Ramjet Intakes

T Cain
Gas Dynamics Ltd.,
2 Clockhouse Road, Farnborough, GU14 7QY
Hampshire, UK

tcain@gasdynamics.co.uk

ABSTRACT

Intake design for supersonic engines, in common with other engineering design problems, is application dependent and the true challenges are in meeting performance targets over the required Mach and Reynolds number ranges while complying with the multitude of constraints imposed by the aircraft/missile and its mission. The fundamentals and limitations of efficient ram compression are well understood and since NASA, DTIC, RTO and AERADE provide free public access to a large database of intake experiments conducted in the 1940s to 1970s, the designer should be aware of problems encountered and the fixes applied during previous testing of isolated intakes. The outline of this lecture is a brief tour of some historic supersonic intakes discussing the features that enable the intake to meet its requirements and applying some reverse engineering to deduce how the designers appear to have approached the problem. The tour is combined with an introduction to tailoring compressive flow fields by exploitation of one and two dimensional flow elements.

1 INTRODUCTION

There is an established format for lectures, books and reviews concerning intakes, with a large section allocated to taxonomy, distinguishing types by: the number and location on the aircraft/missile; the degree of external and internal compression; whether they are based on two dimensional plane flows, axisymmetric flows, or are three dimensional; and whether they are outward turning or inward turning. The function and design of the supersonic diffuser is then dealt with before a discussion of the subsonic diffuser. Methods of accounting for, and controlling boundary layers are necessarily included.

This lecture approaches the subject from a different perspective, here we are less concerned with intakes in general, and far more concerned with the details of selected intakes. The difference in approach reflects a difference in philosophy, and teaching styles. I think it is easier to extrapolate and expand from a detailed small study, than to imagine or reinvent what has not been revealed in a general overview. Should a less focused approach be preferred, there are good references that are free to download. Reference [1] is a fine example, summarising what was known about intakes in 1964, which is practically everything known about them today. It is not that work done after that time is redundant, but since the ground work was complete, later intake studies tend to be either learning exercises for the individuals involved, or are focussed on an application and remain unpublished for commercial and/or military reasons.

Fortunately with the elapse of time and the retirement of aircraft and missiles, the sensitivity of the applied design work reduces, and some details enter the public domain. Consistent with the perspective outlined above, this lecture takes advantage of the information available on historic intakes and draws conclusions regarding the design approach taken. Of particular interest are the features that enable the intake to function over the range of Mach numbers and the angles of attack to which it was subjected. There is considerable risk that some of these conclusions are erroneous but the consequences of a misinterpretation are small, the aim is not to recreate the system, but to explore the design drivers and the response to them. If I have drawn the wrong conclusions, I apologise to the designers for misrepresenting their creations, but

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14. ABSTRACT
Intake design for supersonic engines, in common with other engineering design problems, is application dependent and the true challenges are in meeting performance targets over the required Mach and Reynolds number ranges while complying with the multitude of constraints imposed by the aircraft/missile and its mission. The fundamentals and limitations of efficient ram compression are well understood and since NASA, DTIC, RTO and AERADE provide free public access to a large database of intake experiments conducted in the 1940s to 1970s, the designer should be aware of problems encountered and the fixes applied during previous testing of isolated intakes. The outline of this lecture is a brief tour of some historic supersonic intakes discussing the features that enable the intake to meet its requirements and applying some reverse engineering to deduce how the designers appear to have approached the problem. The tour is combined with an introduction to tailoring compressive flow fields by exploitation of one and two dimensional flow elements.

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at least I should still have succeeded in introducing the problems to be addressed and the elements that are the keys to the solution.

2 TROMMSDORFF RAMJETS

2.1 Mach 4+ ramjet powered flight during WW2

Trommsdorff's ramjet powered projectiles made the world's first supersonic air-breathing flights. About 260 of the experimental 15cm diameter E series, figure 1, were fired from a gun with muzzle velocities of about 1000m/s, accelerating to 1460m/s during a 3.2s burn. Trommsdorff's modestly written, detailed account of their development [2], is unclassified but unfortunately rather difficult to obtain, and his work has not received the recognition it deserves. He was fortunate to be able to call upon the Kaiser-Wilhelm-Institute in Gottingen and the Institute for Aerodynamics in Braunschweig for advice on intakes, gas dynamics and combustion. At the first he consulted with Prandtl, Betz, Ludwieg, and Oswatitsch and at the second with Busemann, Schmidt and Damkohler. Most researchers in gas dynamics will be familiar with those names and will appreciate Trommsdorff could not have been better advised.

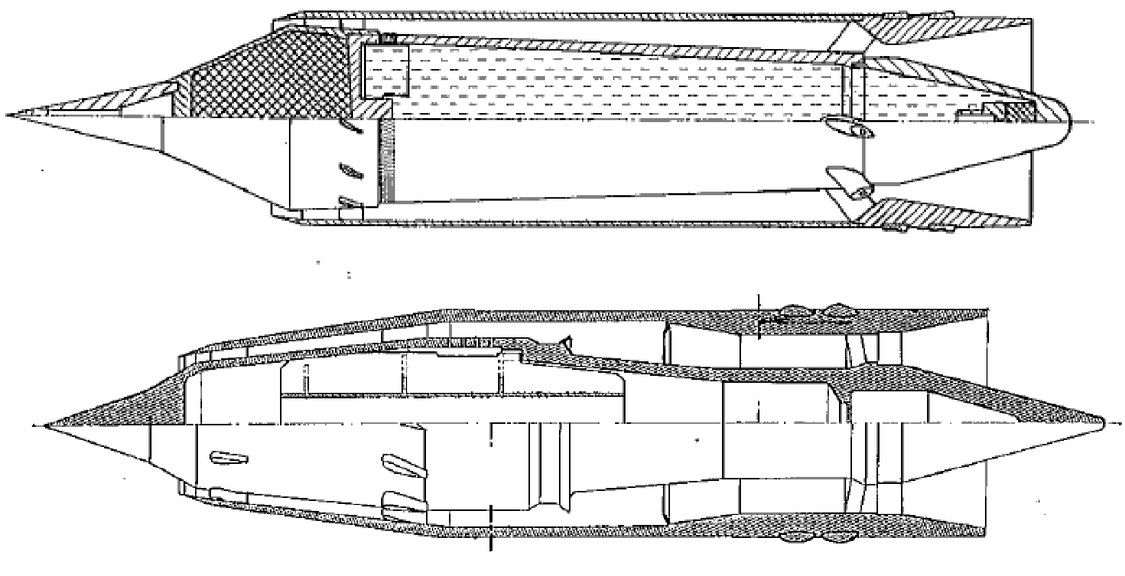


Figure 1: Trommsdorff projectiles, the 28cm calibre, diesel fuelled C3 (top) and the 15cm calibre, CS_2 fuelled E4.(bottom). The original version of this figure was published by the Advisory Group for Aerospace Research and Development, North Atlantic Treaty Organization (AGARD/NATO) for Trommsdorff [2].

The 170kg C3 was designed to be fired from the German K5 gun at 1223m/s, accelerate to 1860m/s, and then cover a distance of 350km in free ballistic flight. The war ended, and Trommsdorf was taken to Russia before the C3 could be tested. After his release and return to Germany, Trommsdorf reported that the C3 achieved the calculated performance when tested elsewhere [2].

2.2 Oswatitsch's intake research

Oswatitsch is now synonymous with the multiple shock external compression intakes of the type exhibited above and Busemann with all internal compression intakes particularly those utilising isentropic compression based on a conical flow (a one dimensional flow, with properties being only a function of the angle from the vertex) the existence of which he hypothesised and proved. The differential form of the equations that describe such flows were formulated and published by Taylor and Macoll with reference to Busemann.

There is a limit to internal contraction above which an intake will not start, and instead flow will spill around the cowl with the amount entering the engine simply set by choked flow through the intake throat as determined by stagnation conditions downstream of a normal shock standing in front of the cowl. This limit, now known as the Kantrowitz limit, was first defined by Oswatitsch in the study he made for Trommsdorff. A translation of his report is available as reference [3] and it also contains the proof of the result for which he is best known: shocks of a multi shock diffuser should have equal strength for maximum pressure recovery. One does not need a mathematical proof to understand this result, it is due to the fact that entropy rise increases rapidly with the temperature ratio across a shock and if two shocks within a sequence did not have the same temperature ratio the entropy gain over the stronger shock will outweigh the decreased rise on the weaker shock when the flow is compressed to the same Mach number.

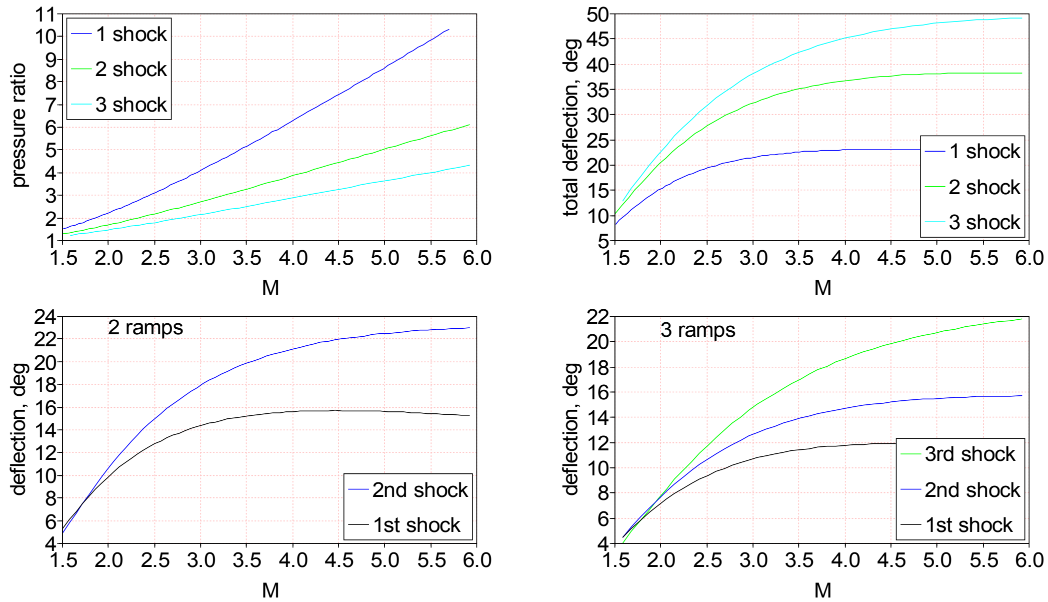


Figure 2: Oswatitsch's optimal multi-shock intake parameters as a function of flight Mach number

Oswatitsch's optimal intake ramp angles are most easily found using his procedure, which starts with the Mach number upstream of the terminal normal shock. Upstream oblique shocks all have this normal component of Mach number and one can determine a corresponding freestream Mach number by simply stepping upstream through the chosen number of shocks. Results from this calculation are presented in figure 2. There are three points to note from the figure: At Mach numbers above three, pressure ratio across each shock is relatively high and in most cases would be sufficient to separate the boundary layer; total deflection of the two and three ramp intakes is very high and unless the deflections are in opposite sense which implies internal compression, the cowl will be at a steep angle implying high drag; successive deflections increase in magnitude, somewhat like C3 in figure 1 and unlike E4.

Oswatitsch experimented with a biconic intake at Mach 2.9 [3], exploring and defining super and subcritical operation (started and unstarted in today's parlance), buzz (the noise from flow pulsation during unstable subcritical operation), subsonic diffuser losses, and the effect of boundary layer bleed and angle of attack. Concerns over self-starting, cowl drag, and flow stability, immediately relegated his shock strength optimisation to the role of guidance. From the beginning, the choice of intake ramp angles for a ramjet was known to be influenced by much more than just shock losses.

2.3 The E4 intake flow field

To explore the design drivers and the resulting E4 intake, figure 1 is assumed drawn to scale and the supersonic diffuser flow calculated using the Method of Characteristics (MOC). The calculation is rotational (allows for the variation in entropy throughout the flow field) and assumes axial symmetry.

