Projecting and Tracking Advanced Technology Improvements in L/D

Configuration Aerodynamics Workshop

NASA Langley
February 26 - March 1, 1996

R.M. Kulfan
Boeing Commercial Airplane Group

TOPICS

- **Importance of Accurate and Consistent L/D Projections**
- **Subsonic Aircraft Tops Down L/D Analyses**
- **Supersonic Drag Components**
- **“Tops Down” versus “Bottoms Up”**
- **Fundamental Aerodynamic Concepts**
- **Define “Acceptable” Aerodynamic Design Space**
- **Apply the process to the TCA Configuration**

11/18/2002 12:59 Page 2

We will discuss the importance of being able to make consistent and accurate of projections of the expected aerodynamic performance improvements that might be achieved by aggressive technology development programs.
“Tops Down” aerodynamic projection charts are often used to compare the aerodynamic efficiency of subsonic transports. The drag of a subsonic configuration is not highly dependent on the detailed geometric shape or on the streamwise distribution of lift. Hence, the Lift / Drag ratio can be related to a single parameter on a universal chart.
At supersonic speeds the cruise drag is very dependent on the volume, volume distribution as well as both the spanwise and streamwise distribution of lift.
Components of drag for a supersonic configuration will be reviewed. It will be shown that a single simple correlation parameter is not sufficient for supersonic aircraft.
Fundamental aerodynamic concepts based on linear theory will be reviewed. These concepts are valid for HSCT type configurations and are used to develop a “tops Down” process for defining “acceptable” aerodynamic design space.
This process will be applied to the TCA configuration to develop projections of the cruise L/D performance level.

Importance of Accurate and Consistent L/D Projections

- **Determine the Viability of an HSCT**
- **Define Meaningful Technology Development Goals**
- **Measure Technology Development Progress**
- **Proper focus of HSR Research Funds and Activities**
- **Support Correct Configuration Design Decisions**

Current HSCT configuration studies are focused on determining the technical, economic and environmental viability of an High Speed Civil Transport. These studies must by necessity include projections of anticipated technical improvements for all of the key disciplines (e.g. aerodynamic performance, structural materials and weights, propulsion system weights and performance, etc.).

The projections represent current assessments of what is expected to be achievable with aggressive technology development programs.

The emerging developments in aerodynamic non-linear design and analysis methods offer the potential of significant improvements in aerodynamic cruise lift/drag ratio. These improvements will have a major effect on the viability of an HSCT.

It is essential to identify realistic achievable goals and to be able to measure the progress to achieve these goals for cruise Lift/Drag ratio.. This is necessary to insure a properly focused

Technology Projection Approaches

1. Bottoms up “Guesstimates”

- Based on “experience” and / or previous successes
- Assume “We can do as good or better”
- Very dependent on initial baseline performance
- Requires similar geometry for direct application
- Not systematically adjustable for geometric differences
- No consistent process
- Can not be used to determine efficiency of initial design
- Lacks “Fundamental” basis
- Projection is an estimated increment to new baseline

11/18/2002 12:59

Page 4

There are two different approaches for making projections of potential improvements in aerodynamic performance lift to drag ratio, L/D.
The first approach is a “Bottoms Up Guesstimate” method. Based on previous experiences or successes we *assume* that we can do even better.
These estimates are very dependent on near similarity between the pervious baseline configuration and the new configuration geometry for which projections are being made.
 The projections are not absolute but related to the performance level of the new design or to some assumed achievable level.
This approach lacks a fundamental basis and is highly dependent on the prophetical wisdom of the individual. This certainly does not lead to consistent meaningful projections.

Technology Projection Approaches

2. Tops Down Estimates

- **Based on aerodynamic “fundamentals”**
- **Independent of initial or current aerodynamic performance**
- **Can apply process to any configuration**
- **Process is rigorous and consistent**
- **Useful for determining efficiency of initial design**
- **Projection is a calculated “achievable upper bound”**

The second approach is a “tops down Approach” based on fundamental aerodynamic principles.
The projections do not depend on the current aerodynamic performance of specific configurations are being made. They do, however, depend on the basic geometric features of the configuration.
This is the approach that will be presented in this presentation. The process is both rigorous and consistent. The projection is a calculated “achievable” upper bound.
This approach will first be illustrated for subsonic transport aircraft and then will be extended to supersonic configurations.

SUBSONIC DRAG

- **Not a Strong Function of Shape or Volume Distribution**
- **Depends on Spanwise Distribution of Lift**
- **Generally Thick Airfoils and High Cruise CL**
- **Cruise at Mach for M(L/D) max**

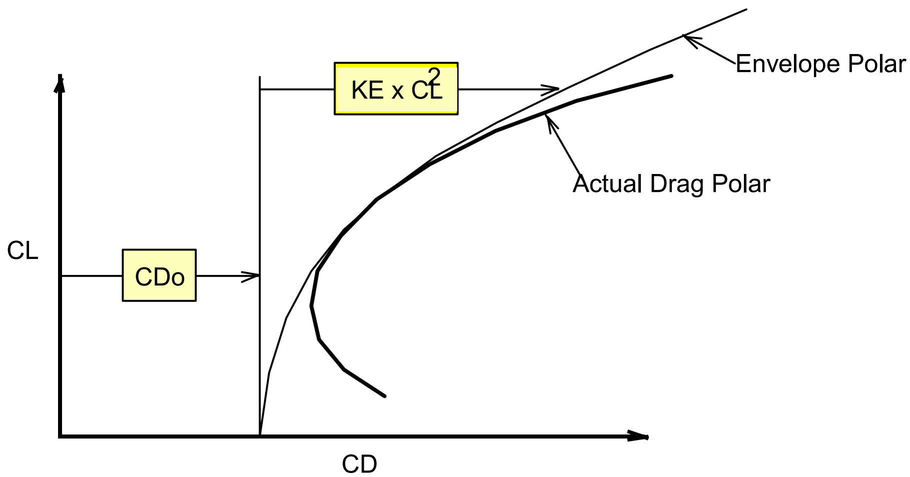
11/18/2002 12:59 Page 6

The drag of subsonic transport configurations is not highly dependent on airfoil shapes or volume distributions at conditions below drag rise as along as the flow remains attached.
The lift dependent drag depends on the spanwise distribution of lift and not on the chordwise lift distribution.
The zero lift drag is primarily friction and profile drag and is very dependent on the overall wetted area of the configuration
The aerodynamic efficiency for subsonic transports is usually specified at the Mach number for long range cruise. This Mach number is very dependent of the fundamental airfoils shapes of the wing.
Subsonic configurations tend to relatively thick wing sections and cruise at relatively high lift coefficients.

Subsonic Drag Polar Approximation

$$CD = CDo + KE \times CL^2$$

$$L/D\ max = \frac{0.5}{\sqrt{KE \times CDo}}$$



sub-plr2.flo 2/21/96

11/18/2002 12:59

Page 7

We can approximate the subsonic drag polar by a simple parabolic equation.

$$CD = CDo + KE \times CL^2$$

CDo is called the zero lift drag.

KE is the drag due to lift factor.

Using this simple expression for drag, the L/Dmax value is dependent on both KE and CDo by a very simple expression.

$$L/Dmax = \frac{0.5}{\sqrt{KE * CDo}}$$

Subsonic Drag Polar Approximation

$$L/D\ max = \frac{0.5}{\sqrt{KE \times CDo}}$$

$$CD = CDo + KE \times CL^2$$

Lift Dependent Drag

* Induced Drag
* Profile Drag Due to Lift
* Drag Rise Due to Lift
* Trim Drag
* ??

Non Lift Dependent Drag

* Friction Drag
* Profile Drag Due to Volume
* Drag Rise
* Interference Drag
* Excrescence Drag
* Miscellaneous Drag

The Non-lift dependent drag consists of:
- 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

Equations for Minimum Induced Drag

$$\frac{Di}{q} = \frac{1}{2*\pi} \int_{-\frac{b}{2}}^{\frac{b}{2}} \int_{-\frac{b}{2}}^{\frac{b}{2}} \frac{d\Gamma}{dy} \frac{d\Gamma}{d\eta} \ln|y - \eta| dy\, d\eta$$

Lift: $L = \rho V\infty \Gamma$

Minimum Drag: Elliptic Load Distribution

$$\Gamma opt(\eta) = \Gamma o \sqrt{1 - \eta^2}$$

11/18/2002 12:59

Page 9

The equation for induced drag at subsonic speeds is shown in the familiar integral form in terms of the wing circulation, Γ.

It will be shown there is a great similarity between this equation and the wave drag equations at supersonic speeds.

The wing lift distribution is elliptic for minimum induced drag.

Tops Down L/D Analysis

Lower Bound Drag:

- Fully Turbulent Flow Friction Drag
- Elliptic Load Induced Drag

$$CD = CFave \frac{Awet}{Sref} + \frac{CL^2}{\pi AR}$$

$$Awet\ adj = Awet \frac{CFave}{0.0021}$$

$$L/D\ max\ pot. = 19.34 \frac{b}{\sqrt{Awet\ adj}}$$

11/18/2002 12:59

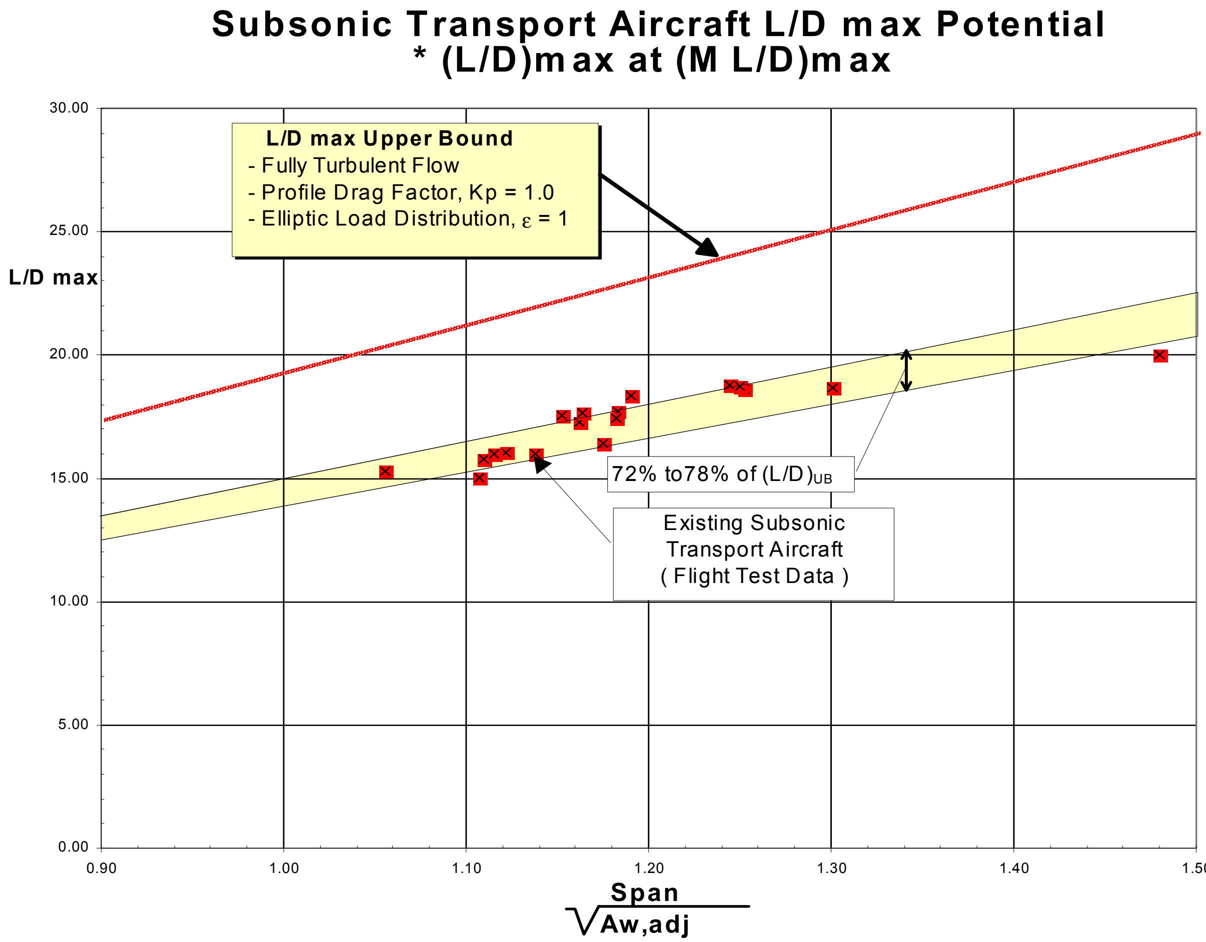
Page 10

For subsonic transport aircraft the lower bound drag components are usually 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 used to normalize out the effects of Reynolds number.
The adjusted wetted area is equal to the actual wetted area times the ratio of computed average skin friction coefficient to an average skin friction coefficient of 0.0021.

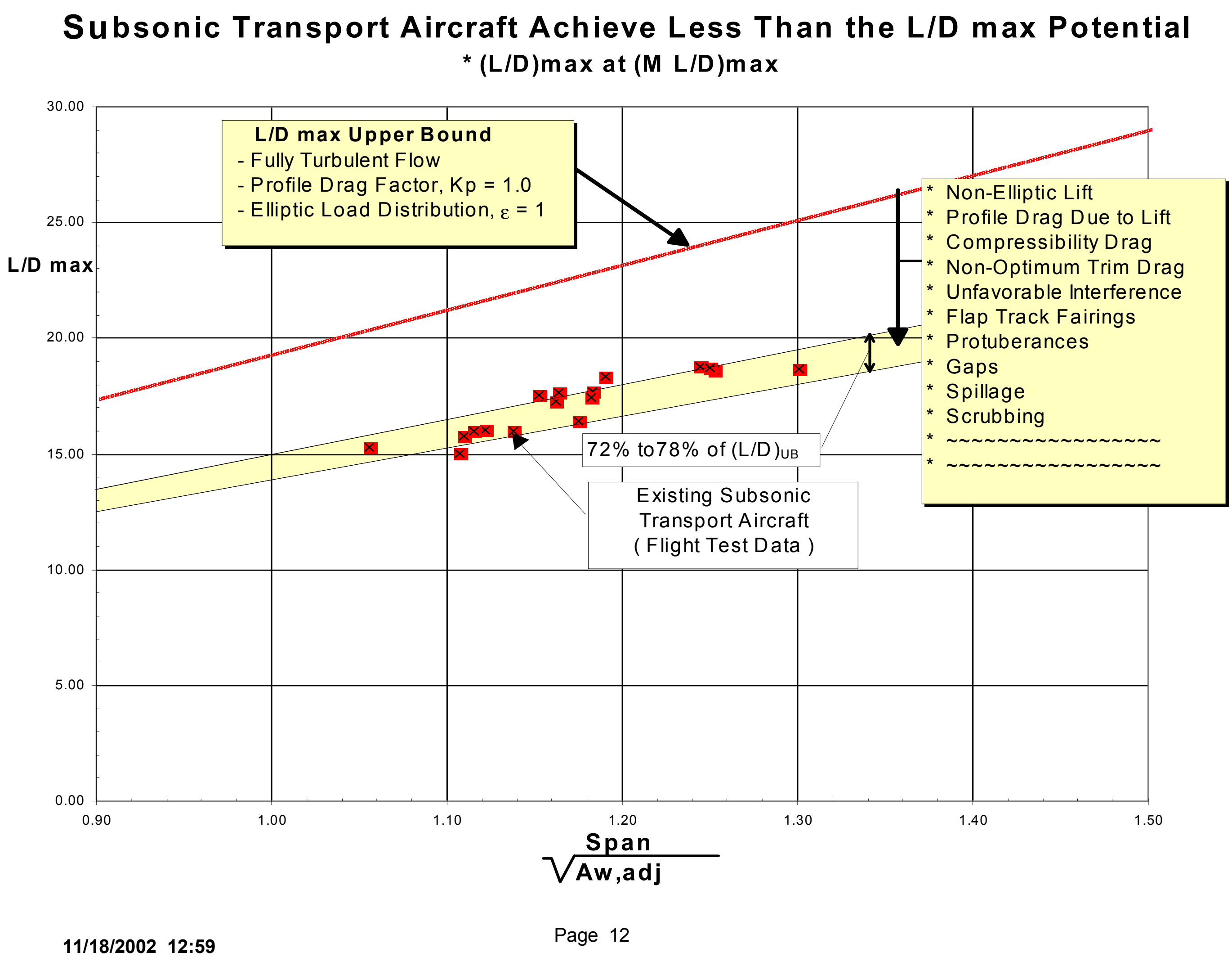
The "Tops Down" L/D max for subsonic transports is then equal to 19.34 times the wing span divided by the square root of the adjusted wetted area.
An "effective" span is used for aircraft having non-planar wing geometries such as tip fins. The "effective' span is the span of an equivalent planar wing that has the same induced drag as the non-planar wing.



11/18/2002 12:59

Page 11

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 aircraft achieve about 72% to 78% of the “achievable upper limit”



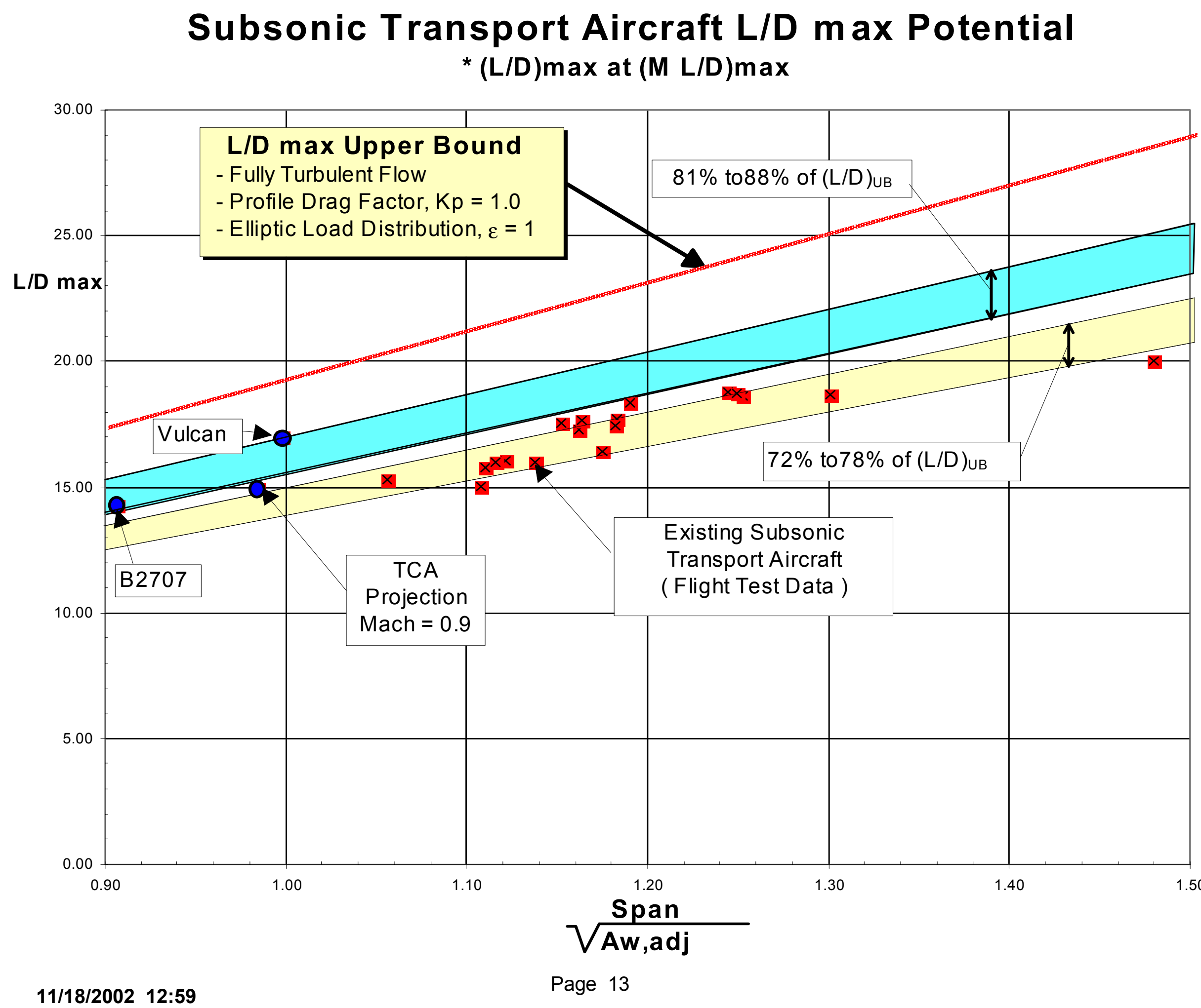
The Subsonic Configurations fail to achieve this Upper Bound Lift / Drag level because of a number of additional drag items as shown in the figure. The most Significant of these additional drag items include:

- The relatively thick airfoils and wide fuselages result in a profile drag increase over the viscous friction drag by approximately 20% to 25 %.

