Journal of Aircraft

VOLUME 5 NOVEMBER-DECEMBER 1968 NUMBER 6

Design and Development of an Air Intake for a Supersonic Transport Aircraft

I. H. RETTIE*
British Aircraft Corporation (Operating) Ltd., Bristol, England
AND
W. G. E. LEWIS†
National Gas Turbine Establishment, Pyestock, England

The reasons for choice and the characteristics of an external-compression intake geometry for operation at Mach numbers up to and beyond 2.0 are described, and the problems of application to SST aircraft are discussed. Mounted under the wing of the Concorde, this inlet is divided to provide independent supplies of air to a pair of engines, an arrangement that introduces particular problems in allowing for the wing flowfield and avoiding interaction between the twin inlets. The precise definition of an intake geometry for a supersonic transport should have regard for the over-all performance of the propulsion system. The choice of controlling parameters and the design of the control system must give good performance and engine handling in a wide range of off-design conditions without demanding excessive complexity. The aerodynamic and other development tests required to make the appropriate decisions are described in detail. The results underline the suitability of this basic geometry in association with the other components of the propulsion system for SST operation.

Nomenclature

A_M = secondary-nozzle shroud entry area (Fig. 11)
A_J = primary-nozzle exit area (Fig. 11)
e = bypass sonic throat area/intake capture-area ratio
DC_{60} = $(\bar{P}_{\min} - \bar{P})/q$
$\bar{P}$ = mean total pressure at the compressor face
$\bar{P}_{\min}$ = lowest mean total pressure over any 60° segment of the compressor face
q = mean dynamic head at the compressor face
M = Mach number
N_L = engine L.P. compressor rpm
P_S = secondary flow total pressure
P_J = primary jet total pressure
T_1 = freestream total pressure
δ_2 = forward ramp angles (Fig. 3)
ϵ_1 = main duct mass flow ratio
ϵ_B = ramp bleed mass flow ratio
μ = secondary flow coefficient = $W_S T_S^{1/2}/W_J T_J^{1/2}$
W_S = secondary mass flow
W_J = primary jet mass flow
T_S = secondary air temperature
T_J = primary jet air temperature
η_1 = main duct recovery
η_B = ramp bleed pressure recovery

Presented as Paper 67-752 at the AIAA/RAeS/CASI 10th Anglo-American Aeronautical Conference, Los Angeles, Calif., October 18-20, 1967; submitted November 10, 1967. The authors thank the British Aircraft Corporation and the Ministry of Technology for permission to publish this paper. They are also indebted to D. P. Morriss of BAC and M. C. Neale of NGTE for valuable assistance during the preparation of the paper.

* Chief Aerodynamicist, Filton Division; now with the Boeing Company, Seattle, Wash.
† Head of Department, Engine Research Department.

I. Introduction

IN this paper we shall review the aerodynamic aspects of the design and development the engine air intake for the Anglo-French Concorde aircraft. The manufacture of the Concorde is being undertaken jointly by Sud Aviation at their factory in Toulouse, France, and by the British Aircraft Corporation's Filton Division at Bristol, England; division of work between the two firms gives the British team responsibility for the over-all design of the power plant. The joint organization of the project has, of course, enabled the French team to contribute wherever it was advantageous; both British and French wind tunnels have been used extensively.

Consider first the constraints imposed upon the intake design by the aircraft as a whole. The general arrangement drawing (Fig. 1) shows the mounting of the engines in twin-underwing nacelles. Each powerplant, including intake, engine, nozzle and the associated control gear, is independent in all important aspects. Aerodynamic independence is achieved by means of a projection forward of the center wall (splitter plate) separating the pair of inlets. Maximum intake efficiency is an economic requirement for SST aircraft; Table 1 shows the relationship between intake performance and payload.

A detailed discussion of the reasons for the under-wing semi-span position for the engines is outside the scope of this paper. Suffice it to say that a suitable layout for the aircraft has been

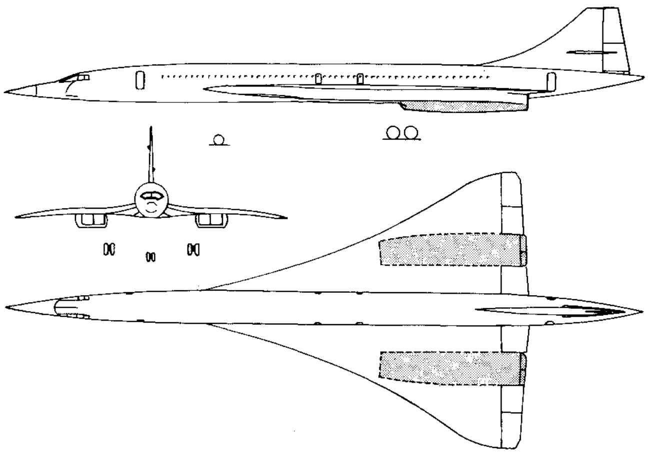

Fig. 1 Aircraft general arrangement.

achieved and that it satisfies the following requirements: 1) adequate length for the complete powerplant, 2) adequate separation of the intake and the wing leading edge to avoid interference between the intake flow and the vortex development on the upper surface of the wing, 3) feasible undercarriage geometry with adequate ground clearance for lift-off and touchdown design cases. We do not wish to protract here the arguments concerning the choice between two-dimensional and axisymmetric intakes. The European choice is obvious, and is justified at least in part by the way it has fitted neatly into the over-all aircraft design.

The third major consideration in selecting the intake design is that of flexibility. The intake must provide good engine face-pressure distributions and high efficiencies for all likely ambient temperatures to be met in cruise, preferably without demanding an extremely complex high-response control system. It must also provide adequate subsonic performance, especially since the fuel for diversion cruise has to be carried all the way across the Atlantic. The ways in which the intake design chosen for the Concorde meets these requirements are our main concern here.

II. Choice of Compression System and Intake Geometry

Confining attention now to the two-dimensional arrangement, both the external/internal and all external compression systems merit consideration for a Mach number of 2.2.[1] At first sight, Mach 2.2 appears a rather awkward speed; it is certainly too low to make a cast-iron case for partial internal compression, but considerable fears were entertained at the outset regarding the cowl drag of a conventional external-compression geometry. External compression was chosen because of the inherent stability of the associated shock pattern and the simplicity of the associated bleed arrangements and control systems.

