

Aerodynamic of Sonic Flight



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TOPICS



Part 2: Fundamental Physics of Transonic Aerodynamics

Part 3: Prior Studies and Technology Needs

The recently announced Boeing "Sonic Cruiser" Program has sparked renewed interest in the characteristics of configurations designed to cruise at near-sonic or transonic speeds. Technology spin-off from the recent High Speed Civil Transport, HSCT, program has lead to what appears to be a paradigm shift in the configuration features for a near sonic configuration as well as it's potential as a viable aircraft concept. Consequently, a paper was presented at the recent Boeing Technical Excellence Conference to discuss "Aerodynamic Issues Involved in Sonic Flight".

The material presented in this paper has been expanded into a series of three parts that include:

- Part 1: Effect of Speed on Airplane Shape
- Part 2: Fundamental Physics of Transonic Aerodynamics
- Part 3 Prior Studies and Related Topics
 - Sonic Boom Considerations
 - Previous Transonic Airplane Studies
 - Paradigm Shift in Near Sonic Concept
 - Aerodynamics Needs and Challenges for Sonic Cruise

This paper contains part 1.



The earliest type of modern commercial aircraft is the "classical design". The basic design philosophy inherent in this type of aircraft configuration was to fly at speeds below which strong shocks can occur on the wing. The airplane speed was therefore limited by the drag rise characteristics of the basic airfoil inherent in the wing design. The wing planform was unswept and utilized large wing spans to minimize the induced drag.

The fuselage was typically a simple tube shape cylindrical body with a rounded nose and a slightly upswept aft body. The propeller driven engines were mounted on the wing without significant aerodynamic considerations

The various components of the configuration including the wing, the body and the engine nacelles were designed in isolation quite independently of the other components. The components were sized and positioned to provide a balanced configuration with the required stability and control characteristics as well as the desired takeoff and landing performance.

Typically the span of the wing was about equal to the length of the fuselage.



The parameter most often used to assess the aerodynamic efficiency of a transport configuration is the product of the cruise lift to drag ratio, L/D, times the cruise Mach number, (M L/D). This parameter is directly related to the range of an airplane and represents how high a cruise speed that a configuration can retain a large value of Lift / Drag ratio.

For a "classical" type of aircraft, the L/D is approximately constant until the "Critical" Mach number of the basic wing airfoil is reached and a significant drag increase due to compressibility effects occurs. The increase is typically due to local pockets of supersonic flow terminated by local shocks that produce wave drag and / or shock induced separation.

Therefore, the value of M L/D will vary approximately linearly with Mach number until the critical Mach number is reached. The M L/D then decreases quite suddenly because of the drag increases rapidly above the critical Mach number.

The slope of the M L/D curve increases with wing span and wing span load efficiency ,e, and decreases with increases in the wetted area of the configuration.

The critical Mach number is set by the airfoil technology. The critical Mach number is reduced by increases in either the lift coefficient or the wing thickness. The effects of an increase in airfoil technology typically includes a slight change in the slope of the M L/D curve and a continuation in the straight portion to the higher critical Mach number.

The gain in the aerodynamic efficiency with increased critical Mach number with advanced airfoil technology is then proportional to the increase in the critical Mach number: Δ (M L/D) ~ Δ Mcrit

Supercritical Flow Development



The schlieren pictures in the figure shows the flow features over an airfoil at subcritical and at supercritical flow conditions.

The upper surface shock is very evident at the supercritical flow condition. The corresponding local Mach number distributions are also shown. In the examples shown, the lower surface Mach numbers also exceed Mach one at the supercritical flow condition and indeed the formation of slight compression waves are also evident in the schlieren photo.

Examples of "Classical" Commercial Airplanes



Douglas DC-3 1935







Boeing B-307 Stratoliner 1938





Douglas DC-2 1934

Boeing 377 Stratocruiser 1947







This figure shows a number of "classical" commercial airplane configurations.

Examples of "Classical" Military Airplanes

Boeing B-17 1935



North American F-82

B-29 Superfortress 1942



North American P-51



Douglas B-26 1940



Consolidated B-24 Liberator 1943









Douglass C-133 1956



This figure shows a number of military fighter, bomber and transport designs based on the "classical" aircraft concept.

Effect of Wing Sweep on Oblique Airfoils Pressure Distributions



In the early 30's, Busemann in Germany and R.T. Jones of the NACA, discovered the beneficial effects of wing sweep in delaying the onset of the compressibility effects.

There results indicated that on Infinite swept wings, as shown in the figure above, the flow over the airfoil is dependent only on the component of Mach number that is normal to the leading edge and on the corresponding normal angle of attack.

The pressure distributions over the swept airfoils are essentially identical when related to equal normal Mach and angle of attack conditions

The wing critical Mach number on an infinite swept wing is therefore set by the basic airfoil technology and the wing sweep. According to this simple sweep theory, the critical Mach number for a swept wing is delayed by the inverse of the cosine of the wing sweep angle.



Swept Wing Aircraft were developed to exploit the beneficial effects of wing sweep in delaying the onset of the compressibility effects. However in order to fully capitalize on the speed increase offered by wing sweep the configurations utilized the emerging jet engines which also benefited by the increased speed capability.

The initial swept wing configurations were tailored designs in which the wing, the body and the nacelles were basically designed in isolation with local tailoring in the wing / body junction and the nacelle strut / wing intersections. The body typically is a simple tube shape.

Typically the wind structural span is held constant as the wing is swept. The actual wing span decreases by the cosine of the wing sweep angle.

The use of the modern CFD design and analysis tools now allow design optimization and integration of the entire wing, body, nacelle, empennage configuration and thereby includes the mutual interference effects automatically in the design process.



Variation of Aerodynamic Efficiency With Mach Number

The "best" wing span for any swept wing configuration is the result of a multi-discipline design optimum process. This process includes many elements such as the configuration design definition, and structural design optimization, structural materials, propulsion system concepts and engine performance characteristics, aerodynamic design features as well as the design mission and engine / airframe matching process.

However, in order to understand the benefits of wing sweep let us assume that the wing structural span is to be held constant as the wing is swept. This is a reasonable assumption since this tends to maintain a constant wing structural weight.

As the wing is swept the aerodynamic span decreases proportional to the cosine of the sweep angle. The L/D of a subsonic airplane is directly proportional to the span. Hence:

L/D ~ cos(Sweep)

The critical Mach for a swept wing is set by the normal Mach number equal to the airfoil critical Mach number:

Mcrit ~ I / [cos(sweep)]

Therefore, the maximum value of M L/D for any sweep angle is about constant.

M L/D ~ Constant

The use of wing sweep, therefore, allows a value of "M L/D" to be delivered to higher cruise Mach numbers. Conventional swept wing transport configurations with moderate tailoring are typically limited to approximately: Mcrit < 0.85

B-52 DESIGN DEVELOPMENT

"EVOLUTIONARY' PLUS "REVOLUTIONARY"



This illustrates the development process leading to the B52. This B52 evolved from a "classic" aircraft to a "swept wing" aircraft during it's design process. In this instance, the benefits of wing sweep were first utilized followed by the incorporation of the jet engine and additional increased wing sweep to maximize the speed benefits offered by the jet engine.

Commercial Swept Wing Aircraft

Boeing Dash-80 1954







Boeing B727 1964









Boeing B747-100



Boeing B767-400ER





This figure shows examples of current modern aircraft that are all based on the "swept Wing" concept.

Military Swept Wing Aircraft



KC-135 Stratotanker 1956



F-86 Sabre 1947

B-47 Stratojet 1947



C-17 Globmaster III



Grumman A-6 Intruder 1960









This figure shows a number of military fighters, bombers, tankers and transports that utilized the swept wing concept.

Near Sonic Aircraft: Mach = 0.9 to 0.98

POSTPONE DRAG RISE



- Basic Design Philosophy: Avoid or Delay the Formation of Shocks at high subsonic Mach numbers.
- Highly Swept Wing Design for Low Induced drag and to Delay the Drag Rise
- Components Well Integrated
- Transonic Area Ruled Body to delay drag rise
- Airplane Speed Limited by Basic Airfoil Shape Plus the Wing Sweep Plus Body Area Ruling
- Body length Larger than the Wing Span.

Near sonic aircraft are designed to push the cruise speed beyond the the capability set by wing sweep and airfoil technology alone. This class of airplanes typically have had highly swept wings carefully located on a contoured area ruled fuselage. The nacelle installations must also be carefully tailored as in the example shown here.

The physical effect of the body contouring and nacelle relocation, is to reduce the local Mach number distribution over the wing. This allows an increase in cruise speed. The basic design philosophy is therefore, to delay the compressibility effects until very high subsonic Mach numbers. The complete configuration is both highly integrated and quite slender.



This illustrates the M L/D for a near sonic design. Relative to a swept wing design, the near sonic M L/D slope is reduced by the increased skin friction since the body wetted area is quite a bit larger then that for a simple tube shape body. The critical Mach number is increased to a higher speed then that for basic swept wing concepts.

The actual critical Mach number limitation for this type of configuration is typically dependent on the ultimate wave drag development on the fuselage and is on the order of $M_{CRIT} \sim 0.95$ to 0.98

"Near Sonic" Configurations

Boeing Early Near Sonic Study Configuration ~1970

F8 With Supercritical Wing Flight Test Vehicle ~ 1971



This shows a picture of an artist conception of an early Boeing near sonic study configuration. Also shown is the NASA F-8 that was flight tested with an early near sonic wing design.

Low Supersonic Aircraft: Mach = 1.1 to 1.4

REDUCE DRAG AT LOW SUPERSONIC SPEEDS
EXPLOIT THE POSSIBILITY OF SUPERSONIC BOOM FREE FLIGHT



- Basic Design Philosophy: Reduce Drag at Low Supersonic Speeds
- Supersonic Swept Wing Design to Minimize Wave Drag and Drag Due to Lift
- Subsonic Area Ruled Body to Reduce Low Supersonic Wave Drag
- Components Well Integrated
- Airplane Speed is selected for the overall design condition
- Oblong Confining Box : Body length Larger than the Wing Span.

At supersonic cruise speeds, wave drag is inevitable. The basic design philosophy for a low supersonic aircraft is therefore, to reduce this wave drag.

At low supersonic mach numbers, the wave drag is primarily due to the volume distribution and wing thickness distribution of the configuration. The wing is designed to reduce the combination of volume wave drag and total drag due to lift. The body has a sharp nose to reduce the body wave drag. The body is highly area ruled to minimize the wing/ body wave drag. The overall design is highly integrated.