- 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 20% to 25% reduction in L/D from the Upper Limit L/D levels.



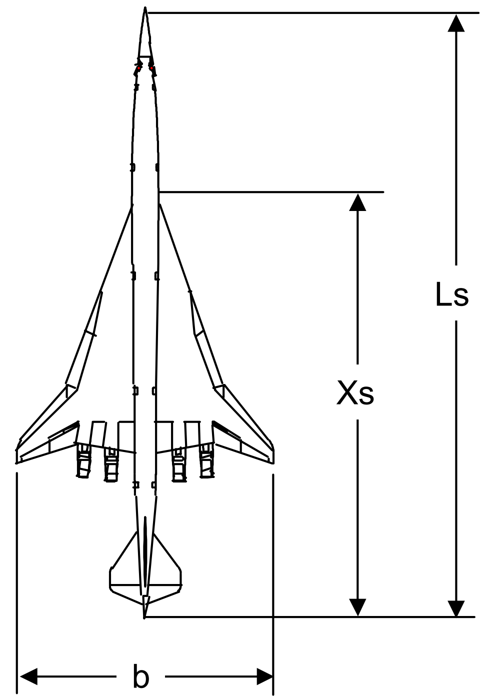
Supersonic type configurations tend to be long, thin and slender and cruise at relatively low lift coefficients. The subsonic viscous drag is essentially equal to flat plate skin friction drag.

The typical over land subsonic cruise Mach number for an HSCT of approximately 0.9. The is well below the drag rise. Mach number.

Consequently, it is expected that an HSCT cruising with optimized flap settings should achieve well in excess of 80% of the corresponding upper limit for L/Dmax at subsonic cruise conditions.

SUPER SONIC DRAG COMPONENTS

- $CD = CD_F + CD_{WV} + CD_{WL} + CDI + CD_{MISC}$
- CDF ≈ WETTED AREA ==> FRICTION DRAG
- CDWV ≈ Vol^2/Ls^4 ==> VOLUME WAVE DRAG
- CDWL ≈ $(LIFT/Xs)^2$ ==> LIFT WAVE DRAG
- CDI ≈ $(LIFT/b)^2$ ==> INDUCED DRAG



11/18/2002 12:59

Page 14

The drag components of an HSCT flying at supersonic speeds consists primarily of friction drag, wave drag due to volume, wave drag due to lift, induced drag and other miscellaneous drag items.
The friction drag is typically equal to flat plate skin friction drag on all of the component surfaces. The friction drag, therefore, depends primarily on the wetted area.
The volume wave drag primarily varies with the volume squared divided by the configuration length raised to the fourth power.
The induced drag varies with the ratio of lift over wing span squared.
The wave drag due to lift varies with lift over the streamwise length of the lifting surface squared. The wave drag due to lift vanishes as the supersonic Mach number approaches one.
It is evident that for low drag, supersonic configurations tend to be long, thin and slender.
The drag at supersonic speeds is very dependent on the shape of the configuration, and on the relative size and locations of the configuration components. It is therefore, not possible to define the maximum L/D max potential for an HSCT configuration in terms of a single universal parameter as is the case for subsonic transport configurations.

Supersonic Drag Polar Approximation

$$L/D\ max = \sqrt{\frac{0.5}{KE \times CDo}}$$

$$CD = \underbrace{CDo}_{\text{Non Lift Dependent Drag}} + \underbrace{KE \times CL^2}_{\text{Lift Dependent Drag}}$$

Lift Dependent Drag

* Induced Drag
* Wave Drag Due to Lift
* Lift Interference
* Trim Drag

Non Lift Dependent Drag

* Friction Drag
* Wave Drag Due to Volume
* Interference Drag
* Excrescence Drag
* Miscellaneous Drag

11/18/2002 12:59

Page 15

Similar to the subsonic condition, we can represent the supersonic drag polar as a two term parabolic equation consisting of the non-lift dependent drag, CDo, plus the lift dependent drag KE x CL^2.

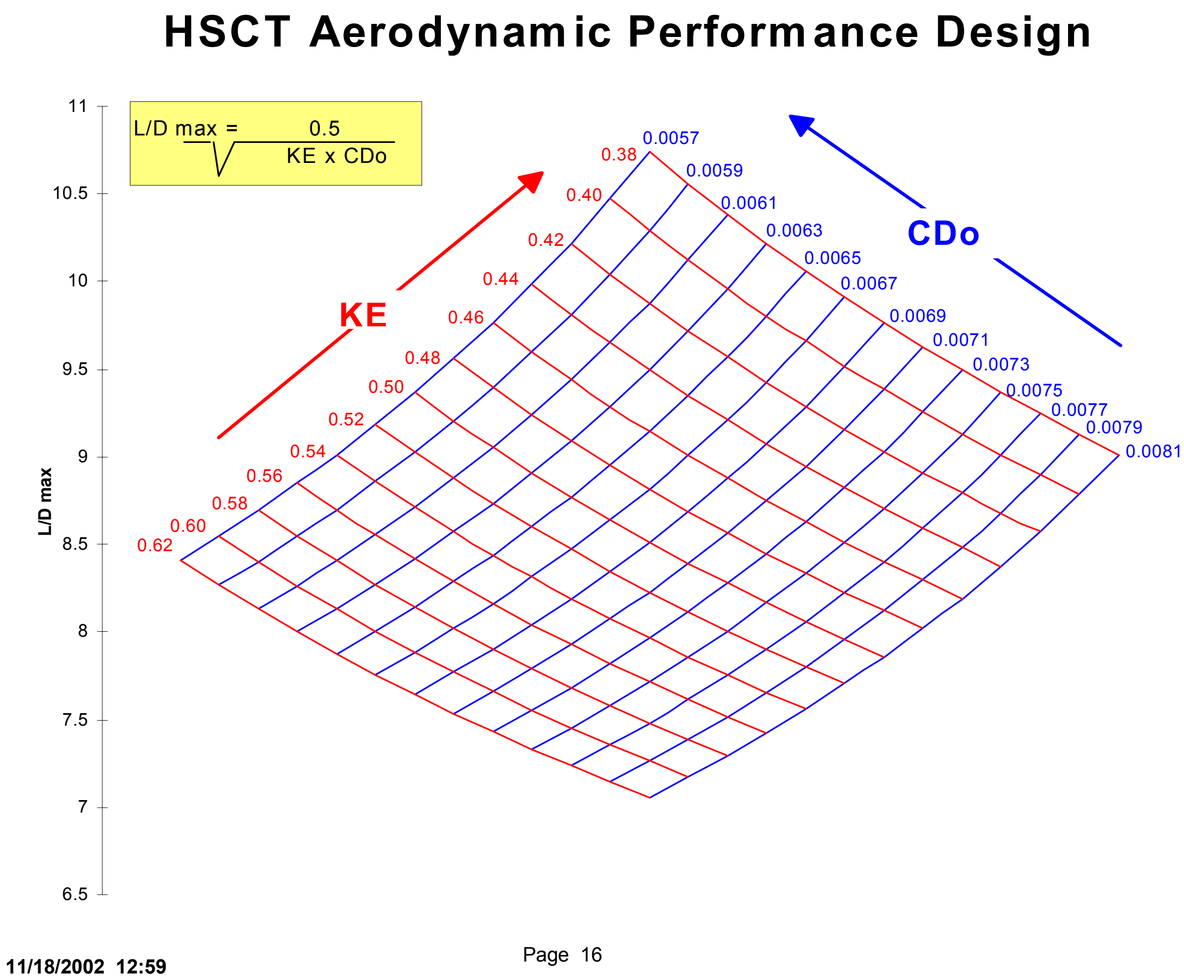
The non-lift dependent drag includes:

- Friction drag
- Wave drag due to volume
- Volume interference drag
- Excrescence and other miscellaneous drag items.

The lift dependent drag consists of :

- Induced drag
- Wave drag due to lift
- Lift interference effects
- Trim drag.

Based on the parabolic drag polar representation, it can be shown that L/Dmax varies inversely with the square root of the product of CDo and the drag due to lift factor KE.



A way to view the dependency of L/Dmax on CDo and KE is in the form of a carpet plot. This is the form that we will use to develop the region for acceptable designs of a specific configuration. This is a two dimensional representation of the design space for supersonic configurations.

In the discussions that follow, it is assumed that the gross overall features of any configuration remain fixed. These include such things as wing area and location on the body, nacelle overall size and locations, planform shape and critical design constraints.

What we wish to determine is the region of acceptable designs that could be developed by different methods and techniques. We will then determine what is considered to be the overall upper limit of achievable L/Dmax for that specific configuration.

To do this we will identify values of CDo and KE that are considered too high for an acceptable design. We will then use fundamental aerodynamic concepts to determine lower bounds of achievable CDo and KE.

CDo "TOO HIGH" LIMIT FOR ACCEPTABLE DESIGN

$$CDo < CD_F + \Sigma\, CD_{W\ ISOL} + CD_{MISC} + CD_{EXCRES}$$

- **FULLY TURBULENT FRICTION DRAG**
- **SUM OF COMPONENT ISOLATED WAVE DRAG [NO FAVORABLE INTERFERENCE]**
- **CURRENT TECHNOLOGY EXCRESCENCE AND MISCELLANEOUS DRAG**
- ***** DRAG CAN BE WORSE THAN THIS *****

11/18/2002 12:59 Page 17

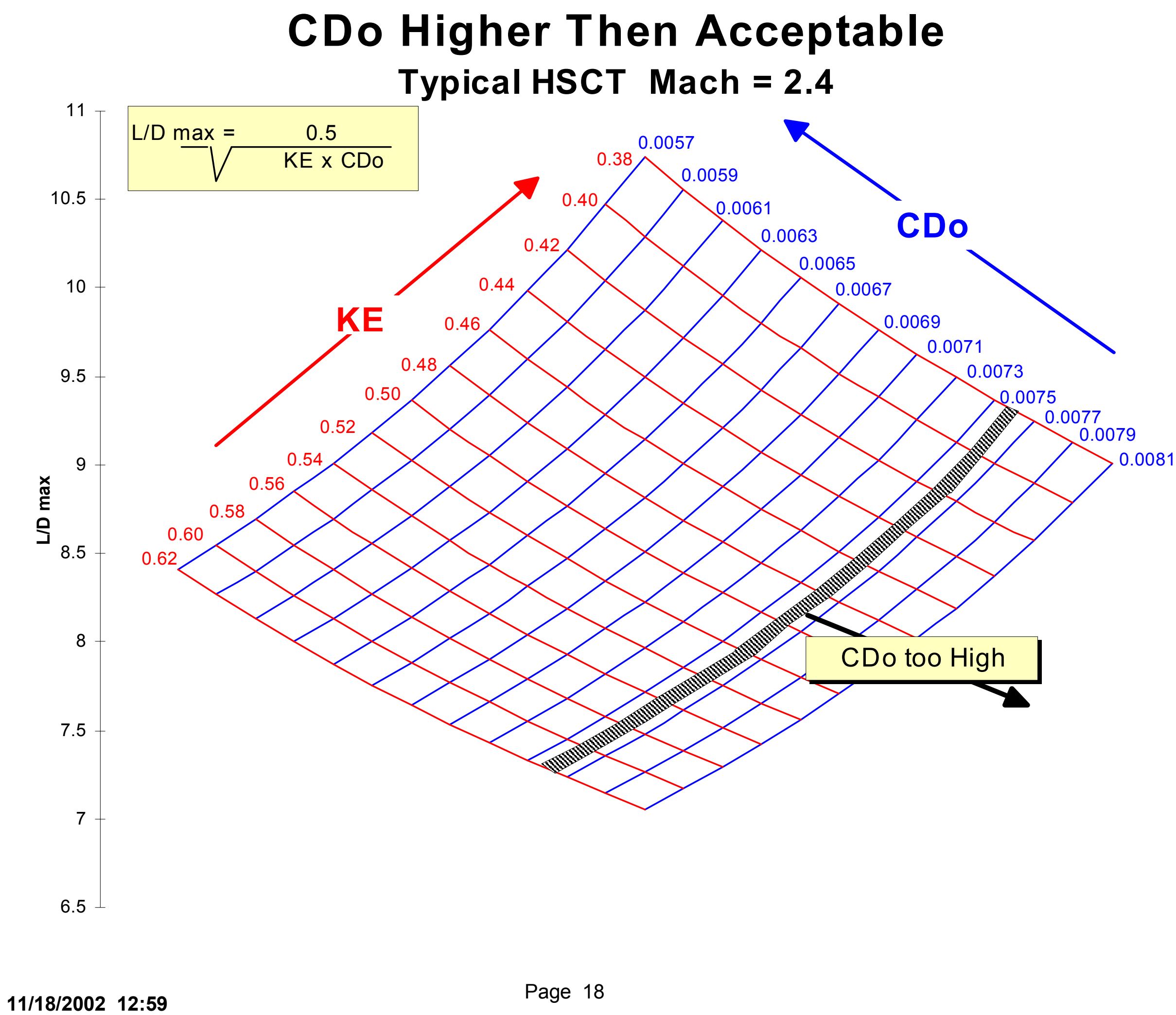
CDo is considered "too high" if the non-lift-dependent drag exceeds the sum of:

- CDF = Fully turbulent flow flat plate skin friction drag.
- CDW = The sum of the isolated wave drag of each of the configuration components. This corresponds to a design with no net favorable aerodynamic interference.
- CDmisc = Current technology miscellaneous drag including excrescence drag.

The most common causes of CDo being too high are:

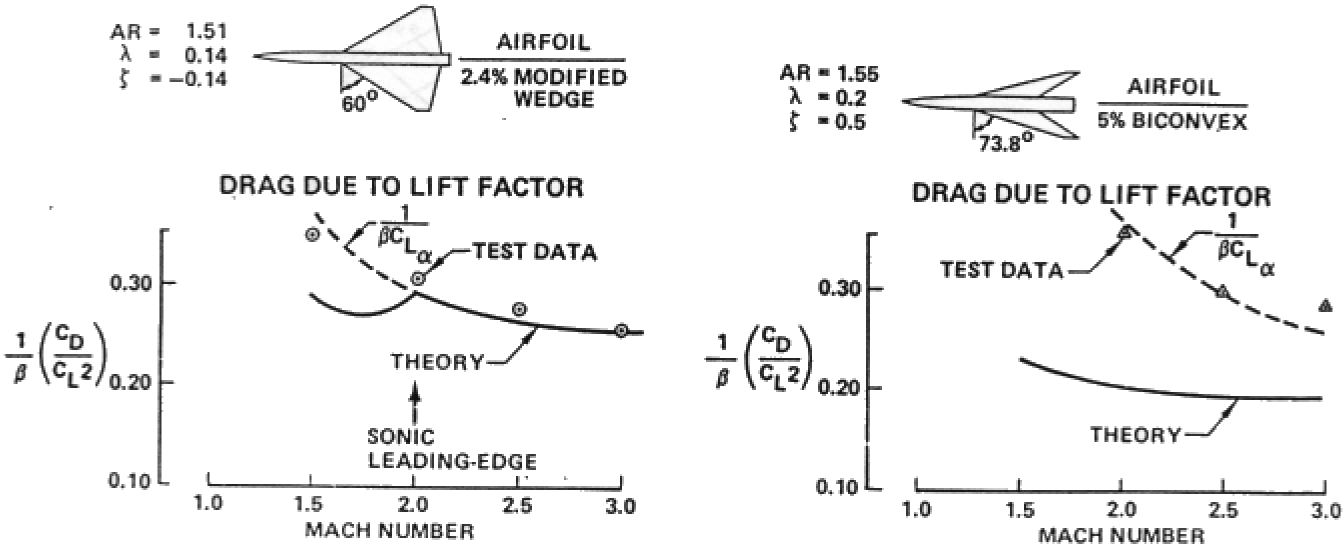
- Unfavorable wing / body interference drag for a non-area-ruled body.
- Nacelles designed and / or located to produce volume wave drag interference.
- Large out of contour bumps such as landing gear fairings
- Separated flow over the wing upper surface or in the vicinity of the nacelle / diverter intersection with the wing.

The zero lift drag can be worse then this acceptable upper limit for CDo.



By calculating the friction drag, the wave drag of the isolated components and the miscellaneous drag items, we can then locate on this chart a boundary beyond which CDo is considered to be “too high” for an acceptable design.

DRAG DUE TO LIFT OF FLAT SUPERSONIC WINGS



This figure compares the experimental drag due to for some flat supersonic configurations with the predicted theoretical drag due to lift. The test data matches the theoretical drag due to lift with zero suction (s=0). The drag due to lift for zero suction as shown in this figure is equal to one over the lift curve slope.

The "KE too high" limit corresponds to the drag due lift that could actually be achieved by a thin flat symmetric wing design.

KE " TOO HIGH" LIMIT FOR ACCEPTABLE DESIGN

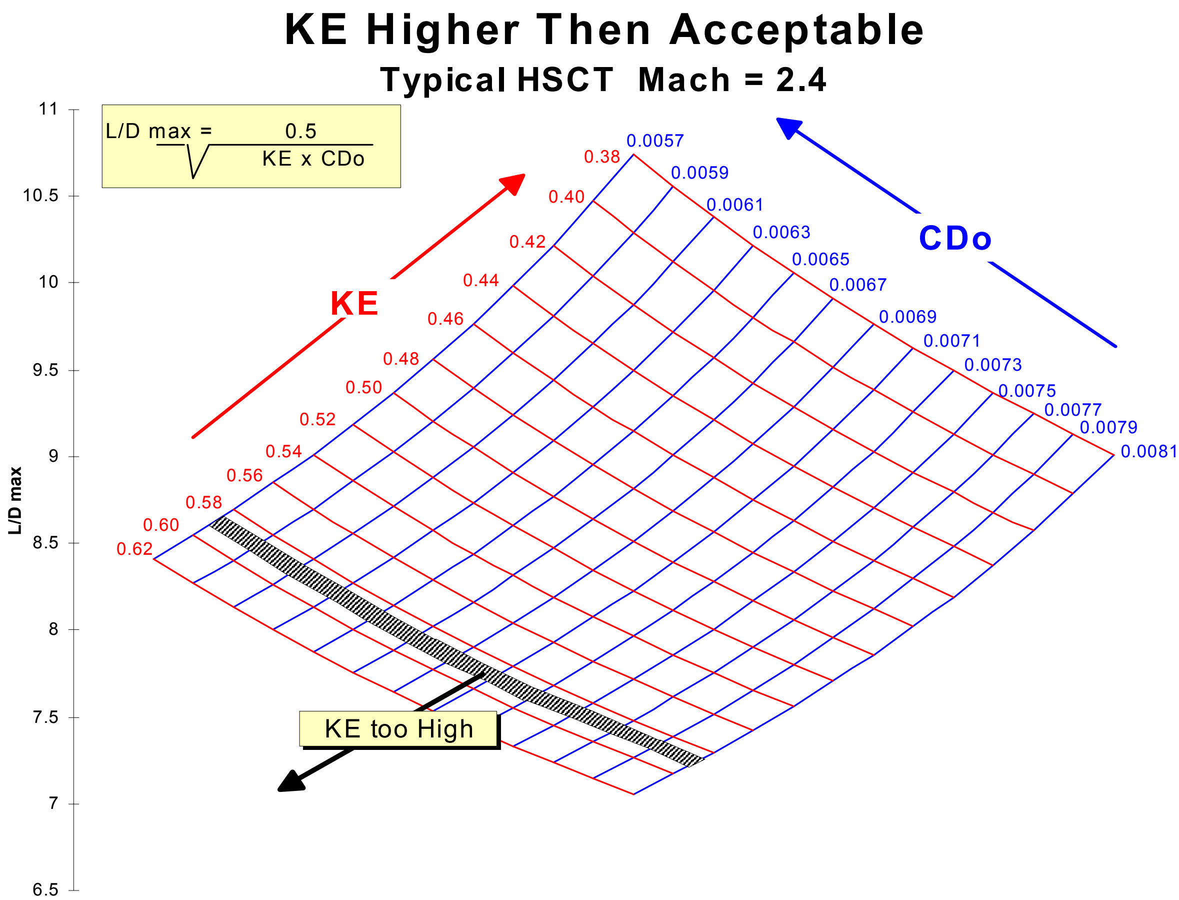
$KE < KE_{S=0}$

- **EQUIVALENT TO DRAG OF FLAT WING CONFIGURATION**
- **NO TRIM DRAG**
- **NO LIFT INTERFERENCE DRAG**
- ***** DRAG CAN ACTUALLY BE HIGHER *****

11/18/2002 12:59 Page 20

As an upper limit for KE we assume that the drag due to lift should be no worse the drag of a flat symmetric wing design with no leading edge suction.
We also assume no favorable interference lift or trim drag.