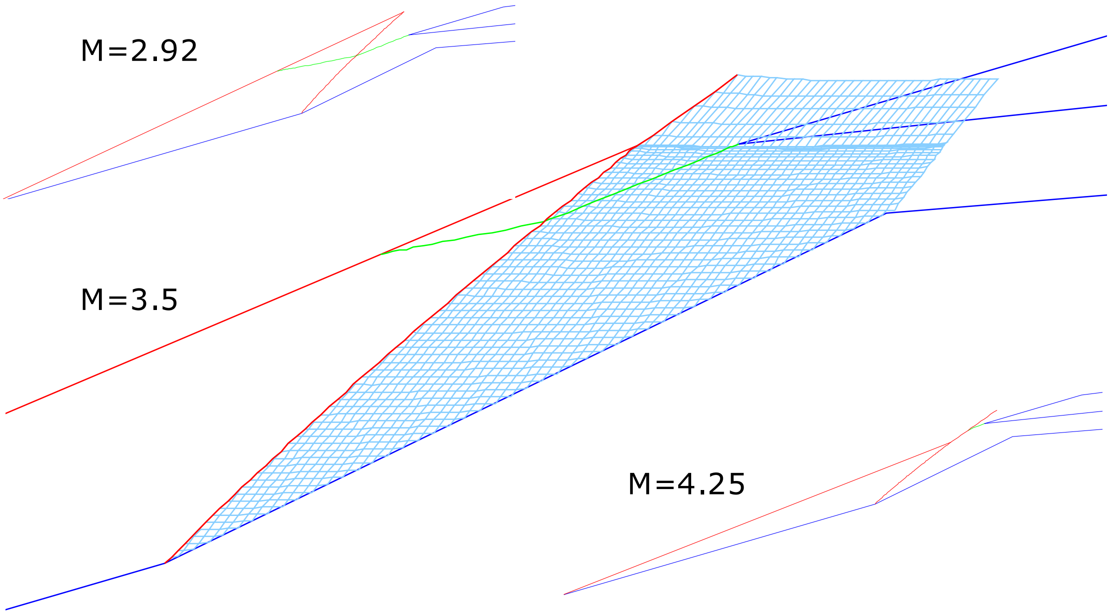


Figure 3: MOC solutions for the E4 19.5°/30.5° biconic intake

Blue lines in figure 3 define the biconic surface and the cowl lip. Red lines are the calculated shocks, the one from the leading edge being straight and the one originating at the cone junction is curved as it propagates through the conical flow over the upstream cone. The green line traces the streamline that intercepts the cowl lip. The light blue lines are the characteristic mesh via which the flowfield solution has been developed. These characteristics are Mach lines within the flow, running both to the left and right of a streamline at the local Mach angle. The right runners are directed inwards, towards the centreline, in this solution and the left runners are propagating outwards, as are the shocks. The compatibility relations that hold at the intersection of the left and right runners enable the flowfield to be defined in a stepwise process that completes in less than a second on a mediocre personal computer.

Note that in this solution we are not yet concerned with the internal flow and the interaction with the cowl. The characteristics have been developed as if the cowl was not present. At the muzzle velocity (M=2.92) the two shocks merge above the lip and their interaction has no effect on the captured flow. Mid-acceleration (M=3.5) the shock interaction generates an expansion fan that enters the intake. The triple point defined by the intersecting cone shocks and the resulting strong shock, sets a limit to the amount of external compression that can be obtained. When the deflection is too high, such that the flow is subsonic behind the strong shock, its position will depend on downstream conditions and the flowfield generally becomes unstable. Charts that define limits to external compression set by the triple point behaviour, are presented in reference 1.

At the peak Mach number of 4.25, flow that has passed through the single shock is entering the intake. This is normally regarded as very undesirable because the stagnation pressure of the flow on the strong shock side of the slip line (the streamline emanating from the triple point) is much less than that of the flow that has passed through two shocks. Whether this truly is a problem is dependent on what back pressure is being applied to the intake (by the combustor) when it is in this state. If this is in excess of the the lowest stagnation pressure then one is reliant on mixing between the two streams, within the isolator, in order for the higher entropy air to pass. The back pressure must always be less than that obtainable by stagnating the mixed flow and how this is calculated is demonstrated in the next section, before returning to a discussion of the E4 intake design.

2.4 Stream thrust and "extra to shock losses"

Most supersonic diffusers produce non uniform flows, either as a result of skin friction creating boundary layers, or non uniform compression such as that produced by a conical compression surface. The E4 intake at Mach 4.25 is a rather extreme example, with non-uniformity inherent in the compression on the first cone (although the straight shock guarantees uniform entropy increase, the streamlines near the surface have been subject to isentropic compression within the shock layer as they are turned to be asymptotically parallel with the surface), a resulting curved shock leading to the second cone, and most significantly the single strong shock to the flow above the slip line.

Various methods have been suggested to account for the non-uniformities on diffuser performance, but only one is rigorous, and not reliant on empirical correction. Wyatt [4] is credited with applying basic thermodynamics to the intake problem and first arguing that the equivalent one dimensional flow is that with the same stream thrust, mass flow, and total enthalpy as the integrated non-uniform flow. The somewhat mystical "extra to shock losses" which are so often modelled empirically are revealed and quantified by this method.

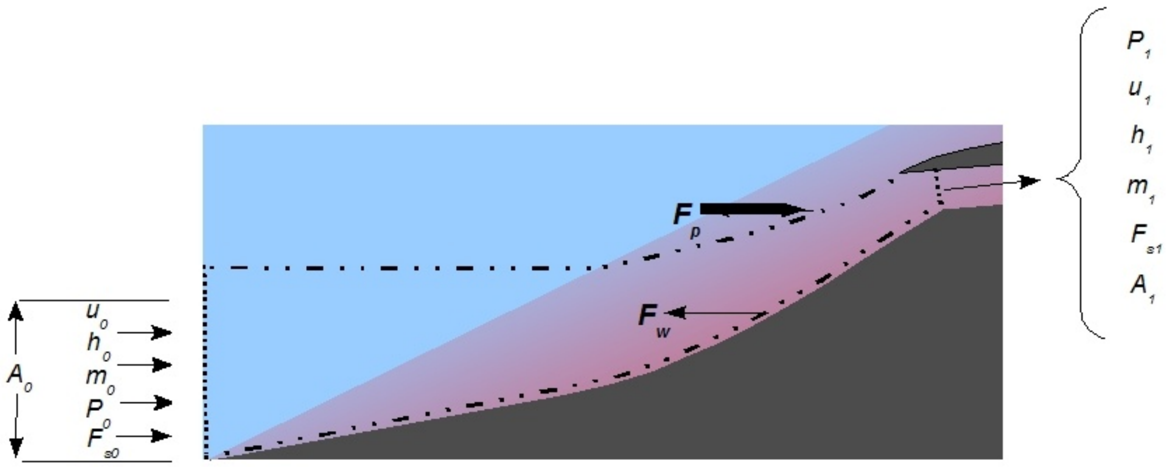


Figure 4: Supersonic diffuser control volume

Consider the control volume of figure 4, the conservation of mass, axial momentum and energy require,

$$\rho_0 u_0 A_0 = \rho_1 u_1 A_1 \quad [1]$$

$$p_0 A_0 + \dot{m}_1 u_0 + F_p - F_w = p_1 A_1 \cos\theta + \dot{m}_1 u_1 \cos\theta = F_1 \cos\theta \quad [2]$$

$$h_0 + u_0^2/2 = h_1 + u_1^2/2 \quad [3]$$

where: ρ, u, A, p, θ, $\dot{m}$ and h are density, velocity, area (normal to u), pressure, stream angle, mass flow, and enthalpy respectively. The area at the outflow boundary is drawn normal to the internal cowl profile and intersects the shoulder. The axial momentum balance described by equation 2, defines the stream thrust F_1 at the outflow boundary. The energy balance described by equation 3, could have included the heat loss to the wall, but that is not essential for the purpose of this discussion. Note that wall stress is included within the wall force F_w and its effect on F_1 is indistinguishable from pressure drag. The ideal intake is the one with the lowest possible drag for a given contraction as this maximises stream thrust F_1.

The pre-entry force F_p has a positive influence on F_1 but the sum of F_p and F_w will always be in the direction of F_w and has a minimum value set by the drag required for isentropic compression to the same contraction ratio. In the case drawn F_p includes the axial force from just inside the cowl lip, but the external component is equivalent to the intakes pre-entry (or additive) drag. Should one wonder where pre-entry drag acts on the airframe, it is an excess in F_w.

When specific heat is constant, equations 1 to 3 reduce to the quadratic,

$$\frac{\gamma+1}{\gamma}\left(\frac{u_1}{u_0}\right)^2+\frac{F_1}{q_0 A_0}\frac{u_1}{u_0}+\left(\frac{2}{\gamma M_0^2}+\frac{\gamma-1}{\gamma}\right)=0 \qquad [4]$$

where q_0 and M_0 are the free stream dynamic pressure and Mach number respectively. Note that the stream thrust at the isolator entrance (F_1) is the only parameter that distinguishes one intake from another in equation 4. Its value is calculated from equation 2, for which it is necessary to know the pre-entry and wall forces. One of the roots of equation 4 is subsonic (ramjet intake) and the other supersonic (scramjet intake). The roots are equal, and correspond to Mach 1 when,

$$\left(\frac{F_1}{q_0 A_0}\right)^2=4\frac{\gamma+1}{\gamma}\left(\frac{2}{\gamma M_0^2}+\frac{\gamma-1}{\gamma}\right) \qquad [5]$$

When the stream thrust is below the value given by equation 5, no solution is possible and the intake will not start. The flow will be choked at the isolator entrance and the excess flow will be spilt.

It should be remembered that the value of F_1 and the one dimensional parameters it is associated with, are calculated from a two (or three) dimensional flow field. Parameters like u_1 and p_1 are equivalent to the values that would be measured if the flow was allowed to become fully mixed, and uniform within a constant area, frictionless isolator. The mixing is associated with a loss in total pressure, but no change in F_1, and it is understandable why some intake designers are reluctant to ascribe the mixing loss to their intake. However, it is stream thrust that determines whether the flow will be able to pass into the combustor and it is not coincidental that this method is able to predict maximum allowable back pressure, while other averaging techniques such as mass or area averaging require empirical factors to account for "extra to shock losses".

2.5 E4 intake performance

Using MOC to calculate the wall and pre-entry forces and equation 4 to determine the equivalent one dimensional isolator state, the isolator static and stagnation pressures were sought for the E4 over the flight Mach number range of 2.92 to 4.25. There are three aspects of this procedure that are worth consideration before reviewing the results:

- The flow local to the cowl lip has particular significance in intake design. When the internal surface is not aligned with the dividing streamline, the local Mach number must be sufficiently high that the internal shock will not detach. The same considerations apply to the external flow. This provides another limit to the degree of compression. The E4 cowl appears to turn the flow back from approximately 27° (local streamline angle) to 7.3°, and the internal shock remained attached over the flight Mach number range. The cowl's contribution to axial stream thrust (due to the finite length between lip and control volume outflow boundary, figure 4) was calculated using the pressure downstream of this internal shock, and included within F_p.

- The isolator appears to be choked (the condition set by equation 5) at M_0=3.17, and the model predicts that below this flight Mach number the intake would run sub critically. However, if it is assumed that the pressure on the inside of the cowl is that given by a strong oblique shock (subsonic downstream) rather than the weak oblique shock, then the stream thrust is sufficient for the intake to operate super critically at Mach 2.92.

- The turbulent boundary layer on the spike was modelled using the momentum integral equation and the flat plate relationships between: momentum Reynolds number and local skin friction coefficient; and edge Mach number and shape factor. The reference temperature method accounts for compressibility. The technique has proved itself to be sufficiently accurate when applied to many different internal and external flows modelled by the author, despite the presence of large pressure gradients under which flat plate closure might reasonably be challenged.

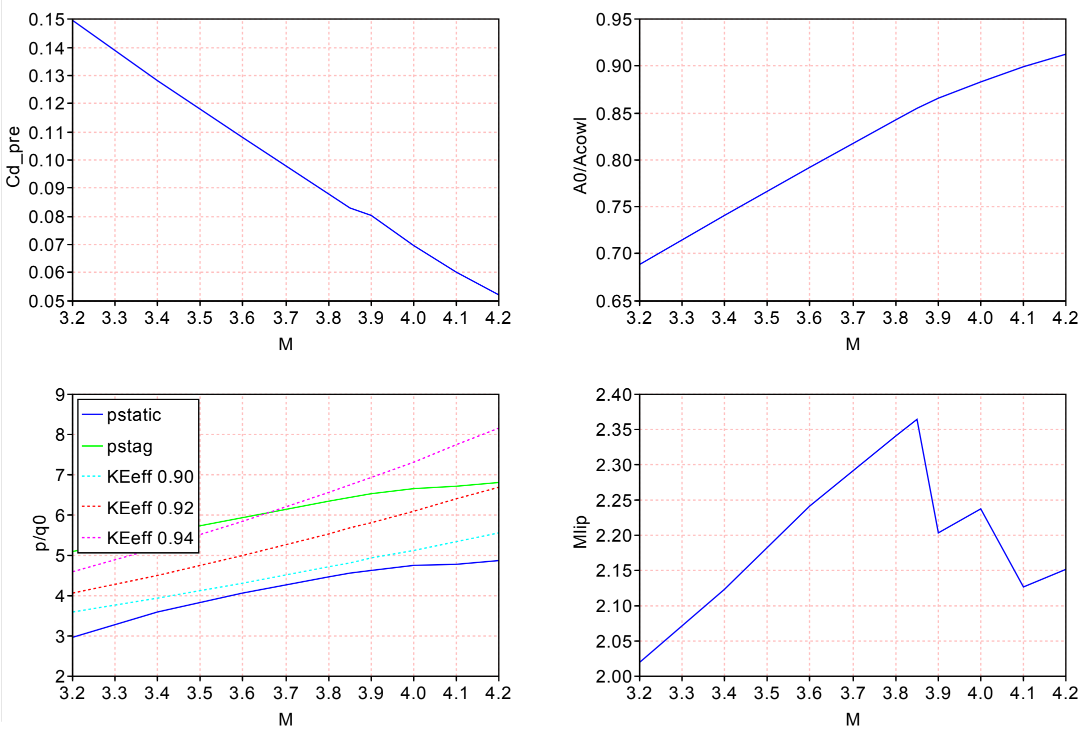


Figure 5: Calculated E4 intake performance

Pre entry drag coefficient (based on cowl area), mass capture ratio and lip Mach number are presented in figure 5. The significance of lip Mach number has just been discussed, but note the discontinuous drop as the slip line intersects the lip and the noise (random component) in the lip Mach number plot at flight Mach numbers greater than 3.85. In the present calculation the discontinuity in entropy at the slip line has been allowed to numerically diffuse through the flow, in a non-physical way, that can only be excused on the basis that it is computationally convenient and has no bearing on the key lessons to be drawn from the E4 study.