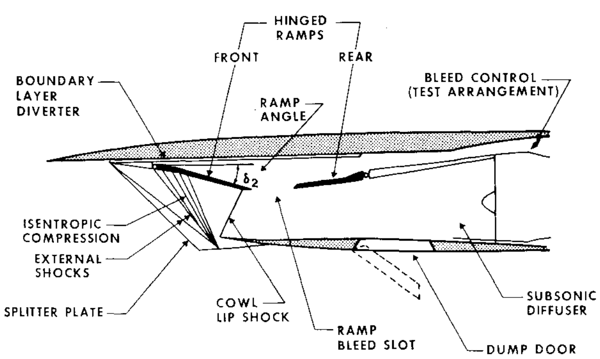

Fig. 2 Intake geometry.

Table 1 Effect of aerodynamic loss on payload

Aerodynamic loss	Reduction of Payload
1% Intake efficiency (subsonic)	1%
1% Intake efficiency (supersonic)	2½%
1% Aircraft drag (subsonic)	1%
1% Aircraft drag (supersonic)	4½%
1% Intake efficiency (takeoff)	5%

In parallel with work on the external-compression design, the external/internal compression system was brought to a high state of development; an early product of this research was the "self-starting" internal-compression system which, if properly exploited, eliminates one major difficulty of that shock geometry. More recent work suggests the possibility of eliminating the incipient instability always associated with the partial internal-compression system operating at its critical point. Those avenues are worth following, but were acknowledged to be incompatible with a time scale appropriate to a first flight in the spring of 1968.

The choice of external compression has been well justified by the simplicity of intake control in flight. There is, for example, no "unstart" mode; there is a generous margin between any operating condition and buzz; best performance over a wide range of flight conditions may be obtained by allowing the bleed flow to compensate naturally for changes in engine air flow demand, without the intervention of any artificial control system; good engine-face velocity distributions are obtained with an elementary throat bleed geometry. These virtues have allowed the use of a low-gain control system which is actually dormant during the majority of supersonic flight. We hope that development of the system during prototype flying will lead to proposals for a manual standby system, eliminating much complex monitoring and duplicated equipment.

It would be wrong to assume that this high degree of stability and simplicity has been obtained merely by sacrificing aerodynamic performance; in fact, the final geometry (Fig. 2) results from a detailed development program. Tests demonstrate the value of isentropic compression in the ramp, not only for its effect on the theoretical shock recovery, but also for the smooth ride it gives the sidewall boundary layers through supersonic compression. The aerodynamicist will note that this cowl generates the "strong solution" oblique shock, across the capture flow; for a practical low-drag design, the limit of drag reduction is imposed by shock detachment at the cowl lip. This design thus combines a high standard of internal recovery and flow quality with low external drag. The purists can (and do) argue whether this development of the bleed slot and cowl-lip geometries has introduced a measure of internal compression.

Note that the bleed slot, whose aerodynamic operation is described fully by Leynaert in Ref. 2, is ideally situated for dealing with the boundary-layer growth on the front ramp and also with the impingement of the cowl-lip shock on the upper duct surface. As pointed out by M. Leynaert, large variations in the flow extracted by the slot can occur without significantly affecting the external shock pattern of the intake. The way this feature simplifies powerplant management will be discussed later.

Other noteworthy features of the intake geometry are the splitter plate, which provides aerodynamic independence of the twin intakes; the hinged ramps, which vary their position according to Mach number and engine setting; and the dump door, which opens to spill excess intake air in cases of engine throttling or shut down.

It was an aircraft requirement to provide safe operation in upset maneuvers from cruise at Mach 2.2. Optimization of intake performance has, however, been carried out at Mach 2.0, approximately the speed at which best aircraft range is, at present, obtained. Modifications to provide optimization for Mach 2.2 cruise are being studied.

III. Characteristics of the Isolated Intake

Uniform-entry "isolated" tests (divorced from wing airflow) have laid the foundation for present Concorde intake design. When the basic design was frozen, intakes were manufactured for testing; they included $\frac{1}{15}$ scale models, a $\frac{1}{3}$ scale twin intake for tests at Centre d'Essais des Propulseurs (Saclay, France), and a full-scale version running in Cell 4 at the National Gas Turbine Establishment. The importance of Reynolds number was fully appreciated early in the test program, and many of the smaller scale models were tested in wind tunnels at stagnation pressures of 5 atm, thus very nearly simulating full-scale conditions.

Performance

Because of the sensitivity of aircraft economics to intake performance, the experimental derivation of performance characteristics demands great precision. Figure 3 shows typical pressure recovery/mass flow characteristics at a test Mach number simulating the mean underwing conditions at Mach 2.0 (freestream). Mass flow measurements shown are based on the performance of standard nozzles with accurately known discharge coefficients. Detailed traverses of the flow ahead of the intake are also necessary. The sensitivity is such that an error of 0.005 in the estimated capture-plane Mach number leads to an error of $\frac{1}{2}\%$ in the capture-mass flow ratio.

Test results show a well-defined critical point near the point of maximum pressure recovery. Calculations of theoretical shock recovery show that the "extra-to-shock" losses are reduced to only 1% of the freestream total pressure. Thus, pressure recoveries in the region of 0.95 are achieved. An excursion into the supercritical region causes the throat-shock structure to move downstream; the consequent reduction in local static pressure simultaneously reduces the bleed flow. Note that the supercritical leg of the characteristic is vertical, indicating that the main duct flow increases by a compensating amount during this process.

When the shock translation is complete, supersonic conditions are fully established in the throat of the intake; thereafter, on the characteristic, bleed conditions are constant. Throughout the whole process the total capture flow is unchanged, so that changes in the bleed are compensated by equal and opposite changes in the engine flow. In the subcritical regime, the cowl-lip shock becomes detached; total capture flow is reduced by the upstream movement of this shock in accommodating the required forespillage. The flow in the throat is then completely subsonic.