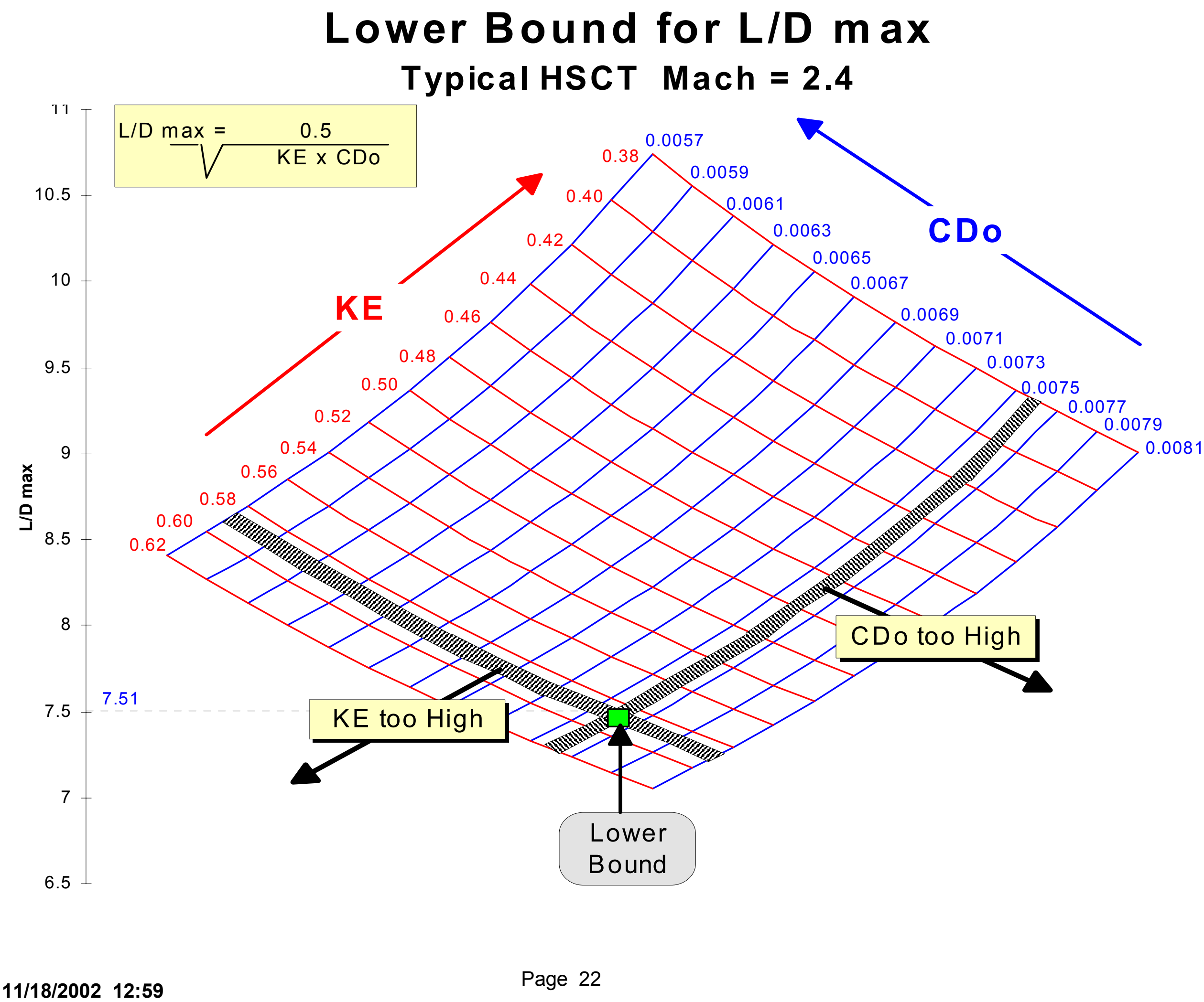
Again the drag for a very poor design can exceed this limit



11/18/2002 12:59

Page 21

This shows the “KE too high” boundary corresponding to the inverse of the lift curve slope.



The intersection of the “CDo too high” boundary and the “KE too high” boundary determines the lower bound for L/D max. This lower bound for L/Dmax essentially corresponds to the Concorde aerodynamic efficiency level.

LINEAR THEORY IS:

- **LONG ON IDEAS**
- **SHORT ON ARITHMETIC**

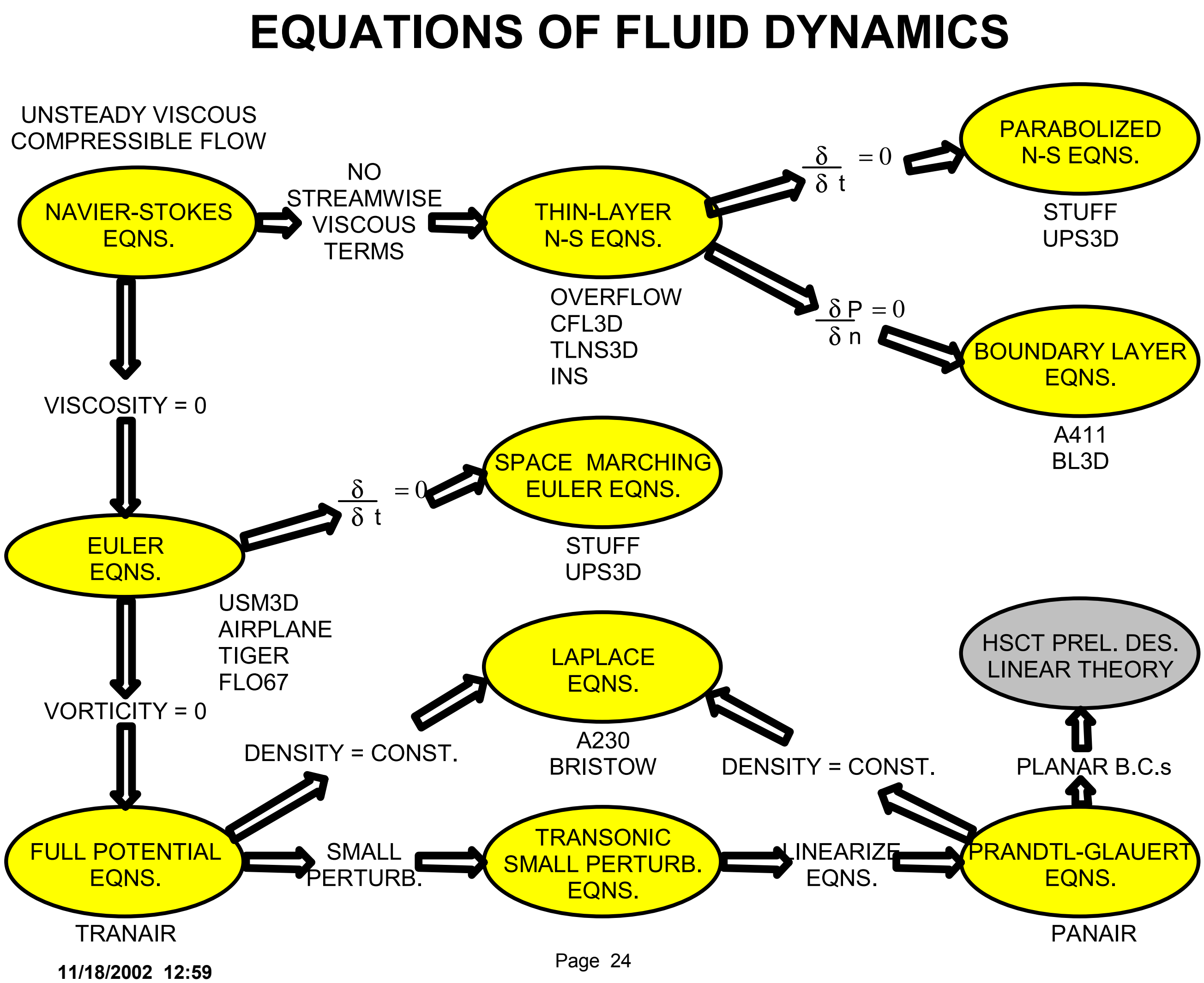
CFD IS:

- **SHORT ON IDEAS**
- **LONG ON ARITHMETIC**

R. T. Jones

In order to determine meaning lower limits for both CDo and KE we will use fundamental aerodynamic concepts based on linear theory.

R. T. Jones has said: "Linear theory is long on ideas but short on arithmetic. CFD is short on ideas bur long on arithmetic."
Linear theory formulations utilize elegant mathematical solutions of simplified flow equations. These solutions often provide insight into the nature of the flow fundamentals.
CFD utilizes powerful numerical solutions of complicated nonlinear flow equations. The solutions can provide details of the flow features for the analysis conditions.



This shows the hierarchy of fluid dynamic equations starting from the unsteady viscous compressible flow Navier-Stokes equations.

The key assumptions in reducing the complexity of the equations to move to the next lower level are also shown.

Some of the various CFD codes in use by NASA and industry are shown next to basic set of equations that are solved by the codes. The Navier-Stokes flow solvers also can be used to solve the Euler equations.

The The HSCT preliminary design linear theory methods reside at the bottom of the hierarchy.

Equations of Fluid Dynamics (continued)

There are a number of simplifying assumptions that are inherent even in the Navier-Stokes equations. The Navier-Stokes equations assume that the fluid medium is a single component perfect gas that can be treated as a continuum in which stress is proportional to strain, and pressure is proportional to density times temperature. Relative to the Navier-Stokes equations, the HSCT linear theory equations assume:

1. inviscid flow: The viscous effects are included in the skin friction drag. This requires care in applying the linear theory to avoid conditions leading to separated flow.

2. Irrotational Flow: The irrotational flow assumption greatly simplifies the numerics of a flow field solution since a single scalar equation is solved in terms of a velocity potential. The vector flow field can be obtained from the velocity potential scalar function. This limits the flow to moderate strength shocks, and non-rotational flow. However, these favorable flow conditions correspond to those on a low drag HSCT Configuration.

3. Small Perturbations. The Assumptions of small perturbations allows the velocity potential equations to be linearized. The solution process to becomes much easier. In addition, linearization allows the powerful concept of superposition to be used. This allows separation of the volume and the lifting effects and provides fundamental understanding of the flow phenomena. Again, the assumption of small perturbations is quite valid for HSCT low drag configurations which tend to be thin and slender, and operate at low lift coefficients.

4. Planar boundary conditions. The assumption of planar boundary conditions further simplifies the solution process. The sources / sinks and lifting elements that represent the geometry must lie on the axes of the fuselage, or the nacelles; and in the plane of the wing. These planar boundary conditions restrict the geometry to circular body and nacelle cross sections, and mid wing / body configurations. It is therefore easy to misapply the theory. In addition design details such as wing / body intersections or nacelle diverter geometry can not be analyzed directly.

Elegant numerical and analytic solutions are possible. These solutions can provide insight and a fundamental understanding of key design variables, design sensitivities and potential performance levels.

The major difficulty with the planar boundary conditions is that numerical singularities can occur in the solution processes. The numerical analyses methods must properly treat these localized numerical singularities.

Currently, the linear theory methods most commonly used for designing optimum wing camber and twist, result in singularities in the camber / twist definitions. The smoothing process significantly reduces the potential benefits of camber optimization. It is felt that non-linear optimization will be able to achieve drag benefits identified by the linear theory predictions but are unachievable by the linear designs.

TWO FUNDAMENTAL QUESTIONS

1. WHY CAN LINEAR THEORY DEFINE LIMITS ?

2. WHY CAN'T LINEAR DESIGNS ACHIEVE LIMITS?

11/18/2002 12:59 Page 26

We must answer two fundamental questions:

- Why can linear theory be used to define lower bounds for both CDO and KE that could be obtained using advanced non-linear CFD methods?

- Why can linear theory designs not achieve these lower limits ?

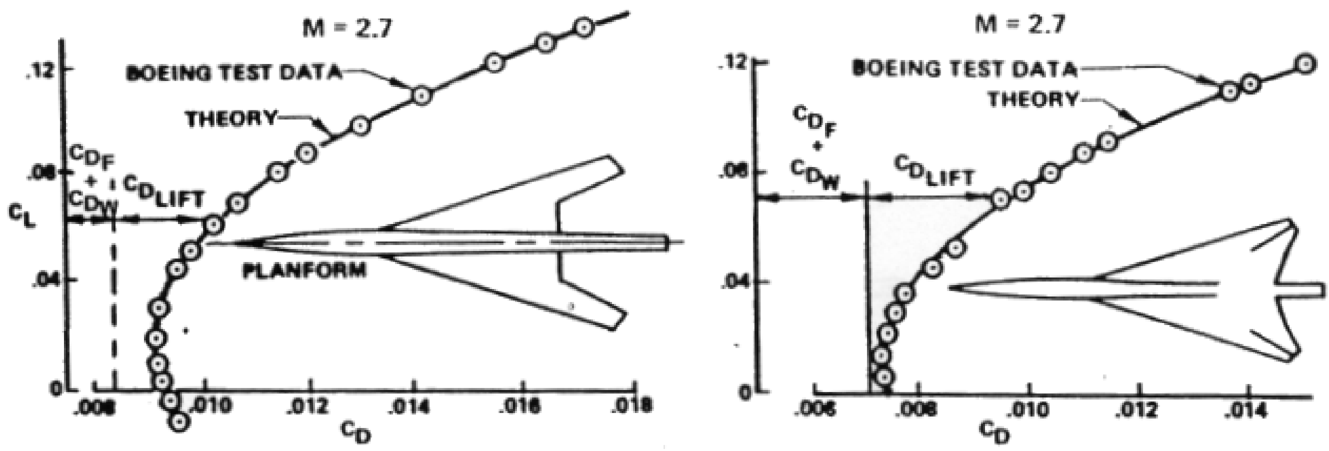
Linear Theory Analyses vs CFD Analyses

- **Underestimates Compression Pressures**
- **Overestimates Expansion Pressures**
- **Disturbances Carried Along Free Stream Characteristics**
- **Easy to incorrectly Apply Linear Theory**
- **Does not capture interference between lift and volume**
- **USES LINEAR PLANER BOUNDARY CONDITIONS**
- *************   ARE THESE SIGNIFICANT ?? ******************
- **FACT: Drag Predictions do Match Test Data for Good Designs**

11/18/2002 12:59   Page 27

Let us examine what the differences are in the results of linear theory analysis tools and results of corresponding non-linear CFD analysis. Not all linear theory methods are the same. The specific linear theory method used in the Boeing HSCT Preliminary design studies is the "Middleton / Carlson" program developed under a NASA contract in the mid 1907's time period. This methodology is a linear theory with planar boundary conditions. Consequently it is easy to incorrectly apply the theory by application to configurations for which planar boundary conditions are not adequate.
Linear theory under estimates compression pressures and over estimates expansion pressures. In addition, the linear theory disturbances are propagated along free stream Mach lines and therefore can not adequately predict shock formations. Linear theory does not predict interferences between lift and volume.
These are not significant effects for long slender , thin configurations at low lift coefficients. These are the conditions for low drag supersonic configurations.
The major restriction is in the use of planar boundary conditions. It is very easy to misuse the theory and produce significant errors. Properly used linear theory can predict the drag characteristics of well behaved configurations very accurately.
The following few charts show typical linear theory vs test data comparisons that have been made for a variety of supersonic configurations at or near the design conditions.

Early US SST Test vs Theory Comparisons



11/18/2002 12:59

Page 28

These wing / body configuration were two of the early Boeing SST baseline concepts. The configuration on the left is the Boeing variable sweep concept that was selected as the winner of the SST competition.

The configuration on the right is the Boeing variable sweep integrated wing / empennage concept. This was the last variable sweep design before the B2707-300 delta wing concept was developed as the final US SST design.

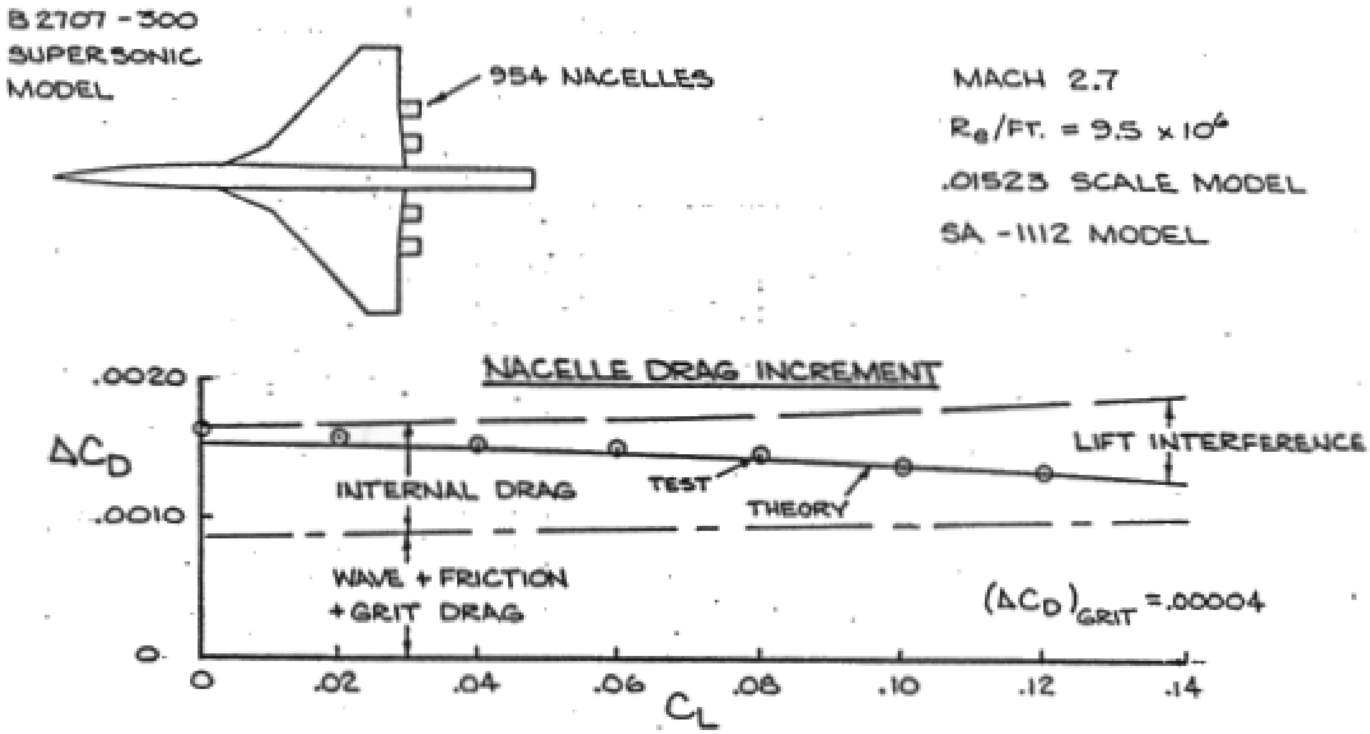
The theoretical drag predictions included friction drag, wave drag due to volume, and drag due to lift.

The skin friction drag was calculated as flat plate skin friction drag. The volume wave drag was calculated by the “Harris” far-field wave drag program. The drag due to lift was calculated by the Middleton /Carlson near field pressure integration method. These calculations were made in the early 1960’s.

These are the same methods used today for the Boeing preliminary design studies.

The theoretical predictions agree very well with the test data.