The increase in mass capture (A_0/A_{cowl}) as the ramjet accelerates was a major design consideration for Trommsdorf [2] as it allowed the ramjet with its fixed nozzle throat to be running at near optimal conditions over the Mach number range. Pre entry drag was a penalty worth paying in order to keep the intake operating near critical as will be discussed in the next section.

Isolator static and stagnation pressure, calculated from the subsonic root of equation 4, are also presented in figure 5. As are curves for kinetic energy efficiency defined as the square of the ratio of exhaust velocity to free stream velocity, with exhaust velocity calculated by assuming the captured air is expanded isentropicaly back to ambient pressure with no prior heat addition (or subtraction). The range of 0.90 to

0.94 would encompass most ramjet intakes with 0.92 being typical for an intake with no boundary layer bleeds and the compromises required for reduced cowl drag, self-starting, and the ability to operate at angle of attack. The E4 curves demonstrate why kinetic energy efficiency is a good descriptor as a fixed geometry intake tends to have a constant efficiency over its Mach number range. The difference between a good intake (0.94) and a mediocre one (0.90), amounts to the difference between static and stagnation pressure in the E4 isolator. Thus the subsonic diffuser can play a significant role in intake performance. It is important to recover a good percentage of the subsonic head, particularly when the solution to equation 4 is approaching sonic and there is a large difference between stagnation and static pressure.

The decline in kinetic energy efficiency evident in the E4 pressure curves as it accelerates is due to the influence of the triple point, and at high M_0 the swallowing of flow above the slip line. But we shall now see that this should have had no influence on the projectiles performance.

2.6 Back pressure and c^*

Intakes can only be understood in relation to the engine they are designed to feed. The quantity c^* defined by,

$$c^* = \frac{p_c A_{nt}}{\dot{m}_c} \qquad [6]$$

neatly expresses the relationship between the mass flow, $\dot{m}_c$, exhausting through the nozzle throat (area A_{nt}) and the combustor stagnation pressure p_c.

Calculation of c* is a problem of equilibrium chemistry, and Gordon and McBride's Chemical Equilibrium Analysis (CEA) program [5], which is widely used and free to download, greatly simplifies this task. Recognising that the mass leaving the combustor is the captured air mass plus the fuel in a proportion described by the fuel air ratio, *fa*, equation 6 may be written as,

$$\frac{p_c}{q_0} = (1+fa)\frac{2 A_0 c^*}{A_{nt} u_0} \qquad [7]$$

CEA results for burning carbon disuplhide in air at 28bar and with stagnation temperatures corresponding to sea level flight are presented in figure 6. Chemical equilibrium, and c^*, are not overly sensitive to pressure and provided one guesses the right order of magnitude, its actual value can be calculated from equation 7, using its estimated value within the calculation of c^*.

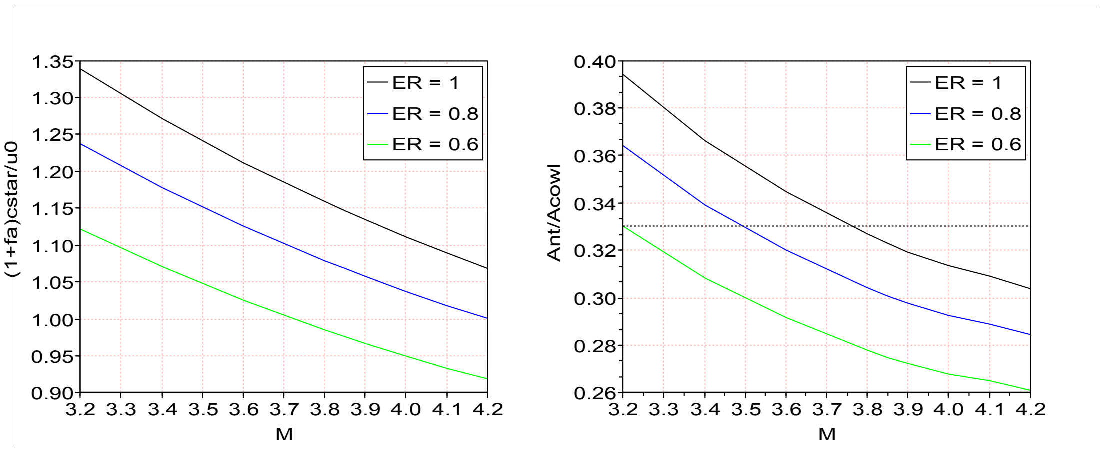


Figure 6: E4 combustor model

The left hand side of figure 6 utilises no information about the intake, but is proportional to p_c/q_0 when A_0/A_{nt} is constant. In that case, p_c/q_0 would have decreased by approximately 20% as the E4 accelerated from M=3.2 to 4.2, unless equivalence ratio was increased. Increasing ER from 0.6 to 1 would have held p_c/q_0 constant but this does not exploit the full potential of the E4 intake which was capable of tolerating a 40% increase in back pressure over this Mach number range as evident in figure 5.

The right hand side incorporates the Mach number dependence of the ratio of capture area A_0 to cowl area, and intake p/q_0 capability presented in figure 5. For p_c/q_0 it is assumed that 80% of the difference between static pressure and stagnation pressure is recovered in the subsonic diffuser. One normally would consider pressure drop across the flame holder and the pressure drop due to heat addition but that detail adds nothing here.

The right hand side of figure 6 is the objective of this E4 study, because it illustrates what is arguably the most important aspect of any ramjet intake design. The ratio of nozzle throat to cowl area is normally fixed. Let us assume that in the case of E4 the value was 0.33 as drawn on the figure. In that case adding fuel at an equivalence ratio (ER) greater than 0.6 would unstart the intake at M=3.2. However as the projectile accelerates ER can be increased, rising to 0.8 at M=3.5 and 1 at M=3.75. At higher Mach numbers the back pressure applied to the intake even with ER=1 is lower than the intake is capable of delivering. The intake is said to be running super critically and in this mode the terminal shock does not sit within flow determined by the supersonic solution of equation 4 (if it did, conditions downstream simply correspond to the subsonic solution) but moves downstream in the diverging subsonic diffuser to where the Mach number is higher and the shock losses will be just that required for the stagnation pressure to satisfy equation 7.

Building on the fundamental studies at Braunschweig and Gottingen, Trommsdorf added perhaps the most important factor that must enter the design compromise and that is that the manner in which mass capture varies with Mach number as the ramjet accelerates should be tailored to maintain efficiency. Defining the optimum mass capture characteristic is made possible by simulation of the flight. Total fuel burn to achieve a given state, is one metric by which the coupled problem of: thrust requirement; pre-entry drag; cowl drag; pressure recovery; angle of attack requirements; and even structural weight implications can be judged and subsequently optimised.

2.7 Lessons drawn from the E4 study

Although created in a time when there was little prior art, the intake for Trommsdorf's E4 is remarkably sophisticated, and serves to illustrate the following design features:

- Mass capture characteristics are tailored by appropriate positioning of the cowl lip relative to the cone tip and biconic junction;
- The degree of turning is set by the lowest flight Mach number, and in particular by the manner in which the flow interacts with the cowl lip;
- Pressure recovery at high Mach number was compromised by the previous two constraints with no effect on system performance, because back pressure is determined by the engine and this was sufficiently low.

3 ROLLS ROYCE THOR AND ODIN

3.1 Bloodhound

Bloodhound is a British surface to air missile that entered service in 1958 as the Mk 1, to be superseded by the more capable Mk 2 in 1963. The 180km range Mk 2 was powered by two Thor BT3 ramjets, figure 7, mounted above and below the body, which accelerated the vehicle from a boost Mach number of 2.15 to cruise at Mach 2.5 [6]. The Bloodhound used twist to steer and a variable incidence wing which in theory kept the angles of attack low for the ramjets even during intercept manoeuvres. In practice, fuel flow to the leeward ramjet had to be reduced at angles of attack greater than 4° in order to avoid combustion instability [7]. This is an example of an installation problem and not too surprising if familiar with the flow over cylinders at incidence. A second problem revealed during test flights, that has wider implications for intake design and in particular the design of subsonic diffusers, would be difficult to predict without the benefit of the hindsight afforded by the bloodhound experience.

Figure 7: Thor BT3, courtesy of Rolls Royce Heritage Trust

3.2 The BT3 intake flow field

The BT3 has a 24°/31° biconic intake as measured from the inset photograph in figure 8. MOC flowfield computations reveal a textbook design, with a design Mach number of 2.5. Both shocks appear to be focused on the lip at the cruise Mach number. Tailoring mass capture is clearly less significant when the flight Mach number range is limited. Cowl drag and weight would have been primary drivers in this podded engine application, and these encourage the use of a short radius of curvature for the isolator (to minimise drag) and a relatively high rate of divergence in the subsonic diffuser (to minimise length and weight). At M=2, the streamline angle at the lip is 22° and the local Mach number is 1.28. The angle has a direct effect on cowl drag and, as will be shown, the lip Mach number sets the minimum isolator radius and thus controls the projected area of the cowl. The choice of the second cone angle is likely to have been determined by these lip parameters. Following Oswatitsch's principle, one might have expected a more slender first cone, but there is only a 2.5% drop in total pressure across the first shock (at M=2) and the angle of attack sensitivity is much reduced with high cone angles.

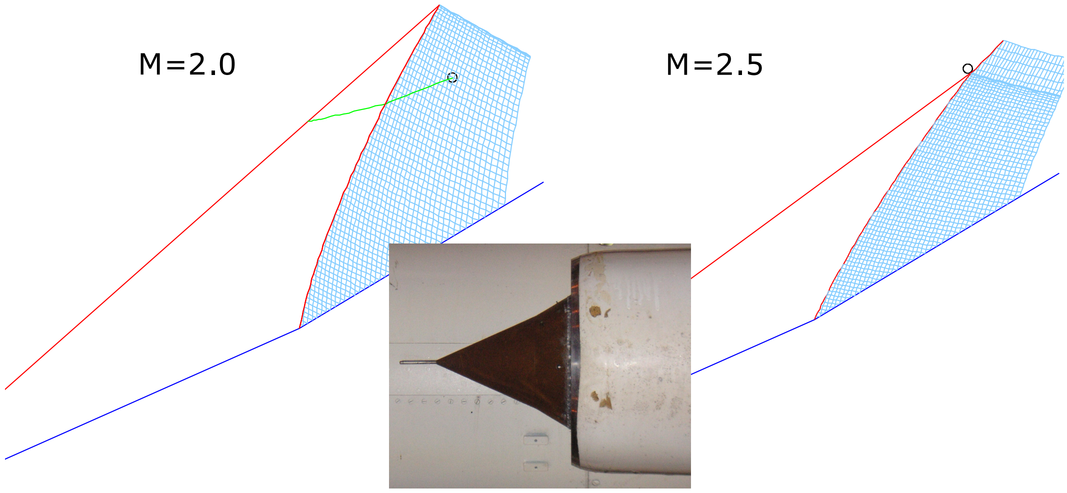
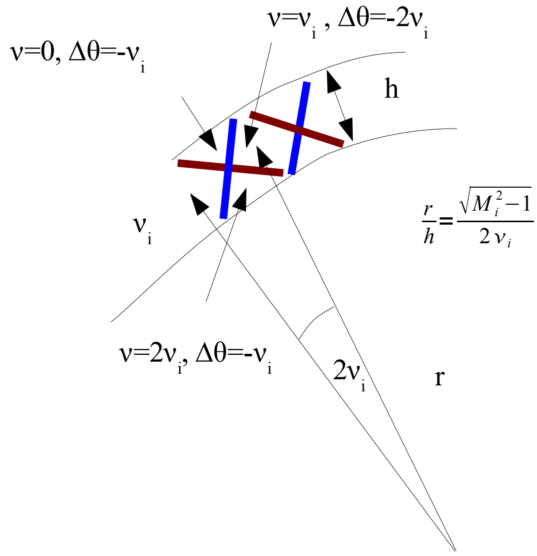


Figure 8: MOC solutions for the Thor BT3 24°/31° biconic, the circle marks the cowl lip position

3.3 Isolator curvature

The isolator, also known as the intake throat, is a section of near constant area that links the supersonic and subsonic diffusers. At maximum back pressure, the isolator contains the terminal shock system that converts the supersonic root of equation 4 to the subsonic root. These roots are related by identical mass flow, energy and stream thrust and therefore they are connected in precisely the same way as the conditions upstream and downstream of a normal shock. No matter how complicated the isolator flowfield, the upstream and downstream states are essentially linked by the Rankine Hugoniot relations. Only wall forces and heat transfer can alter this relationship and a small decrease in stream thrust due to skin friction is unavoidable, but clearly minimised if the isolator is kept short. Losses due to wall pressure forces are more difficult to estimate except in the trivial (but not uncommon for scramjets) case in which wall pressure forces do not have a component in the direction of the inlet/outlet momentum balance.