The airplane cruise speed is selected rather then being constrained to avoid the drag rise as for subsonic designs. These designs can also exploit the possibility of "boom" free supersonic flight which depending on the season and the jet stream, allows cruise Mach numbers between Mach ~ 1.04 to Mach ~ 1.4



The linear portion of the M L/D curve for the low supersonic configuration is less then that for a swept design because of the reduced wing span and the increased wetted area associated with a long slender area ruled fuselage. The critical Mach is delayed because of the body contouring. The dip in the curve results from the wave drag build up with Mach number.

It is possible to fly free of the sonic boom for cruise Mach numbers on the order of 1.05 to 1.40 depending on temperature and weather conditions. This is possible as long as the local ground speed is less then Mach 1.0.

Variable Sweet Oblique Wing: Mach = 0.9 to 1.5

MINIMIZE DRAG OVER WIDE SPEED REGIME Basic Design Philosophy: Minimize Drag For All Cruise Speeds Highly Sweep Oblique Wing for Minimum Wave Drag and Drag Due to Lift For All Flight Mach Numbers Variable Wing Sweep to Increase Takeoff Lift / Drag Ratio Cannot Exploit Area Ruled Body Concepts Airplane Speed Limited by Basic Airfoil Shape and Associated Wing Normal Mach Number (0.73 to 0.78) Critical Design Issues: Engine Location, Wing Pivot Location. Landing Gear Design $b/L \approx 0.4$ Oblong Confining Box : Varies with Wing Sweep $b/L \approx 0.7$ Can Fly Free of Sonic Boom at Low Supersonic Speeds WELL INTEGRATED DESIGNS

The variable sweep oblique wing, VSOB, concept is an unconventional approach to the design of a low supersonic aircraft. The wing is unswept for low subsonic speeds, and is highly swept at the highest cruise Mach Number. In between, the sweep schedule is quite simple: the sweep is selected to provide a constant normal Mach for all cruise Mach numbers:

$$\Lambda_{sweep} = \cos^{-1} \left(\frac{M_{Norm}}{M_{\infty}} \right)$$

The design normal Mach number is set by the airfoil technology, the wing normal thickness to chord ratio and the design sectional lift coefficient and is typically on the order of:

At the supersonic speeds an elliptic planform with a uniform lift distribution results in both minimum induced drag and in minimum wave drag due to lift.

The wing also provides very low levels of wing volume wave drag provided that the wing is swept greater then the free stream Mach angle. Because the wing planform streamwise length is long compared to the body length. The VSOB does not derive significant benefits from body area rule contouring.

The critical design issues for the VSOB include engine location and integration of the wing pivot and the landing gear.



The variable sweep wing allows the configuration to achieve optimum aerodynamic performance for all Mach numbers. As such the VSOB, embodies the performance features of the "classic", the "swept", the "near sonic" and the "low supersonic" designs.

The wing is unswept for low subsonic Mach numbers up to the critical Mach number and hence, the M L/D curve is initially linear.

Beyond the critical Mach, the wing sweep varies with the cruise Mach number such that the normal Mach number is equal to the critical Mach number. Therefore, beyond the Critical Mach number and up to Mach 1.0, the M L/D is approximately constant. The wing sweep schedule is such as to maintain a constant normal Mach number equal to the critical Mach number of the wing design. This is typically equal to about $M_{NORMAL} \sim 0.73$

At Mach 1.0 ML/D drops because of the wave drag that ultimately will occur.

At the low supersonic Mach numbers, the L/D varies primarily with the cosine of the sweep angle. Since the normal Mach is constant the M L/D tends to be nearly constant up to a limiting Mach number. This occurs when the wing sweep has reduced the aerodynamic span so much that the induced drag starts to increase significantly. This typically occurs for Mach numbers on the order of: $M_{limit} \sim 1.4$ to 1.6

Variable Sweep Oblique Wing Configuration

NASA Ames Research Center AD-1 Flight Test Vehicle 1980



Boeing Oblique Wing Study Configuration 1973



This figure shows pictures of the NASA Ames AD-1 experimental Aircraft that was used to demonstrate the flying capability of a variable sweep oblique wing concept. Although, the flight Mach number was restricted to subsonic speeds, the flight test program did demonstrate that flight was possible without the need of a sophisticated control system to decouple the lateral and longitudinal forces and moments.

This flight test program was also a classic example of a very low cost flight test program that provided otherwise not attainable test data and flight information.

The figure also shows the Boeing variable sweep wing concept model 5-3 that was developed during the NASA sponsored High-Transonic Speed Transport Aircraft Study*.

* Kulfan, R. M., "High-Transonic Speed Transport Aircraft Study - Summary Report", NASA CR-2465, & Boeing Document D6-60231

Slender Supersonic Aircraft: Mach ~ 1.5 to 2.7

MINIMIZE SHOCKS AND EXPLOIT FAVORABLE INTERFERENCE



 Basic Design Philosophy: Minimize Shocks to Reduce Drag For Supersonic Cruise Speeds

- Attached Flow Over Wing and Body At Cruise Condition
- Highly Integrated Configuration Design
- Highly Constrained Wing Thickness Distribution Designed for Minimum Volume Wave Drag
- Optimized Camber and Twist Distribution and to Minimize Drag Due to Lift For Cruise or Multi-Point Design Conditions
- Area Ruled Body Design To Minimize Volume Wave Drag, Body Camber and Shaping to Reduce Drag Due to Lift
- Optimized Nacelle / Airframe Design Integration
- Highly Swept Slender Wing with Variable Leading and Trailing Edge flaps to reduce Drag at Off-Design Conditions and Produce Necessary Landing and Takeoff Performance
- Requires Multi-Point Design
- Airplane Speed Generally Set by the mission requirements.

The configuration arrangement of a supersonic transport configuration is highly dependent on the technology embodied in the design as well as the mission objectives. This is similar to the US SST and the more recent HSCT concepts.

The basic design philosophy is to minimize shocks to reduce drag for supersonic cruise speeds. Consequently, the configurations tend to have long slender bodies and thin wings with a great deal of shaping and careful integration. The wing thickness tends to be thin but are highly constrained to provide the necessary depth and volume requirements. The body is area ruled to reduce the volume wave drag. Both Body shape and camber can influence the wave drag due to lift. The nacelles are carefully shaped and located to maximize the favorable interference effects.

Supersonic wing planforms tend to favor both leading and trailing edge flaps to reduce the drag at off design conditions as well as to provide the necessary high lift takeoff and landing performance requirements.. The shape of the wing leading edge tends to have a significant influence on the requirement for, and deployment of the leading edge flaps.

The supersonic configuration requires multipoint design optimization including takeoff and landing conditions, subsonic and supersonic design conditions and transonic acceleration considerations.

The airplane cruise speed is generally set by the design mission requirements.



Variation of Aerodynamic Efficiency With Mach Number

The M L/D curve for a supersonic transport aircraft tends to have a reduced slope at subsonic speeds relative to a subsonic aircraft because of the reduced wing span and relatively large wetted area. This is followed by a rather sudden "drop" due to the rapid build up of wave drag in the lower supersonic speed regime. At the higher supersonic speeds the wave drag tends to level off and the M L/D tends to increase linearly but with a slope less then the subsonic slope.

Since the supersonic configuration does encounter wave drag at its design condition the concept of a "critical Mach" does not apply.

Slender Wing Aircraft







This figure shows pictures of the Russian TU-144 and the British / French Concorde supersonic transport configurations both of which are based on the "slender Wing" concept. The F-22 and F-15 are supersonic slender wing fighters. The Vulcan is an application of the "slender" wing concept to a subsonic aircraft.

Variable Sweep Supersonic Aircraft: Mach ~ 1.5 to 2.7

MINIMIZE CRUISE AND OPTIMIZE SUBSONIC AND HIGH LIFT PERFORMANCE



The use of "Variable Wing Sweep" is another configuration option for a supersonic aircraft design. As shown in the example in the figure, The use of variable sweep allows the aircraft to perform as a higher aspect ratio swept wing at subsonic flight conditions and as a low aspect ratio slender wing at supersonic mach numbers.

The overall basic design philosophy is to minimize shocks to reduce drag for supersonic cruise speeds as for the "slender Wing" concepts. Consequently, the configurations tend to have long slender bodies and thin wings with a great deal of shaping and careful integration. The wing thickness tends to be thin but are highly constrained to provide the necessary depth and volume requirements. The body is area ruled to reduce the volume wave drag. Both Body shape and camber can influence the wave drag due to lift. The nacelles are carefully shaped and located to maximize the favorable interference effects.

At off-design transonic and subsonic design conditions, the wing is unswept to increase the aerodynamic span and thereby improve the overall aerodynamic performance. The leading and trailing edge flaps are also used to further reduce the drag at off design conditions as well as to provide the necessary high lift takeoff and landing performance requirements.

Variable Sweep Wing Configurations

NASA X-5 Flight Test Vehicle (1952)



Rockwell B-1 Lancer (1974)

(1964)

Boeing Variable Sweep SST Design B2707-190 Model (1962)

Low Speed Subsonic Cruise Supersonic Cruise



Supersonic Cruise → Subsonic Cruise → Low Speed



Supersonic Cruise → Subsonic Cruise → Low Speed

The NASA X-5 experimental flight test vehicle was an early demonstrator of flight capability with variable sweep. The X-5 was based upon the design of a Messerschmitt P. 1101 airplane discovered in Germany at the end of World War II, although the P. 1101 could vary its sweep only on the ground. The X-5 could fly only at subsonic speeds, but it did demonstrate the ability to vary the wing sweep from 20 to 60 degrees in flight.

A number of modern day variable sweep supersonic fighters are also shown in the figure.



This shows a composite of the aerodynamic efficiencies for the previously discussed configurations. Beyond Mach 1 there is typically a dramatic drop in aerodynamic efficiency. The speed gain in this "drag bucket" region has in the past, not been found to be sufficient enough benefit to overcome the penalty of the wave drag build up on the overall economics of an airplane.

However, this speed regime may allow the achievement of sonic-boom-free flight which may help offset the adverse effect of the relatively low aerodynamic efficiency.

Hypersonic Aircraft: Mach = 4.0 to ?

EXPLOIT SHOCKS



- Basic Design Philosophy: Exploit Shocks to Produce Compression Lift and Drag Reductions
- Fully Integrated Configuration Design
- The Volume , Lift and Thrust Producing components May be Indistinguishable
- Many Concept Options



The final class of aircraft which we will discuss are those designed for hypersonic flow. Shock waves and expansion waves are a predominate feature of the flow, and cannot be avoided. The shock wave or compression wave strength per unit deflection angle is significantly greater than the corresponding expansion wave strength. This is contrary to supersonic flow where expansion wave and shock wave strengths are essential equivalent.

Consequently, hypersonic aircraft are designed to exploit the compression lift. In addition, the concept of cancellation of compression waves by interacting expansion waves is often employed.

The wave rider concept and the flat top wing / body concept are example configurations that tend to exploit the compression lift. Where as the busemann biplane configuration utilizes wave cancellation techniques. The ring wing and parasol wing concepts exploit both wave drag cancellation and shock capturing for lift effects.