Boeing B2707-300 US SST Configuration



11/18/2002 12:59

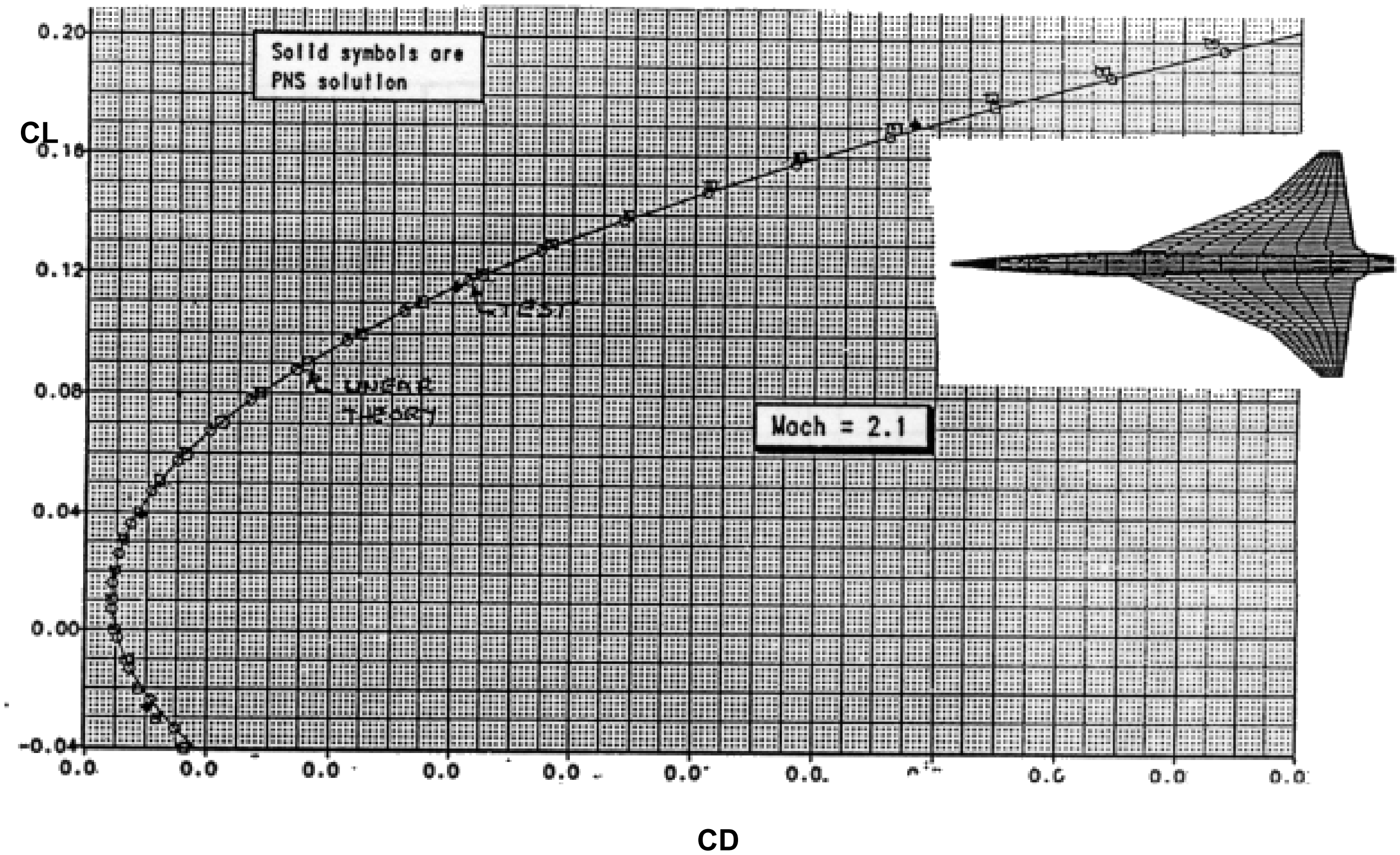
Page 29

This compares measured and calculated nacelle installed drag increments for the US SST configuration.
Near field wave drag methods and drag due to lift methods were used together with flat plate skin friction calculations for the theoretical nacelle drag increments.
Linear theory was able to predict the significant amount of favorable interference drag that was actually achieved.

These calculations were made in the early time 1970 time period.

Boeing 1807 WT Configuration

Mach = 2.1



BSWT 623 data
11/18/2002 12:59

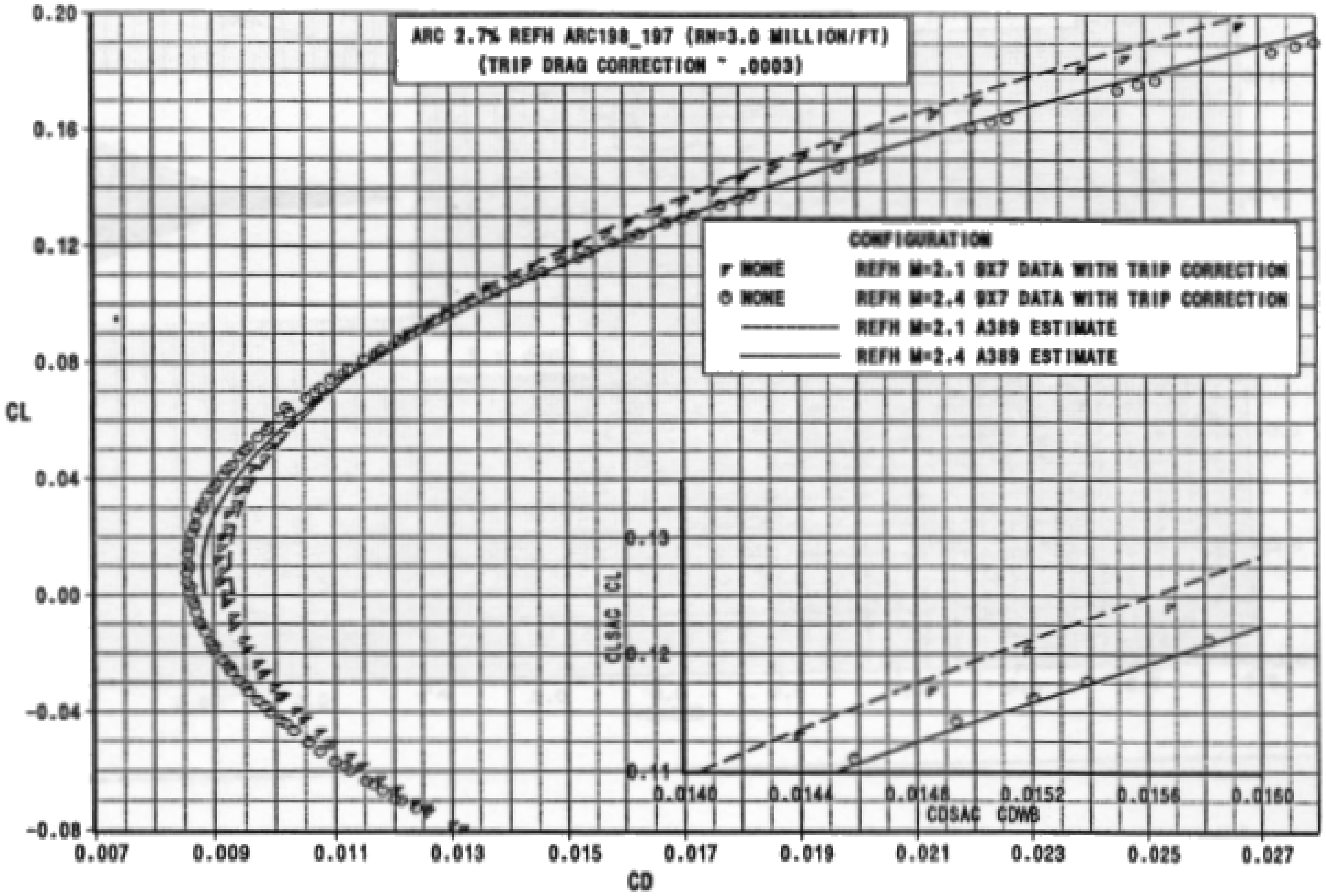
Page 30

This was the first HSCT wind tunnel configuration that incorporated the unique Boeing developed blunt leading edge radius design. The design was developed by an iteration procedure of linear theory design and nonlinear theory analysis.

Linear theory drag predictions are compared with the test data at the original design Mach number of 2.1.

The test data are the circles with the curve drawn through them. The linear theory predictions are indicated by the squares. The predictions are in excellent agreement with the test data. This configuration was the predecessor to the Boeing developed Ref H geometry.

Comparison of Ref H Test Data With Predictions

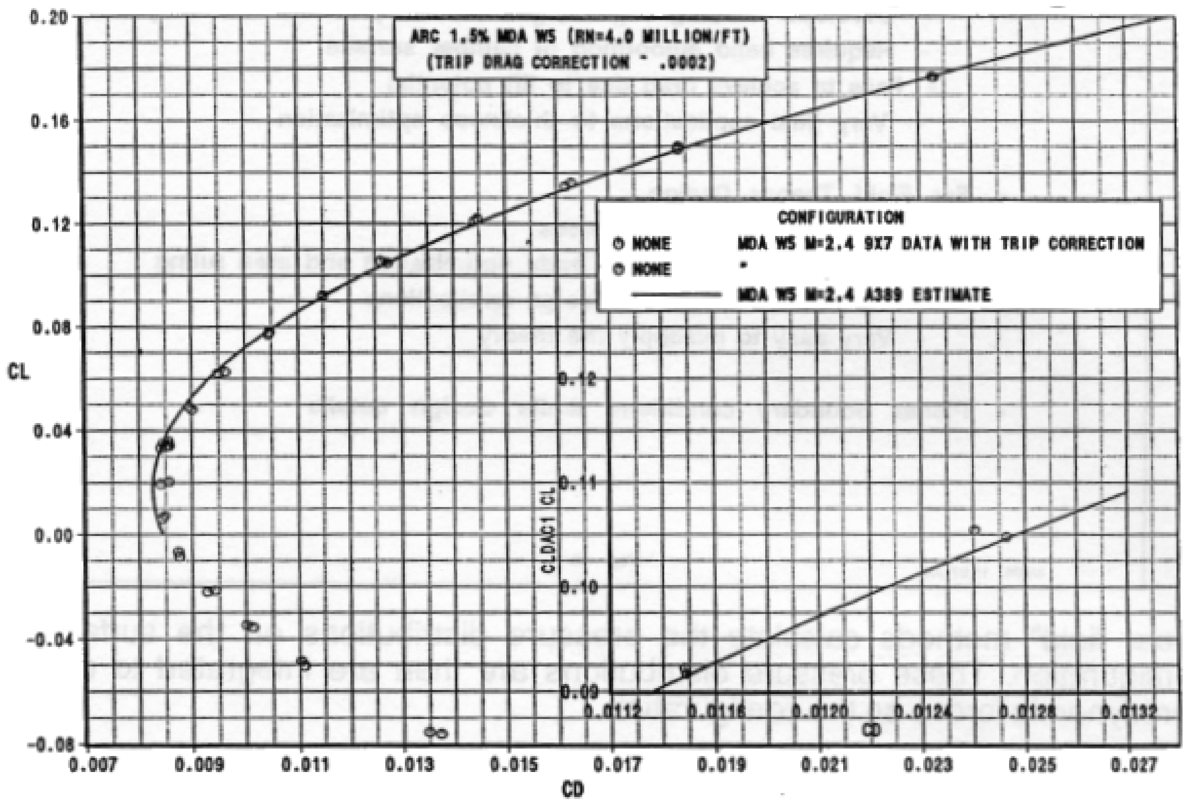


11/18/2002 12:59

Page 31

The comparisons also indicate very good agreement with the linear theory predictions and the NASA Ames test data for the 2.7% Ref H wind tunnel model.

MDA Optimum W5 Arrow Wing Design



11/18/2002 12:59 Page 32

This figure shows a comparison of linear theory predictions with the test data for the NASA Ames / MDA non-linear arrow wing configuration, W5. This model was the result of a very successful NASA/ industry joint nonlinear design optimization activity involving NASA Ames and McDonnell-Douglas,

The test and theory agree very well. The comparisons in this figure and the previous figures were made at or near the design Mach number. The linear theory predictions typically do not agree as well with test data at off-design Mach numbers The theory does not properly account for the leading edge forces that typically occur at the off design Mach numbers.

However for the purpose of establishing meaningful performance improvement projections, we are concerned about the usefulness of linear theory predictions at the design Mach number conditions.

Linear Theory Design

- **Near Field Design for Camber and Twist**
  - **Develops leading edge singularities in pressures or slopes**
  - **Requires hand smoothing of camber surface**
  - **Fails to achieve drag due to lift potential**
  - **Very little applications to thickness optimization**

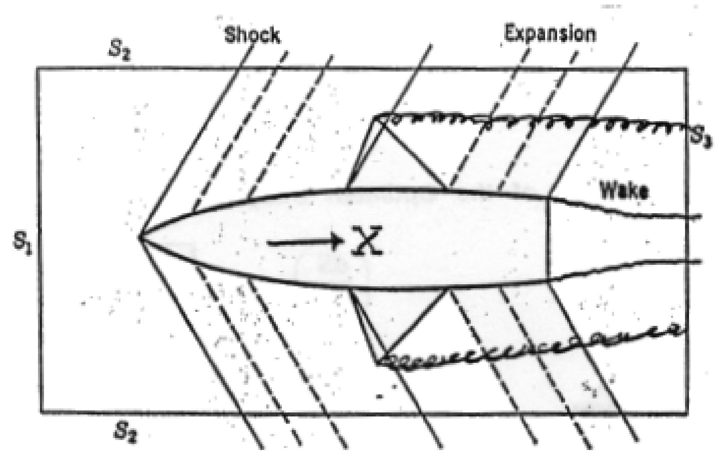
- **Far Field Theory Design**
  - **Not bothered by edge forces**
  - **Has been restricted to body optimization and area ruling**
  - **Verified by non-linear design applications**
  - **Very easy to misapply the theory**

- **Planar boundary conditions limits design details**

11/18/2002 12:59 Page 33

"Near field" methods calculate the pressure distributions on the surface of a configuration. These pressure distributions are then are integrated to obtain the aerodynamic forces on the configuration. Near field methods can be used develop optimized linear theory camber and twist distributions. Mathematical singularities in the solutions can produce localized infinite slopes or pressures. The designs that produce the pressure singularities are difficult to evaluate properly and can often lead to leading edge separated flow. The designs that result in singularities in the surface slopes require that the linear theory designs be hand modified in the regions where the singularities occur. These regions include the wing root and near any break in the leading edge sweep . This smoothing process has a rather significant adverse effect on the drag. Consequently the linear theory designs fail to achieve the theoretical low drag potential. Very little has been done in the area of wing thickness optimization using linear theory until very recently. A new far field approach has been developed and appears quite promising.
"Far field" theory has been used to optimize body area distribution and to develop area rule body shapes to minimize wing / body interference effects.
Linear theory concepts and methods have been very successful in developing low drag nacelle installations.
Because of the planar boundary conditions, linear theory cannot capture design details. This can also be a significant limitation.
The linear theory fundamentals are considered reasonable to identify meaningful lower bound drag levels, however, linear theory cannot produce the designs that achieve these levels.

FAR-FIELD THEORY DRAG CALCULATION



- MOMENTUM CHANGE THRU S_1 AND S_3 :
  - FRICTION DRAG
  - WAKE DRAG
  - VORTEX DRAG
- STREAMWISE MOMENTUM CHANGE THRU S_2
  - VOLUME WAVE DRAG
  - LIFT WAVE DRAG

11/18/2002 12:59 Page 34

Far field theory drag calculation methods are derived from a control volume approach. The configuration is enclosed in a large cylinder that extends both radially and downstream a great distance from the configuration. The streamwise momentum change through the ends of the cylinder is equal to the friction drag, any base or wake drag plus the induced drag. The induced drag equation for supersonic flow is exactly equal to the subsonic induced drag equation.

At supersonic speeds, the shock waves and expansion waves that are generated by the configuration pass through the cylindrical surface. The resulting streamwise loss of momentum through the cylindrical surface is equal to the volume wave drag plus the wave drag due to lift.

BODY WAVE DRAG

$$\frac{D_W}{q} = \frac{1}{2\pi}\int_0^L\int_0^L S''(x)S''(\xi)\ln|x-\xi|\,dx\,d\xi$$

This Similar to the Induced Drag Equation with: $\Gamma \approx \frac{\mathbf{dS}}{\mathbf{dx}}$

The Slope of the Optimum Body Area Distribution must be Elliptic

$$\left(\frac{dS}{dx}\right)_{Opt} \approx \sqrt{1-\left(\frac{x}{L}\right)^2}$$

Therefore: $S_{Opt} = A\,max\left[1-\left(\frac{x}{L}\right)^2\right]^{3/2}$

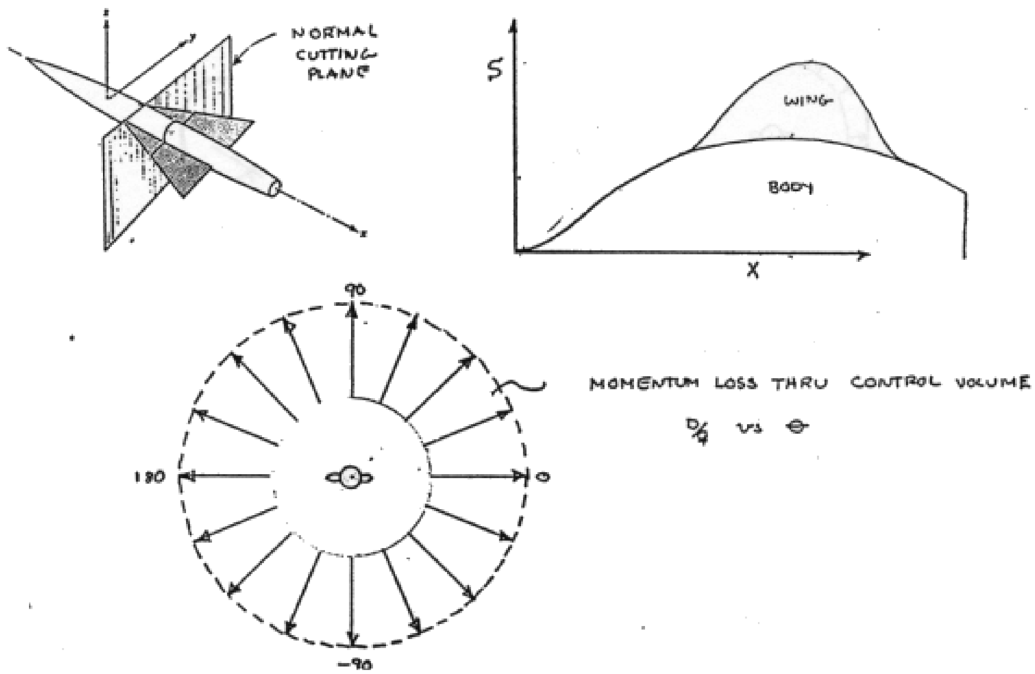
MINIMUM DRAG FOR GIVEN VOLUME : SEARS-HAACK BODY

11/18/2002 12:59 Page 35

Von Karman represented a body of revolution as line of sources and sinks. He obtained the above equation for the wave drag of a body of revolution.
This form of this equation is very similar to the induced drag equation shown earlier. The function in the induced drag equation is the spanwise derivative of the circulation or lift distribution, while the similar function in the wave drag equation is the second derivation of the area distribution.
The optimum lift distribution for minimum drag is elliptic. Because of the mathematical similarity of the wave drag equation, we can immediately note that the slope of the optimum body shape must also be an elliptic distribution.
The optimum body area distribution for a given body is then obtained by integration of the elliptic slope distribution.
This resulting shape is called a Sears-Haack body, which is the minimum drag body shape for a given volume.

TRANSONIC AREA RULE

$\mathbf{M} \approx 1.0$



- **MINIMUM DRAG: COMBINED NORMAL AREA DISTRIBUTION IS A SEARS-HAACK BODY**

11/18/2002 12:59 Page 36

A symmetric wing plus body , in linear theory, can be represented by line sources for the body and a planar sheet of sources for the wing.

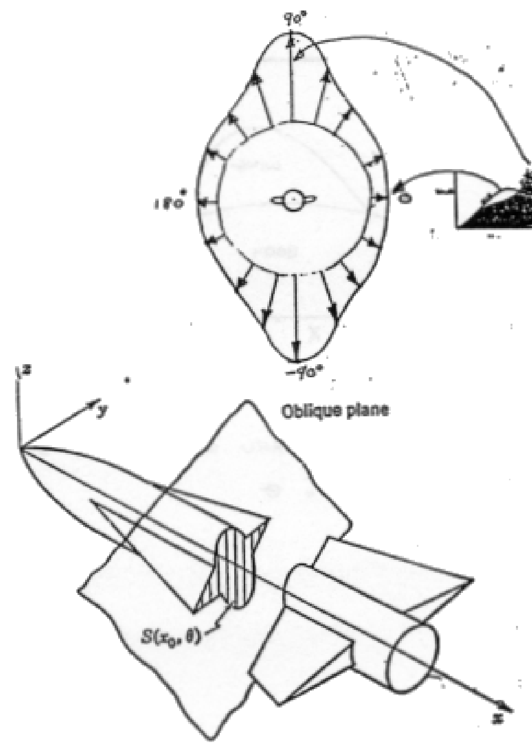
Whitcomb discovered experimentally and Hayes discovered analytically that near Mach one, the wave drag of the wing plus body is the same as that of an equivalent body which has an area distribution equal to the wing body cross sectional area distribution.

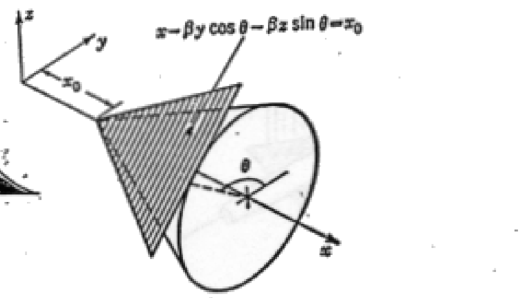
The physical interpretation of this result is as follows. Near Mach one the disturbances caused all of the sources radiate out in planes normal to the body axes. Linear theory allows superposition. Hence. all the sources in the same plane can be slide to the body axes without changing the drag. This results in the equivalent cross sectional area body.