Bloodhound's short, high-curvature isolator is typical of intakes designed for Mach 2 to 3. Such intakes normally exhibit a region of stable subcritical operation when tested in isolation, but it would be very surprising if any ramjet intake was allowed to operate this way when coupled to a combustor. Allowing air mass flow and pressure to be coupled to fuel flow and combustion would simply be inviting instability. Therefore we should expect a supersonic/transonic flow within the curved isolator and wall pressure forces to play an important role. Wall force is required to increase the axial stream thrust from $F_l\cos\theta$ (equation 2) to F_l as the flow is turned horizontal, if the potential pressure recovery at the cowl plane is to be realised after the turn. One could imagine with a sufficiently large turn radius the difference between surface pressures on the spike and cowl induced by the centrifugal acceleration of the air flow would result in precisely the required thrust. However in order to minimise the external drag the designer will try to turn the flow as tightly as possible without significantly compromising the pressure recovery. The empirical rule that the radius of curvature should be a minimum of four throat heights [8] has limitations and a physical basis that are revealed by the following analysis.



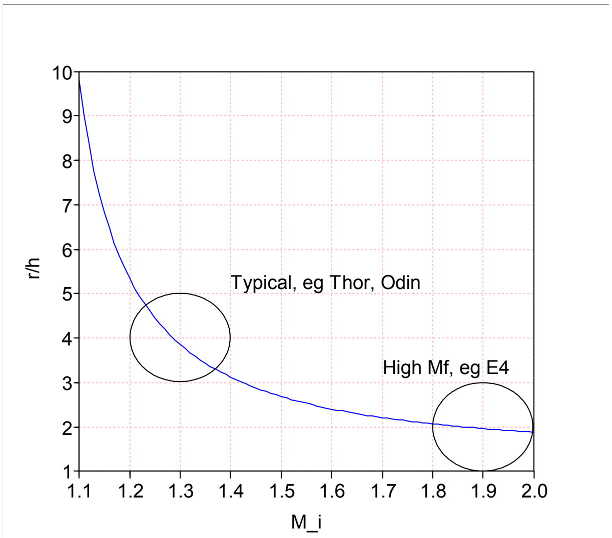


Figure 9 A model to determine the curvature at which an isolator chokes

Consider the flow entering the isolator at Mach number M_i and corresponding Prandtl Meyer angle υ_i in figure 9. If at the entrance both the cowl and spike are given a small deflection $\Delta\theta = -\upsilon_i$ through a weak shock (red line) and expansion (blue line) respectively, then the flow near the cowl will be at Mach 1, the minimum allowable if the decoupling function of the isolator is to be maintained, and the flow near the spike will be at a Mach number corresponding to $2\upsilon_i$. Downstream of the intersection of the shock and expansion, in the region where the flow has encountered both, the Mach number is returned to M_i but the streamline angle has been reduced by $2\upsilon_i$. This process takes a distance set by the gap height, h, and the average propagation rate of the trailing edge of the expansion fan. Since the shock and fan trailing edge meet close to the spike surface, the trailing edge propagates most of the way at the Mach angle for M_i and the shock expansion process takes a distance of approximately $h\sqrt{M_i^2-1}$ to complete. The streamline angle has changed by $2\upsilon_i$ over this distance and the radius of curvature r is therefore,

$$\frac{r}{h} \approx \frac{\sqrt{M_i^2-1}}{2\nu_i} \qquad [8]$$

This function is consistent with the “four throat heights” rule of thumb when the entrance flow Mach number is 1.3 which is typical for intakes designed for flight at Mach 2 to 3 (figure 9). Note the functions strong sensitivity at low Mach number has implications for operation at angle of attack, because the extra compression on the windward side can easily lead to subcritical operation if the combination of isolator curvature and nominal M_i are too close to the limit for zero angle of attack.

3.4 Subsonic diffuser

A flow straightener is a surprising feature to see in a ramjet engine, particularly one that would appear to be unnecessary given that it is placed just upstream of a colander flame holder, as revealed in the sectioned Thor engine of figure 7. One might have expected that the head loss across the flame holder, needed to drive the fuel/air mixing and stabilise the flame, would have been sufficient to encourage flow uniformity. Fuel is introduced via the radial spray bar visible just downstream of the honeycomb flow straightener, and the fuel air mixture must pass through the square cut outs in the conical flame holder. Flame stabilisation is achieved by leaving the tab formed by cutting three sides of the square hole, bent internally and hanging from the downstream forth side- a rather beautiful piece of practical engineering. Indeed, the straightener was only introduced as a fix to the problem encountered during bloodhound's acceptance trials. As the engine was throttled for cruise, the intake operated deep in its supercritical regime with a strong terminal shock sitting in the subsonic diffuser. This resulted in flow separation and a narrow high velocity stream, disrupting the distribution of fuel within the air.

The flow straightener solved the problem by virtue of having a head loss that is proportional to local dynamic pressure: a narrow high velocity stream results in a high pressure drop and this feeds upstream to reduce the size of the separation and widen the stream. That is how flow straighteners (also known as aerodynamic grids) work, but its function in Thor had two additional attributes. The first was that its presence resulted in the intake running closer to critical in cruise and thus reducing the strength of the terminal shock, however during acceleration when the intake was operating critically it did not have a strong adverse effect because in that state, the Mach number (and dynamic head) at the straightener was very low. The second attribute is associated with a rather cunning integration of the pilot flame air supply into the base of the straightener. The annular gap at the inner diameter of the straightener feeds air to a centrally located pilot flame and the proportion that passes to the pilot increases with the head loss at the straightener. This allowed the cruise fuel requirement to be fed to the pilot, allowing stable combustion within a primary stream prior to mixing with the secondary stream in the main combustor.

It would be remiss not to mention that the main concern of subsonic diffuser design is preventing boundary layer separation in the adverse pressure gradient, and that keeping diffusion rates equivalent to that within a 3° to 5° half angle cone has proved effective. This is a problem common to many applications of fluid mechanics and the evidence for this result is presented in textbooks such as that by Massey [9]. The bloodhound experience adds to the story, by reminding us that the diffuser must also provide an acceptable level of uniformity to the combustor when the intake is operating supercritically, and that when the back pressure applied by the combustor is low, it may be possible to exploit the excess pressure that an intake can provide to help optimise the airflow within the combustor.

3.5 Seadart

Seadart is a much smaller surface to air missile than Bloodhound, having been designed to be stored vertically between decks of the Royal Navy Type 42 destroyers. It is boosted to Mach 2^+ (here we assume 2.1) and can accelerate to Mach 3 but the flight Mach number is a function of total temperature (an airframe limit) and thus depends on altitude [6]. Seadart's intake is an isentropic spike integrated with the missile forebody. A central air transfer duct feeds the Odin engine at the rear. The central schematic in figure 10 shows the spike is part of a large central body that contains the warhead and this dual function combined with the need to create missile volume allowed a long isolator with limited turning to be combined with a subsonic diffuser of low divergence rate. The lower right figure is the combustor viewed through the nozzle, showing an inverted flame holder in comparison to that of Thor. The main airflow passes from the central air transfer duct through the colander which generates longitudinal vortices. Fuel is injected through radial spray bars near the transfer duct exit. The pilot zones are hidden behind the small outward facing tabs at the base of the colander. The multiple small holes in the combustor liner provide cooling air which, like the pilot air is drawn from the outer edge of the transfer duct.

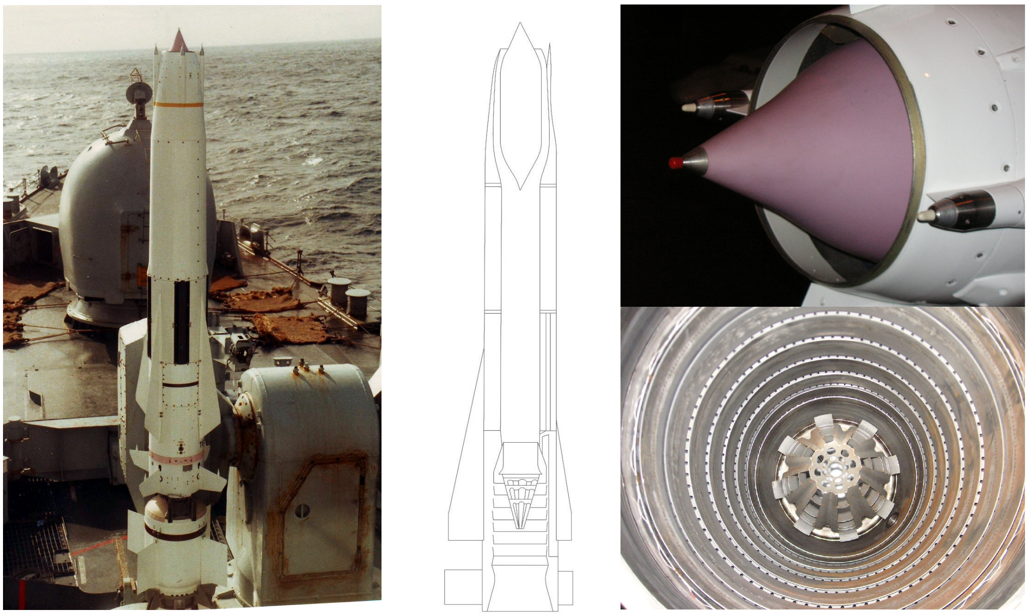


Figure 10 Seadart. Left hand photograph copied from Wikipedia.org, right hand photographs were taken at the Kemble museum, UK.

3.6 Seadart intake flowfield

Measurements made from photographs reveal the intake has much in common with Thor's, having a 24° fore cone and a cowl lip positioned along the ray that is very close to the bow shock angle at Mach 2.5. Placing the focus of a Prandtl Meyer fan at the cowl lip and turning the flow to 24° there, develops the isentropic turn bringing the surface angle to 33°. The resulting contour, figure 11, matches that measured from photographs. Thus the design appears to be another textbook example, with the design Mach number chosen mid range, and both the shock and fan focused on lip at that Mach number.

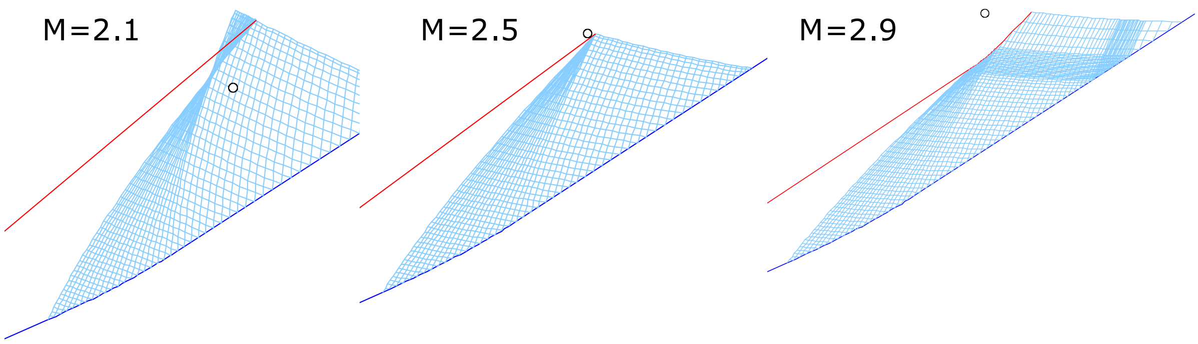


Figure 11: MOC solutions for the Seadart 24°/33° isentropic spike, the circle marks the cowl lip position

An incomplete MOC solution for Mach 2.1 is shown on the left hand side of figure 11. The left running characteristic originating at the end of the turn has overtaken the one originating at the start. Wherever a characteristic catches another of the same family it merges to form a weak shock which propagates at the mean of the upstream and downstream Mach angles, that is, it bisects the characteristics on which it was formed. Away from its origin, as the shock grows in strength, the shock angle is best determined directly from the Rankine Hugoniot relations, while ensuring compatibility with the downstream flow field. An example of this is given later. For current purposes, the solution is left incomplete as it provides a revealing physical picture of the compression process above an isentropic spike when operated below design Mach number. The right hand figure for Mach 2.9, shows the flow field at above design Mach number, and no such complication occurs.