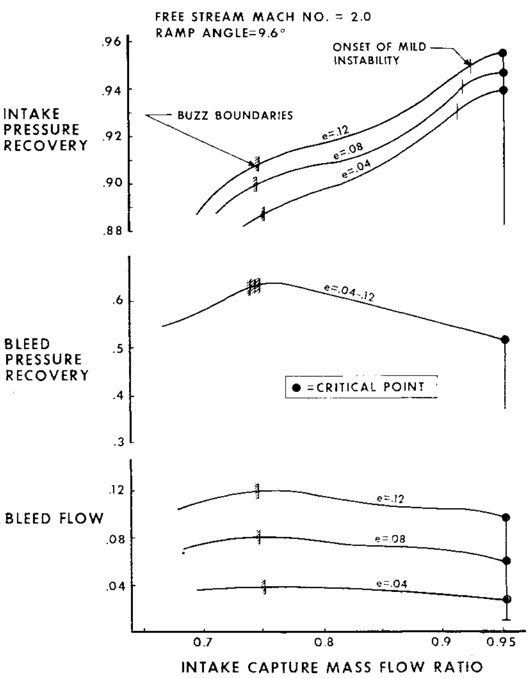

Fig. 3 Typical intake characteristics.

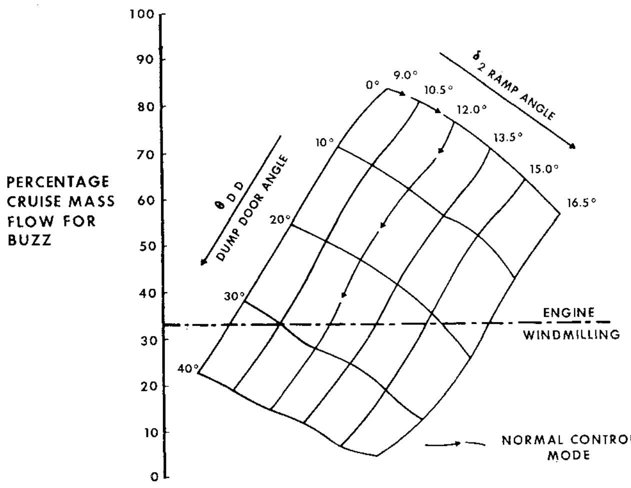

Fig. 4 Cruise margins to buzz.

The characteristics of Fig. 3 are drawn for three different values of bleed control area, i.e., three values of the ratio of bleed pressure to bleed-mass flow. Note that to a good approximation, the level of bleed recovery uniquely defines the intake operating condition relative to the critical point, independently of throat bleed mass flow. For peak internal performance at the aircraft design point it is necessary to size the intake and bleed passages for operation at an elevated bleed pressure, close to critical conditions.

Stability

A significant feature of the characteristics is the "mild instability," which occurs as a result of shock oscillation when the intake is operating slightly subcritically. The oscillation is probably initiated according to the Ferri criterion. It commences upon the intersection of a vortex sheet with the cowl lip; the vortex sheet emanates from the intersection of the near-normal shock, upstream of the cowl lip, with the ramp hinge shock. A quiescent region exists between this region of mild instability and the onset of buzz, the latter being precipitated by intersection of the first wedge shock and the expelled shock.

The onset of mild instability could provoke surge if the engine is sensitive to flow oscillation at the compressor face. This aspect of engine sensitivity will remain obscure until flight experience is obtained; it is considered prudent at the present time to avoid intake operation in this condition. This clearly imposes an upper limit to the level of ramp bleed-pressure recovery for steady-state operation. Operation at ramp bleed pressure recoveries consistent with near-critical intake conditions provides an adequate margin to cope with rapid transient variations in mass flow.

Only remote circumstances such as, for example, undetected failure of both control system channels followed by failure of the affected engine can produce buzz conditions. Nevertheless, it is a design requirement that the intake structure withstand buzz conditions for a reasonable time. Buzz pressures in the intake and bleed ducts were recorded in a model equipped with a rotating valve to simulate buzz, surge, and hammer-shock conditions. Changes in ramp angle and dump door position accommodate reductions in engine mass flow without incurring buzz (Fig. 4). With full deflection of each,

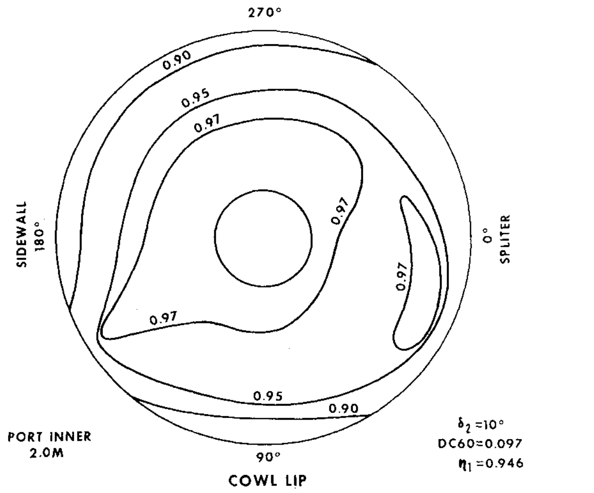

Fig. 5 Engine-face distortions total-pressure contours—view looking downstream.

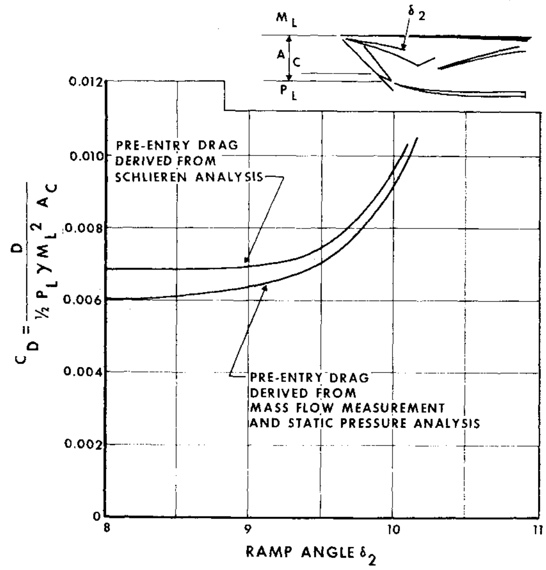

Fig. 7 Pre-entry drag at a local Mach number of 1.9.

the engine mass flow can be safely reduced to 5% of the maximum continuous value at Mach 2.0 (ISA+5°C). The figure also shows the schedule of ramp and dump-door movements in engine throttling, that will ensure acceptable engine-face distributions.