Therefore, the wave drag at Mach one can be calculated by the isolated body wave drag equation with the area in the equation equal to the cross sectional area obtained by a cutting plane normal to the body axis.

Thus the minimum volume wave drag for a symmetric wing plus body near Mach one occurs if the combined area distribution is equal to a Sears-Haack body.

SUPERSONIC AREA RULE





$$S_n(x_0,\theta) = \frac{1}{M_\infty} S(x_0,\theta)$$

$$\frac{D}{q} = \frac{1}{2\pi}\int_0^{2\pi}\left[\frac{-1}{2\pi}\int_0^{\ell(\theta)}\int_0^{\ell(\theta)} S_n''(x,\theta)\, S_n''(\xi,\theta)\, \ln|x-\xi|\, dx\, d\xi\right] d\theta$$

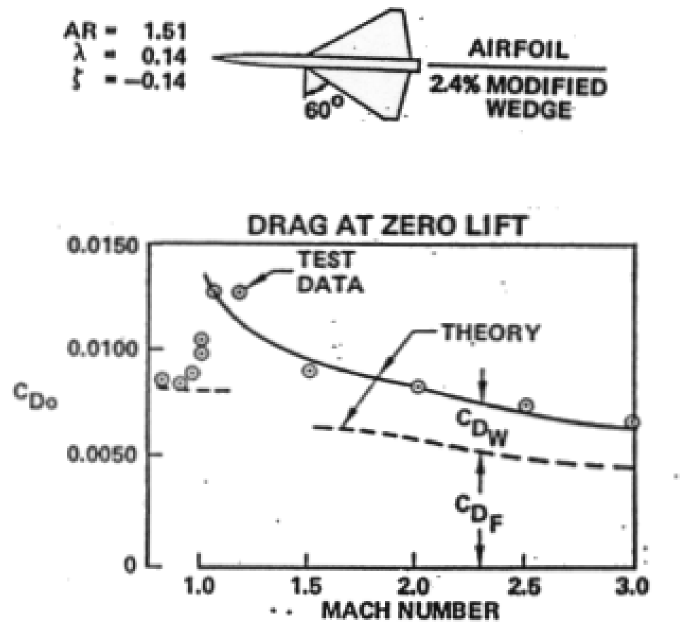
- **WAVE DRAG CAN BE CALCULATED AS A SUM OF "Θ" DEPENDENT EQUIVALENT BODY AREA DISTRIBUTIONS DRAGS**

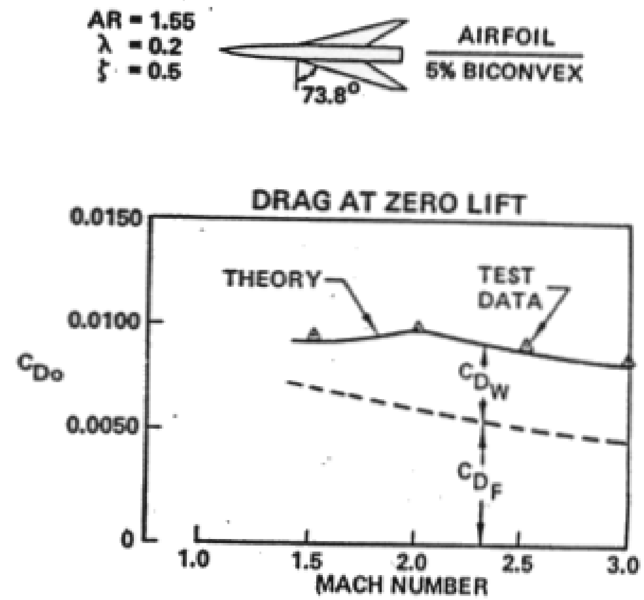
At supersonic speeds greater than Mach one, the the disturbances from the sources and sinks that represent the wing / body propagate in the down stream Mach cones.

The momentum loss around the configuration is no longer symmetric. However the concept of sliding all sources / sinks in the same propagation plane still applies .

The propagation planes are tangent to Mach cones with vertices on the axes of the body. The propagation planes are identified by the angle theta. Theta zero represents momentum loss on the plane of the wing. Theta 90 represents momentum loss in the Z axes above the configuration. All the sources / sinks in the cut through the wing / body planar surface corresponding to the intersection of the propagation plane at a given angle theta are slid along the intersecting cut to the axes of the body. This creates a theta dependent equivalent body for each cutting plane angle from 0 to 360 degrees. The wave drag of the configuration is then calculated from the sum of the drags of the theta dependent bodies.

TYPICAL ZERO LIFT DRAG ASSESSMENTS
- SYMMETRIC CONFIGURATIONS





11/18/2002 12:59 Page 38

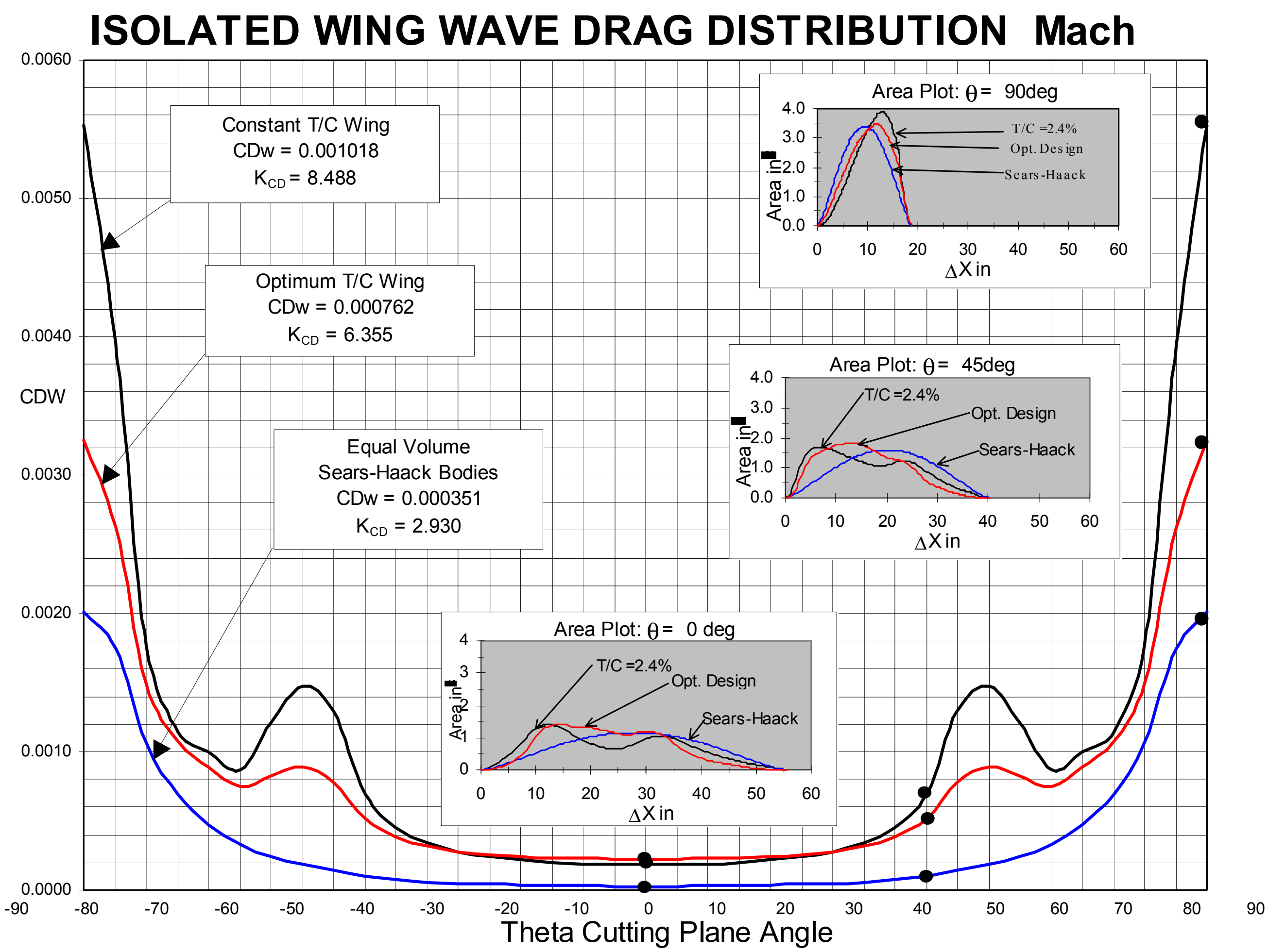
This is a typical test versus theory comparison of drag at zero lift (CDo) for two symmetric wing / body configurations.
The theoretical predictions include fully turbulent flow flat plate skin friction drag plus the volume wave drag calculated by the supersonic wave drag program.
The test versus theory agreement is very good and shows that the far field wave drag method can give valid drag predictions.

“LOWER BOUND” SUPERSONIC WING / BODY WAVE DRAG

$$\frac{D_{WV}}{q} = \frac{1}{2\pi} \int_0^{2\pi} \int_0^{l(\theta)} \int_0^{l(\theta)} S''(x,\theta) S''(\xi,\theta) \ln|x-\xi| \, dx \, d\xi \, d\theta$$

- **“MINIMUM” DRAG: EACH “Θ” EQUIVALENT BODY IS A SEARS-HAACK BODY**
- **USUALLY IMPOSSIBLE TO ACHIEVE**

This is the far field for a symmetric non-lifting configuration. The volume wave drag is the average of the theta dependent equivalent bodies.
Because of the similarity of this equation with the transonic wing / body equation, it follows that the lower bound zero lift wave drag for any symmetric configuration occurs if each of the theta dependent equivalent body is a Sears-Haack body for the same length and maximum area.
This lower bound is exact for a yawed elliptic wing with a circular arc wing section and constant spanwise curvature. However, it is generally impossible to define such a volume distribution for an arbitrary wing / body configuration.
Thus we need a more realistic lower bound for zero lift wave drag.



11/18/2002 12:59

Page 40

This figure shows the equivalent body wave drag as a function of the cutting plane angle. The angle is from -90 deg which is below the configuration around to the right of the configuration at 0 deg to the top of the configuration, theta = 90deg. The drag variation is symmetric around to the left of the configuration and is not shown in the figure.

This is the drag for an isolated cropped delta wing with a supersonic leading edge (60 deg sweep) at Mach 3.0. Three drag levels are shown and correspond to:

- Constant T/Cmax = 2.4% wing
- Optimized spanwise T/Cmax wing with the same wing volume.
- The drag at every theta angle if the body shape was a Sears-Haack body

The small insert figures compare the equivalent body shapes at theta angles of 0, 45 and 90 deg.

The wave drag for the constant T/C wing is approximately CDw = 0.00102. The wave drag of the optimum wing is CDw = 0.00076. The lower bound wing drag level is CDw = 0.00035 which is half of the optimized wing drag.

The difference between the lower bound and optimum drag levels would be less than a factor of two for wing plus body.

Consequently a factor of 1.75 times the lower bound drag is used for our achievable lower bound limit on zero lift wave drag.

CDo "TOO LOW" LIMIT

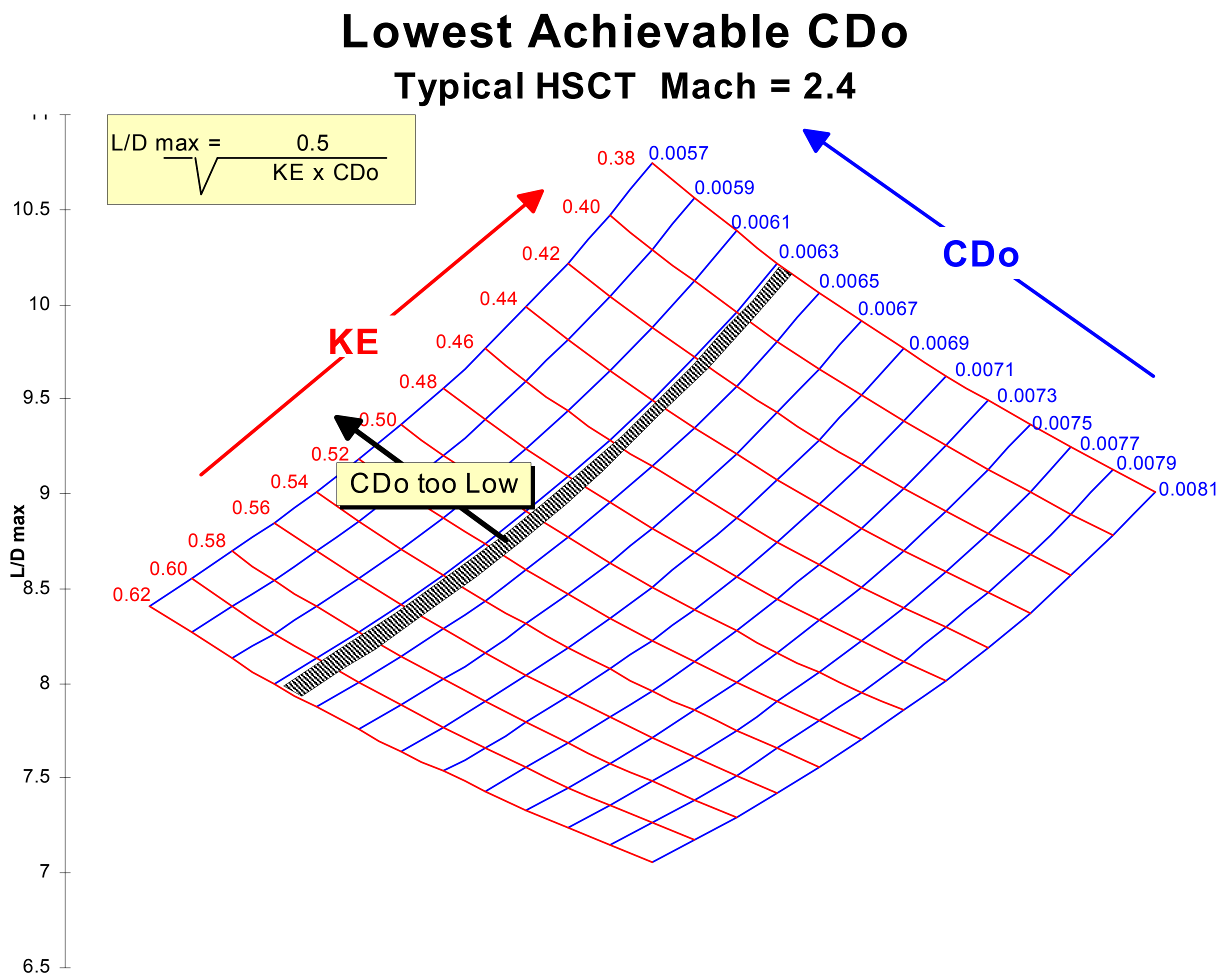
$$CDo > CD_F + 1.75\ CD_{W\ SH}$$

- **FULLY TURBULENT FLAT PLATE SKIN FRICTION**
- **WING / BODY WAVE DRAG = 1.75 X EQUIVALENT SEARS-HAACK BODY**
- **ZERO INSTALLED NACELLE WAVE DRAG**
- **EMPENNAGE WAVE DRAG INCLUDED IN WING / BODY WAVE DRAG**
- ***** DRAG "*CAN'T*" BE LOWER THAN THIS *****

11/18/2002 12:59 Page 41

The "too low" limit for zero lift drag is equal to the sum of:
- Fully turbulent skin friction drag
- Wing / body volume wave drag equal to 1.75 times the drag of an equivalent Sears-Haack body having the same maximum area as the combined wing plus body area distribution and the length of the fuselage.
- The empennage drag is included as part of the wing / body drag.
- Zero installed nacelle wave drag

The zero lift " can't be lower" then this level for the given configuration.



11/18/2002 12:59 Page 42

This show's the CDo " too low " boundary for the example HSCT configuration.

Wave Drag Due to Lift

$$\frac{D_{WL}}{q} = \frac{\beta^2}{4\pi^2}\int_0^{2\pi}\sin^2(\theta)\int_0^{l(\theta)}\int_0^{l(\theta)}\frac{dl(x,\theta)}{dx}\frac{dl(\xi,\theta)}{d\xi}\ln|x-\xi|\,dx\,d\xi\,d\theta$$

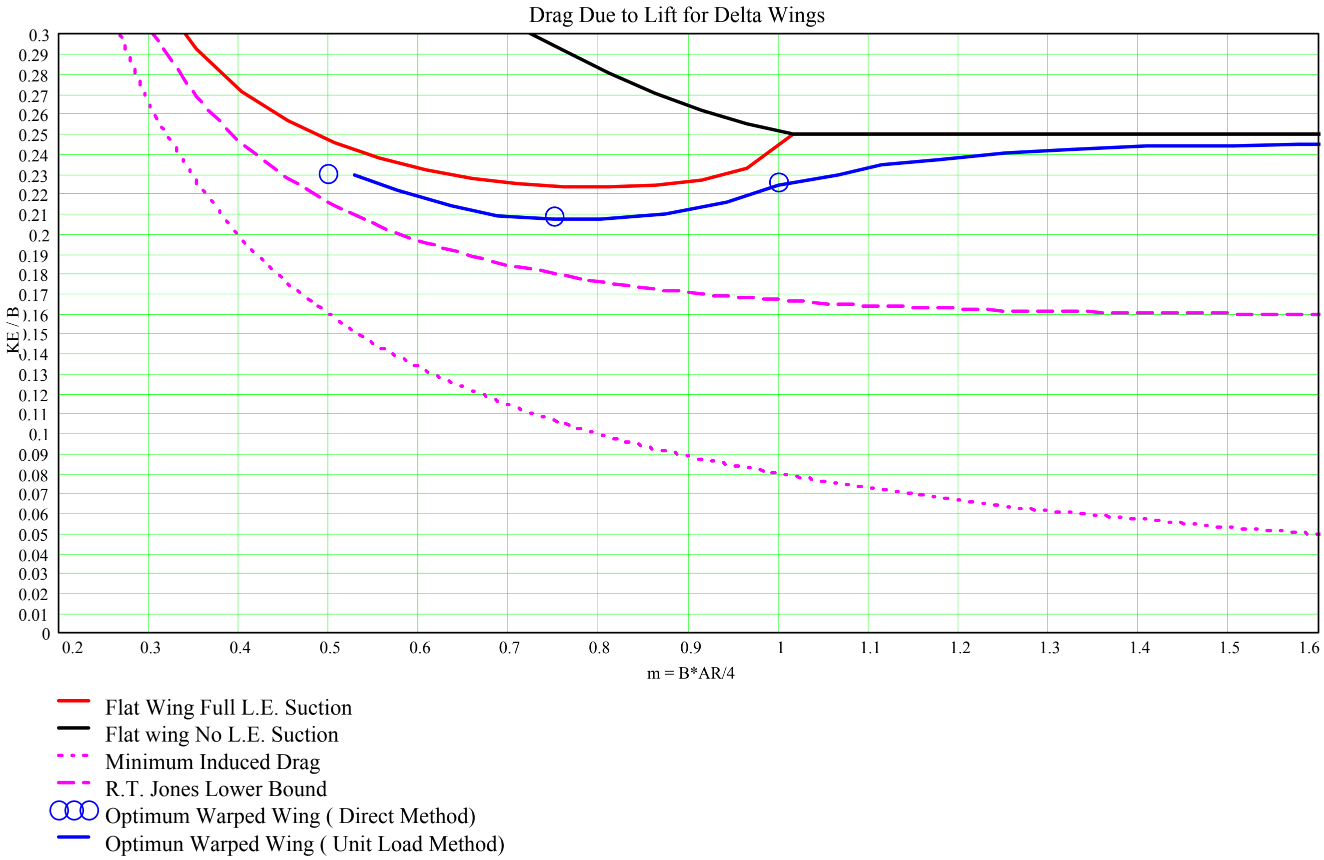
$$\beta = \sqrt{M^2 - 1}$$

- **Lower Bound Drag: Each "Θ" Lift Distribution is Elliptic**
- **Usually Mathematically Impossible to Define such a overall lift distribution on the wing.**

11/18/2002 12:59 Page 43

The far field theory drag equation can be extended to the calculation of wave drag due to lift by replacing the volume producing source elements by lift producing vortex elements. The resulting wave drag due to lift equation is shown in the figure.
Notice the $\sin^2\Theta$ term in the equation. This indicates that the contribution to wave drag due to lift is zero in the theta = 0 plane.
Because of the β^2 term, the wave drag due to lift is seem to equal zero at Mach =1 and increases rapidly with Mach number.
Since this equation is very similar to the previously shown induced drag equation and the supersonic wave drag equation, It is obvious that the lower bound for drag due to lift would occur if every "theta" lift distribution were elliptic.
This is exactly the case for a uniform load yawed elliptic wing. However it is usually impossible to prescribe a load distributions far any arbitrary planform that would be elliptic for all theta angles.
Hence this lower bound is in general not a realistic lower limit.

