning of aerodynamic shrouds,[4,97] or of auxiliary fuel tanks and bombs, and also the firing of air-to-air rockets from an airplane; here, the common problem is to predict and then to avoid catastrophic collisions in flight.

It seems interesting to conclude these tunnel/flight comparisons with some aerodynamic results obtained on the "Gemini" capsule. Models of various sizes were tested in NASA facilities at Ames (counterflow supersonic tunnel) and at Langley (conventional hypersonic tunnel), and in those of Arnold Engineering Development Center-von Karman gas dynamic facility (AEDC-VKF) (Hotshot F). The results obtained on L/D and on the trimmed angle of attack for a given center of gravity location are shown on Fig. 53 compared to flight data from Gemini GT-3. Based on the observations of Ref. 89, the flight data were plotted as a function of 1) normal shock Reynolds number (Re_{2d}) until the spacecraft had decelerated to a relatively low flight Mach number-high Reynolds number condition and 2) Mach number in the lower Mach number-high Reynolds number flight regime. A close inspection of Fig. 53 shows that viscous effects change the trim angle of the spacecraft by 4° between an entry altitude of 265,000

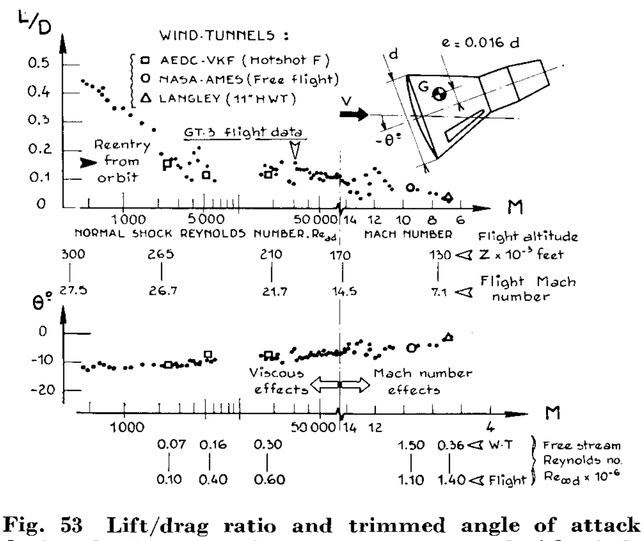


Fig. 53 **Lift/drag ratio and trimmed angle of attack during the "Gemini" GT-3 re-entry compared with wind-tunnel predictions.**

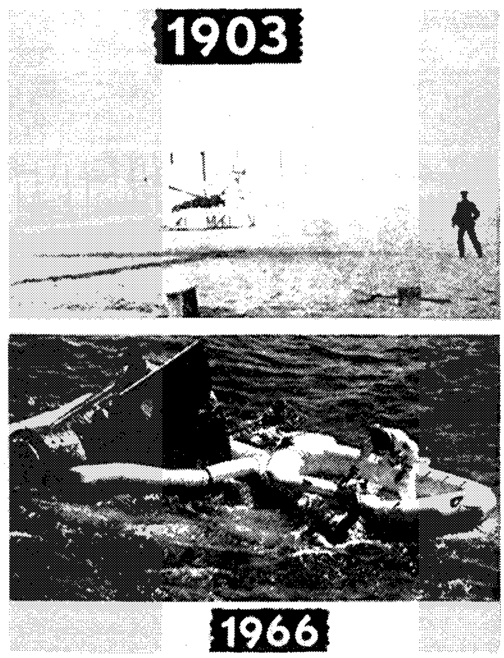


Fig. 54 The first and the latest contributions of the United States to the aerospace development.

and 160,000 ft. Excellent agreement between the flight and tunnel data is noted.

Thus, at least for the Gemini-type shape and for the force data, Mach and Reynolds number, when correctly interpreted, are adequate similarity parameters even at orbital velocities. The same has been shown for slender conical shapes at velocities up to 18,000 fps (comparison of tunnel and free flight range drag measurements[99]).

6. Conclusions

In this lecture I have tried to show that it is often possible to predict with sufficient accuracy the aerodynamic characteristics of an airplane or a missile from laboratory results. We have also seen that the disagreements between wind tunnel and flight are often due to an excessive difference in the Reynolds numbers and that this is particularly serious with respect to maximum lift, minimum drag, and heating rates; however, some discrepancies can only be explained by errors in measurements, or by ineffective testing methods used both in wind tunnel and in flight. The stability and control derivatives are generally well predicted from tunnel results, but more accurately in steady than in unsteady regimes.

As regards predictions of the flying qualities from tunnel tests, previous experience with similar configurations is required; ultimately, it is obvious that familiarization of the pilot with the dynamic behavior of a new airplane (detection of shortcoming in its flying qualities, optimum setting of the artificial dampers, etc.) requires a ground practice on a 6 degree-of-freedom ground simulator.[100]

The difficulties experienced in collecting the few comparisons presented in this lecture show that such analyses are very rarely published, and perhaps seldom performed. The responsibility for this situation seems to be split among 1) the government technical agencies, which should request, in the prototype contract, that a comparison between the laboratory predictions and the flight results be furnished; 2) the manufacturers, who often no longer show any interest for such comparisons when the airplane, or missile, is accepted and when they start working on the next project; 3) the laboratories, which are often more interested in accelerating the rate of testing than in analyzing the meaning of the results; and 4) the flight test centers, which have too little contact with the aerodynamic laboratories before the first flight test, as well as during the development flights.

Consequently, it seems that, starting with the preliminary design stage, more liaison is highly desirable between all technical groups, so that the laboratory engineer, the designer, and the flight engineer speak the same language. Working together, they have to detect, and cure, all the aerodynamic troubles they may expect from a careful analysis of the tunnel tests.

Another important duty for government agencies is to foresee the requirements for development test facilities, to have them operational when they are needed, for new projects[101]; we have seen that this has not always been the case, particularly for the specialized facilities required for studying V/STOL aircraft, transonic aircraft at large Reynolds number, and for the tests of complete hypersonic ramjets with full simulation of temperature, pressure and Mach number.

Finally, let us note that the ground test facilities not only contribute greatly to the design and development of airplanes, spacecraft, and missiles, but also are the birthplace of advanced research,[102] which has opened the way to the present aerospace developments. The United States contribution in this field has been considerable and is summed up, over the period of sixty-four years, in Fig. 54: Let us pay particular tribute to the Wright Brothers, and to the latest two astronauts of the Gemini program; although the vehicles used by these pioneers were not particularly "streamlined," they will nevertheless remain famous in the history of our time.

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