At Mach 2.1 the calculated mass capture is 85%, and the flow at the lip is at 24° and Mach 1.29 while the Mach number at the surface is closer to 1.24. Judging from figure 9 the radius of curvature should be between four and five throat heights to maintain supersonic flow, and this is at least consistent with the external views of the intake. Thus on the evidence available there is little to distinguish the Thor and Seadart intakes at low Mach number or suggest an improved angle of attack capability. Further MOC analysis indicates that below Mach 2.0 the intake runs with the isolator either choked or subsonic, depending on back pressure. This is due primarily to the high degree of external turning and is a phase all ramjets must pass through during boost. The tandem Seadart booster attachment is designed to allow through flow and ignition during boost (the ignitor breech is in the 4 o'clock position in the combustor photograph of figure 10) and this probably occurs with the isolator choked.

At Mach 2.9 the flow at the cowl entrance has an entirely different character and it is in this state that one should expect robust performance with tolerance to high angle of attack. The bow shock passes inside the cowl lip but the majority of the captured mass is compressed within the shock layer around the spike. Free stream air is also passing through the entrance and must undergo all its compression internally. Given that the internal lip angle is probably only slightly less than 24° the air would first be expanded around this corner were it not for the internal separation bubble that forms in such cases. The isolator flow, as normal, is complicated by shock boundary layer interactions and flow separations, but has this marvellous ability to adjust itself to make the outlet compatible with the inlet provided the isolator is made long enough and its divergence rate is zero or very low. This flexibility is only exhibited when: entrance Mach number is high enough that the flow is not choked by curvature; and stream thrust, F_I, is sufficient to tolerate the applied back pressure.

3.7 Lessons drawn from the Thor and Seadart studies

Two lessons from these related ramjets are:

- Separations within the subsonic diffuser can result in unacceptable air flow distribution to the combustor. Thor required a flow straightener to fix the problem it encountered in acceptance trials. Seadart adopted a very low divergence rate in the subsonic diffuser, akin to having an abnormally long isolator. The thought behind this approach was that if isolators can contain strong shocks without severe flow distortion then so should subsonic diffusers of sufficiently low divergence, and the approach proved effective [7].

- Angle of attack capability is affected by cross flow on the spike and this effect is reduced by choice of a sufficiently large fore cone angle, and operation at high Reynolds number. However a more fundamental limit is associated with choking of the isolator when entrance Mach numbers are too low. If the windward side flow is compressed to too low a Mach number, then the flow will choke if subject to either internal contraction or isolator curvature and the intake unstarts. Because of this an intake designed to operate over a Mach number range, will have better angle of attack performance at high Mach number.

4 VARIABLE GEOMETRY AND BLEEDS

4.1 Introduction

The intakes examined above are all fixed geometry supplying engines with fixed nozzle throat areas. None had boundary layer bleeds, primarily because they didn't need them. Although skin friction was taken into account in the calculation of stream thrust and E4 intake performance, no adjustment was made to account for boundary layer displacement thickness in any of the flowfield calculations. This neglect was based on convenience and a desire not to introduce unnecessary complications. However, in practice complications are often unavoidable and various measures and ingenious devices have been devised that allow intakes to function over large Mach number ranges and at relatively low Reynolds numbers. The NASA Hypersonic Research Engine, the BAC/SUD Concorde and the Lockheed SR-71 will serve as an introduction to this aspect of intake design.

4.2 NASA Hypersonic research engine

The Hypersonic Research Engine (HRE) project to design, develop, construct and flight test a high performance ramjet/scramjet was an exceptionally well documented research programme that provides a rare (perhaps unique) view of intake development from engine concept to trials review. Recommended reading for this purpose are: the project review by Andrews and Macklay [10] as an introduction; followed by the AiResearch report [11] on their engine concept; the AiResearch final report on the intake programme [12]; and finally the analysis of the intake experimental results by Andrews and Macklay [13].

To some extent, the ready availability of this documentation makes commentary on the design process redundant. However continuing with the theme of the lecture we will look at particular details of the final (phase II) intake with the objective of learning something general. Fortunately in this case reverse engineering is not required.

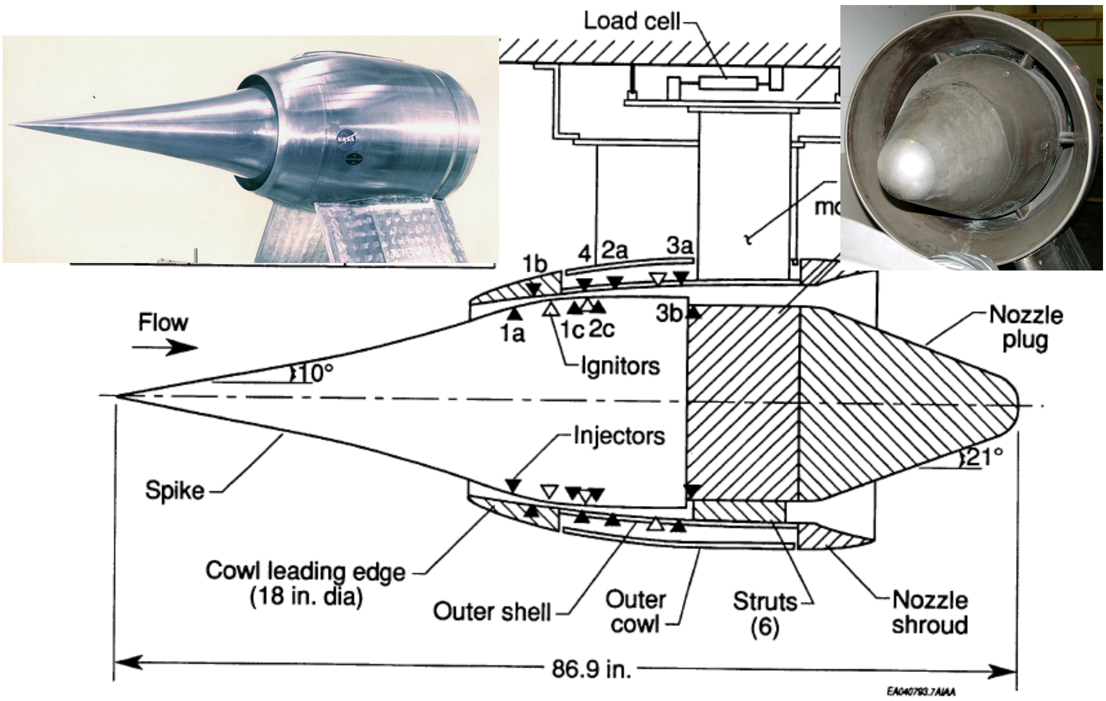


Figure 12: The Hypersonic Research Engine, courtesy of NASA

4.3 Tailoring mass capture and contraction ratios

The HRE intake has a translatable spike and a 5.645° (not measured from a photograph!) up sloping throat. The combination of the two allows the throat area to be varied. In most translating spike intakes, the throat is formed between the rear, down-sloping, surface of the spike and the cowl, and the spike is retracted to reduce the throat. On the the HRE the spike was extended to reduce the throat, while simultaneously allowing shock on lip to be maintained from Mach 6 to 8. The appropriate sign and magnitude of the slope depends on the mass flow characteristics of the engine. Turbojets are best supplied by an intake that increases both capture area and contraction ratio as Mach number increases, as will be evident from the Concorde and SR-71 discussion. The HRE intake/engine coupling was unusual and more intimate: the gap between the cowl and spike downstream of the throat was the scramjet engine, while that upstream was the intake, and both geometry vary as the spike translates.

For the HRE intake to have full capture from Mach 6 to 8 and simultaneously increase contraction, the up sloping throat was an elegant solution that also allowed the intake to meet the requirement that it be closed during the acceleration phase of the test flight. The spike was simply extended until the throat area was almost zero which occurred near the cowl lip. One disadvantage of an up sloping throat is high cowl drag as the engine cowl has to grow to accommodate the growing spike (and its matching internal contour). While the HRE is sometimes portrayed as a naïve design that failed to recognise the significance of cowl drag, maximising net thrust by minimising cowl drag was not a project requirement. The objective was to demonstrate good internal performance in a pod that could be tested on the X-15 [10], and the intake reflects the requirements and constraints it was designed to meet [12].

An unusual feature of the spike contour for the final intake "T", was the use of two distinct isentropic turns each with its own focus and design Mach number. The phase-I contour had been a 10° cone with an isentropic turn to 20.5° focused on the lip at Mach 8. This intake was found to have insufficient contraction at Mach 6 and 8 and too much at Mach 4. A lack of contraction is potentially easy to rectify but since the spike was to remain fixed at the Mach 6 position for flight between 4 and 6 (to maintain the engine geometry), increasing contraction at Mach 6 while simultaneously decreasing it at 4 necessitates a change in contour that will result in increased spill at Mach 4 while still retaining full capture at 6. This is the same challenge Tromsdorff faced and the Lockheed engineers (on subcontract to AiResearch) found a similar solution. They moved the focus of the turn to the cowl lip at its Mach 6 position, and turned through the angle (5.8°) to produce the required spill at Mach 4. Further turning was needed to achieve the required contraction at all Mach numbers, and this was delayed until the last point possible without requiring high internal contraction at Mach 8, and so the second turn was focused on the Mach 8 lip position at Mach 8.

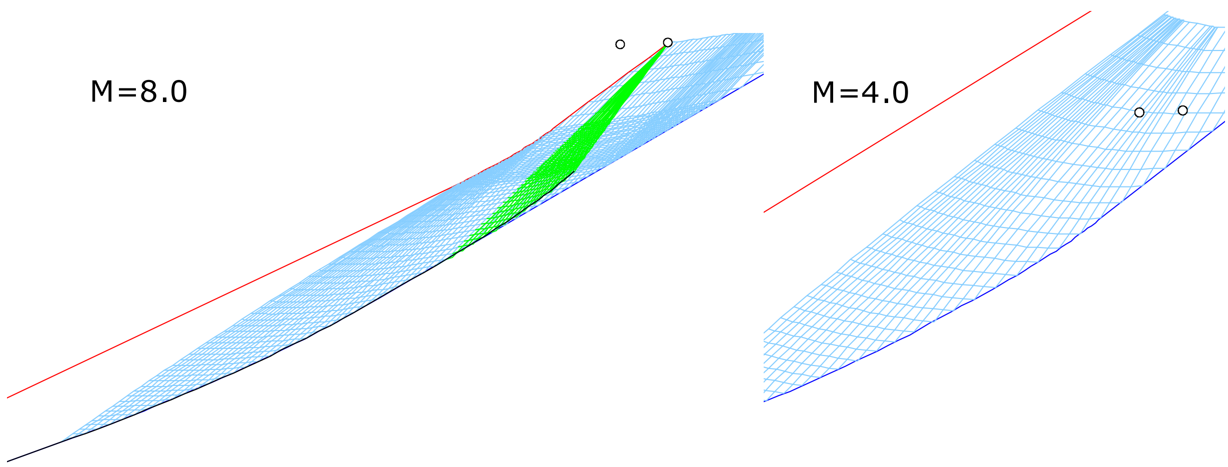


Figure 13: MOC solutions demonstrating the development and function of the intake "T" contour (see text).

The design process and its Mach 4 result are illustrated by the MOC solutions in figure 13. The first step in the process, the generation of the Mach 6 turn, is omitted as this is substantially the same as the lip focused turn for Seadart, in figure 11. Downstream of the turn the contour continues as a straight conical section, and this is the blue line that is the lower boundary on the left side of figure 13. The flow over this contour at Mach 8 is then calculated with MOC and that generates the light blue mesh on the left. The compression from the Mach 6 turn is strengthening the bow shock and deflecting it away from the surface. The new Mach 8 lip position is determined by where the shock reaches cowl radius, note that it is forward of the undeflected shock position reducing the stroke required for the spike. This had the disadvantage that the throat angle (up-slope) had to be increased to meet the specified throat area variation between Mach 6 and 8.

The third step in the process is to identify the left running characteristic that intersects the shock at the new Mach 8 lip position, as it forms the upstream boundary of the flow through the next turn. The flow downstream of this characteristic, calculated in step 2, will no longer be valid. A Prandtl Meyer fan is centred on the lip and traced back to find the new surface, continuing the turn until the required external contraction is reached.