Engine-Face Pressure Distributions

A typical distribution of engine-face pressures in cruise (Fig. 5) shows one of many distributions that were simulated at an early stage by gauzes in an engine-test cell. No single parameter can uniquely define how an engine reacts to a given distortion pattern; experience in Great Britain with the DC60 parameter[3] has shown a fairly high correlation with the effect of flow distortions on engine-surge margin. The distortions are very small indeed, and the test work carried out in Cell 4 at NGTE gives every reason to expect trouble-free engine operation in flight. Figure 6 shows the variation of the distortion parameter with throat bleed conditions. Note the sensitivity of compressor flow distortion to bleed conditions for extreme values of the bleed flow; both an upper and a lower limit to throat bleed flow must be defined in order to ensure satisfactory operation.

Pre-Entry Drag

The positioning of the ramp shocks ahead of the cowl-lip leads to a mass-flow ratio of less than 1.0 at all flight conditions. The maximum value of this ratio depends, of course, upon the particular setting of the variable ramp: associated with the adjustment of the capture flow are changes in the external flow conditions. The pre-entry drag rises as the spill is increased, at first at a rate corresponding with purely supersonic flow ahead of the cowl lip. Subsequently, the pre-entry drag rise steepens as the cowl-lip shock detaches and causes a region of partially subsonic flow. There are simultaneous changes in the cowl drag.

Tests were made with intakes featuring two identical sides, both having leading edges lying on the straight line joining ramp tip to cowl lip. Other tests used models with the "splitter plate" that separates the two intakes forming each powerplant nacelle. Tests with the "nonsplitter" arrangement and large sidewall windows permitted the pre-entry drag to be estimated from the observed shock pattern (Schlieren analysis) which is deflected upstream of the inviscid pattern because of boundary-layer effects. The shock pattern, and hence the pre-entry drag, could also be deduced from measurements of the static pressures on the ramp surface and the total capture-mass flow ratio. The two methods were in excellent agreement (Fig. 7).

The introduction of the splitter plate renders deductions drawn from experimental observations of the shock geometry less certain. The problem was therefore approached from a theoretical standpoint; the spanwise effect of the splitter plate in deflecting the ramp shock geometry was computed theoretically, and the consequent reduction in capture mass flow compared with the experimental measurement. The excellent agreement of computation and experiment (Fig. 8) lends confidence to the computed effect of the splitter plate on pre-entry drag. Before applying this pre-entry drag data to aircraft performance predictions, corrections accounting for lift effects and for the nonuniform wing flowfield must be applied.

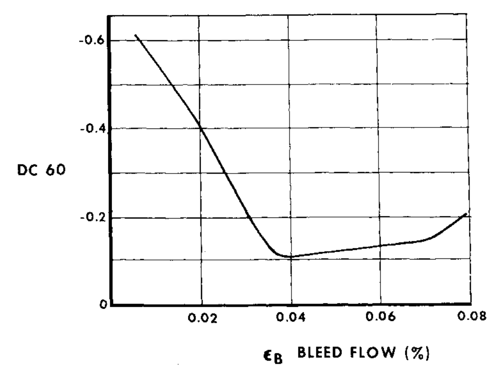

Fig. 6 Effect of bleed flow on engine-face distortion.

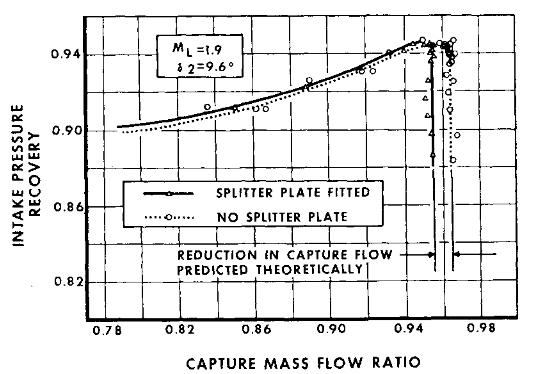

Fig. 8 Effect of splitter plate on capture mass flow.

Use of Large-Scale Intakes

Because of their size, tests of the full-scale twin intakes with engines and intake-control systems were impossible. However, Cell 4 at the NGTE (Pyestock, England) was able to accommodate a single Concorde intake with Olympus 593 engine, making possible full-scale investigation of intake characteristics and the effects on engine handling.

In addition to the full-scale single-intake test, French officials decided to test an approximately one-third scale twin intake with General Electric J85/J2 engines, using the R5 facility at Centre d'Essais des Propulseurs (Saclay, France). This made possible the study of twin duct interaction effects, including those resulting from engine surge, albeit with engines whose surge sensitivity and characteristics were different from those of the Olympus engine.

These two installations are equipped with versatile automatic intake control systems, so that it is possible to explore the stability and response characteristics of the complete controlled powerplant and to check the validity of an analog simulation of the dynamics of the system. Operating limits can be assessed, and effects of sidewash, ramp travel and dump-door opening can all be investigated at various engine settings in both steady-state and transient conditions.

A gratifying aspect of isolated engine testing is the good agreement between different test configurations. Figure 9 shows the agreement between the characteristics obtained from full-scale tests in Cell 4 at NGTE and from other tests at five atmospheres on a $\frac{1}{15}$-scale model. The compressor-face flow distortions also agree; at a typical operating point, a DC60 value of -0.09 obtained from a model test compares with a figure of -0.10 in the full-scale test.

IV. Complete Aircraft Testing

Wing Flowfield Effects

To study the effects of the wing flowfield, a $\frac{1}{15}$-scale model of the complete aircraft was constructed, complete with twin nacelles fully instrumented for intake testing. First, a full exploration of the underwing flowfield was made without nacelles to determine boundary-layer depths and profiles, intake-face Mach numbers, and flow directions. To account for the sidewash near the underwing surface at the appropriate cruise incidences, the intakes were toed-in on the wing of the aircraft (Fig. 1), giving a banana-like curvature to the diffuser. The amount of toe-in is 4° for the outer intake and 2° for the inner.