Delta Wing Supersonic Drag Due to Lift Potential



Page 44

11/18/2002 12:59

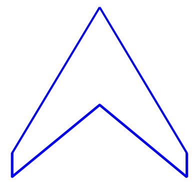
In order to arrive at a more meaningful achievable lower bound for supersonic drag due to lift lets look a various drag due to lift levels for a delta wing planform as shown in the above figure.

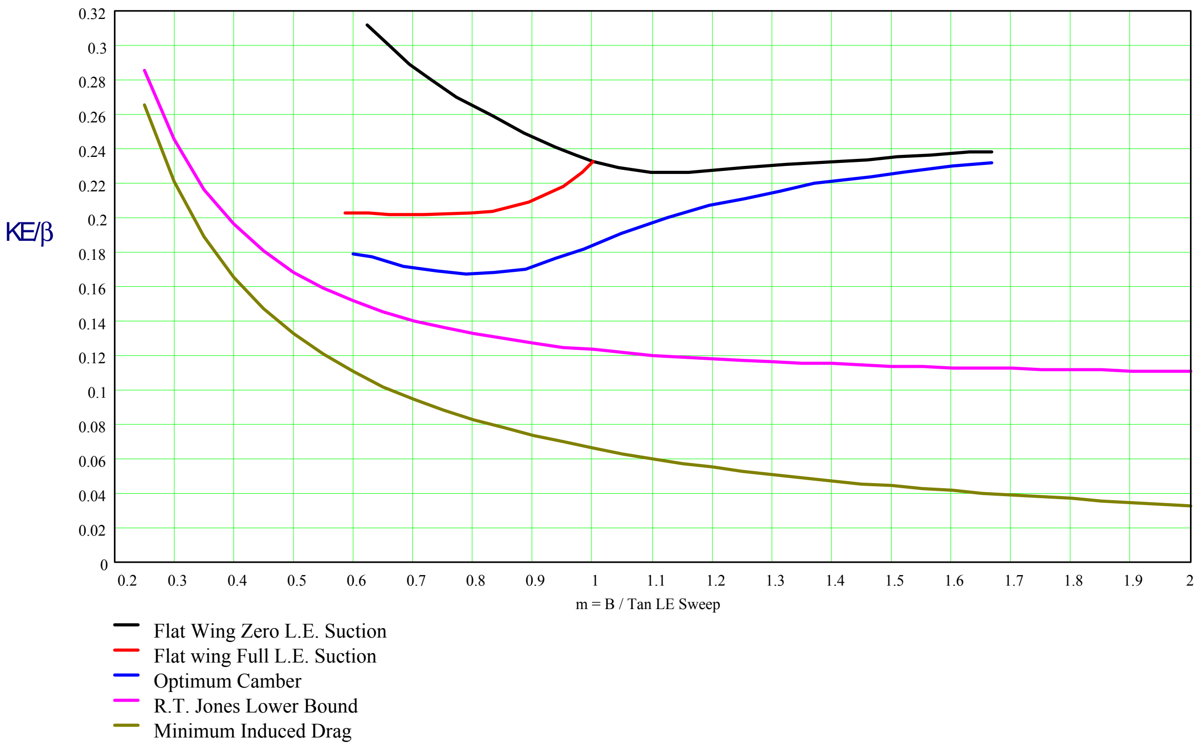
The horizontal axis variable m is the ratio of the tangent of the free stream Mach angle to the tangent of the wing leading edge sweep. A value of m less than one indicates that the leading edge is swept behind the Mach line (subsonic leading edge).

The lower “dotted line” is the minimum induced drag corresponding to an elliptic spanwise lift distribution. The “ dash “ line is the sum of the minimum induced drag plus the previously discussed lower bound wave drag due to lift. The upper curve is the upper bound for drag due to lift corresponding to a flat wing with zero leading suction. So the meaningful achievable drag due to lift must be somewhere between the “dash” curve and the upper curve.

The curve that starts at m=1 flat wing curve is the drag due to lift for a wing with full leading suction. The remaining drag due to lift curve is the minimum drag due to lift level calculated by near field linear theory. This drag level is about 95% of the flat wing with full leading edge suction. This is the simple criteria that is used for the achievable lower bound drag due to lift.

Arrow Wing Supersonic Drag Due to Lift Potential





11/18/2002 12:59 Page 45

This figures shows similar drag due to lift calculations for a classic swept arrow wing.
In this case, it is seen that the linear theory near field optimum drag due to lift potential is about 85% of the flat wing with full leading suction. As previously mentioned, the linear theory designs to achieve this drag level are physically impossible. However, it is felt that the drag level is achievable, but not by linear theory.
We will use for the achievable lower bound for wing body drag due to lift a level equal to 95% of the flat wing with full leading edge suction.

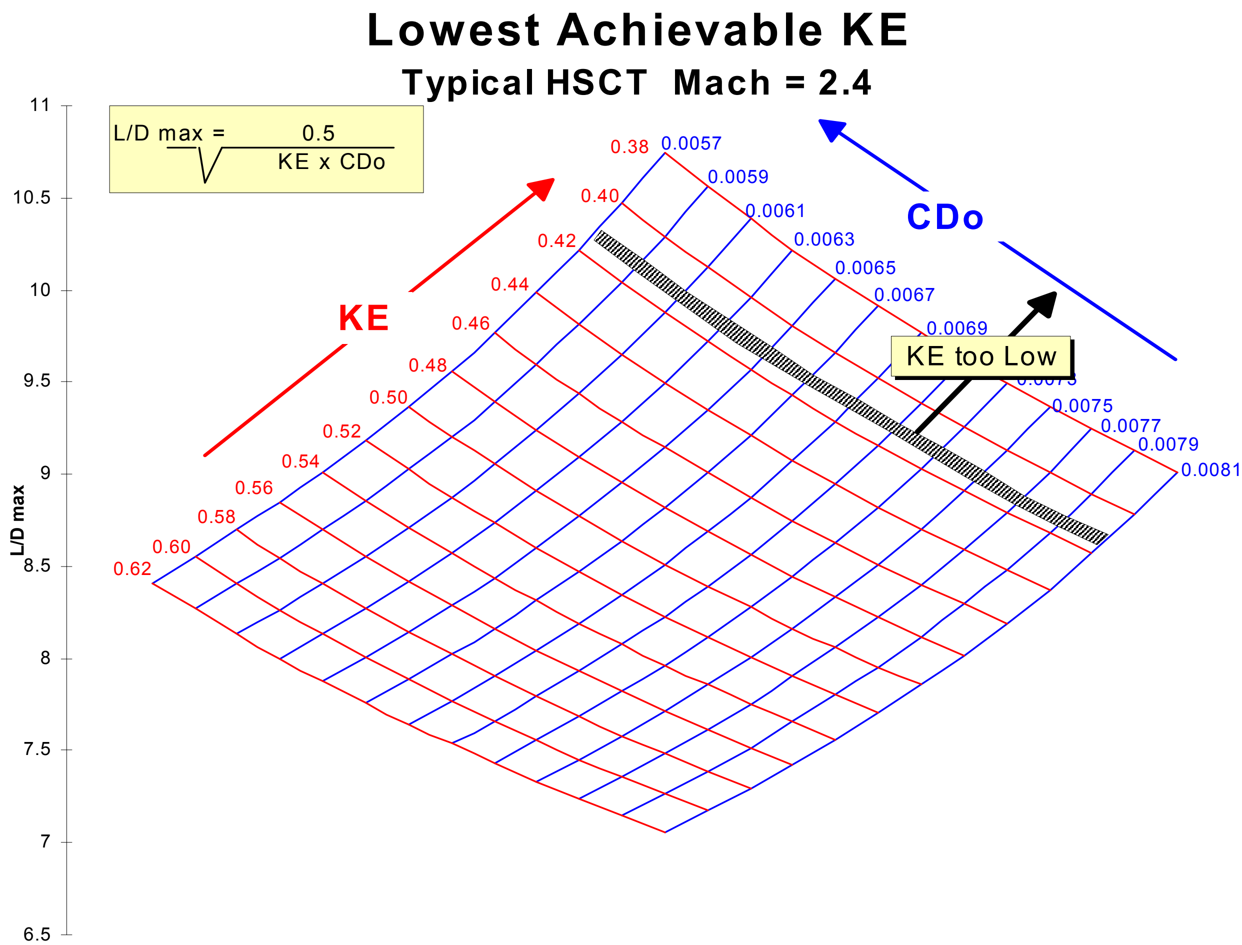
KE "TOO LOW" LIMIT

$$KE > KE_{S=1} \left\{ KE_{FAC} - 2\,K_{NAC}\,(\Delta CL_N/CL) - K_{TRIM}\,(KE_{S=1}/K_{TAIL})\,(S_{HT}/S_{REF}) \right\}$$

- **WING / BODY KE 5% LOWER THAN "FULL SUCTION" DRAG LEVEL ==> $KE_{FAC} = 0.95$**
- **FAVORABLE LIFT INTERFERENCE : 65% OF "IDEAL" ==> $K_{NAC} = 0.65$**
- **FAVORABLE TRIM DRAG : 80% OF "IDEAL" ==> $K_{TRIM} = 0.80$**

11/18/2002 12:59

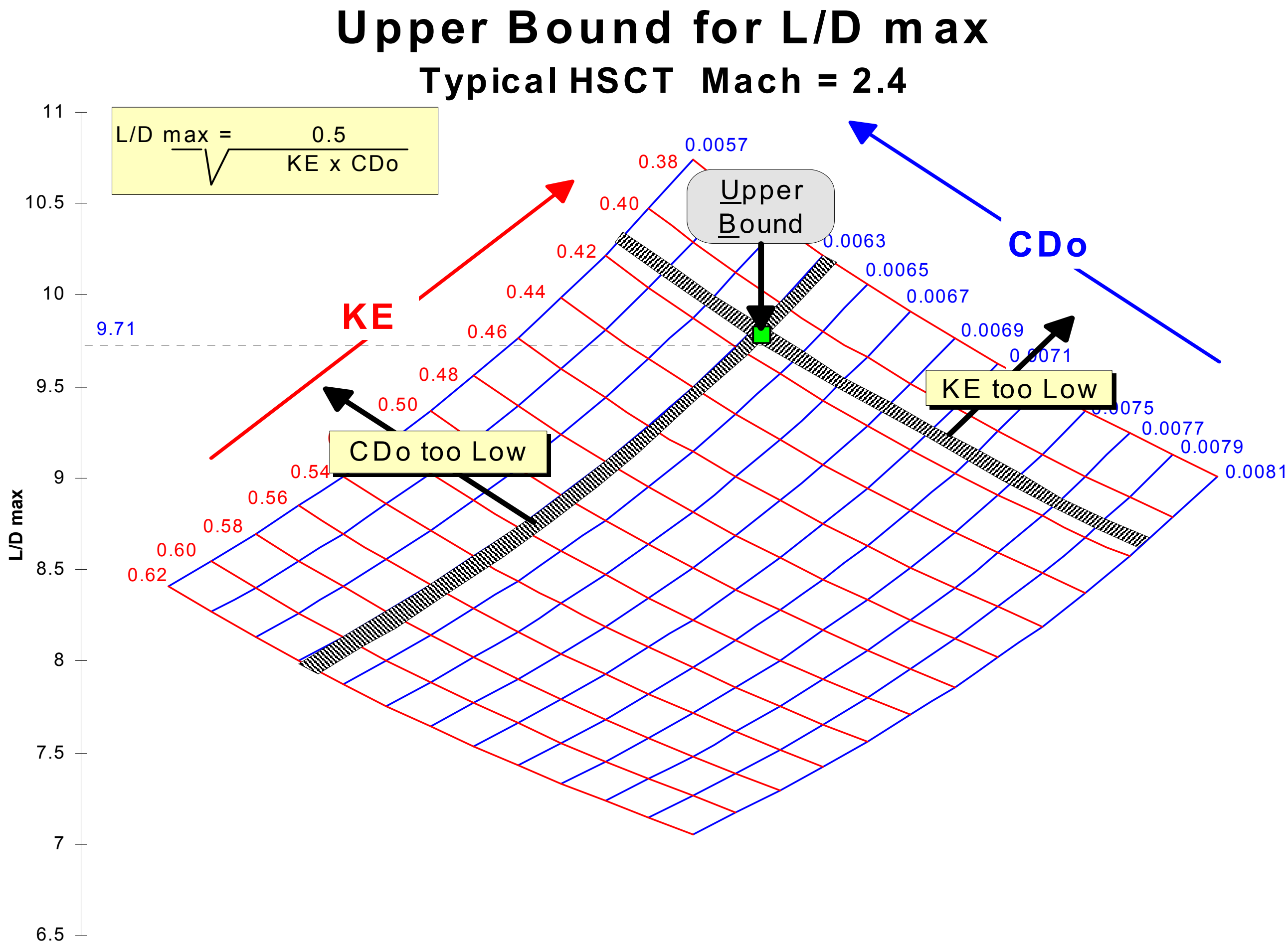
Nacelles designed properly to produce a positive pressure field on the lower surface of the wing can create a favorable interference lift that reduces the necessary wing / body lift for a given overall lift coefficient. This results in a reduction in wing / body drag due to lift. However, the nacelle pressure field acting on the wing camber surface produces a drag increment and the the wing lifting pressures acting on the nacelles produce an adverse buoyancy drag. On current nacelle installations about half of the ideal lift interference favorable interference is lost because of these two adverse effects. For the lower limit drag due to lift we assume that it is possible to achieve 65% of the ideal nacelle lift interference effects.
At supersonic speeds a horizontal tail upload will also result in a reduction in drag due to lift. The ideal level occurs when the tail upload is not reduced by any wing downwash effects. A favorable trim drag equal to 80% of the ideal level is considered to be achievable.
As previously mentioned, the achievable wing / body drag due to lift level that is used is equal to 95% of the flat wing with full leading suction.



11/18/2002 12:59

Page 47

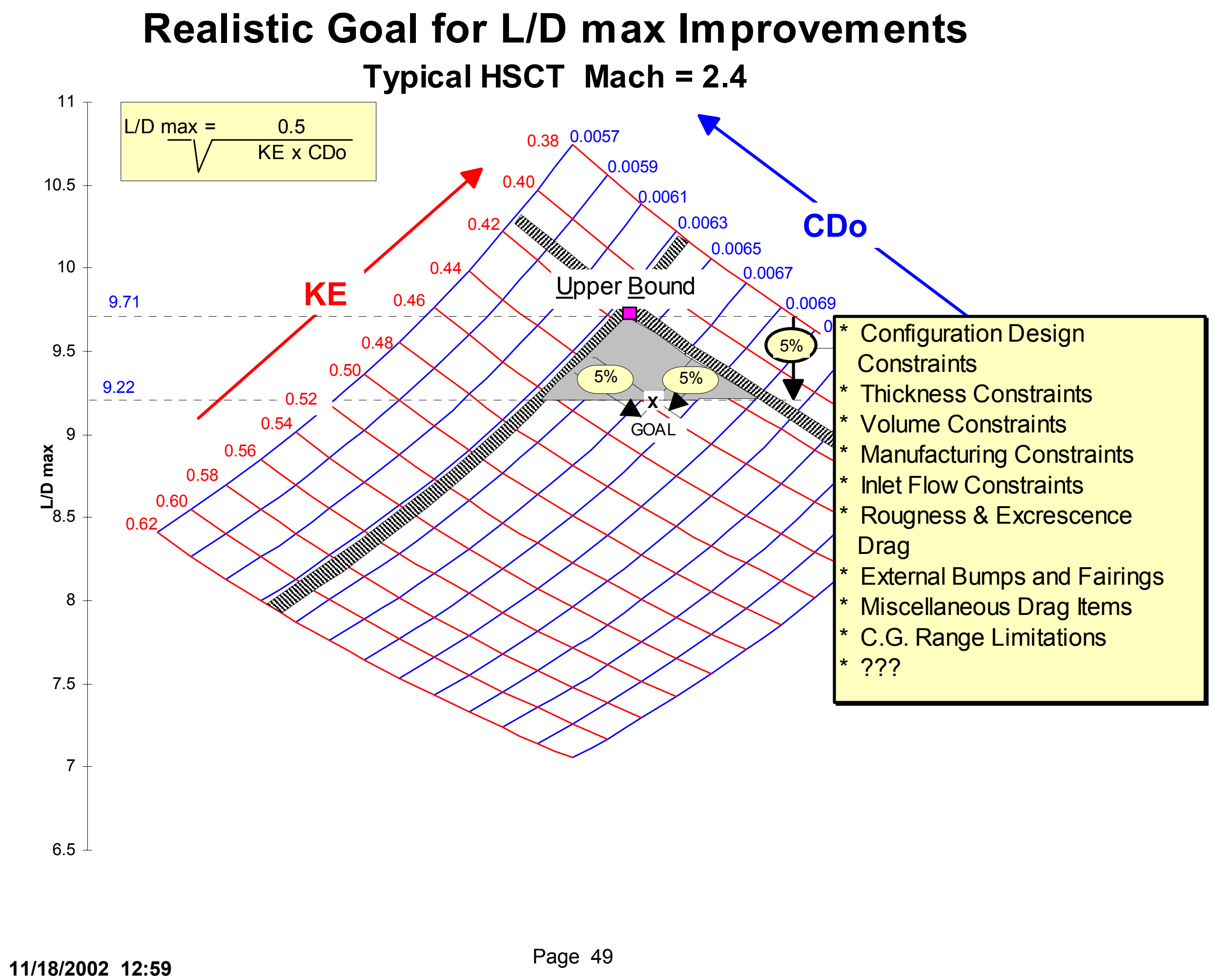
This figures shows the “KE too Low” boundary for the example typical HSCT configuration.