The flow over this contour at Mach 4 is shown on the right side of figure 13. The regions of influence of the two turns are made evident by the bunching of the left running characteristics at the top of the mesh. The second turn has not influenced the spill at Mach 4 since the leading (upstream) characteristic from this turn intersects the lip in the Mach 4 position. Thus confirming the strategy of setting Mach 4 capture ratio by the degree of the first turn. The second turn does influence Mach 4 spill with the lip in the Mach 8 position and this proved to be very important. The intake could not be started with the lip in the forward position because the internal contraction was well above the limit established by Oswatitsch (and later Kantrowitz). To start the intake, the spike was extended to reduce internal contraction, and then retracted to the running position once the intake had started. A major obstacle to the design was that the ratio of capture area to throat area was not monotonic and at Mach 4 it peaked between the starting and running positions. Endeavouring to keep the peak below a value of 6, which was regarded as a safe working limit, was a serious challenge that was never quite satisfied. The phase-II T intake, just described, had a peak capture-to-throat ratio of 6.1 at Mach 4, and its starting proved unreliable and very sensitive to wall temperature (a boundary layer effect). Most of the other contours examined by the Lockheed engineers had much higher peaks, and intake T's lower value is due, in part, to the second focus.

4.4 Concorde

A turbojet places different demands on an intake than those of a ramjet, as noted in the previous section. But if a ramjet were to be part of a combined cycle propulsion system, and/or was required for an application where the mass of fuel used was sufficient to favour engine efficiency over structural weight saving, then some of the sophisticated features of turbojet intakes would likely be adopted for the ramjet. The Concorde intake has set a very high standard, having high performance, being robust in operation, mechanically simple, and having an uncomplicated control system that was dormant for most of the flight. So it would make a very good case study, for anyone endeavouring to meet similar requirements.

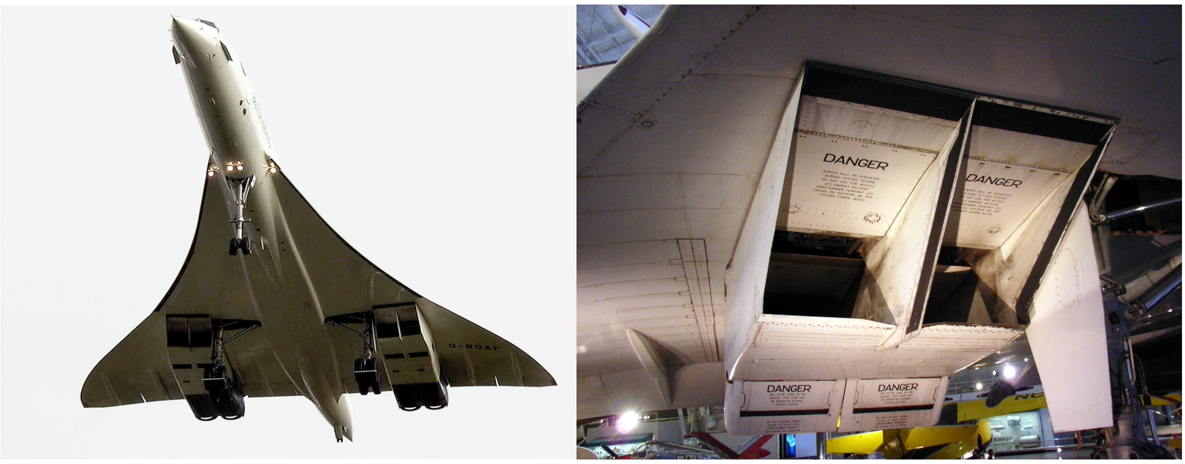


Figure 14: Concorde and its nacelle, courtesy of A. Pingstone and Wikipedia.org

Before Concorde's first flight on the 2nd March 1969, and eight years before it entered service in January 1976, Rettie and Lewis described the design and development of the supersonic transport's intake at the

10th Anglo-American Aeronautical conference. Their paper [14] gives a fairly complete description of the intake omitting some geometric details that have since entered the public domain [15-17]. The square intakes (unity aspect ratio) are mounted within under wing nacelles and consist of a 7° initial fixed ramp followed by a movable ramp that incorporates a 5.75° isentropic turn. At the design flight Mach number of 2.0 the second ramp is set to δ_2=9.6° turning the flow to a total angle of 15.35°. The internal cowl contour is a circular arc inclined at 12° to the under wing flow at the lip, requiring a 3.35° turn of the captured flow for shock attachment. The upstream section of the subsonic diffuser is a moveable ramp, hinged at its downstream end and its actuation is mechanically linked to the upstream (supersonic) ramp so the ramps move in sync. The gap between the ramps forms a wide bleed slot that on design, spans a distance of approximately 60% of the capture height, *h* and bleeds 6% of the captured mass.

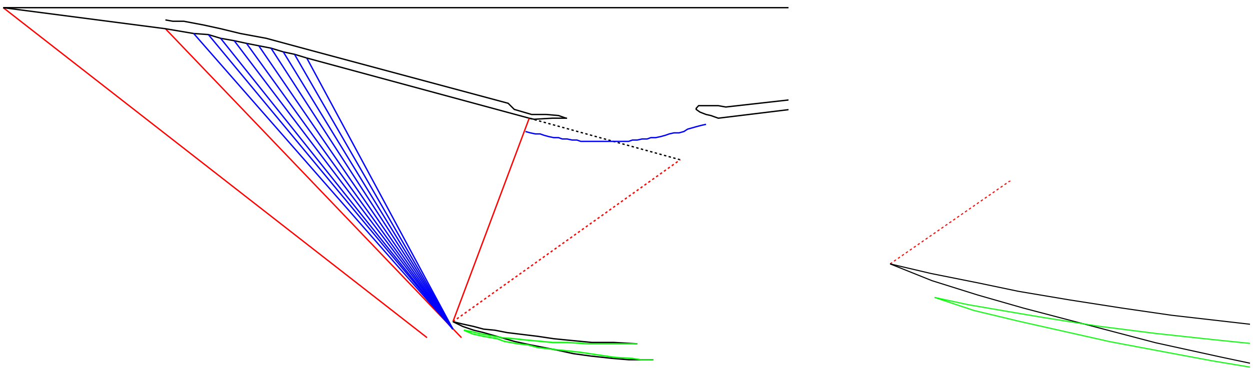


Figure 15: Concorde's intake flowfield on design at Mach 2.0, as calculated and drawn to scale. The enlarged detail of the cowl on the right compares the new and old contours.

The isolated intakes were tested at Mach 1.915 which was said to be a good approximation to the local Mach number for the outboard pair. The calculated two dimensional inviscid flowfield at this Mach number is drawn to scale in figure 15. The first ramp is positioned 1.43*h* ahead of the lip and the leading edge of the second is 0.52*h* downstream so that the hinge shock intersects the lip with δ_2=9°. The compression fan appears to have been focussed 0.02*h* directly above the lip when on design with δ_2=9.6°. The two cowl profiles drawn in figure 15 were obtained from SUD report C379 [17] describing aerodynamic improvements for the production aircraft that had been proven by 1978. The lowered and thinned cowl lip resulted in a 1100kg reduction in fuel burn on the London New York flight which is equivalent to approximately 11% of the payload (i.e. 11 passengers). The cowl profiles and the wide bleed slot are unique features that warrant greater attention and are the focus of the following section. However we shall first note some of the other essential elements of the intakes. The dump doors, open in the right hand photograph of figure 14, are actively controlled and opened to spill excess air when the engine is throttled for descent or should it have to be shut down. The dump doors also partially open on days 25K warmer than the international standard atmosphere (ISA+25), when the engine air demand is reduced by the high total temperature, and δ_2 would need to be increased above 12° in order to create sufficient fore spill. Greater ramp angles lead to excessive flow non-uniformity at the engine face.

On days between ISA+5 and ISA+25 the ramp angle is actively controlled between 9.6° and 12° in order to provide the required spill to match engine demand. On days with temperatures below ISA+5, which is 90% of the time on the intended routes, the intake runs supercritcally with the wide bleed passively accepting the difference between engine air demand and that captured by the intake. This appealing attribute of this type of bleed was studied in detail by Lenyaert at ONERA, and her AGARD paper [18] contains sketches of the transonic flow over the constant pressure slot and the manner in which the flow reattachment at the leading edge of the subsonic ramp adjusts to vary bleed mass. The subsonic ramps leading edge was also modified in 1981 as part of the intake improvement package [19]. The end profiles of the ramps drawn in figure 15 were traced from the Structural Repair Manual [19].

Although the design was focussed on cruise the intake must also satisfy the engine demands during take off, subsonic cruise and acceleration. The dump door contains a spring loaded flap that opens inwards to provide extra air during take-off and further aft on the nacelle there is another spring loaded flap that admits outside air to the engine bay when sub atmospheric. During subsonic cruise, pressure recovery in the bleed slot is sufficient that all the cooling air including that for the nozzle is supplied by the bleed. In subsonic flight the ramp angle δ_2 is set to approximately 2° and increased progressively during acceleration above Mach 1.3, as illustrated in figure 16b of reference [14].

4.5 Terminal shock strength

Pressure recovery at Mach 2 was 95% with ≈1% of the loss attributable to the subsonic diffusion, 0.5% to the first shock, 0.02% to the second, and 3.7% to the third and final, strong oblique shock. Loss of stream thrust via skin friction on the supersonic ramps is confined to the boundary layer which is extracted through the bleed slot. On design the bleed pressure recovery was 45%. If the forward ramp had been lengthened as indicated by the dashed line in figure 15, with a weak oblique shock to align the flow with the cowl and a normal shock to bring the flow subsonic the 3.7% loss could in theory have been reduced to 1.5%. Now since a 1% increase in pressure recovery would result in a 2.5% increase in payload [14] there was strong motivation to seek all possible gains, so the fact that this loss was tolerated highlights the significance of cowl drag, isolator curvature, nacelle length, and self starting requirements, in a way that no generic discussion could. After the isentropic turn the Mach number was 1.38 and with reference to figure 9 one might expect to turn the flow in an isolator with a radius of three throat heights. If the weak shock solution had been adopted the entrance Mach number would have been 1.26 and a radius of curvature of 4.5 throat heights is indicated. A cowl contour with a gradual turn back after an extended run at 12° would have been excluded very quickly, but compression to such low Mach numbers in order to minimise the terminal shock loss is a common working assumption of generic intake studies.

An additional advantage of a strong terminal shock is that the downstream Mach number is reduced, thereby reducing the potential subsonic diffusion losses. Concorde's subsonic diffuser had the relatively light job of decelerating the flow from Mach 0.76 to Mach 0.5 at the engine face, corresponding to a 23% increase in static pressure, putting the 1% loss of this process in clearer perspective.

4.6 The wide bleed slot

Three distinguishing features of the intake are: the relatively modest flow turning; the short forward ramp; and the wide bleed slot. The motivation for the first two is explained above but the practical implementation is intimately connected with the third, ingenious device. A bleed gap between the subsonic and supersonic ramps, simplifies the mechanisms of both, dispenses with the need for a flexible surface, and removes the boundary layer upstream of the terminal shock to increase shock stability and decrease flow distortion. In these respects Concorde's intake has much in common with those of fighter aircraft like the F-4 and F-15. The unusual feature is the length of the gap, which spans the full turn of the isolator (and cowl). Since flow momentum within the bleed space is negligible, it imposes a constant pressure on the shear layer bridging the two ramps, with the pressure level being a function of the bleed mass flow and bleed outlet area. (Concorde's bleed control area is set by matching with the primary exhaust jet in the dual stream nozzle [14].) The effective contour presented by the bleed gap naturally adjusts itself to simultaneous satisfy the main duct flow and bleed flow, without inducing large losses in the former. This is particularly important in the supercritical regime, allowing engine mass flow to increase in response to a decrease in atmospheric temperature, rather than accepting the decrease in pressure recovery that results if constant mass flow is maintained to the turbojet.

Another favourable effect of the wide bleed is its demonstrated ability to accommodate a high curvature cowl. The modified cowl in figure 15 maintains the 12° initial angle of the original but the radius of curvature is reduced from 3.4*h* to 2*h* (throat height is 0.69*h*). The simple theory developed in section 3.3 can't tell us how tightly subsonic flow may be turned without inducing high losses, but since it only takes a 6% stream tube contraction to accelerate the flow from Mach 0.76 to sonic, it seems likely that tightly turning high Mach number subsonic flow would result in choking and increased spill just as it does at low supersonic Mach numbers. In fact at ramp angles of 11° and 12° the lip shock is detached [15], despite the fact that ramp Mach number remains sufficient for attachment, and the intake spills about twice the mass flow it would with attachment. Note that a choked isolator does not mean the intake is running subcritically, this only occurs when the back pressure is sufficient to influence the spill. The spill due to shock detachment is a positive attribute because the ramps are only set at these higher angles in order to force spill on warm days. It would be interesting to know if the decreased radius of curvature of the new cowl did force shock detachment on design with the 2% increase in capture height compensating for the loss. In any case, it is clear that there was a significant decrease in cowl drag and that the compliant wide bleed slot was able to accommodate the new contour.