It was expected that when boundary-layer diverter drag was accounted for, an over-all optimum performance would be found with some degree of immersion of the first wedge of the intake in the wing boundary layer. Tests on both the isolated intake and the complete aircraft model show a small loss in capture coupled with a drop in main duct pressure

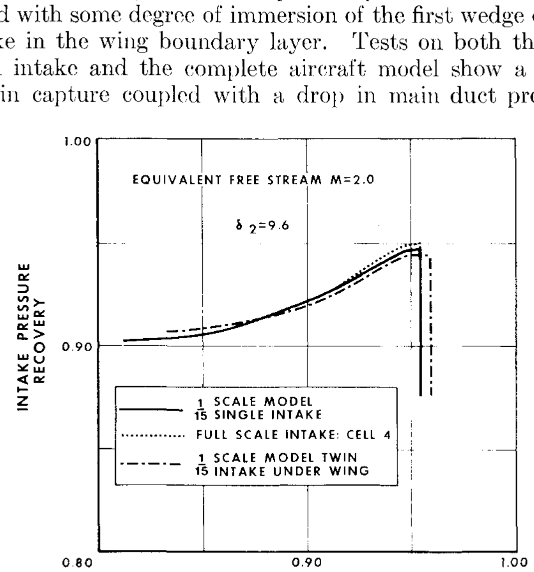

Fig. 9 Capture mass flow ratio.

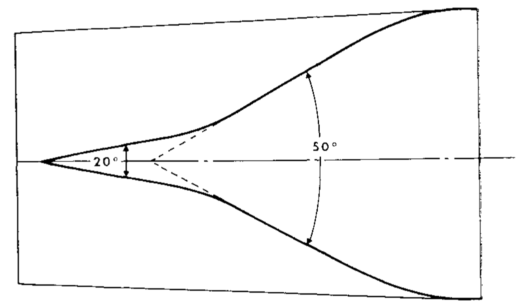

Fig. 10 Diverter geometry.

recovery. The reduction in diverter drag, however, more than offsets this for degrees of immersion which lead to at least the outer 30% in depth of the boundary layer being swallowed by the intake. A small increase in ramp bleed flow is sufficient to regain the main duct recovery of the non-immersed case.

The diverter planform was developed as a result of considerable testing of both isolated diverter models and the complete aircraft model. Early diverter designs showed an "unstarted" diverter flow where forespillage from the diverter was interfering with the intake shock pattern and causing a significant loss of intake capture. It is clearly essential to position the apex downstream of the diverter inlet, and it was found necessary to reduce the included angle to not more than 20° (Fig. 10).

Intake Characteristics

Figure 9 shows a comparison between the intake characteristics measured on the complete aircraft model with those of isolated intakes. In general, measured recoveries agree very well and the complete aircraft model shows an improved maximum capture and a much-reduced level of instability, particularly in the region just below the critical point.

The splitter plate was designed to ensure independence of the shock patterns in front of each intake when one or other was operating in subcritical conditions; complete aircraft testing has confirmed the design. In stable running, there is no interaction between the two intakes. When buzz conditions are reached in one duct, a very small sympathetic disturbance is observed in the other intake; the condition becomes progressively more important as the buzz regime is penetrated. In practice, this means that when buzz is fully established in one duct, oscillations of about 30% of the buzz amplitude will be experienced in the neighboring duct.

Another form of interaction results from induced yaw following an engine failure, causing sidewash at the face of all intakes by very nearly the sideslip angle attained by the aircraft, and a consequent deterioration of the engine face pressure distributions. A modification to the sidewall geometry of the intake is being incorporated in the third and subsequent aircraft, and will significantly improve the tolerance of the intake to sidewash. Tests in Cell 4 at NGTE on the prototype geometry have shown that the current engine surge margin is adequate for the expected amounts of sidewash.

V. Integrated Powerplant Concept

General Arrangement of the Powerplant

Each Concorde nacelle houses a twin powerplant, each member of the pair being independent in all possible aspects. The internal air system (Fig. 11) has two principal streams of air, denoted as primary and secondary.

The secondary stream is derived from the intake throat at the ramp bleed slot and passes through the engine bay, to be exhausted in the dual stream secondary nozzle. This arrangement combines considerable flexibility of powerplant

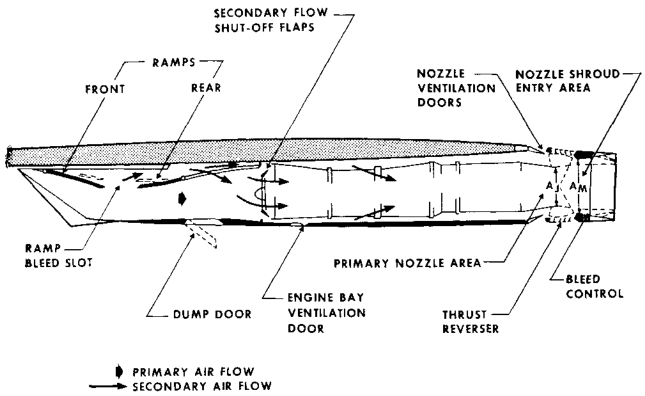

Fig. 11 Powerplant layout.

operation with good over-all performance. A correctly matched dual-stream nozzle provides high exhaust efficiency for the mixed primary and secondary flows. The added advantage of being able to use this air supply for engine bay ventilation and cooling makes the arrangement much superior to any other.

An important development problem was the reduction of the pressure losses in secondary flow ducts and engine bay to an acceptable level. Internal flow tests on an accurate half-scale model of the powerplant were carried out, and the arrangement was progressively modified until an efficient system was obtained. The greater part of the losses were arising at the entry to the engine bay; in the bay itself, Mach numbers were low and the drag of the engine and its accessories was not important.

Recall that the intake characteristics were presented for various values of bleed control area, this being the test configuration (Fig. 3). The effective bleed control area (Fig. 11) is determined by the choking effect of the expanding primary jet in the dual stream nozzle. The detailed implications of this arrangement have been dealt with at length by Talbot and Furness in Ref. 3, and are only summarized here. It is clear that it is the engine and exhaust nozzle which normally dictate how the intake and its associated bleed flow operate. At a given speed, specific range will vary with bleed flow; an optimum value for any cruise condition can be determined only after a comprehensive study of intake, engine, engine bay, and nozzle characterisitics.