Page 48

11/18/2002 12:59

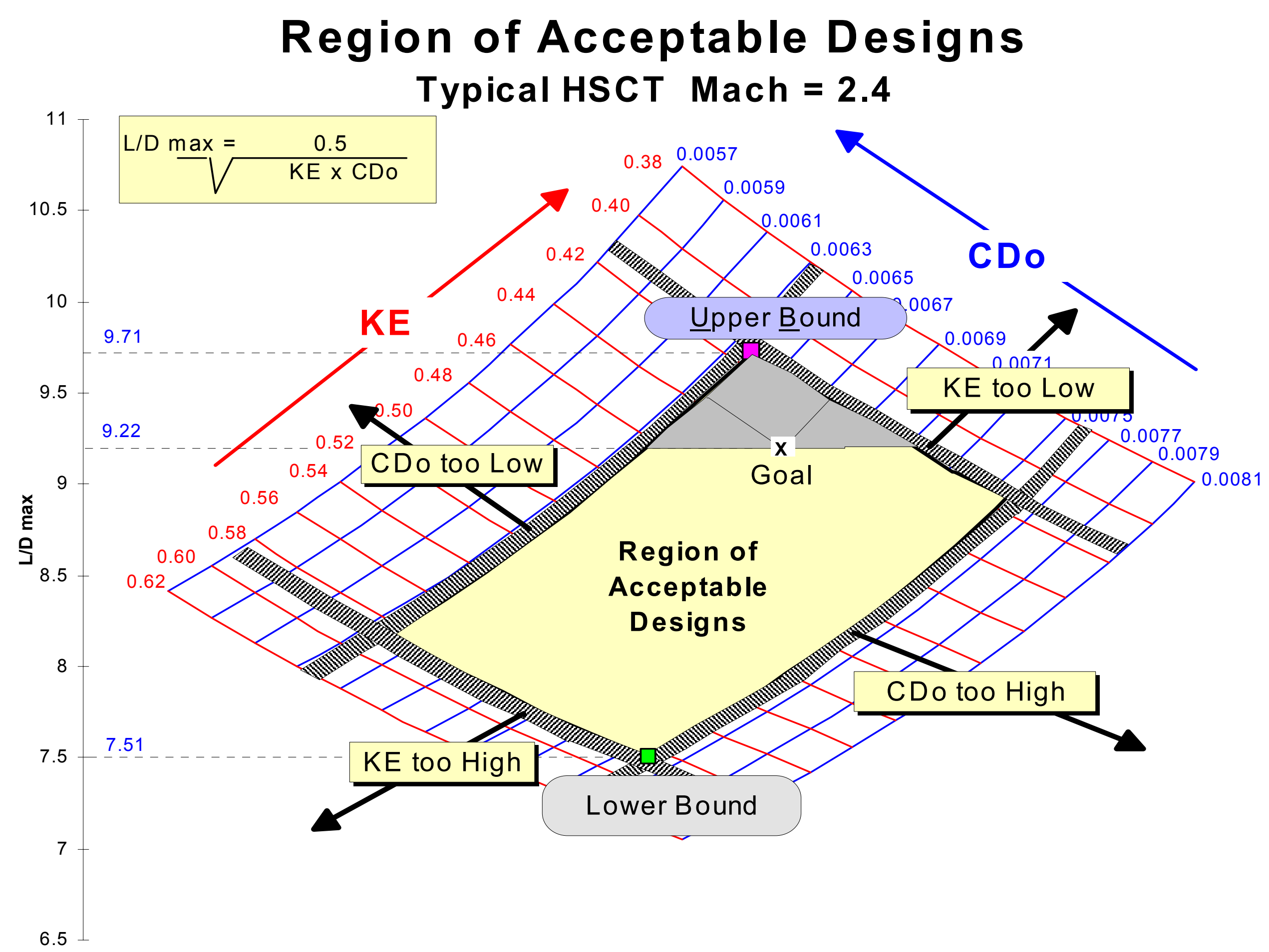
The intersection of the CDo “too low” boundary with the KE “too low” boundary define the upper bound for L/Dmax



Similar to the subsonic aircraft “tops down”, the L/Dmax is practically not achievable because of other configuration design considerations.
These factors for a supersonic transport aircraft include such factors as:

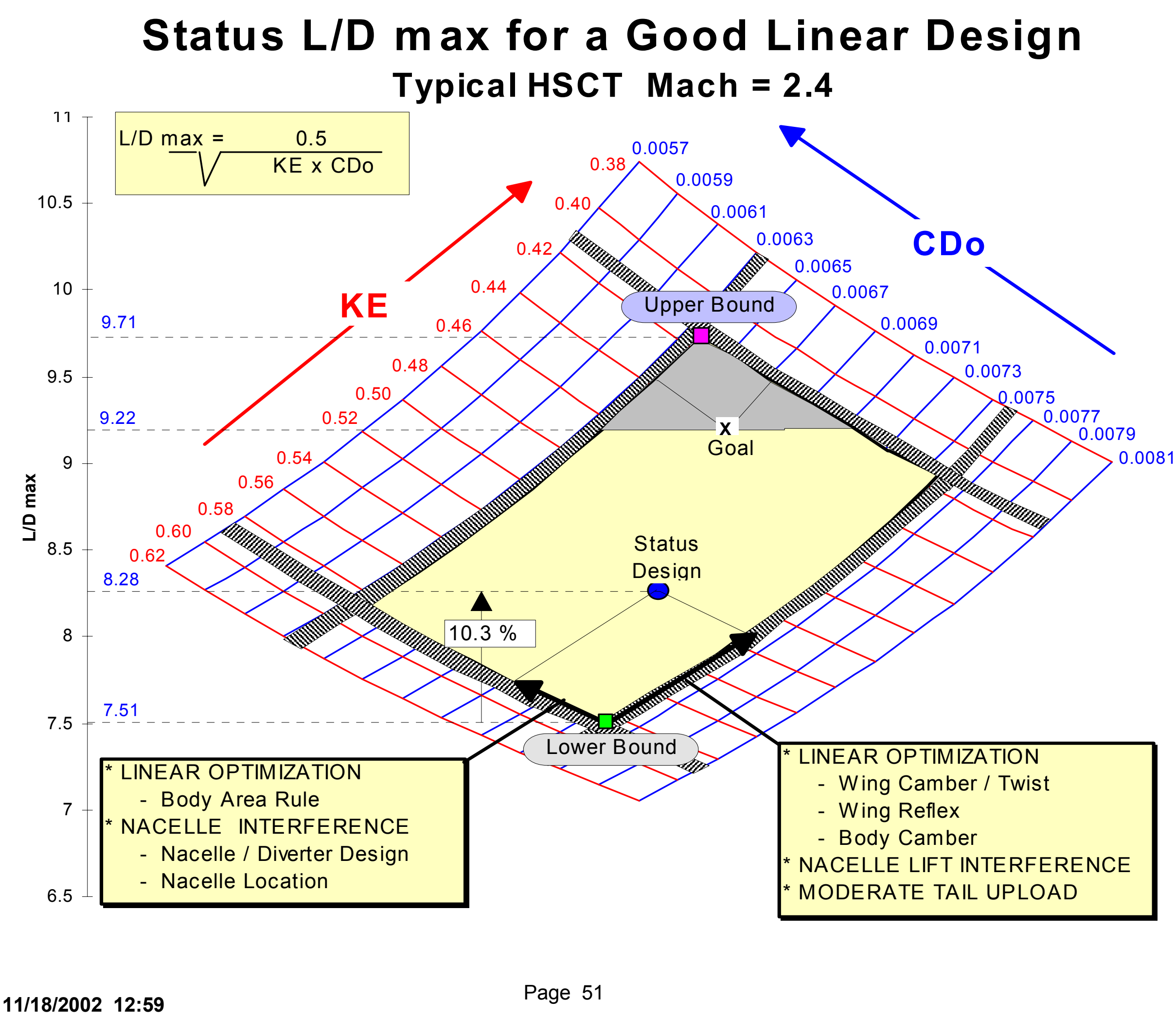
- Configuration thickness and volume constraints
- Manufacturing and surface curvature constraints
- Inlet flow constraints
- Ground clearance effects on aftbody upsweep
- External bumps and fairings
- Roughness and excrescence drag
- Cruise center cg gravity limitations
- Miscellaneous drag items

A “goal” L/Dmax equal to 95% of the achievable L/Dmax is used to account for these effects.



11/18/2002 12:59 Page 50

Combining the upper and lower boundaries for zero lift drag, CDo, and for drag due to lift factor, KE, defines the region of acceptable designs for a specific configuration. This acceptable design region is shown for the example HSCT in the figure above.



This figure can be used to identify the level of aerodynamic efficiency relative to the upper and lower L/D bounds.

In the above example, the linear theory status design has an L/Dmax that is 10.3% greater than the lower bound corresponding to the Concorde technology level. This configuration achieved favorable aerodynamics effects from a combination of:

- Reduced wing / body drag from body area ruling interference effects
- Favorable nacelle / airframe volume wave drag effects
- Reduced drag due to lift from the linear theory camber / twist design plus wing reflex to reduce the adverse nacelle on camber effects.
- Favorable nacelle lift interference effects.
- Favorable trim drag

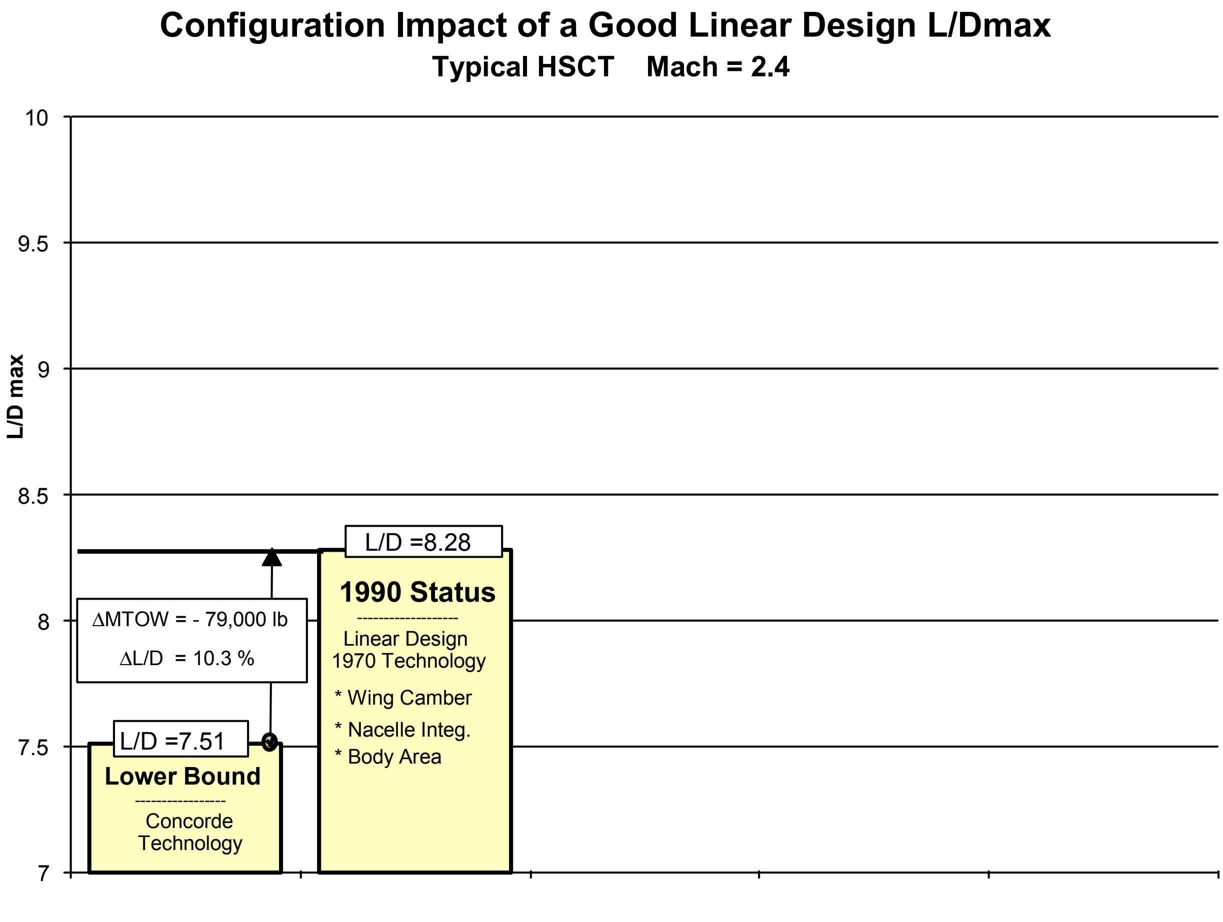
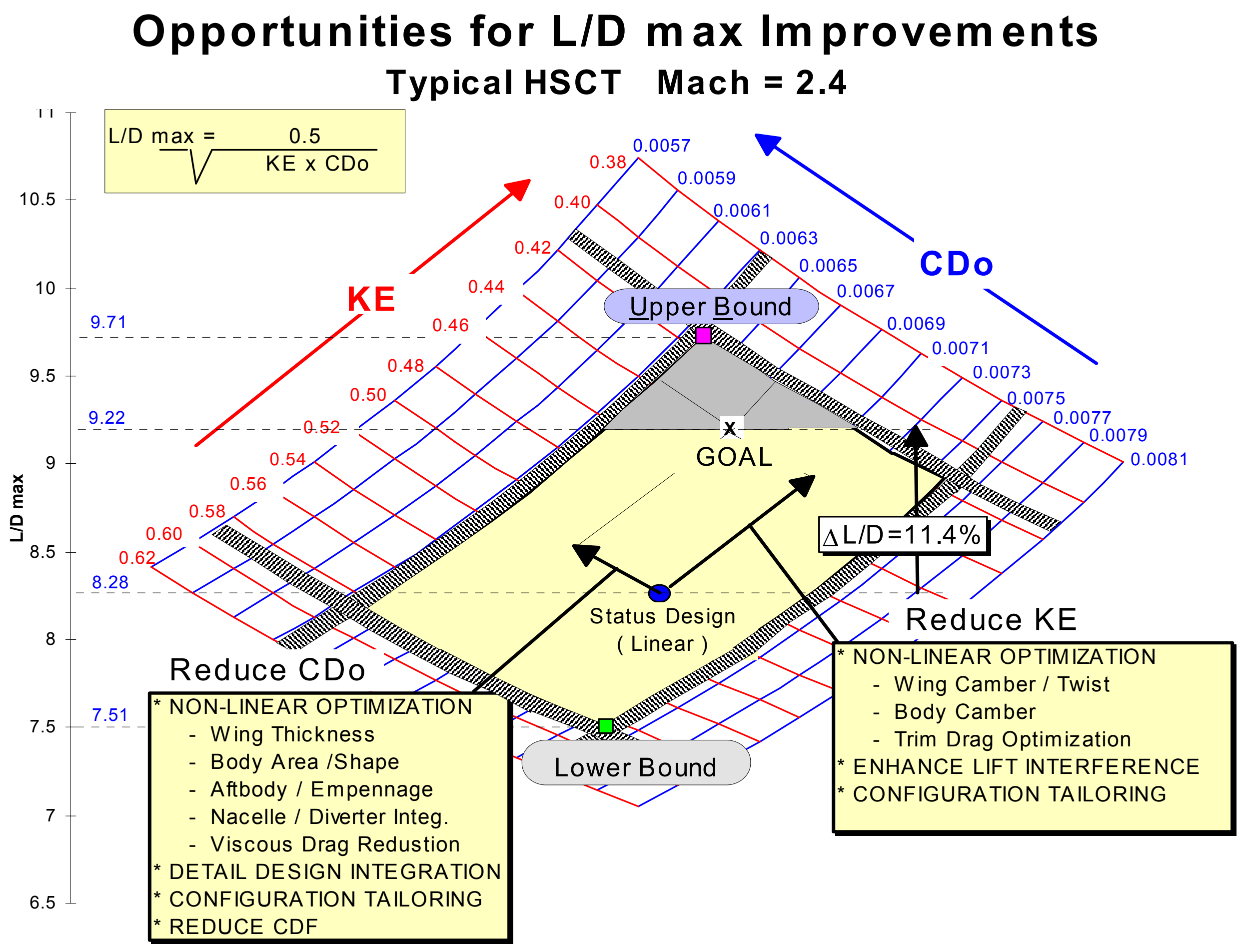


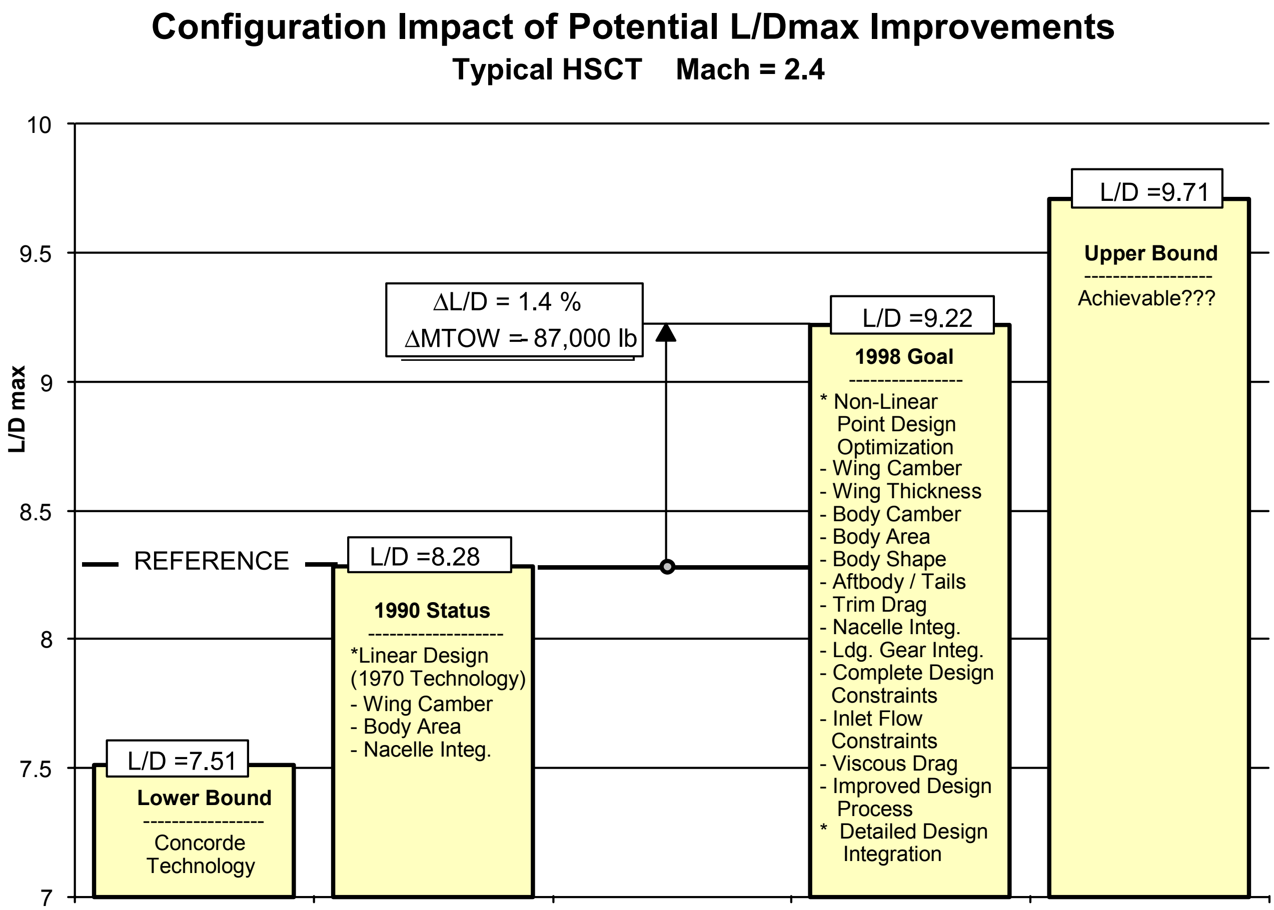
figure results in a savings of 79.000 lbs in max take off weight, MTOW. This is representative of 1990 technology linear theory design capability.



Page 53

11/18/2002 12:59

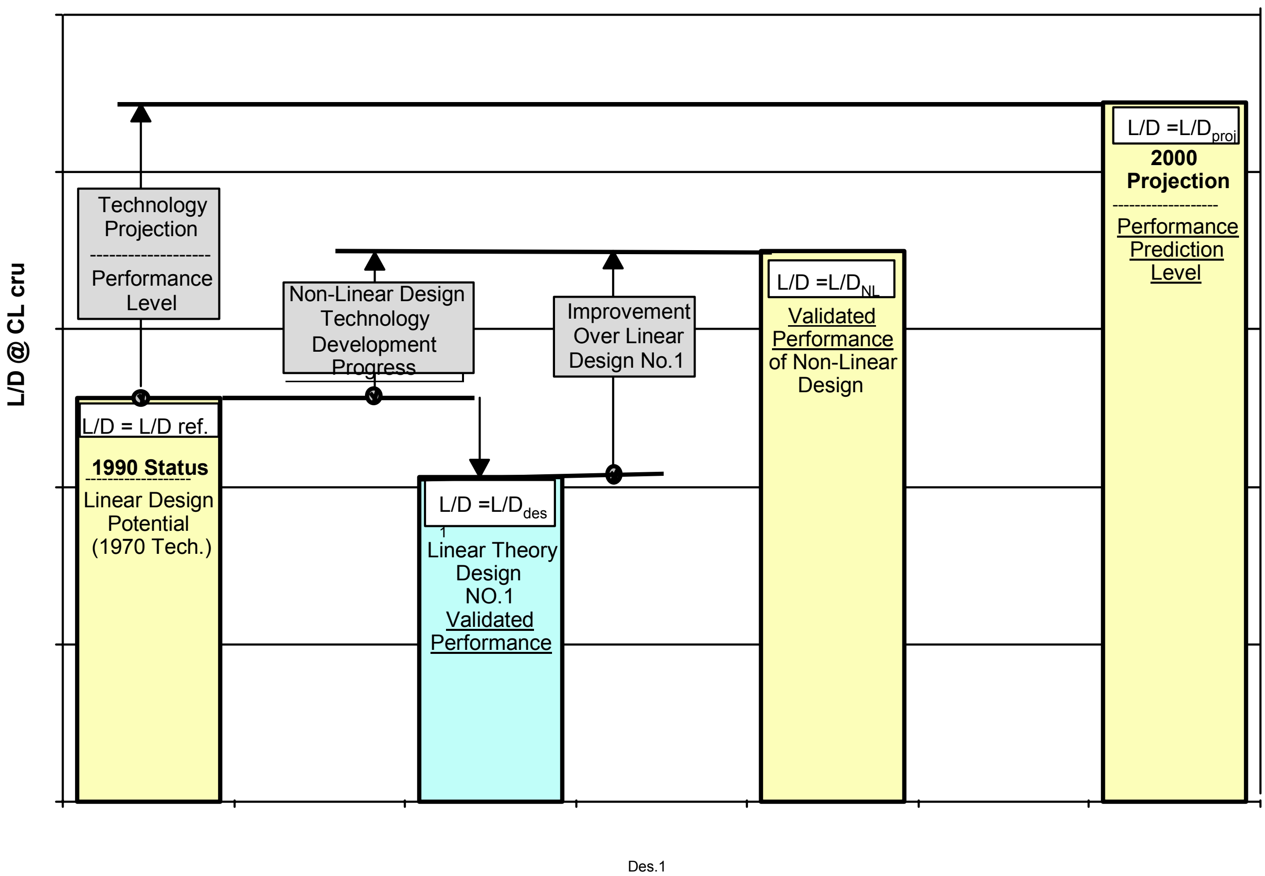
The difference between the goal L/Dmax level and the L/Dmax of the linear theory status design is the projected benefit of design optimization and design development using the emerging advanced nonlinear design and analysis methods.
The figure factors that are expected to contribute to reductions in both CDo and the drag due to lift factor, KE.