4.7 Blackbirds A-12, YF-12, SR-71

Designed to fly at Mach 3.2 at altitudes from 75 to 85kft for reconnaissance, the existence and purpose of the single seat CIA's A-12 and its close relative the two seat USAF SR-71, figure 16, were initially kept secret. A cover story was developed in which only the interceptor version of the aircraft was revealed when it was thought necessary to explain the public expenditure and aircraft sightings [20]. Three two seat interceptor versions, designated the YF-12, were built in comparison to twelve A-12 and thirty two SR-71. Technical articles relating to the aircraft type generally refer to the YF-12 (or F-12), but as far as propulsion is concerned there is no (known) difference between the types.

According to recently released reports, the intake (including its control system) was the single most important problem pacing the flight development [21]. Kelly Johnson says in his history of the Oxcart program "*Before we had a usable inlet, we had to collect two million data points in the wind tunnel, and later we had to do at least that many in flight*" [22]. The mice which are the mouse shaped lumps downstream of the throat visible in the right hand photograph of figure 16 were introduced some two years after the first flight, in order to control "duct roughness at Mach 2.4" [21]. Presumably this refers to flow distortion and the mice probably improved the subsonic diffusion by reducing the rate of expansion.

Figure 16: An SR-71 at Edwards courtesy of NASA and a close up of the spike and cowl internals of the YF-12 in the USAF museum courtesy of J. Kurzke.

Some wind tunnel and flight data, were published relatively early on, providing schematics and surprisingly detailed descriptions of the intake function and control system but omitting geometric details [23,24]. Additional information from the flight manual [25], some NASA reports [26-29], and photographs of the aircraft now on static display, make it possible to obtain a more complete picture of the intake. The purpose being, as with the other studies, to learn something from a successful applied design.

The general arrangement as described by Campbell [23], the flight manual and elsewhere, is as drawn in figure 17. The intake throat is formed between the rear inward sloping surface of the translating spike, and the internal contour of the cowl. The spike is retracted as Mach number increases which simultaneously increases capture area and decreases throat area, thereby increasing stream tube contraction. The spike boundary layer is extracted through a slotted surface near maximum diameter and passes through the centre-body support struts to be vented overboard. The cowl boundary layer is bled through a ram scoop known as the “shock trap”, passes through the engine compartment to provide cooling and then into the ejector nozzle, shielding the nozzle from the afterburner exhaust and contributing to thrust. The forward bypass is an actively controlled extraction of air from the subsonic diffuser through “doors” on the cowl surface located in between the mice. The forward bypass air is vented through the most upstream louvres on the nacelle, visible in figure 16, while the downstream louvres are those for the spike bleed. The thrust/ drag penalty of forward bypass was considerably greater than that of the shock trap air or the manually controlled aft bypass. To reduce the flow through the forward bypass the pilot could select one of three aft bypass open positions: 15%, 50% and 100%. According to Graham [31], the selection was based on the indicated position of the forward bypass doors. As the aircraft accelerated between Mach 1.7 and 3, the forward bypass begins to open excessively due to excess air mass capture with respect to engine demand and the pilot first selects 15% and then 50% open. By Mach 2.6 the forward doors begin to close tight and the pilot shifts the aft doors back to 15%. Above Mach 3.05 the intake capture and engine demand were well matched and the aft bypass was normally closed.

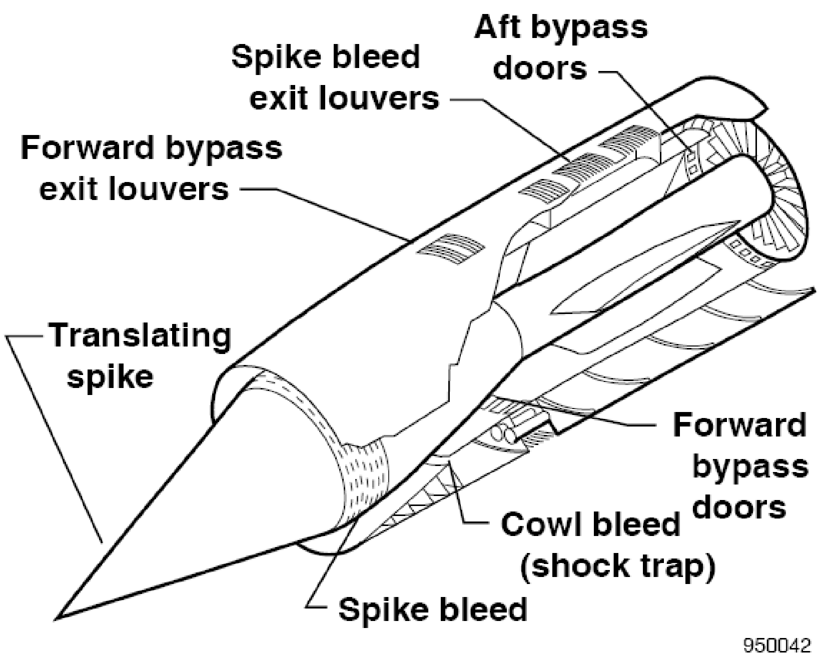


Figure 17: SR-71 intake general arrangement, figure courtesy of NASA [30]

This awareness of the intake function and the human interaction with it and its control system, explains in part why intakes are spoken of with apparent affection in accounts of supersonic aircraft. Decelerating air, exchanging momentum for pressure, and kinetic energy for internal energy, would be a much less interesting process were it not for the fact that the air within the lower regions of the boundary layer does not have sufficient momentum to negotiate the adverse pressure gradients. It is only viscous momentum exchange with the outer layer that makes it possible for the air close to the cowl and spike surfaces to be dragged further into the nacelle. When this is insufficient the boundary layer separates from the surface forming a recirculation bubble between what is now a free shear layer and the surface. Such bubbles tend to be unsteady and radiate acoustic noise through the air and surface. In Graham's account of flying the SR-71 he writes- “*As you became more experienced in the aircraft you could sometimes feel when an unstart was about to occur if the forward bypass door was closing down too tightly. A very subtle inlet duct rumble manifested itself throughout the airframe and gave you a clue that an unstart was imminent,* *unless the [forward bypass] doors were about to open up, or you took corrective action by shifting the aft bypass doors closed".* We can infer from this corrective action that high back pressure was not the source of instability because: (1) the intake control system would have sensed high pressure just downstream of the shock trap and opened the forward bypass and (2) if the separation was due to excess air (high back pressure), closing the aft bypass would be certain to trigger the unstart. This highlights the additional function of the forward bypass, both bypass regulate airflow to match intake to engine (albeit one is automatic), but the forward bypass is also an essential bleed within the subsonic diffuser, and at intermediate spike positions it needed to be kept slightly open even with the intake running supercritically.

4.8 Mixed compression with shock on shoulder

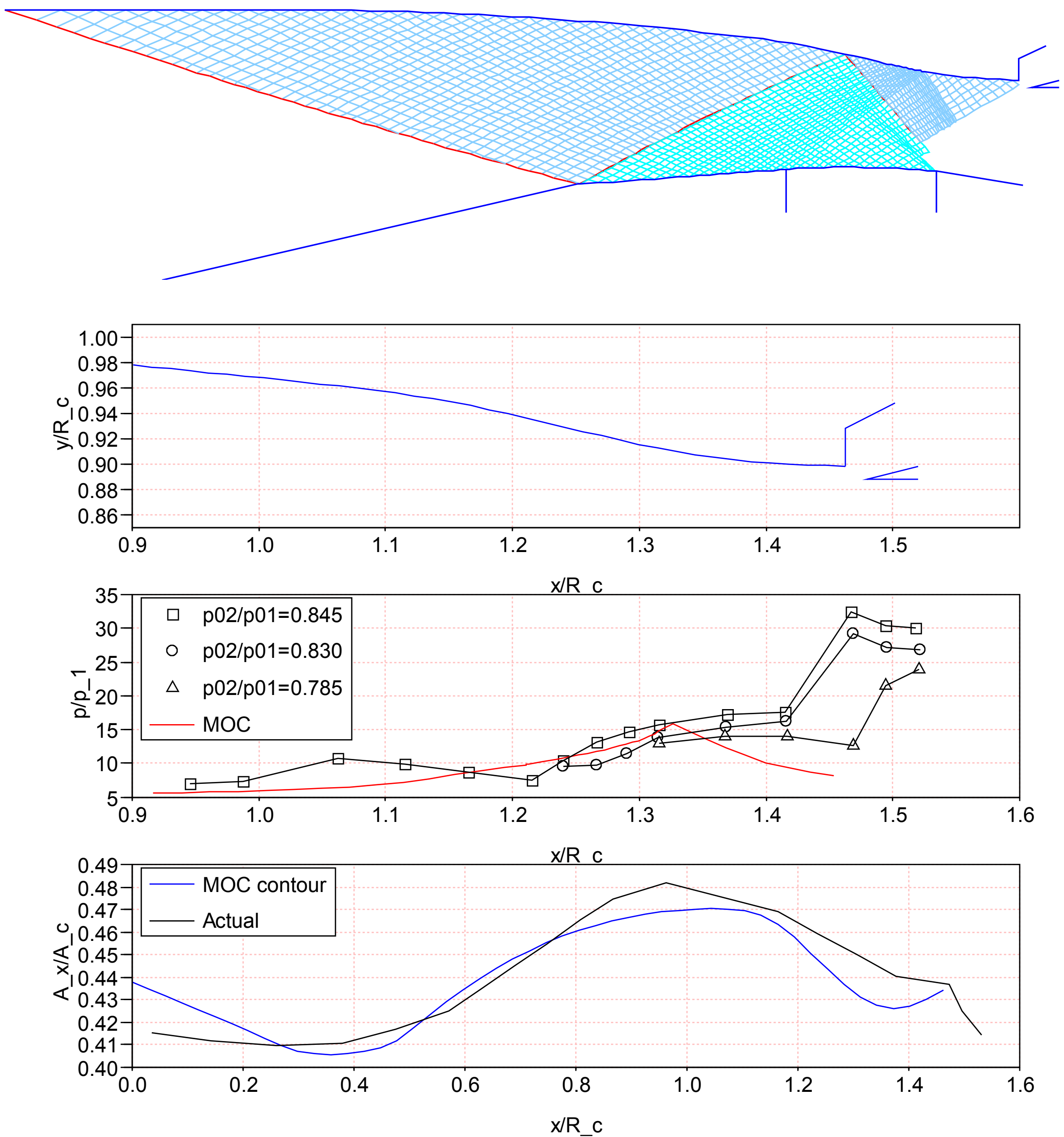


Figure 18: MOC solution for M=3.2 and comparison with measured cowl pressures from Blausey et al [26], noting at M=3.2 the minimum dynamic pressure is 310KEAS [25]. The lowest figure compares calculated duct cross sectional area distribution with that from Bangert et al [28] for the spike in the forward position.

Cowl drag is reduced by turning the flow to axial as rapidly as possible. By using a relatively slender 13° half angle cone [27], and an internal lip angle that appears to be zero, the captured flow may be turned back to near axial immediately by the cowl shock. But this is only the first stage of compression. At Mach 3.2 the Mach number downstream of the internal cowl shock varies from 2.56 at the cowl to 2.35 at the spike and to obtain the pressure recovery of 0.785 further deceleration is required prior to the terminal shock. This is achieved in what is essentially a two dimensional supersonic nozzle operating in reverse, compressing the flow to approximately Mach 1.47. This second compression process requires approximately 27° of turning (the difference in Prandtl-Meyer angles between entrance and exit Mach numbers) half of which is inward and the other half outward so that the outflow is practically axial at the throat, thus maximising stream thrust. The MOC solution in figure 18 is based on the section of the cowl in drawing 16 of Blausey *et al*. [26], scaled for a radius of $0.9R_c$ at the shock trap and extrapolated by cubic spline to the cowl lip $1.462R_c$ upstream [29] with the slope forced to zero there. The spike contour was developed simultaneously with the MOC solution: the shoulder is placed where the cowl shock intersects the 13° cone; an $0.3R_c$ conical section at 4.2° is added to produce a very weak reflected shock given the calculated streamline angle downstream of the incident shock of 4.13°. The length is based on visual inspection, figure 19 (and counting rivets!), but its function is to reflect the compression generated by the inward turning cowl thus enabling the flow to be redirected axially without expansion and the estimated length is consistent with this supposed objective; A constant pressure section follows to represent the slotted bleed surface that is delineated in figure 18 by the vertical lines. Although not strictly necessary when using a porous (slotted) cover, for maximum momentum recovery (minimum bleed drag) the bleed plenum will operate at the maximum possible pressure. Since this must be lower than the minimum pressure over the bleed surface one would seek to place the bleed in a region of constant pressure when possible. Defining the bleed as a region of constant pressure is sufficient for MOC to develop the contour which turns inward under the influence of compression from the cowl; The inward turn of the spike is terminated at -9° and thereafter held constant until the translating spike meets its supporting cylinder. The choice of spike angle in the subsonic diffuser is discussed with respect to off design operation in a following section.