Choice of Intake Area

Beyond a brief reference to the effect of flow distortions, nothing has so far been said about the factors governing the choice of intake size in respect to a given engine-flow characteristic. There are three questions to be answered. First,

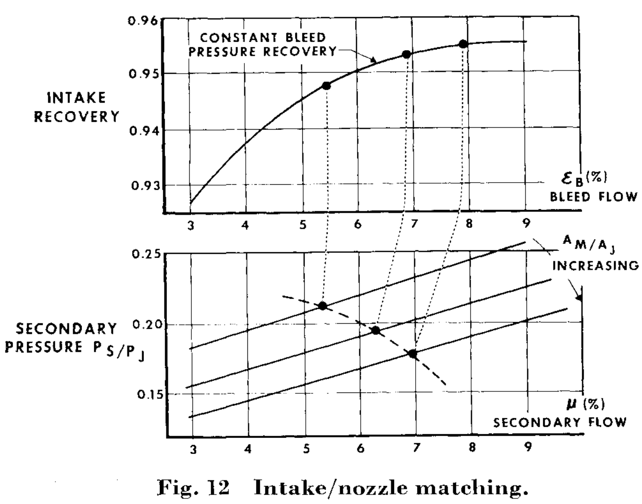

Fig. 12 Intake/nozzle matching.

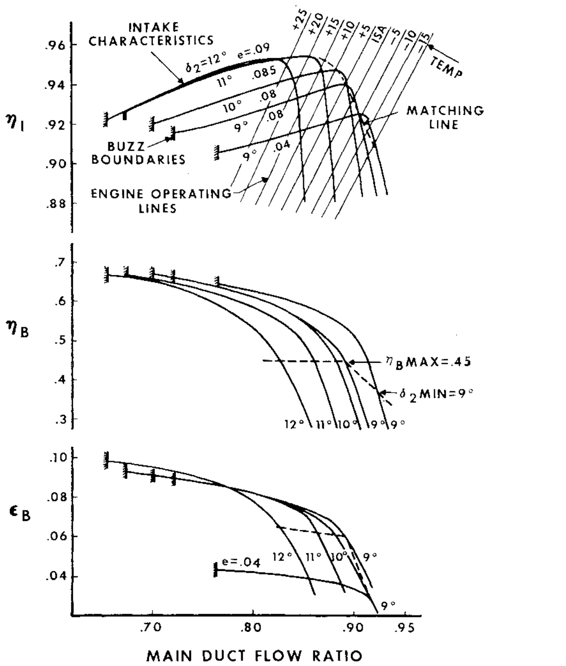

Fig. 13 Intake/engine matching.

what ambient temperature should be assumed for the cruise point? Second, where on the intake-recovery-vs-bleed-flow characteristics should the engine operate? Third, what margins should be built in to provide good stability? The specified design point for the intake is Mach 2.0 at an ambient temperature of ISA+5°C (as a mean temperature, this is exceeded in only about 15% of occasions in the North Atlantic route).

The presentation of intake characteristics given in Sec. III showed that operation at near-critical conditions with high duct recovery and an operating margin from the mild instability would be ensured by choosing a unique upper bound for the ramp bleed pressure. Now the geometrical parameter which is most significant in determining the bleed control area is, for a fixed engine setting, the nozzle shroud entry area. Figure 12 shows the variation of main duct recovery with bleed-mass flow for constant bleed pressure, and also the variation of nozzle pumping characteristics with shroud entry area.‡ The relationship between matching intake and nozzle shroud areas can be obtained from this diagram and the variation of net thrust obtained as a function of bleed flow. The result of this calculation shows a fairly constant net thrust for values of the bleed flow between 3 and 6%. Beyond this figure, the increasing secondary momentum loss and deteriorating nozzle performance are not offset by improvements in intake efficiency. This range of bleed flow is compatible with acceptable engine face distortions (see Fig. 6). The arguments detailed below concerning flexibility of operation over the likely range of flight conditions led us to associate an ambient temperature of ISA+5°C with a high value of bleed flow. The intake and nozzle shroud areas were therefore chosen so that at the design point the intake operates at maximum bleed pressure, with a bleed flow of about 6%, and with a ramp angle at the minimum consistent with adequate stability.

Effect of Ambient Temperatures

The flow characteristics for a supersonic intake operating at a constant Mach number are such that its airflow varies inversely as the square root of the ambient temperature; thus a rise in ambient temperature leads to a drop in airflow.

‡ Secondary flow in the nozzle is composed of the ramp-bleed flow plus cabin-air heat exchanger and other cooling flows exhausted in the engine bay.

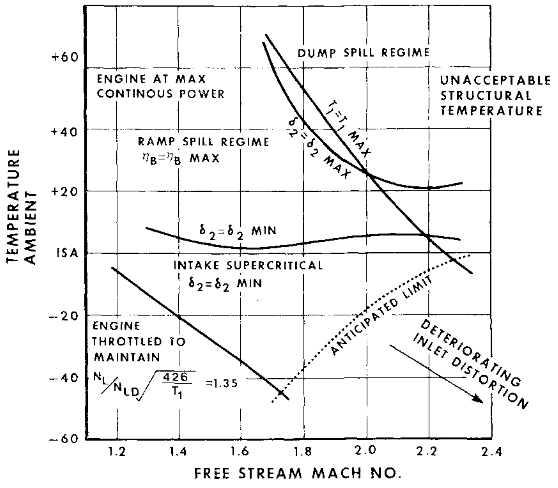

Fig. 14 Intake operating regimes.

Similarly the engine will also respond to ambient temperature changes, but in a manner which approximates the direct inverse of ambient temperature (assuming constant rotational speed and intake efficiency). It is therefore possible to match the powerplant for optimum performance at only one ambient temperature. Maladjustments can also arise from differences in engine flow characteristics from mean brochure figures, and other sources. The manner by which differences are adjusted is an important aspect of supersonic intake design.