Hypersonic Configurations



This shows some examples of hypersonic configurations.

THE X-15 was a air launched rocket powered research vehicle. The X24B was a member of the lifting body configurations that investigated un-powered reentry leading to the space shuttle.

The X-43 is an air-launched unmanned hypersonic research configuration that demonstrated scram jet technology for hypersonic flight.

Many Aircraft Configuration Options

 $M_{\text{DESIGN}} = 1.05$ to 1.25



For most any aircraft design objectives, there are usually many different design options as shown in this figure. In the example above it is seen that aircraft derived from the "swept wing", "slender wing", "oblique wing" and "variable sweep" concepts are conceptually feasible for low supersonic speed applications

AERODYNAMIC CONFIGURATION INNOVATIONS



Messerschmitt (Lippisch) Variable Sweep Wing Patent (1941)

Blohm and Voss (Vogt)

Oblique Wing Fighter

Concept (1944)



Lippisch Delta Wing Supersonic Fighter (1944)

Swept Wing

Busemann (1935)

R. T. Jones (USA)



Junkers Swept Forward Wing Bomber Testbed (1944)



X-21A Laminar Flow Control Airplane (1960) - Fundamental Research 1940 - 1950



Northrup Flying Wing

Most Innovative Aerodynamic Concepts were established 50 to 65 Years Ago

The general features and shape of the aircraft has typically been associated with technological progress in aeronautics.

It can be argued with some degree of validity, that the vast majority of existing aircraft concepts were established (al least on paper) 50 to 65 years ago. The swept wing, conceived by Busemann (1935) and about the same time by R. T. Jones, and the development of the delta wing planform by Lippisch together with the Jet engine conceived by Whittle are the basis for almost every modern high-speed airplane configuration.

During the decade between 1935 and 1945, visionaries foresaw a range of configurations that even today can be considered very modern. These include the swept forward wing pioneered at Junkers in the early 1940's, the oblique wing originally conceived by Vogt at Blom and Voss in 1943, and variable sweep wings, a patent for which was issued to Messerschmidt in 1941 for the concept by Lippisch.

The variable sweep oblique wing was first proposed nearly 60 years ago and was first tested in a wind tunnel in 1945. In recent years the oblique wing proposed by R. T. Jones led to the oblique wing demonstrator program in 1978.

The flying wing, conceived by Jack Northrup over 65 years ago is an example of an aerodynamic concept that may be very dependent on developments in avionics for stability, control, load alleviation and flutter suppression for a practical application.

Other aerodynamic technologies also have their basis in work conducted more then 60 years ago. Early investigators in the achievement of low drag by suction included Holstein (1940), Loftin and Burrows (1949) and Pfenninger (1946, 1949). This led to the pioneering work with the X-21A aircraft by Northrop and the Air Force in the 1960's. More recent flight test development activities in the 1990's include the laminar flow control experiments on the Boeing 757 and the F-16XL supersonic flight test program.

MULTI-DISCIPLINE AIRCRAFT DESIGN



DESIGN: TEAM OF EXPERTS WITH IDEAS AND EFFORTS TOWARDS A COMMON GOAL.

The previous discussions have related the general features of an aircraft to the general nature of the aerodynamics at the cruise conditions. The design of any airplane, however, is always the result of a multi-disciplinary optimization process. This process attempts to balance the often conflicting demands of the various technical disciplines in order to achieve the most favorable solution for the specific design mission and objectives

Any successful airplane development program requires multi-disciplinary design optimization and integration . The design team is made of a team of experts with ideas and efforts directed towards a common goal. This requires many trade and benefits studies and trades since the visions of the "perfect" airplane are often different for the team members. An essential ingredient for a successful program is to identify the most critical technology options from all of the various technical disciplines.

OVERVIEW

Session 1: Effect of Speed on Airplane Shape

Session 2: Fundamental Physics of Transonic Aerodynamics

Session 3: Prior Studies and Technology Needs

The recently announced Boeing "Sonic Cruiser" Program has sparked renewed interest in the characteristics of configurations designed to cruise at near-sonic or transonic speeds. Technology spin-off from the recent High Speed Civil Transport, HSCT, program has lead to what appears to be a paradigm shift in the configuration features for a near sonic configuration as well as it's potential as a viable aircraft concept.

A three part Paper has been written to discuss the aerodynamics of sonic flight.

In part 1, we will examine how speed affects the basic aerodynamic design philosophy and the corresponding shape of an aircraft configuration.

In part 2, simple concepts will be used to examine the fundamental physics of transonic flow and the implications on the aerodynamic shape and design integration for a near-sonic / transonic airplane.

In Part 3, previous near-sonic airplane design studies will be briefly reviewed to illustrate the types of configurations that have been considered in these earlier studies. The fundamental characteristics of sonic boom will be discussed including the possibility for over land boom free flight

A "Tops Down" assessment of the aerodynamic performance of high aspect ratio conventional subsonic commercial aircraft will be compared with assessments of low aspect ratio HSCT type configurations. The results suggest the possibility for an alternate approach for a near transonic configuration, then those previously considered that is similar to the "sonic cruiser" configuration.

Enabling technology aerodynamics needs for the development of a viable nearsonic / transonic configuration will be summarized.

The author has not been directly involved in the sonic cruise program. The



There are a number additional "tools" in the aerodynamist's "tool box". These are either sub-elements or are related the CFD and EFD. These include:

- VFD, (Visual Fluid Dynamics) obtained either from CFD calculations or from special test measurements. These include such things as surface flow streamline patterns and oil flow pictures, and streamline traces and flow field measurements. The visual ques are very powerful in developing an understanding of the flow phenomena.
- LFD, (Linear Fluid Dynamics), This is a numerically simpler version of CFD but can provide unique fundamental understanding of the flow phenomena directly from the nature of the equations. I n regions where the linear theory results are valid, the linear theory tools are very efficient for conducting both trade and sensitivity studies and for generation of large amounts of data as required for airplane performance calculations.
- SFD, (Simplified Fluid Mechanics), These are usually calculation procedures based on simplified flow analogies and empirical relations. Most flat plate skin friction calculations fit into this category, as well as some techniques for prediction the development of leading edge vortices on highly swept wings.

Analytic Theory and CFD



One of the great joys of my career was to meet and become friends with Dr. Robert T. Jones. He possessed, among many other things, the remarkable ability to share his wisdom which he often blended with a touch of humor.

One time when discussing analytic theory and CFD, he said with a twinkle in his eyes : "Analytic theory is long on ideas but short on arithmetic, and CFD is long on arithmetic but short on ideas".

Although, analytic theory can provide some unique insights and ideas, it does require both understanding and care to correctly apply the theory because of its numerical and flow physics limitations.

Similarly, the use of non-linear aerodynamic methods also requires a basic understanding of the inherent limitations of the methodology. In addition, one should be aware that current CFD solution processes can result in significant differences in the predictions obtained by the different numerical methods.

The advanced CFD codes, however, can provide a detailed insight in to the nature of the flow characteristics on complicated configurations and for conditions where wind tunnel testing is not possible. The non-linear design optimization methods also offer the promise of substantial performance gains.

The best strategy is to use both CFD and analytic theory and then exploit the unique strengths and benefits of each technology. This will provide both the "ideas" and the "arithmetic" plus the added bonus of increased synergistic understanding and design capability resulting from the integrated usage of both technologies.

This is the strategy that is used in this paper.

Analytic Theory and CFD



The fact that classic analytic theory is "old" does not mean that it is of little use. In fact, analytic theory is one and will remain one of the important tools in the aerodynamist's tool box. Fundamentals are not time dated.

Some of the achievements as shown in the figure, that were obtained with analytic theory, are still of great use today and will also be for the near term and the far reaches of the future.
CFD is Required for Configuration Development

- Insight into Flow Features
- Non-Linear Design Optimization
- Detailed Design Integration
- Unique Concepts Evaluations



The use of CFD is a critical element throughout a configuration development process. The results of CFD analyses can provide insight into the nature of the flow features over a configuration. Significant performance gains can be achieved by the use of non-linear CFD design optimization processes. CFD has many other uses such as in detailed design integration studies and in the evaluation of unique configuration concepts.

This figure shows examples of the valuable insight that CFD can provide into the details of complicated flow phenomena.

The figure on left is a comparison of the predicted surface flow patterns with the colored oil flow measurements that were obtained in the Boeing Supersonic Wind Tunnel, BSWT. The model geometry was the Ref H configuration developed during the HSCT studies. This design was developed with linear theory design methods followed by non-linear design analyses in an iterative process. The predictions were made with the parabolized Navier-Stokes program , STUFF.

The viscous flow predictions appear to accurately model the local surface flow phenomena. The CFD particle traces show the inboard turning of the wing upper surface flow near the wing apex. Flow over a mild wing upper surface shock is also apparent. It is also seen that the wing upper surface particle traces are pushed outboard by flow off the body.

The figure on the right shows predictions of the flow characteristics over a 2D nozzle at a transonic Mach number of 1.1. The predictions were made using the OVERFLOW Navier - Stokes program. The CFD results show that the flow over the upper nozzle flap separates near the aft end with a classical singular saddle point and forms an open separation bubble. The flow over the side nozzle surfaces separates and rolls up into a vortex separation surface.

Flow Development on a Round Nose Airfoil



This illustrates the typical effects of increasing Mach number on the lift, drag and pitching moment of the airfoil shown in the previous picture.

The drag, lift and the pitching moment tend to increase as the flow becomes increasingly supercritical and the region of local supersonic flow grows. The upper surface shock strength continues to increase with increasing Mach number and ultimately may become strong enough to separate the boundary layer.

When flow separation occurs, there is an increase in drag, a sudden loss of lift near the aft part of the airfoil and a resulting dramatic nose-down pitching moment. The loss of lift is due to the boundary layer separation on the upper side of the airfoil. This phenomenon is called "shock stall". It has some analogies with the airfoil stall, past the angle of CLmax . Shock stall is quite dependent on the Reynolds number and occurs earlier at typical wind tunnel conditions then in flight.

As the Mach number continues to increase, the upper and lower surface shocks move downstream and the amount of separation decreases, and the airfoil recovers part of its lift, until the free stream becomes supersonic. Beyond that point, the lift decreases again, although more gradually.

The precise effects of increasing Mach number is very dependent on the geometry of the wing. Wing thickness, camber, sweep are among the most effective methods of postponing the shock stall to higher speeds.

Transonic Flow Over an Airfoil



This figure contains pictures of the flow past an airfoil at a fixed angle of attack. These were obtained with schlieren optics which make strong density changes such as shocks, boundary layers and wakes visible.