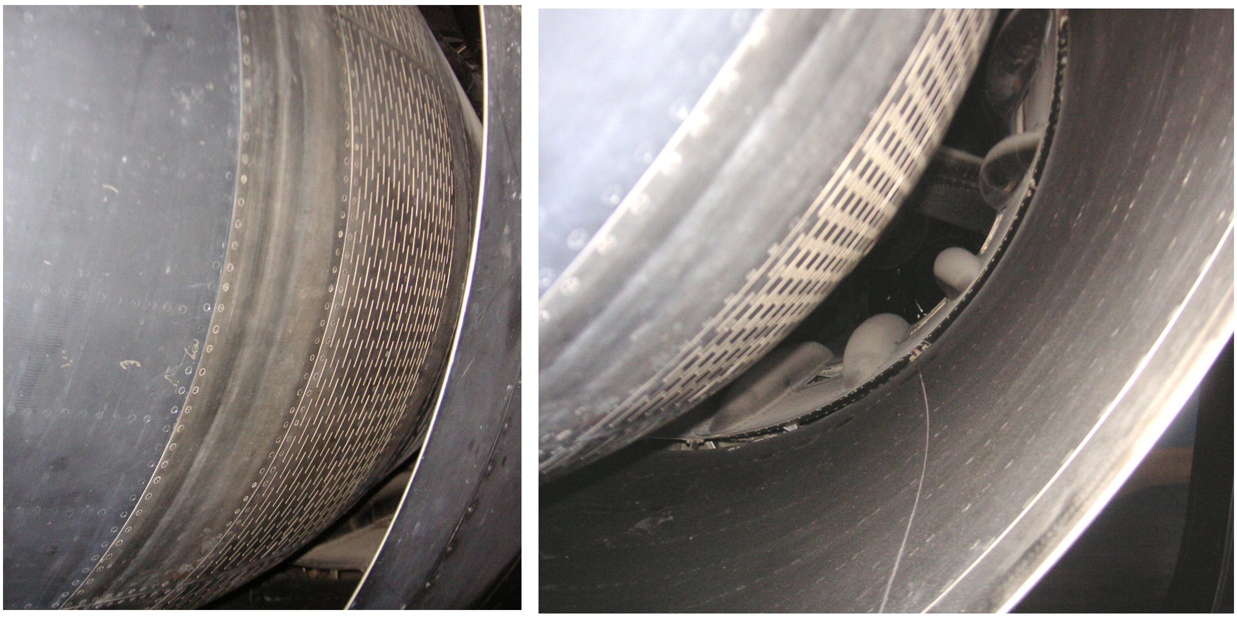


Figure 19: Photographs taken at the USAF museum revealing: the (seemingly) conical section downstream of the abrupt turn from 13°; the spikes slotted bleed; and the cowl's shock trap and mice.

Although not providing a perfect match with the measured pressures, the inviscid MOC solution with the assumed contour results are close enough to give confidence in this interpretation of the design (at least to the author). The duct area distribution after an $0.862R_c$ translation to its forward position is also in reasonable agreement with that from Bangert et al [28]. A viscous solution along with some tweaking of the spike and cowl contours would enable the pressure and area measurements to be matched, but this is unlikely to prove more instructive and the effort therefore unwarranted, especially if it is to be superseded by release of reports containing the actual contour.

4.9 Porous bleeds and the shock trap

For the assumed intake geometry, right running characteristics merge as they approach the aft end of the bleed (figure 18). These will form a shock that will reflect off the solid conical subsonic diffuser, but as the shock is weak and the boundary layer thinned by the porous bleed, the interaction will be slight and confined. This is one function of the spike bleed, its primary function would be to limit the extent of upstream propagation of any separation caused by the much stronger terminal shock (a pressure ratio of 2.5 for Mach 1.5).

The "shock trap" visible as the forward facing slot in figure 19 and sketched at the end of the cowl contour is a bleed type credited by Campbell [23] to Luidens and Flaherty [32]. They define it as "*a scoop bleed with area expansion starting upstream of the leading edge of the scoop*". Expansion in the Blackbird shock trap begins 1.8*h* ahead of the leading edge of the scoop that is a distance *h* off the surface [26]. The primary function of the shock trap is to remove the cowl boundary layer prior to the terminal shock in the subsonic diffuser. At cruise, 8% of the intake air is captured by the trap with a pressure recovery of 0.27 [26] and ducted to the ejector as previously described. The trap pressure recovery is equivalent to $13p_1$ which is consistent with the measured static pressure near the trap for p_{02}/p_{01}=0.785. Note that as pressure recovery is forced above this value (by throttling the wind tunnel model's outlet) a separation is produced ahead of the shock trap as evident by the increase in cowl static pressures, figure 18. Bleed mass flow is also reduced in this state [26]. This is an unstable situation that leads to unstart as the turbojet must swallow the extra mass which it can only do by increasing density and therefore increasing back pressure. The higher static pressures on the forward facing surface of the cowl decreases the stream thrust as would any increase in boundary layer mass flow entering the throat, and a decrease in stream thrust corresponds to a decrease in potential pressure recovery. The virtues of Concorde's wide bleed slot are apparent by contrast, as it had the opposite and stable characteristic with an increase in back pressure leading to an increase in bleed mass flow without disturbance on the upstream ramp. This is not meant to imply that a wide bleed slot would have been a better solution for the SR-71, the intakes operate under different constraints, but studies such as those by Lenyaert [18] and Blausey et al [26] that define the characteristics of different bleed types should benefit future designs.

4.10 Spike schedule, aerodynamic contraction and mice

Spike translation is scheduled with Mach number, adjusted by small offsets that are a function of side slip, incidence, and normal acceleration [23]. Figure 20 presents mass capture ratio, cowl lip conditions and the radius at which the internal shock would intersect the 13° ray, when following the nominal, near linear, spike schedule obtained from the flight manual (noting that the spike tip is at $2.409R_c$ at Mach 3.2 [28]). During acceleration the internal shock intersects the spike downstream of the shoulder, as demonstrated by the right hand top figure which indicates the theoretical radius at the intersection with a 13° ray is always greater than that at the shoulder. The expansion over the shoulder at the lower flight Mach numbers followed by re-compression from the internal shock, clearly must create a stable interaction with the boundary layer upstream of the bleed flowfield. Perhaps the dark bands visible on the second conical section of the spike, figure 19, are evidence of heat stress from such an interaction.

The intake starts between Mach 1.6 and 2.1 [23] with the actual value probably determined by engine mass flow demand and therefore dependent on ambient temperature. The lowest Mach number corresponds to shock attachment at the lip as made clear by the bottom right plot in figure 20 in which the conditions at the lip, determined from the Taylor-Macoll equations, are compared with the angle for shock detachment from the Rankine-Hugoniot shock relations (black line). Note the very large increase in capture area from $0.42A_c$ at M=1.6 to $0.99A_c$ at M=3.2, which suits the turbojet, given some fine adjustment using the bypasses as described previously.

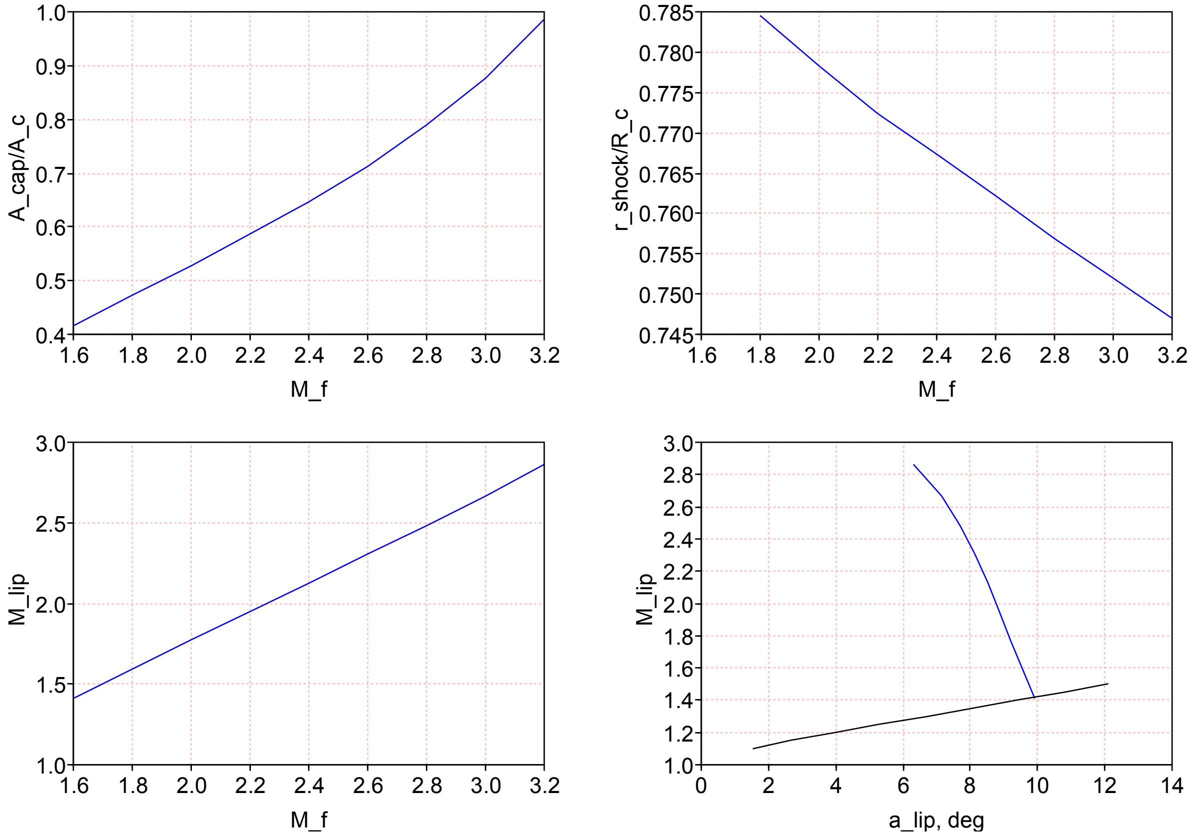


Figure 20: Calculated off design characteristics of the SR-71 intake (blue lines). The black line in the bottom right is the deflection at shock detachment.

A real advantage of this increase in capture area is that it reduces the variation in throat area required to achieve the necessary aerodynamic contraction. At M=3.2 isentropic compression to Mach 1 requires a contraction of 5.12 whereas at M=1.6 the value is 1.25. The increase in capture area reduces the required throat area variation in this hypothetical, perfect, intake from a factor of 4.1 to just 1.73, which is mechanically far simpler to achieve. In fact, the actual throat area at M=3.2 is 54% of the area at M=1.6, a factor of 1.85 [23].

With the full stroke of the spike primarily determined by the Mach 1.6 and 3.2 mass capture requirements, the throat area variation is obtained by choosing the right slope of the spike in the subsonic diffuser. The combination of a -9° conical diffuser with an $0.9R_c$ radius at the shock trap entrance, allows the throat to vary between $0.41A_c$ and $0.22A_c$ with an $0.862R_c$ stroke. With throat geometry having imposed a tight constraint on the spike angle, subsonic diffusion rate must be controlled with the cowl contour in this region. However any forward facing surface in a subsonic diffuser reduces axial stream thrust (there is no net radial component) and this has the potential to destabilise the terminal shock. Figure 19 reveals that the main cowl surface in the subsonic diffuser converges very slightly but this must have been insufficient to compensate for the rapid increase in duct area due to the receding spike. The mice solved the problem and although this three dimensional approach may have been forced by a need to retrofit, it might have wider application, particularly if it was found to be less destabilising then an axisymmetric convergence.

5 CONCLUSIONS

Applied intake design is a great aerodynamic challenge, from determining the specifications that enable it to best match the engine and aircraft/missile, right to up the final stages in which fine tuning is accomplished to compensate for things that didn't go quite to plan. Creating a geometry that directs flow exactly where one wants it and in the state it needs to be, is made the more satisfying by the fact that an analytically based subtle change to a contour can have a large effect in a compressive decelerating flow.

However, esoteric intake studies are less rewarding than those with application: the basic design techniques are well known, and there already exists large, freely available, databases of wind tunnel experiments on generic intakes. It is hoped that by focussing this lecture on some historic intakes, and highlighting the features that I think were critical to their operation the subtle beauty of these devices and the true accomplishment of their designers can be appreciated.

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