The primary manner in which the present intake is adjusted to match engine conditions is by variation of supersonic forespillage, effected by variation of the ramp angle. At ambient temperatures above the design condition, the ramp angle is increased in order to maintain near-critical conditions. At temperatures below the design condition the intake is allowed to operate in the supercritical regime.

Intake adjustment is illustrated in Fig. 13, where the intake characteristics are plotted against main duct flow so that the intake and engine conditions can be directly related. In the supercritical regime, note that the recovery/flow characteristics are no longer vertical. Since the total of duct and bleed flow which is constant, the powerplant air-flow systems are self-compensating (at ambient temperatures below that for minimum ramp angle) in that changes in engine demand can occur without changing the total intake capture flow. Studies show that it is more efficient to maintain maximum continuous engine conditions and operate the intake supercritically than to throttle the engine to maintain intake recovery. The throttling mode which is most favorable for cold-climate operation is a reduction of rpm at constant turbine entry temperature; on the Olympus engine, this involves a reduction of primary nozzle area leading to a loss of installed nozzle performance.

Turning now to flight conditions in which the ambient temperature is higher than that for minimum ramp angle (the ramp-spill regime), near-critical intake operation is achieved with good intake pressure recovery over an extensive range of temperature variation when the ramp is controlled so as to maintain a constant ramp bleed pressure. The effective bleed control area, which increases fairly rapidly with ambient temperature in the supercritical regime, is almost constant in the ramp spill regime.

Powerplant Management

Having chosen the capture area of the intake, it is possible to identify the following four regimes of powerplant operation according to the ambient temperature; 1) very cold conditions, where the large engine air flow demand would lead to

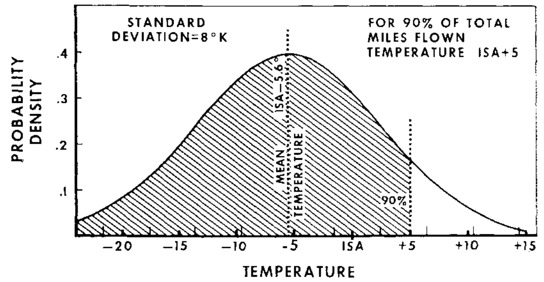

Fig. 15 World-wide frequency weighted temperature distribution.

low ramp bleed flows; engine face distributions would then be poor, and bay ventilation inadequate; the engine rpm must therefore be reduced until a tolerable level of ramp bleed pressure ratio is obtained; 2) intermediate conditions, where the engine may be operated at maximum continuous setting and the ramp is at the minimum angle consistent with adequate stability; 3) warm conditions, where the ramp angle is increased to limit ramp void pressure recovery; 4) very warm conditions, where a large ramp angle would lead to poor engine face distributions; the ramp angle is limited and the dump door opened to maintain ramp void pressure.

Figure 14 shows the ambient temperature ranges defining these regimes with the chosen intake area and the Olympus 593 engine; Fig. 15 shows the probability of enroute ambient temperatures, frequency-weighted over worldwide long-distance routes. Note that the band of temperature in which the control system is (in normal circumstances) dormant covers a large percentage of aircraft-cruise miles. Movement of the ramp and dump door is effected automatically by the intake control system. Throttling of the engine in the very cold regime is manual.

Intake-Control System

The maximum and minimum ramp bleed pressure recoveries and ramp angles as functions of Mach number are shown in Fig. 16. The intake control system senses bleed pressure

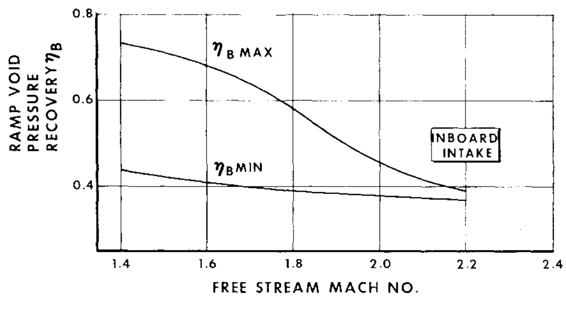

Fig. 16a Ramp void-pressure recovery schedule.

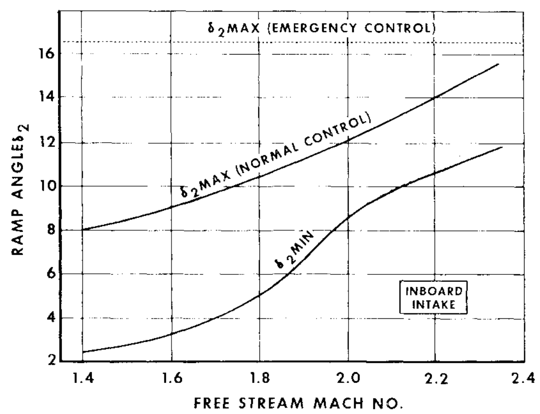

Fig. 16b Ramp schedules.

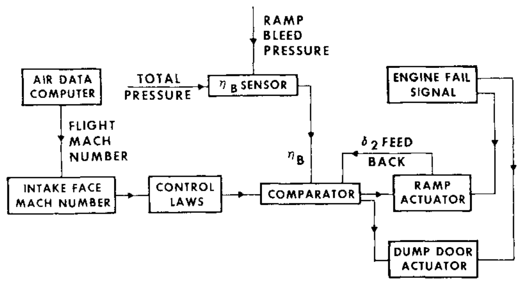

Fig. 17 Intake control system flow diagram.

recovery; if we consider a flight condition where the intake is operating at the maximum value of this parameter, it is clear that a rise in ambient temperature or throttling of the engine will lead first to a rise in bleed pressure recovery and then to a controlled increase in ramp angle. Similarly, an increase in flight Mach number will lead to an increase of ramp angle.

The attainment of maximum ramp angle switches control to the dump door, and this arrangement copes automatically with throttling back or shut-down of the engine. To cover engine failure, an input based upon rate of change of engine rpm increases the rate of operation of the ramp and dump door. An additional ramp overtravel is provided to cover emergency cases, such as fire when the engine-bay flow must be shut off, and the requirement for forespillage is therefore increased.