The shock waves are evident in the pictures as dark lines. These shock waves mark the boundary between a region of supersonic flow in front and the subsonic flow behind. The corresponding pressure and Mach number distributions on the airfoil are shown below each picture.

It is readily seen that even small changes in free stream Mach number can cause significant changes strength and the location of the shocks and in the nature of the flow over an airfoil. Thus the location of the lift force on an airfoil can change dramatically.

This problem becomes even more obvious in the practical case of cambered wings and bodies at angle of attack at transonic speeds where the stability and control characteristics of an airplane can be greatly affected by the rather sudden changes in the magnitude and location of the lift vector.

For the example shown in the above figure, the flow over the airfoil at Mach 0.70 is free of shocks and has a thin boundary layer and a thin trailing wake.

At Mach 0.80, A rather mild upper shock appears. As the Mach number increases to 0.82, the strength and size of the shock increase and the shock moves slightly aft on the airfoil. A region of supersonic flow also appears on the lower surface as evidenced by a series of mild compression waves.

At Mach 0.83, a strong upper shock is accompanied by a weaker shock on the lower surface. The trailing wake is thickened and a slight loss of lift is seen near the aft portion of the airfoil. As the Mach number is further increased to 0.85, the upper and lower shocks become stronger and start to move aft. The boundary layer appears to separate and results in an unsteady rather thick wake. This results in a significant loss in lift near the trailing edge and a resulting increase nose up unstable pitching moment. These effects are even



Flow Characteristics of Sharp Airfoils

This illustrates the flow development over a circular arc pointed nose airfoil with Mach number from subsonic through hypersonic speeds. The pressure distributions on the airfoil are also shown for the different Mach numbers. Since the pressure drag is the integral of the product of the pressure times the local airfoil we can easily interpret the relative areas of drag and thrust on the airfoil.

At subsonic conditions the areas of thrust and drag balance so that the net pressure drag is zero. At transonic conditions below Mach 1, the the "drag" on the aft end of the airfoil over powers the "thrust" on the front of the airfoil resulting in drag on the airfoil. The drag at this condition is directly related to the recovery shock. If the shock is sufficiently strong, the boundary layer will separate causing additional drag.

At supersonic and hypersonic conditions, the pressure drag is due to the wave drag of the airfoil and is also related to the shock losses over the front and aft shocks.

Schlieren Photographs of Flow Past Airfoils



SHOCKS: FLOW: BOUNDARY LAYER: WAKE: None Steady Thin Small Lambda Type Fluctuating Thick Large and Violent Bow and Trailing Edge Steady Thin Small

This figure shows a number of Schlieren photographs of flow past airfoils at subsonic through supersonic speeds.

The upper series of pictures show the flow around a round nose subsonic type of airfoil. The lower series of pictures show the flow development over a pointed nose supersonic type of airfoil.

The flow over both airfoils at the subsonic condition is attached with a thin boundary layer and small trailing wake.

The high transonic pictures show the existence of very strong shocks that separate the boundary layer and results in large trailing wakes.

The pictures for the supersonic flow condition shows that the shock on each airfoil has moved to the aft of the airfoil. The boundary layer remains attached over the airfoil and the wake behind the airfoil once again is small.

The sharp nose airfoil has an attached bow shock where as the front shock for the round nose airfoil is detached and is located upstream of the airfoil.

This pictures indicate that as the flow increases from subsonic to supersonic speeds, the airfoils encounter rather violent changes in their aerodynamic characteristics.

Comparison of Airfoil Characteristics



Airfoil Designs have evolved from the conventional airfoils used in the early commercial jet aircraft to the more recent supercritical airfoils inherent in today's modern subsonic commercial jet aircraft. In this figure, a classic conventional airfoil is compared with a supercritical airfoil that has been designed to delay and reduce the transonic drag rise, due to both the wave drag of strong normal shock and the shock-induced boundary layer separation drag.

Relative to a conventional airfoil, a supercritical airfoil has reduced amount of camber, an increased leading edge radius, small surface curvature on the upper or suction side, and a concavity in the rear part of the pressure side. The concavity near the trailing edge produces lift to make up for the reduction in lift near the front of the airfoil. The lower surface if bulged near the front portion to produce the necessary wing thickness. Supercritical airfoils tend to have increased aft loading and a relatively large nose down pitching moment. Some of the early supercritical airfoils where designed and wind tunnel tested by Whitcomb at NASA Langley in the 1950s.

On the suction side the steep supersonic acceleration of the conventional airfoil is eliminated. The flow reaches supercritical speed, that is maintained over a large part of the airfoil. Then there is a small pressure plateau behind the shock and a pressure recovery in the trailing edge region. The reduction in local Mach number also reduced the strength of the shock when the flow decelerates below critical speed.

On the pressure side the flow is maintained at subcritical speeds, with a flat distribution with a Stratford-like pressure recovery at the trailing edge.

Design of supercritical airfoils has been made easier in recent years, due to the progress in computational

Supercritical Airfoil Offers Many Design Options:

- Increase Cruise Speed
- Reduce Wing Sweep and Increase Aspect Ratio → Increase L/D
- Increase Wing Thickness → Reduce Weight,
- Increase Wing thickness → More Fuel Volume for Range Growth
- Increase Wing Thickness and Aspect Ratio → Increase L/D
- Increase CL → Airplane Growth to Higher Gross Weights
- Any Combinations of the Above
- Increase Buffet Boundaries
- Adverse Effect on Trim Drag for Aft-Tail Configuration
- Possible Beneficial Effect on Trim Drag for a Canard Configuration

As shown in this figure, the increased speed capability of a supercritical airfoil can be utilized to provide a number of different design options in developing an optimized aircraft configuration.

One-Dimensional Flow Streamtube Flow

Combine The Euler Equation is:
$$U \frac{dU}{dx} = -\frac{1}{\rho} \frac{dp}{dx}$$

With the Definition of the Local Speed of Sound: $\left(\frac{\partial p}{\partial \rho}\right)_s = a^2$

With the Equation of Continuity for a Streamtube: $\frac{d}{dx}(\rho US) = 0$

Where S(x) = Streamtube Cross Sectional Area



Many of the interesting physical features of the transonic flow can be qualitatively understand by considering the simplest case of one- dimensional flow in a stream tube.

The Euler equation for one-dimensional flow, which relates the change of velocity in the stream tube to the pressure gradient along the tube, is: dU = 1 dr

$$U\frac{dU}{dx} = -\frac{1}{\rho}\frac{dp}{dx}$$

$$\frac{d}{dx}(\rho US) = 0$$

The equation of continuity states that the mass flow is constant along the stream tube is;

where S(x) is the stream tube area

The definition of the speed of sound -----

Combining these equations gives the following equations that relate an incremental change in stream tube area, dS to the corresponding incremental change in pressure dP as S

$$\frac{dP}{\rho U^2} = \frac{1}{(1-M^2)} \frac{dS}{S}$$

 $\left(\frac{\partial p}{\partial \rho}\right) = a^2$

Or by rearranging the terms in the equation, yields

These equations defines the relation between the cross sectional area change for a streamtube and the associated pressure gradient pressure gradient.

Basic Features of Transonic Flow

- Local Particle Speeds Both Greater and Less Then the Sonic Speed coexist.
- Large Pressure Changes in Transonic Flow Produce Small Stream Tube Area Changes
 ds
 de

$$\frac{dS}{S} = \sqrt{1 - M^2} \frac{dP}{\rho U^2}$$

- Stream Tube Areas Are Virtually Constant: An Increase In Fluid Velocity Is Balanced By An Equal Decrease In Fluid Density
 - Between Mach = 0.4 and Mach = 0.7 Subsonic Stream Tube Area Change is 31 %
 - Between Mach = 0.9 and Mach = 1.1 Transonic Stream Tube Area Change is 0.9 %
- Streamtubes can Curve and be Displaced by Lateral Pressure Gradients.
- The Cross Flow in Planes Normal to the Free Stream Tends to be Incompressible as in the Case of Flow near a Slender Body

The previously shown relationship between cross sectional area change and pressure gradient

$$\frac{dS}{S} = \sqrt{1 - M^2} \frac{dP}{\rho U^2}$$

shows that for flows near the speed of sound, streamtube areas are virtually constant and the flow will resist any stream tube area changes.

The streamtubes however can be changed in curvature and lateral displacements by lateral pressure gradients.

Therefore the cross flow in planes in planes normal to the free stream direction will tend to be incompressible as in the case of a flow near a slender body.

Near Sonic Streamtube Characteristics



This illustrates the fundamental behavior of a set of initially circular streamtubes of air flowing past a body of revolution at near sonic speeds.

The Streamtubes are displaced radially by the body streamwise area changes. Near Mach one the cross sectional area of the streamtubes remain constant as the flow passes over the body. The stream tube cross sectional shapes, particularly close to the body will vary from the initially circular cross sections. Any cross flow will further distort the streamtube shapes, however the cross-sectional area remains constant.

The cross section shape distortion effect is only close to the body and in the cross flow plane.

Transonic Flow Characteristics



Subsonic Flow

Stream Tube Area Contraction Over Maximum Thickness Point Causes Disturbances to Diminish Rapidly in the Vertical Direction

Transonic Flow

Stream Tube Area Constancy Over Maximum Thickness Point Causes Stream Tubes Remote From The Object to Retain The disturbances Imposed by the Surface

• Disturbances in Transonic Flow Propagate Laterally to Large Distances

- Effects Such as Wing-Body Interference Are Not Localized in Transonic Flow And Encompass the Entire Wing
- Wind Tunnel Wall Interference Effects are Very Strong

Transonic or "Slender Body" Equivalence Rule

- Far From the Configuration the Flow Becomes Axi-symmetric and Equal to the Flow about an Equivalent Body of Revolution.
- The Total Flow Field About a Configuration Can be Obtained as the Sum of Two Parts
 - 1. A Mean Flow Corresponding to That of the Equivalent Body of Revolution
 - 2. A Two-Dimensional Incompressible Cross-Flow Part That Makes the Tangency Condition at the Body Surface Satisfied.
- The Cross-Flow Represents the Effect of Local Flow Divergence Due to Changes in Cross-sectional Area, in Cross-Section Shape or Position of the Section in the Flow Field.
- The Equivalent Body Mean Flow Contains the Mach Number Dependence and Accounts for the Accumulative Effects of the Distance Sources Representing the Equivalent Body.
- The Induced Drag Is Dependent in the Flow in the Cross Flow Plane and is Therefore Independent of Mach Number.
- The Drag Rise of the Entire Configuration is Due to the Equivalent Body Providing that the Wing Does Not Become Critical in the Flow Field Induced by the Equivalent Body

Local Geometry Cross Flow Part

- Lateral Components of a Disturbance have no upstream or downstream effects and Propagate Only Laterally
- Lateral Velocity field about an Airplane can be Calculated by Incompressible Cross flow Concept.
- Cross Flow Lateral Effects are local effects and Decay Rapidly Laterally



The local geometrical cross flow part arises since at transonic speeds the lateral components of a disturbance tend to only propagate laterally. The nature of this flow can be most readily visualized by fixing an observation point within the fluid and watching the configuration fly past.

Consider a thin slab of fluid aligned perpendicular to the flight path and observe what happens to the fluid in the slab as the aircraft flys through it. The fluid will flow laterally to allow the airplane to flow through the fluid slab.

The Contributions to the Flow Field Due to the Lift Distribution and the Volume Shape Effects Are Contained in the Cross Flow Solutions.



Downward "Plunge" of Entire Cross Section or Parts of the Section Represents Angle of Attack Effects and twist / camber effects

The contributions to the flow field due to the lift generated by the aircraft as well as the fine details of the local geometry are all contained within the cross flow solutions.

The fluid being "pushed out" or "flowing into" the configuration represents the changing shape of the configuration. The downward"plunge" of all or part of the configuration represents angle of attack effects as well as the camber and twist effects. The angle of attack effects and the camber / twist effects generate the lift distribution on the configuration.

The cross flow is governed by the incompressible two $\frac{\partial^2 \varphi}{\partial L}$ two $\frac{\partial^2 \varphi}{\partial L}$ is solved by the incompressible two $\frac{\partial^2 \varphi}{\partial L}$.

$$\frac{\partial \phi}{\partial n} = U_{\infty}(\text{local surface inclination} - \alpha \bullet \sin \theta)$$

The boundary condition on the surface satisfies the flow tangency requirement.

Mean Flow Equivalent Body Part

- A Thickness or Volume Element Induces Fore and Aft Velocities if Mach < 1
- A Thickness or Volume Element Induces Aft Velocities if Mach > 1
- At Near Sonic Speeds the Induced Velocities Do Not Diminish Laterally
- Consequently The Far-Field Flow is Not Affected if All the Airplane Cross Sectional Area Is Shifted Laterally to the X-Axis, Producing an Equivalent Body



Lateral Extent of Flow Field Influence

Local Mach Number Distribution



TRAINAIR Study by John Chan

This shows the results of flow calculations about a body of revolution using the TRAINAIR full potential code. The corresponding pressure or wave drag variation with Mach number is also shown.

These results illustrate the large lateral extent of the flow field influenced by the body at the near sonic conditions.

The lateral extent of the flow poses significant challenges both for wind tunnel measurements as well as CFD analyses. The lateral extent of the flow is far beyond the walls of the typical wind tunnel and also larger then the typical outer boundary of the CFD calculation region. Consequently well calibrated porous slotted wind tunnels must be used to obtain experimental measurements relatively free of wall interference effects.

The CFD calculation lateral boundary for near sonic analyses, must be greatly extended relative to analyses at subsonic conditions. This results in increasing both the grid density as well as the resulting computational time.

Mean Flow Equivalent Body Part



The flow over the equivalent body of revolution is governed by the non-linear full potential equation:

$$(1 - M_{\infty}^2 - (\gamma + 2)M_{\infty}^2 \phi_x)\phi_{xx} + \phi_{rr} + \frac{1}{r}\phi_r = 0$$

$$\frac{\partial \phi}{\partial r} = \frac{dR(x)}{dx}$$

on the equivalent body at r =

Subject to the boundary condition: R(x)

The solution of this problem is still a difficult problem though relatively easily handled by existing CFD methods. However, utilizing the concept of the equivalent body does provide a simple means to interpret the near sonic flow phenomena as related to the physical characteristics of the aircraft configuration.

This indeed may be the most useful part of the near sonic general slender body theory.

Mean Flow Equivalent Body Part



- The Entire Airplane Flies at the Free Stream Mach Number M_{∞}
- Each Local Part of the Airplane Feels a Different Effective Local Mach Number
- The Effective Local Mach Number is Dependent on the Rate of Change of the Area Distribution of the Equivalent Body and Depends only on X, the longitudinal Station.

The results of the general slender body theory imply the following²:

- The entire airplane flies at a free stream Mach number equal to the free stream Mach number
- Each local part of the airplane feels a different effective Mach number.
- This effective local Mach number is dependent on the rate of change of the area distribution of the equivalent body and depends only on X, the longitudinal station.

This interpretation will be utilized to aid in the understanding of the main features of the flow about an aircraft configuration at transonic speeds. The interpretation is also useful in identifying desirable features of a configuration designed to fly at near sonic speeds.

Wing-Body and Equivalent Body Flow Relationships



A lifting wing plus body configuration operating at near sonic speeds will typically have two shock systems. These include an area shock that is associated with the overall area distribution of the configuration and the wing lift shocks associated with the basic design of the wing.

The area shock strength, location and wave drag is identical to the area shock on the equivalent body of revolution.

The equivalent area distribution also creates the Mach number distribution that the wing design must operate in and therefore has an influence on the drag rise of the lifting wing.

The drag rise of the configuration is composed of both the equivalent area distribution drag rise, plus drag rise of the wing operating in the equivalent body induced flow field.

Examples of In-Flight Area Shocks



The presence of the area shock waves on aircraft cruising at transonic speeds is quite evident in these pictures.

Transonic Equivalent Body Analyses

What it Can Do:

- Predict the "Lowest" Potential Drag Rise for a Specific Configuration Arrangement.
- Help Identify Configurations Modifications (Component Shapes and Locations) for Performance Improvements.
- Determine the Optimum Fuselage Shape for Lowest Potential Induced Mach Number and Minimum Potential Drag Rise for Given Configuration Arrangement.
- Aid in Comparing Configurations and Sorting Out Design Options.
- Useful in Developing a Good Initial "Seed" Geometry for Non-Linear Design Optimization.

What it Cannot Do:

- Compensate for a Fundamentally Poor Wing Design.
- Guarantee a Certain Level of Aerodynamic Performance.
- Replace the Need for Design Optimization and Detailed Design Integration.

The transonic equivalent body concept, as it will be shown, is an extremely useful concept for both understanding the physics of transonic design integration as well as providing a mechanism for initiating preliminary transonic design integration and also for evaluating alternate configuration designs.

However, as with any simplified aerodynamic tool or methodology, it is essential to understand both the usefulness of the transonic equivalent body concept as well as the limitations.

The figure summarizes both "what it can do" and "what it cannot do ".



Whitcomb based his formulation of the Sonic area rule upon an experimentally verified equivalence between wings and bodies at sonic speeds. It was observed that there does exist a marked similarity between the shock wave patterns around a wing-body combination and that of a body of revolution having the same cross-sectional area distribution.

Comparisons of Wing + Body With Equivalent Body



Experimental studies have been made to compare the drag rise characteristics of various wing-body configurations with the drag rise characteristics of their corresponding equivalent bodies of revolution having the same axial development of cross sectional areas normal to the free stream flow⁴. Some of the results of these experiments are shown above.

The basic body of revolution had a 6 to 1 ogive nose followed by a cylindrical section for an overall body length to diameter ratio of 11.47.

The planforms included:

- Delta wing having an aspect ratio 4, taper ratio of 0, with a leading edge sweep of 37 degrees. The streamwise airfoil sections for this wing are 4 % thick and consist of circular arcs with the maximum thickness at 40 % chord.
- Swept wing having an aspect ratio of 4, taper ratio of 0.6, leading edge sweep of 45 degrees. The streamwise airfoil sections were NACA 65A006 airfoils.

The drag coefficient increments, ΔCD_0 , shown above were obtained by subtracting the drag coefficients measured at a Mach number of 0.85 from those measured at a higher Mach number. This subtraction nearly eliminates the effects of the differences in the skin friction between the various configurations and therefore can be considered equal to the wave drag of each configuration.

The results show that the drag rise of the equivalent body is indeed very close to that of the actual wing plus body.

It is also seen that the addition of the wing substantially reduces the critical Mach numbers of the wing plus body configurations relative to the isolated body.

The critical Mach number with the swept wing is seem to be substantially higher then for the delta wing.



Effect of Wing Aspect Ratio on Mean Flow Characteristics



Let us now use the equivalent body concept to explore the self induced flow characteristics of various wing planforms.

This compares the shapes and area distributions of the equivalent bodies for a low aspect ratio wing and also for a high aspect ratio wing. The calculated local Mach number distributions for the each wing are also shown to illustrate the difference in the flow characteristics on the wings.

The low aspect ratio has a short stubby equivalent body whose radius increases with the thickness of the wing. The local Mach number over much of the equivalent body supersonic and is significantly higher then the free stream level. The region of supersonic flow is terminated by a strong recompression shock.

Since the wing flow can be interpreted as if each individual section of the wing were exposed to a different local Mach number corresponding to the equivalent body mean flow. It is clear that most of the wing is subjected to the local supersonic flow and hence encounters drag divergence much earlier then corresponding to the basic airfoil section properties. The shock wave appearing in the equivalent body basic flow will be reproduced on the wing.

The equivalent body for a medium aspect ratio wing corresponding to aspect ratios of 7 to 8. Is also shown in the figure.

The equivalent body local Mach number distribution results in a supersonic flow region near the nose and near sonic flow near the aft end of the body and hence will encounter drag divergence earlier then other areas of the wing. The local Mach number decreases to a nearly constant level over the mid portion of the body that is equal to the local free stream Mach number.

Typically, then the higher aspect ratio wing will have a higher drag divergence Mach number then a



The use of the equivalent body can also be used to explain why the classic simple sweep theory most often fails to work in full accordance with the theory.

The basic simple sweep relations between the various 3D aerodynamic quantities and the corresponding equivalent 2D values are shown in the figure.

According to simple theory, the critical Mach of a infinite swept 3 D wing would increase by the inverse of the cosine of the wing sweep angle.

Wind Tunnel Pressure Distributions On a Swept Airfoil at $M_{\infty} \cos \Lambda = 0.65$



 $\alpha_{N} = 2 \text{ deg}$



This figure shows that the equivalent 2d normal pressure distributions on infinite swept wings are indeed as predicted by the simple theory for a subcritical normal Mach number of Mn = 0.65 for wing sweeps on 20 and 40 degrees. The corresponding free stream Mach numbers are 0.69 and 0.86 for the two swept wings.

"Infinite" Swept Wing \thickapprox Two Dimensional Airfoil Theory



The equivalent body for an "infinite" swept wing is a "infinite" length constant area cylinder. Consequently, the equivalent area distribution has zero slope and therefore induces no additional Mach number on itself because of it's longitudinal area distribution. The airfoils sections on the wing behave as swept 2-D airfoils at a a free stream Mach number corresponding to the normal Mach number.

The mid-span region on a very high aspect ratio swept wing operates as if it were part of an infinite wing since the Mach number level has decreased to the free stream level because of the long nearly cylindrical mid-portion of the equivalent body. The regions of the wing near the apex and the wing tip would, however, experience higher then freestream Mach numbers corresponding to the nose and aft end of the equivalent body area distribution.