For the prototype Concorde, an extremely versatile intake-control system is fitted (Fig. 17). Atmospheric data is derived from the aircraft air-data system; full provision is made for altering the control laws and, if necessary, elaborating them to take account of incidence or sideslip effects. It is hoped that these laws may be generated aerodynamically when they are finally defined for production aircraft.

VI. Low-Speed Problems

The emphasis in this paper has been on development problems associated with intake characteristics at the cruise Mach numbers; the intake geometry is basically designed for this condition, but many difficulties have been encountered in obtaining satisfactory performance in other flight conditions.

Takeoff

The intake entry area is dimensioned for cruise and means of increasing the mass flow at low speeds must be fitted. Two alternative designs of auxiliary door have been developed (Fig. 18) and the study of engine/intake compatibility has been assisted by the Vulcan flying test bed, a Vulcan aircraft

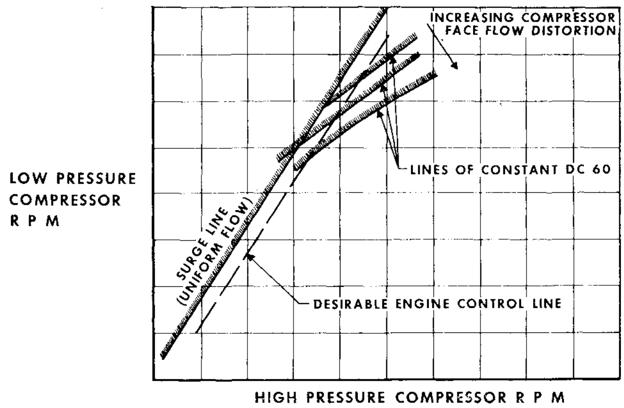

Fig. 19 Effect of compressor face-flow distortion on engine surge.

fitted with a representative Concorde powerplant in a central position under the fuselage.

The difficulty here is the powerful effect of engine-face distortions on engine-surge margin in the high $N/T^{1/2}$ engine conditions when running at maximum take-off thrust (Fig. 19). The simple form of auxiliary inlet might incure risks of poor engine handling; therefore, a sophisticated aerodynamic design incorporating a flap and a cascade was developed in the wind tunnel. As shown in the figure, it can provide extremely good engine-face pressure distributions.

Subsonic Performance

The importance of the secondary-flow system in subsonic flight must not be underestimated. Development of the internal geometry of the ramp bleed slot has enabled bleed-pressure recoveries of around 95% to be obtained, ensuring a good supply of secondary air to the nozzle and allowing diversion cruise (at Mach 0.93) with the nozzle-ventilation doors closed. Decreased reliance on these ventilation doors should simplify the development program and improve the confidence which can be placed on aircraft performance estimates.

However, the particular problem posed by the intake at high subsonic Mach numbers is that of spill drag. There is no special difficulty entailed in securing either good pressure recovery or flow uniformity at the compressor face, but the drag problem arises on two counts: first, an intake sized for supersonic performance has to operate at low capture mass flow ratios during subsonic cruise and the pre-entry drag is thus significant; second, the ability to develop compensating suction forces on the cowl is severely limited by the need for a sharp low-angle lip to minimize drag in supersonic flight. Figure 20 shows schematically the familiar arrangement of the forces involved. These are fundamental facts of life which the designer of an intake for any supersonic airplane has to face.

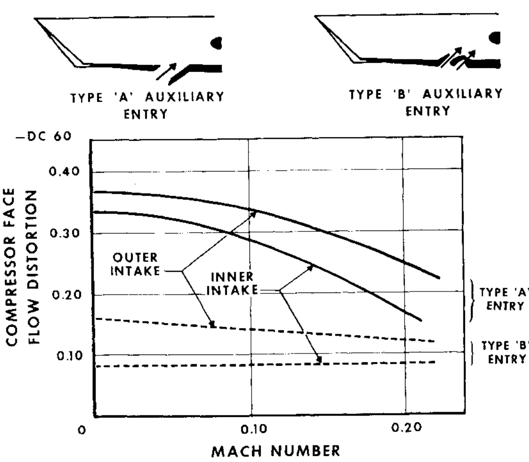

Fig. 18 Compressor face-flow distortion at low speed.

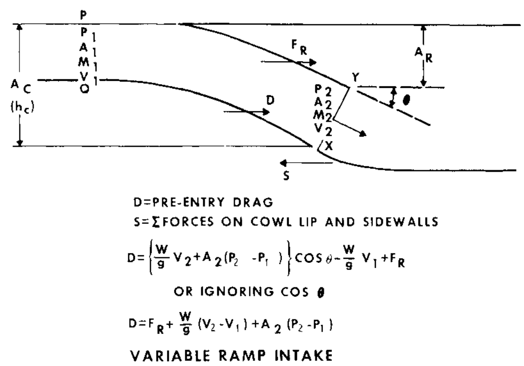

Fig. 20 Ramp intake in subsonic flight (the forces involved).

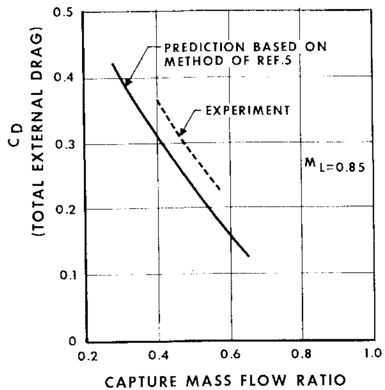

Fig. 21 Drag of ramp intakes at high-subsonic Mach number.

The proper appreciation of the magnitude of these forces and an awareness of the methods by which they might be minimized is important. Two years ago it was possible for Mount to observe that the whole area was one of "the many little technical pockets that has been skipped over in the transition from high-speed subsonic to supersonic design point flight" (Ref. 6). Among the reasons for this situation might be supposed both a general failure to appreciate the severity of the problem, and the fact that in the whole field of experimental techniques, the accurate determination of spill drag presents probably the greatest challenge of all.

During the past two years, much progress has been made. Not only can drag estimates be advanced with confidence, but also the favorable effects stemming from such possibilities as trimming the ramp position are well understood. Figure 21 shows a typical case of the gratifying agreement now achieved between experimental drag levels and those predicted by the method of Ref. 5. The sharp dependence of drag on capture flow ratio is most evident.

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