Effect of Wing Sweep on Wing Equivalent Body Area Distribution

This compares the wing equivalent body area distributions for two finite wings differing only in wing sweep.

The effect of increased wing sweep is to lengthen the equivalent body and to reduce the maximum cross sectional area of the equivalent body. The equivalent bodies, however, both will induce a local Mach number that will be considerably greater then the free stream Mach number over portions of the wing.

The rapid area growth on the 35deg sweep wing results in a strong forward shock and a milder shock near the wing tips. Providing that the inboard flow is not separated by the forward shock, the mid span of the wing will behave much as an infinite swept wing with the local Mach number being equal to the free stream Mach number.

Why Simple Sweep Theory Does Not Always Work "Perfectly"



This figure illustrates why simple sweep theory does not always work "perfectly".

Because of the self induced flow effects, the local sectional Mach numbers may be significantly greater then the free stream Mach number. This is particularly in the wing root and wing tip regions. Physically this corresponds to an unsweeping of the wing isobars near the wing root and near the wing tip. The mid portion of the wing planform may indeed correspond to an infinite swept wing region. Thus it can be seen that the use of wing sweep to delay the onset of subsonic drag rise is never as effective as that predicted for the infinite yawed wing on the basis of the sweep independence principle.

The use a wing strake to increase the wing sweep near the wing root using, along with a raked wing tip to increase the sweep in that region are effective means to increase the drag rise characteristics of a swept high aspect ratio wing.

Wing / Body Self Induced Effects On Critical Mach number



The effect of maximum thickness to chord ratio in the critical mach number for a "classic" NACA airfoil typical to the airfoils used on the previously discussed wind tunnel models are shown above.

For each of the experimental models previously discussed, the critical Mach number elements are shown. The basic airfoil critical Mach numbers corresponding to the wind tunnel models maximum thickness /chord ratio are shown.

The increase in critical Mach number corresponding to the wing sweep for an infinite aspect ratio wing is also shown. This results in a substantial increase in Mach critical that is inversely proportional to the cosine of the sweep angle.

The self induced effect is due to the finite aspect ratio of the wing along with body effects. These are seen to result in a substantial reduction in Mcrit. The critical Mach number of the isolated body is shown for reference

Both of the experimental wings discussed above, had thin wings, (T/C = 4%) compared to typical subsonic and prior near sonic designs ($T/C \sim 10\%$). Consequently the drag rise Mach numbers shown for the configurations above may appear relatively high.



Effect of Sweep on Transonic Drag Rise



Long Body Versus Short Cylindrical Body

- \approx Equal Pressure Drag Rise
- \approx Equal Critical Mach Number
- -More Friction Drag
- Greater Region to Accommodate the Wing +





This illustrates the general characteristics of an equivalent body for a cylindrical body plus wing. The calculated local Mach number distribution for the combined wing equivalent body is also shown along with the local Mach distribution for the isolated wing.

At the nose of the forebody, the body experiences a local Mach number level lower then free stream. The flow expansion around the nose results in a region of supersonic flow followed by a rather strong shock that lies forward of the wing. The local Mach just in front of the wing is actually slightly less then free stream. The area growth of the wing then results in a region of supersonic sonic flow followed by a shock near the front portion of the wing. The mid span region of the wing experiences a local Mach number about equal to the free stream. The presence of the body actually softens the wing front shock and eliminates the mild shock near the wing tips relative to the local Mach number distribution on the isolated wing.

Because of the strong body shock and the wing shock, the wing body configuration will become critical at a Mach number considerably lower then the critical Mach number characteristics of the basic airfoil sections.

Effect of Planform Modifications



Typically a wing with a straight leading edge begins to add suddenly to the combined equivalent body area distribution.

The addition of a glove to the leading edge area can reduce the frontal area slope without adding to the maximum area of the equivalent body. As shown in the local Mach number distribution, the glove actually eliminates the wing front shock This results in approximately a 16% reduction in the drag rise at Mach 0.9.

The addition of a trailing edge extension, typically will add to the the maximum area of the equivalent body. This will increase both the induced Mach number on the wing and strengthen the wing front shock and also results in a 48% increase in the drag rise at Mach 0.9.

Consequently it is obvious that proper tailoring of the wing planform is very important for a transonic configuration.

Effect of Body Contouring on Wing-Body Drag



This shows the drag rise increments for the previously described isolated bodies and the wing plus body configurations along with the data obtained for the wings located on indented bodies. The indented bodies were designed such that the equivalent bodies for the wings plus indented bodies equaled the basic cylindrical body.

The indented bodies dramatically reduced the drag rise associated with the wing. In fact the drag rise for the swept wing plus indented body is nearly equal to the isolated baseline body up to Mach 1.04.
Effects of Body Contouring



Effect of Body Area Ruling



The data in the previous figure showed that contouring a body in the area of the wing can dramatically reduce the drag rise for a wing plus body configuration.

The effect of shaping the overall area distribution of a configuration to match an optimum distribution designed to produce a low induced Mach number might be expected to produce an even greater benefit.

The figures on the left show the equivalent area distribution and the corresponding local Mach number distribution for a wing plus cylindrical body.

The figure on the right shows the effect of shaping the body to match an overall optimum area distribution. This actually required adding substantial volume to the front and aft ends of the body since the minimum mid body area was held equal to that of the cylindrical body.

Contouring or area ruling the body is seen to dramatically reduce the equivalent body local Mach number to a level that is totally subsonic and only slightly above the free stream Mach number.

Comparison of the drag rise characteristics for the two configurations shows that the optimum body has totally eliminated the drag rise up to the last calculated condition of Mach = 0.98.

Transonic Optimum vs Sears Haack Body





An important refinement to the body optimization process is to include what is commonly called lift compensation. The lift compensation area is an allowance for the expansion of the stream tubes in the supervelocity regions on the wing upper surface where local Mach numbers on the order of 1.6 can be reached. The shape and magnitude of the lift compensation area correction has in the past been determined by both Boeing and NASA wind tunnel test programs. As shown in the figure, accounting for the lift compensation is very important on the drag rise characteristics of a configuration. The lift compensation area reduction is typically applied to the fuselage above the wing upper surface since the stream tube expansion occurs on the upper surface.





We can use these fundamental concepts to gain some information concerning the potential interaction between a nacelle and a wing.

The effect of a nacelle can be broken down into it's cross flow effect and it's equivalent area Mach number effect. Locating a nacelle in a longitudinal area where it adds to the maximum cross sectional area is clearly an undesirable location. If the nacelle is also adjacent to the wing surface would compound the adverse effects by adding to the cross flow velocities near the wing . This would both force the wing and the nacelle to fly in higher Mach number environments.

The best locations for the nacelles would be such to smooth out the wing area growth on the upstream and / or down stream sides.



This shows the equivalent body area distribution for a B747.

The area of the wing body intersection is seen to be quite large in the region of the wing and adds substantially to the total area distribution. This most certainly increases the local Mach Number that the wing operates in and there by decreasing the critical Mach number of the configuration.

The results of an exploratory study of a cab extension is also shown. The extended cab smoothes out the front portion of the area distribution. The net effect at cruise lift coefficients is a significant increase in critical Mach.



Boeing wind tunnel tests results for a near sonic transport wing-body combination⁵ are shown in the figure on the left. The incremental drag rise above Mach 0.8 is compared to the corresponding drag for the axi-symmetric equivalent body. The drag rise characteristics are essentially equal even though the wing body is at the lift coefficient appropriate for cruise and the equivalent body is at zero angle of attack.

The figure on the right shows the application of the equivalent body area rule concept to a complete near sonic configuration⁶. The shape of the total combined area plot is seen to be completely smooth thus delaying the appearance of the configuration shock and the associated drag rise to near-sonic speeds. The actual area is somewhat smaller then the total area in the vicinity of the wing to provide compensation for the wing lift. The lift compensation in it's original form was put forth by R.T. Whitcomb for applications to configurations operating at near-sonic speeds. While the lift-compensation postulate was not developed analytically, the basic idea has been extended theoretically and the experimental results have been generally quite favorable.

The fundamental design philosophy applied to these prior near sonic configuration design activities was to develop a supercritical wing utilizing the most advanced airfoils and then utilizing the equivalent body concept to develop a completely integrated configuration. The success of this approach is illustrated in the drag rise chart on the lower right which compares the drag rise of the of the wing body configuration of the near sonic configuration with the corresponding axi-symmetric equivalent body drag and with the drag rise for the B747. The increase in the drag rise Mach number is on the order of Δ Mcrit ~ 0.12.





Another concept that can be applied at transonic speeds is the "Streamline Contouring Concept^{6,14}". The essential idea is to use aerodynamic shapes for certain configuration elements (eg fuselage, nacelles, fairings, etc) that conform to stream lines calculated for idealized flow about the wing.

In the development of the streamline contouring concept, the starting point is the conception of the infinite yawed wing as an ideal lifting wing. The streamlines on a swept wing show a lateral displacement as shown in the figure. The lateral displacement increase with increase lift. Wing thickness moves the streamlines equal amounts to the right, on the upper and lower surfaces.

Thus in order to match the streamlines the wing body intersection must differ on the upper and lower surfaces as shown. The net cross sectional area of the body is held constant and equaled to the area ruled body. Hence the aerodynamic effect is restricted to the cross flow. Because of the limited region of influence of the cross flow, it is apparent that stream line contouring can have little or no effect on the wing flow outboard of the root section.

When applied to a fuselage, the streamline concept tends to have more wetted area, weigh more and cost more to build and these typically negate the potential aerodynamic improvements. In addition, the potential benefits from stream line contouring can also be achieved by tailing of the local wing design. This is quite easily achieved with the modern CFD design and analysis processes.



Stream line contouring can also be applied locally to the integration of nacelles on a wing. An application of streamline contouring to nacelle installation on a wing is shown in the figure. The experimental results for this type of application showed both reductions in drag as well as increases in the critical Mach number.

As in the case with favorable interference of any kind. The beneficial favorable interference is strongly dependent on the quality of the starting design. With a poor initial nacelle installation it is conceivable to achieve a large beneficial effect from streamline contouring. However rather then demonstrating aerodynamic gains from the streamline concept this would just be one of a number of ways to improve an initially bad design.



This summarizes some of the desirable design features for a Near Sonic Configuration that can be identified by application the simple equivalent body concept.

Each of these should be considered in the development of a smooth overall area distribution with a low maximum area.

TOPICS

- Session 1: Effect of Speed on Airplane Shape
- Session 2: Fundamental Physics of Transonic Aerodynamics
- Session 3: Prior Studies and Technology Needs

In this session we will review some basic sonic boom fundamentals and considerations primarily related to the possibility of boom free flight at low supersonic Mach numbers.

Previous Boeing near sonic and transonic configuration studies will be summarized.

A "Tops Down" Approach will be used to suggest a paradigm shift in the features of a near sonic configuration similar to the Boeing Sonic Cruiser that differs from the configurations of the previous near sonic studies.

Some aerodynamic technology needs and challenges for a sonic cruise configuration will be summarized.



The figure above shows the effect of motion of a pulsating sound source on the generated sound waves. This leads to the well known Doppler effect. The Doppler effect is characterized by frequency changes due to a source of sound approaching an observer at speed u. In this case a single frequency (having period) travels a distance cT (c = speed of sound), while the source travels a distance uT. The apparent wavelength is changed to (c - u) T. and the frequency is shifted to its Doppler-value.

Thus, a train approaching a station at a speed of 60 m/h, and emitting a single frequency sound of 8 KHz would be heard by the passengers on the platform at 8.7 KHz.













Sounds emitted by a source traveling at supersonic speed u > c are heard by the observer on the ground in reverse order.

Da, De, Di, Do will be heard as Do Di De Da. At the sonic condition, u=c, all the sounds are heard at the same time.

Subsonic Airplane Ground Pressure Field



When an airplane is flying over the ground the weight of the airplane is transferred to the ground as an increased pressure field on the ground. For a low speed subsonic airplane, the increased pressure field has lateral and longitudinal symmetry with respect to the airplane and extends over a very large area. The supporting pressure field travels at subsonic speeds with the airplane.

The center of pressure of the induced pressure field is directly below the airplane. The maximum pressure occurs at the center of pressure and is given by the expression:

$$p_{\rm max} = \frac{W}{2\pi h^2}$$

The maximum ground pressure is extremely small since it varies inversely with the height of the airplane squared.

For the example given for a typical subsonic transport with a weight of 400,00 lbs at an altitude of 35,000 ft, the maximum pressure is an imperceptible level equal to 0.000052 lbs/ft²



The pressure field generated by a supersonic airplane is confined between two shock cones. One emanating from the nose of the airplane, called the bow shock cone, and a second emanating from the tail which is called the aft or tail shock cone. The intersections of the shock cones with the ground each delineates a hyperbola.







A typical HSCT configuration typically generates an entire system of shock waves as shown in the near field close to the airplane.

At large distances from the airplane, as indicated by the mid field, the shock wave system tends to steepen and begin to coalesce.

At extremely large distances in the far field near the ground, the shock system will typically coalesce into a bow shock and a tail shock.

At the bow shock, a compression occurs in which the local pressure, p, rises rapidly to a value Δp , above the atmospheric pressure. Then a slow expansion occurs until some value below atmospheric pressure is reached, after which there is another rapid compression at the aft shock.

Generally the bow shock and the aft shock are of similar strengths, and the pressure varies linearly between the two shocks. This nominal sonic boom signature is called the "N" wave. This "N" wave moves at supersonic speeds with the aircraft.

The region between the defined by the area between the intersections of the bow shock and the tail shock is known as the primary sonic boom carpet. Receivers in this carpet detect the sonic boom, that is the "N" wave as the airplane passes.

The primary carpet sweeping by a person outside on the ground would result in a audible response as shown in the figure. Since the ear detects changes in pressures only above a certain frequency, it will respond only to the steep parts of the pressure waves and not to the gradual pressure changes in between.

Typically the time between the front and rear shock is of the order of 0.1 sec or greater and the ear will probably detect two loud "booms". The degree of indoor disturbance or structural damage attributed to the pressure signature is believed to be more dependent on the impulse (the integral of the positive portion) and the length of the signature.





Shock waves produced by an airplane flying at speeds only slightly in excess of Mach one (local speed of sound) may not reach the ground. This is caused by warm layers of air below the airplane that refract or bend the path of the shock wave.

Higher atmospheric temperatures below the airplane leads to an increase in the local speed of the shock wave. If the local speed of the shock wave is the same as the airplane speed, at some altitude below the airplane the shock wave will be refracted back into the atmosphere and will not reach the ground. It will be perpendicular to the horizontal at that altitude. Below that altitude, the local speed of pressure disturbance is greater than the airplane speed, and a subsonic pressure disturbance with a gradual pressure rise will occur. When an airplane speed is just equal to the speed of sound at the ground, the airplane Mach number is defined as "the threshold Mach number"

The Threshold Mach number is dependent on variations in temperature and wind gradients below the airplane and the ground¹⁰. As a result, it varies from day to day and place to place. Head winds in the vicinity of the airplane increase the threshold Mach number while tailwinds tend to decrease it.



Boom free supersonic flight is possible as long as the ground speed is ,therefore, less then the speed of sound at the ground. Depending on the flight direction, season and airplane altitude, cruise Mach numbers in the range of 1.04 to 1.25 are allowed for east to west and west to east transcontinental US flights.

The use of advanced Global Positioning Systems, can provide a continuous and accurate measure of the ground speed. This together with local information about the temperature and wind could allow the boom free flight conditions to be determined,



One of the critical elements of sonic boom activities is determining the response to the sonic booms.

There is considerable there is considerable concern about the manner in which people and structures respond to sonic booms and how such responses will affect the community acceptance of supersonic overland flight.the nature of the response problem is illustrated in the figure above.

The sketch suggests two different exposure situations for people.

In one case, the person is outdoors and is impinged on directly by the sonic boom.

In the other case the observer is inside the building and the waves impinge on the building. The building then acts as a filter which determines the nature of the exposure stimuli reaching the inside observer.the ingredients of the indoor stimulation is shown in the chain diagram.

The sonic boom induced excitation which causes the building to vibrate may arrive either through the ground or through the air. It is generally considered that the air path is the more significant one and is called the primary path. Building vibrations may be observed directly by the subject. The person may also sense the vibration induced noise. In extreme cases, superficial damage to the structure may occur.

Where as the outside exposure is related to the strength of the booms, the inside exposure is considered to be related to the impulse of the boom, which depends in the total positive pressure on the signature along with the exposure time.

The chart above lists some of the research needs in the very important area of sonic boom response design criteria.



The results of previous Boeing Near-sonic Transonic aircraft design studies will be discussed. These include the Boeing in house and Boeing / NASA Contract studies that involved both detailed configuration development studies and extensive wind tunnel test development programs.

The Boeing/NASA high transonic speed transport aircraft studies that were focused on low supersonic cruise speeds will be summarized



A NASA engineer, Robert Gilruth devised this way this method of conduction transonic research. He had observed that significant supersonic flow existed on the top of a North American P-51 wing during dives even though the airplane speed was only M=0.75. As the first application of the wing-flow technique, a small airfoil model was mounted perpendicular to the P-51 wing upper surface. Subsequently many other airfoils and aircraft models were tested. This technique provided the most systematic and continuous plots of transonic data assembled by the NACA at that time. This technique also provided the first experimental validation of R. T. Jones' theory which predicted the drag of a wing in transonic and supersonic flight would be reduced significantly if the wing was swept.

A similar technique was used to obtain transonic and supersonic data in subsonic wind tunnels by mounting small models on the top on bumps placed on the wall of a wind tunnel that produced a region of increased speeds including supersonic flow.

Boeing 1969 Near Sonic Wind Tunnel Test Development Program



The extensive wind tunnel development program conducted by Boeing focused on a number of objectives that included:

- development of an understanding of transonic aerodynamic design elements and options
- validation of the transonic design concepts
- Obtaining performance data to support the evaluation of the near sonic transport feasibility assessments.

A variety of wing planforms, nacelle geometry and location, fuselage shapes, and complete configurations were investigated. The Boeing tests, together with complimentary NASA Langley near sonic configuration work provided significant insight into the aerodynamic design requirements of such a configuration.



This shows more details about the various wind tunnel models and model parts that were part of the aforementioned extensive Boeing Wind tunnel test programs.

The test configurations included:

- Axi-symmetric bodies variations to investigate body area distribution shapes, fineness ratios and the position of the maximum area.
- Various wing planforms corresponding to different wing designs, different fundamental wing airfoils, and with and without wing gloves (strakes) and trailing edge extensions (yehudi)
- Various complete model area distributions to investigate forebody shape, aftbody shape, with and without lift compensation
- various external flap track fairings and gear pods on the wing
- · various wing mounted and body mounted nacelle locations

F8 With Supercritical Wing Flight Test Vehicle ~ 1971



Historically, flight research has played a pivotal role in the advancement of aeronautical science and technology and has long been the preferred way to prove new aerodynamic concepts or configurations. Concepts that improve vehicle performance in the transonic speed range have been routinely been validated in flight because of the difficulty in making accurate and precise measurements in wind tunnels at near sonic speeds.

One of the proof of concept programs that was very successful was the application of Dr. Whitcomb's supercritical wing wing concept to the Vought F8A aircraft. This aircraft was chosen because the wing could be easily removed and replaced with a new design, the landing gear retracted into the fuselage and not the wing, and because the F-8 had supersonic capability. The results of the flight test program did demonstrate as much as a 15% increase in performance.





In 1970 the Boeing Company⁵ conducted an analysis of the design, performance and economics of Mach 1.2 and 0.98 transports relative to a subsonic advanced technology transport designed to cruise at Mach 0.84. All three transports were designed for the U.S. transcontinental range of approximately 2800 nmi with equivalent payloads of 175 passengers. The design Mach of 1.2 was selected to represent a boom free supersonic cruise option.

The study configurations are shown above. The near sonic and transonic configurations where derived from subsonic configuration design philosophy extended to the higher design Mach numbers. The Mach 0.98 configuration had slight body contouring to prolong the drag rise. The fuselage of the Mach 1.2 configuration was "area ruled" to minimize the transonic wave drag.

The configuration development was supported by an extensive wind tunnel development program conducted in 1969 and 1970.


This compares the interior arrangements of the study configurations. The area rule constraints for the near sonic and low supersonic configurations dictates a departure from the cylindrical designs used on conventional subsonic transports. Body waisting is incorporated on the faster aircraft.

The subsonic configuration was designed to have a constant 7 - abreast double isle arrangement.

The Mach 0.98 airplane at its minimum section has a double isle 6 - abreast arrangement. Consequently the fuselage is longer than the subsonic configuration for an equal number of passengers.

The longer, thinner and more highly waisted Mach 1.2 fuselage has a single isle and 4 abreast seating at its minimum section.



The aerodynamic performance, L/Dmax comparisons for the transonic study configurations are show in the upper right figure. There is a sudden decrease in aerodynamic efficiency as the Mach number exceeds 1. The range factor, which is the product of the aerodynamic and propulsion efficiencies peaks at the near sonic flight condition.

The payload to gross weight ratio is seen to decrease as the design Mach number is increased. This is the result of increased structural weight associated with increased wing sweep, increased body length, body contouring and a required increase in engine size.

The direct operating costs, DOC, are shown relative to the subsonic baseline configuration. The approximate 1 hour transcontinental time savings obtained by the Mach 0.98 transport is attained at a cost of only a 5 % increase in DOC. The more attractive two-hour time savings offered by the Mach 1.2 configuration lost it's appeal in view of the 50% increase in DOC.

The results of this study resulted in the award of a number of NASA contracts for the study of the application of supercritical wing technology and other advanced technologies to long range transport aircraft⁶.



Other Boeing Near-Sonic Studies

- Boeing Joint Advanced Research Project, JARP Transonic Cruise Study (1985)
 - Updated Assessments of ATTP Configurations Based on Technology Projections for 1995
 - Program Ended Because of Lack of Funding and shift of Focus to the HSCT program



Boeing under contract from NASA conducted a study to assess the benefits of advanced technology to future commercial transports that would be viable from technical, economic and producibility considerations. The study includes design Mach numbers of 0.90, 0.96 and 0.98. The configurations developed for the various design Mach numbers are shown above. The Mach 0.90 was considered to be the highest Mach number practical for a non-area ruled body. The other two Mach numbers represented the selected near-sonic design conditions.

The study results^{7,8} summarized above, show the effect of the increased design Mach number on the relative Direct Operating Cost, DOC. Increasing the design Mach number from 0.90 to 0.98 resulted in approximately and 8% increase in Relative DOC for the advanced technology level identified in the study. This corresponded to a 4% improvement relative to the baseline technology level. It was also shown that additional technology improvements would further reduce the relative DOC for Near Sonic Flight.

In 1985, the Boeing Company as part of it's Joint Advanced Research Project, JARP, initiated another study of the feasibility of a near-sonic transport configuration^{9,10}. The objective was to re-evaluate the technical and economic viability of aircraft designed to cruise at near-sonic speeds. The plan to achieve these objectives included:

- Review and evaluate results from previous transonic cruise studies
- · Define technology improvements relative to the previous studies and identify the impact
- Define and evaluate improved and alternate configurations based on technology projections for 1995
- · Prepare a proposal for subsequent follow on studies

The initial results indicated that the technology projections used in the prior studies conducted in the 1969 to 1972 time period were consistent with the latest projections for the 1995 time period.

Following a reduction in the overall JARP funds, the activity on this task was terminated.

Reference 10 contains a rather extensive list of the references for all of the prior transonic design studies and aerodynamic design activities and experimental investigations.



A series of studies were conducted by Boeing under contract to NASA to evaluate the potential gains that would be derived from the application of advanced technologies to transport aircraft designed to cruise at high transonic speeds and to explore the feasibility of overland boom free supersonic flight.

During the first phase, configurations with five different wing planform concepts were developed and evaluated. These included:

- Fixed wing sweep arrow wing
- Delta wing
- Variable sweep wing
- Single body variable sweep oblique wing
- Double body variable sweep Oblique wing

Subsequent studies focused on further development of the single body variable sweep wing configuration



In the previously described near sonic studies. The configurations were developed by extending subsonic design method to the near sonic speed region through the use of the near sonic equivalence rule.

The configurations in this study were developed utilizing supersonic design methodology extended down to low supersonic speeds, The fuselages of the configurations were all shaped using the transonic area rule to help minimize the cruise wave drag.

The results of the initial phase of the study are summarized in the figure above. The single body oblique wing concept had the lowest takeoff gross weight because of the low fuel requirements because of the high cruise lift/ drag ratios.

The delta wing configuration was nearly equivalent to the single body oblique wing configuration because of its lower structural weight which nearly compensated for its lower lift / drag ratio. The oblique wing configuration, however had the potential to achieve much lower community noise levels with little performance penalty.

Subsequent studies focused on further development of the single body variable sweep wing configuration.



The previous near sonic configuration studies were all developed by extending high aspect wing configurations to near sonic speeds using the near sonic equivalence rule. These all utilized high aspect wings with rather high thickness / chord ratios.

A "tops down" approach will be used to compare the aerodynamic efficiencies at subsonic speeds of subsonic designed configurations and with those of supersonic designed configurations.

The results suggest the possibility of a paradigm shift in the approach to design a near sonic configuration that would indeed resemble the Boeing published pictures of the sonic cruiser

Tops Down Aerodynamics

What It Can Do:

- Serve as a Valuable and Valid Comparison Tool
- Define the Maximum L/D Potential for any Configuration
- Define the Impact of Aerodynamic Performance Improvement Technologies
- Quickly Identify a "Realistic" Potential or Target L/D for Any Configuration
- Provide Ready Comparison of L/D Potential for Different Configurations
- Tracking Mechanism for Following L/D Design Development Improvements.
- Identify How Much More Potential Improvement there is for a specific Configuration Design*
- Identify General Potential Areas of Improvement for a Defined Configuration

What It Cannot Do:

- Predict What the Performance is for a Specific Configuration Design.
- Identify What Specific Changes are Necessary to Improve the Configuration
- Define Which is the Best Overall Airplane Design

If You Don't Know Where You're Going, How Will You Know When You Arrive?

"Tops Down" aerodynamic assessments can serve as a very valuable and valuable comparison tool. It can be used to define the maximum L/D potential that a configuration might conceivably achieve.

It is also a useful approach to illustrate the impact of aerodynamic performance improvement technologies on an configuration. Tops down assessments can be used to quickly identify a "realistic" potential or target L/D for an aircraft configuration and can be used to compare the L/D potential for different configurations.

A tops down assessment can also as a tracking mechanism for following the L/D improvements associated with configuration design developments and can identify how much potential there is for additional aerodynamic performance improvements.

Since the tops down asse4ssments are made with very limited information about a configuration, it can not provide details of the aerodynamic performance for a specific configuration. As such it cannot identify what specific changes are necessary to improve a configuration, nor can it define which configuration is the best overall aerodynamic design.



The drag elements for a subsonic aircraft cruising below the critical Mach number, can be grouped into two categories. These include those that are strongly dependent on lift, and those that are not.

The Non-lift dependent drag Items include:

- Friction drag
- Profile drag due to thickness.
- Compressibility drag
- Interference drag
- · Excrescence drag and miscellaneous drag

The lift-dependent drag items include

- Induced drag
- · Profile drag due to lift
- Compressibility drag due to lift
- Trim Drag

The induced drag varies inversely with the span of the wing. Consequently Subsonic configurations have typically utilized high aspect wings. As a result of the large spans, the wings were relatively thick and had rather high thickness / chord ratios.



In order to compare the relative design efficiencies of various configuration designs it is advantageous to define an upper bound for the lift/drag ratio for a configuration and then to compare the actual lift/drag ratio at the long range cruise Mach number with this upper bound. The long range cruise Mach number is typically defined as the Mach number where the drag rise is equal to 20 counts of drag (Δ CDrise = 0.0020). This process is commonly called a tops down analysis

For subsonic transport aircraft the lower bound drag components can be considered to include:

- Minimum CDo equal to fully turbulent flow flat plate skin friction drag.
- Minimum drag due to lift equal to the induced drag for planar wing configurations
 - with elliptic load distributions .

An adjusted wetted area is typically used to normalize the effects of Reynolds number variations on the viscous drag. The adjusted wetted area is equal to the actual wetted area times the ratio of computed average skin friction coefficient to a reference skin friction coefficient of 0.0021.

The "Tops Down" L/D max for subsonic transports is then equal to 19.34 times the wing span divided by the square root of the adjusted wetted area.

An "effective" span is used for aircraft having non-planar wing geometries such as tip fins. The "effective' span is the span of an equivalent planar wing that has the same induced drag as the non-planar wing.



The values of L/Dmax at the Mach number corresponding to (M L/D)max are shown for existing subsonic transport aircraft based upon flight test data.

The existing subsonic commercial transport aircraft achieve about 72% to 78% of the "upper limit" aerodynamic efficiency¹².

Comparisons of an aircraft design with the upper bound not only is a convenient way to assess the performance efficiency of a particular design, but it is also a useful way to identify potential performance gains due to design improvements.



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The subsonic aircraft configurations fail to achieve this Upper Bound Lift / Drag level because of a number of additional drag items as shown in the figure. The most significant of these additional drag items include:

- The relatively thick airfoils and wide fuselages result in a profile drag increase over the viscous friction drag by approximately 10% to 20 %.
- At the long range cruise Mach number, subsonic aircraft typically have 15 to 20 counts of drag rise (ΔCD = 0.0015 to 0.0020).
- The spanwise load distributions based on structural design trades, tend to depart from the ideal load distribution. The typical spanwise load distributions are more heavily loaded near the wing root. This together with an increase in profile drag due to lift typically increases the induced drag approximately 10% to 12% above the ideal level.

These three drag items account for a 15% to 18% reduction in L/D from the Upper Limit L/D levels.



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Commercial supersonic aircraft configurations tend to be long, thin and slender and cruise at relatively low lift coefficients. The subsonic viscous drag is essentially equal to flat plate skin friction drag.

The typical over land subsonic cruise Mach number for an HSCT of approximately 0.9. The is well below the drag rise. Mach number. Hence there is no drag rise at the cruise mach number.

The Concorde and the US SST subsonic Lift to drag ratios on the order of 64% and 82% of their respective L/D upper bound levels.

The Boeing recent HSCT configuration utilized an innovative wing leading design concept¹³ that resulted in a substantial increase in the subsonic cruise L/D level to about 92% of the upper bound level.

A near-sonic configuration derived from a HSCT concept utilizing a slightly higher aspect yet still relatively thin wing could conceivably result in a L/Dmax level equal to that of current subsonic aircraft but at near sonic speeds.

The configuration should still incorporate all the previously described favorable design features corresponding to application of the near sonic wing equivalent body design guidelines. This indeed appears to be the approach used in the Boeing sonic cruiser as indicated in the published pictures.



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The various published pictures of the Boeing Sonic Cruiser suggest that the configuration has indeed been derived using the previously defined paradigm shift in the near sonic configuration concepts.

It appears that the Sonic Cruiser has benefited from previous supersonic transport studies ideas, Boeing HSCT Studies technology and prior Boeing near-sonic configuration guidelines.

The applications of the desirable features of a near sonic configuration derived from the equivalent body concept are not clearly evident in these pictures but should most certainly be considered in an optimized mature design.



The development of a viable sonic cruiser is indeed an exciting possibility for commercial transport aircraft. Technical advancements in aerodynamics as well as developing appropriate design guidelines will be absolutely necessary to make this possibility a reality.

This chart summarizes aerodynamic research and development activities considered to be important in supporting the development of a viable near-sonic and for boom-free supersonic flight.

The author is not directly involved in the in the sonic cruise program. The information presented in this paper is consequently based on prior experience and not insight gained through this program.