11/18/2002 12:59 Page 54

This figure shows the impacted of the projected improvements in cruise L/Dmax on the MTOW of the mission sized HSCT configuration relative to the current linear design.
The 11.4% projected improvement in L/Dmax will result in a reduction in the maximum takeoff weight of 87,000 lbs.

Tracking Advanced Technology L/Dmax Improvements
Typical HSCT Mach = 2.4

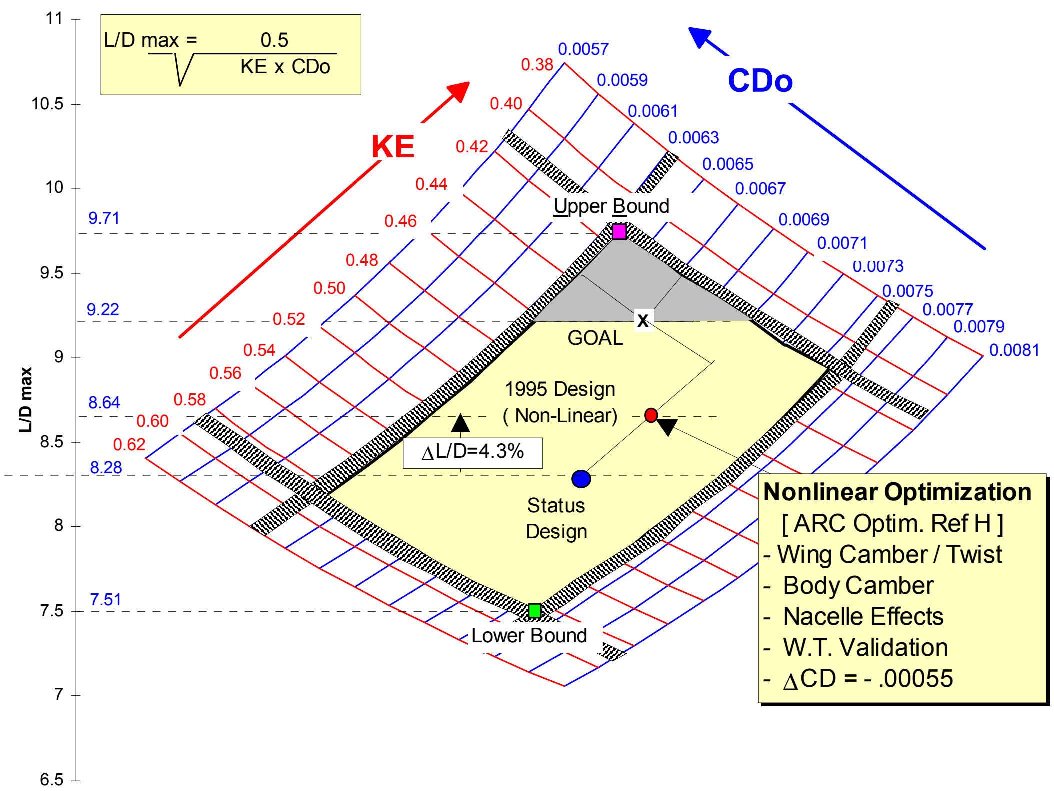


Page 55

11/18/2002 12:59

This chart shows the procedure that should be used to track the progress in improvements in cruise L/Dmax relative to a mission performance baseline.
Experience has shown that the preliminary theory design methods can identify a level of performance achievable by a linear theory design provided that a sufficient number of design iterations between linear design, nonlinear design analysis and modifications to the linear design are made. During this design iteration, the linear theory performance predictions do not very significantly. The linear theory design is considered validated if the nonlinear prediction or wind tunnel test data matches the linear theory design predicted performance.
The figure above illustrates the effect if the design process is not carried to convergence. A successful nonlinear design would show a greater improvement relative to the "poor" linear design. The technology gains must be measured relative to the performance levels of L/Dmax even though the actual performance improvement relative to a poor initial design is greater.
Of course, the ultimate level of success is how close a nonlinear design comes to the predicted target level of L/Dmax.

1995 Non-Linear Design Progress- L/D max



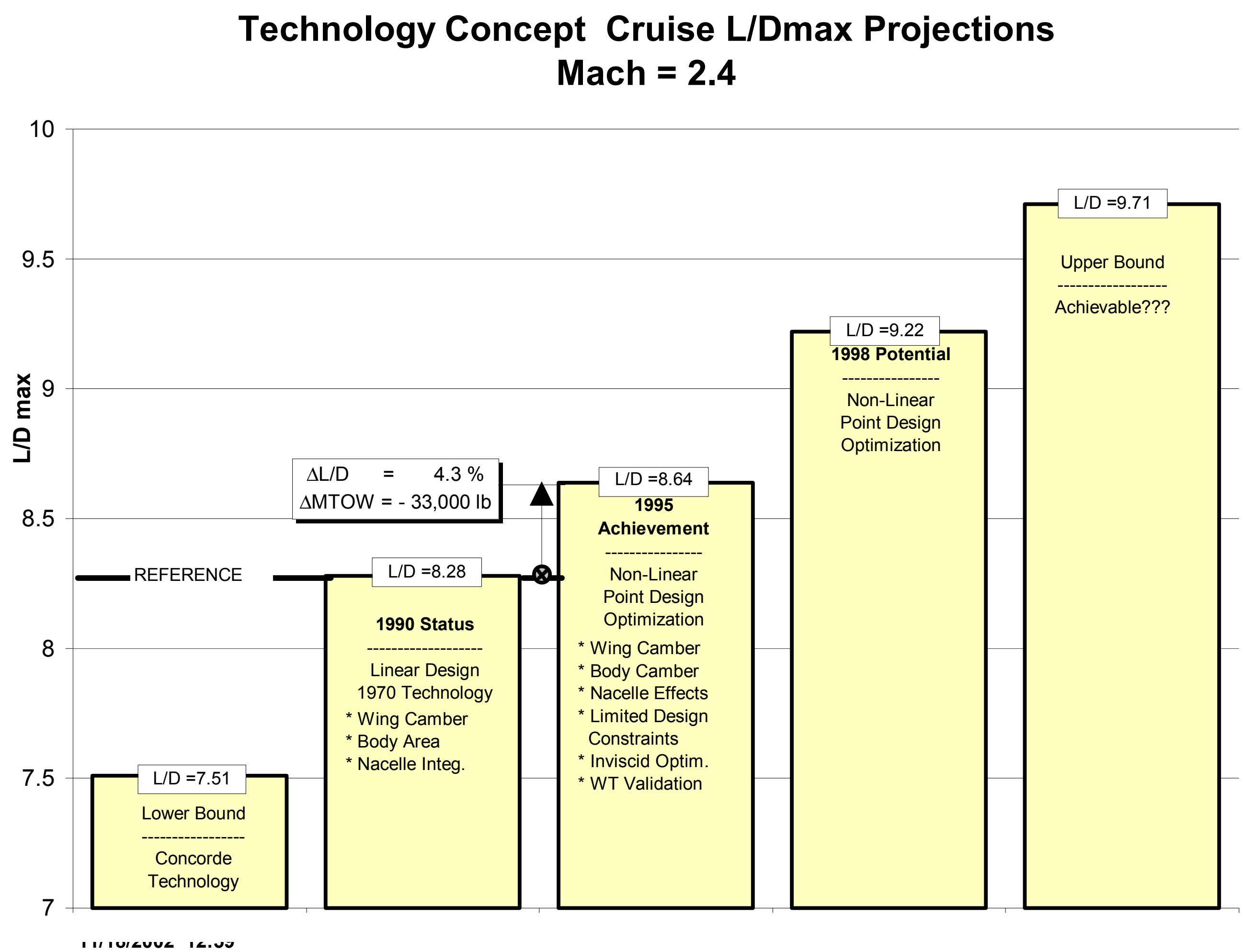
11/18/2002 12:59

Page 56

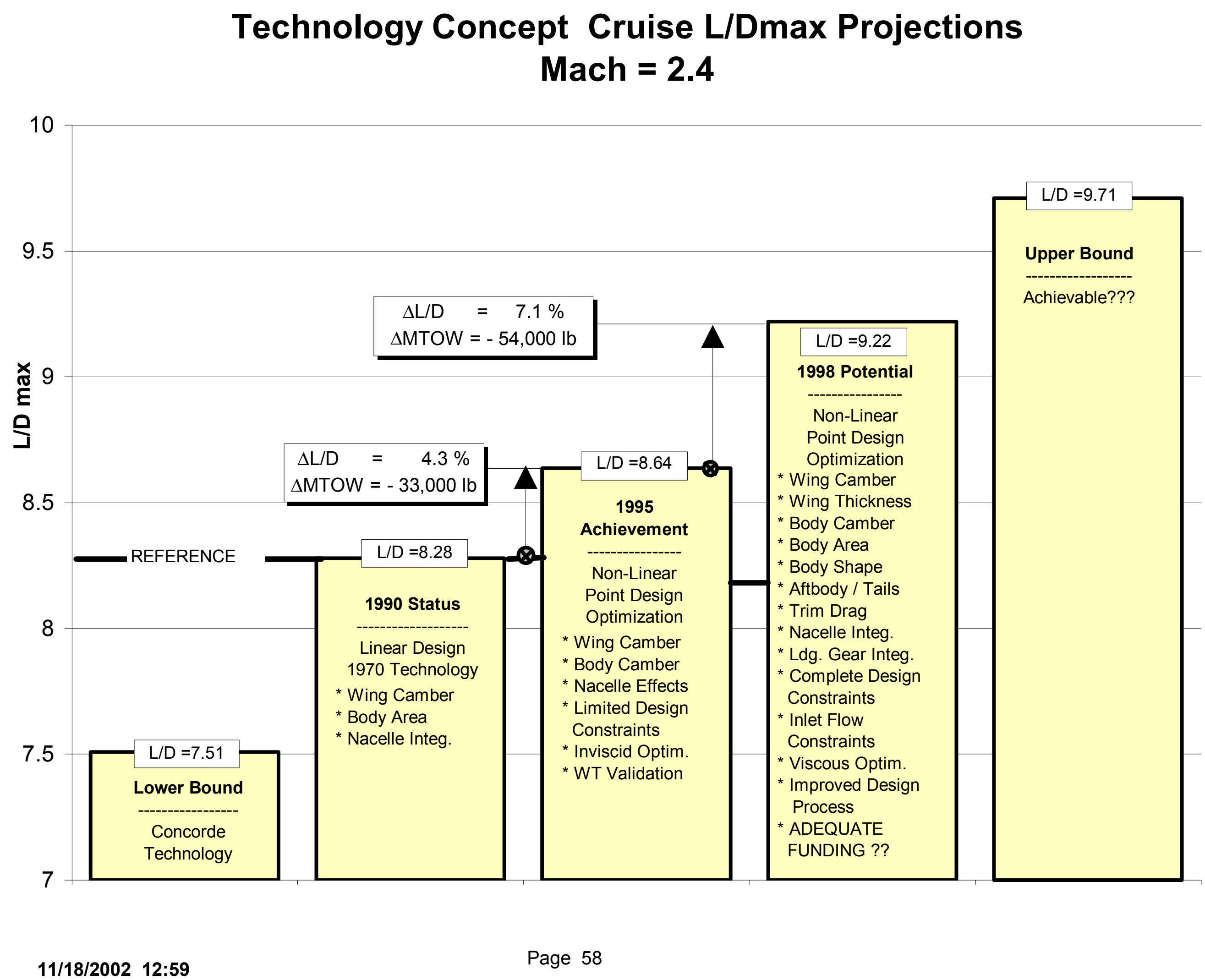
This demonstrates performance improvements achieved to date using nonlinear design optimization. The NASA Ames nonlinear design of the Ref H geometry , the Ames -704 design achieved a drag reduction at cruise of 5.5 drag counts (ΔCD = -0.00055) for the wing / body / nacelle configuration.

The design variables included wing camber and twist, body camber, and some wing inboard leading edge thickness increases.

The performance is applicable to the example HSCT configuration which is similar to the Ref H configuration including the low drag nacelle installation. The increase on L/Dmax is 4.3%.



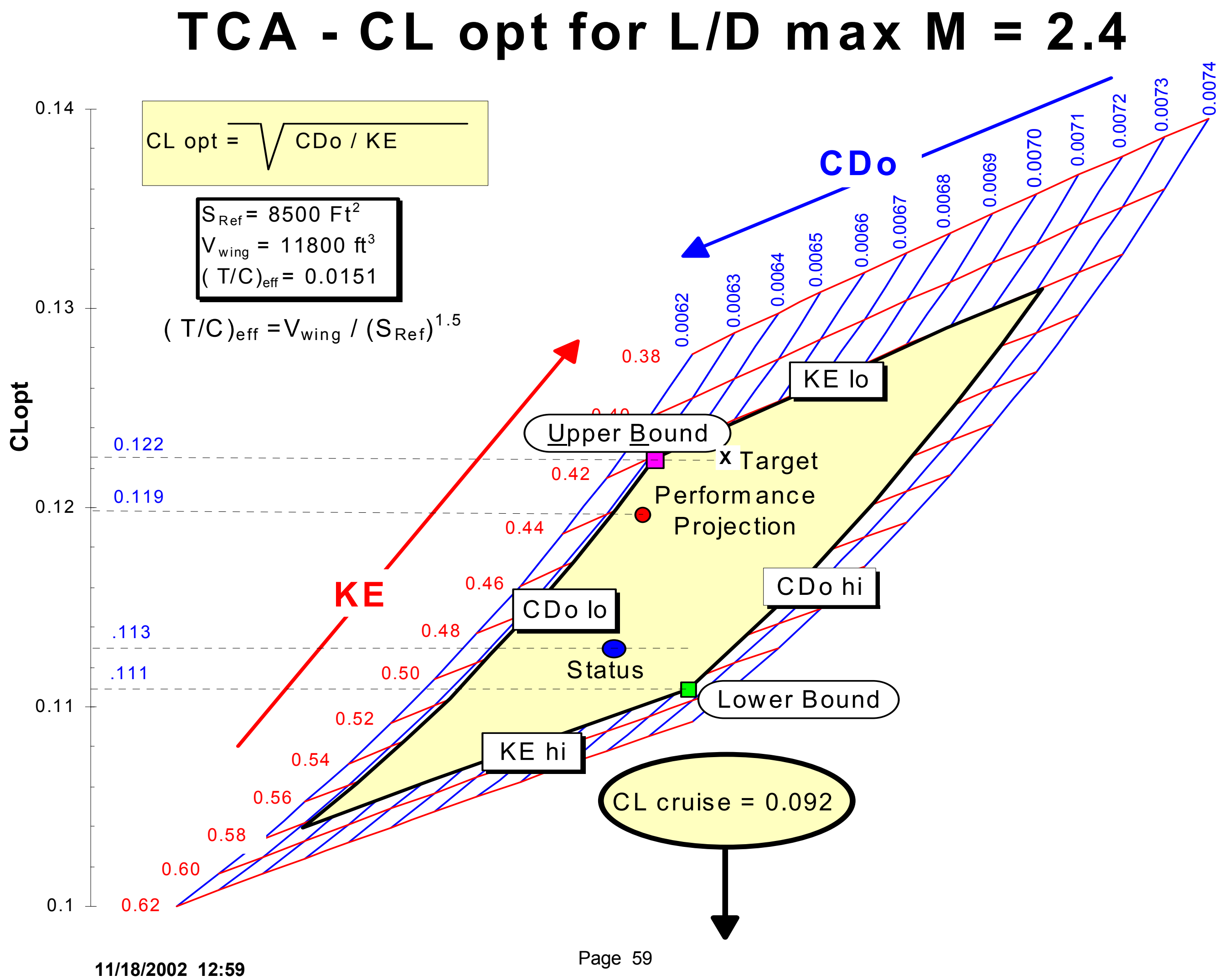
Achieving the 4.3% increase in L/D max, as demonstrated on the Ref H configuration, would result in approximately a 33,000 lb reduction in max takeoff weight for the resized airplane.



The projected further improvements in L/D max will result further developments and enhancements in the emerging non-linear aerodynamic design optimization technology together with improvements in detailed design. Examples of anticipated improvements in the detailed design processes include:

- Nacelle / diverter design integration
- Landing gear design integration
- Wing / body junction design
- Viscous and excrescence drag reduction
- Multi-disciplinary design changes

TCA - CL opt for L/D max M = 2.4



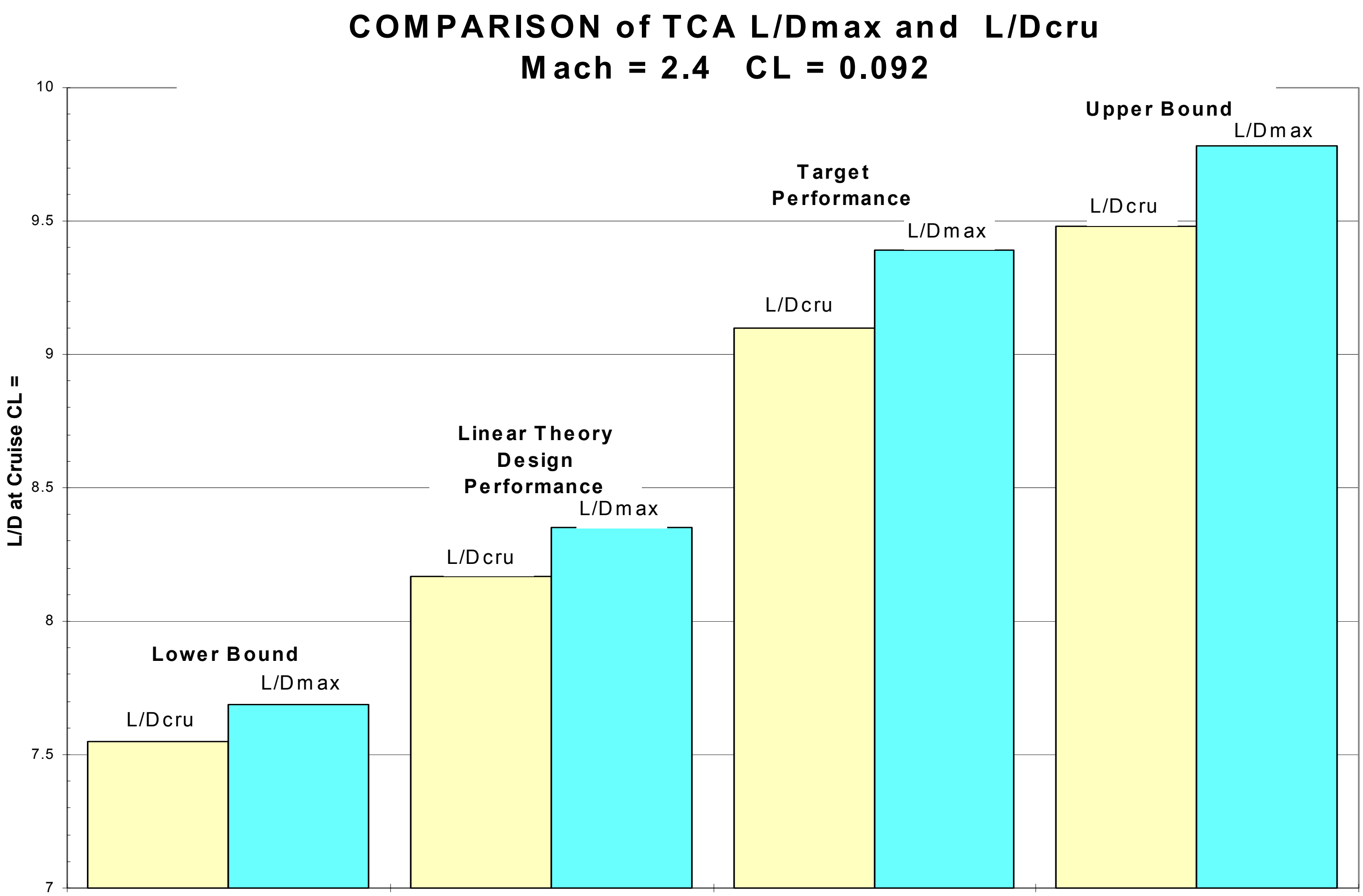
The CL for L/Dmax can be calculated from the zero lift drag coefficient, CDo, and the drag due to lift factor, KE as:

$$CLopt = \sqrt{\frac{CDo}{KE}}$$

Reductions in CDo Reduce the optimum lift coefficient. Reductions in KE increase the optimum CL.

The current TCA configuration has a rather large wing area because of fuel volume requirements and takeoff noise constraints. Consequently, the cruise CL = 0.092 is substantially lower than the optimum design lift coefficient, CL = 0.120.

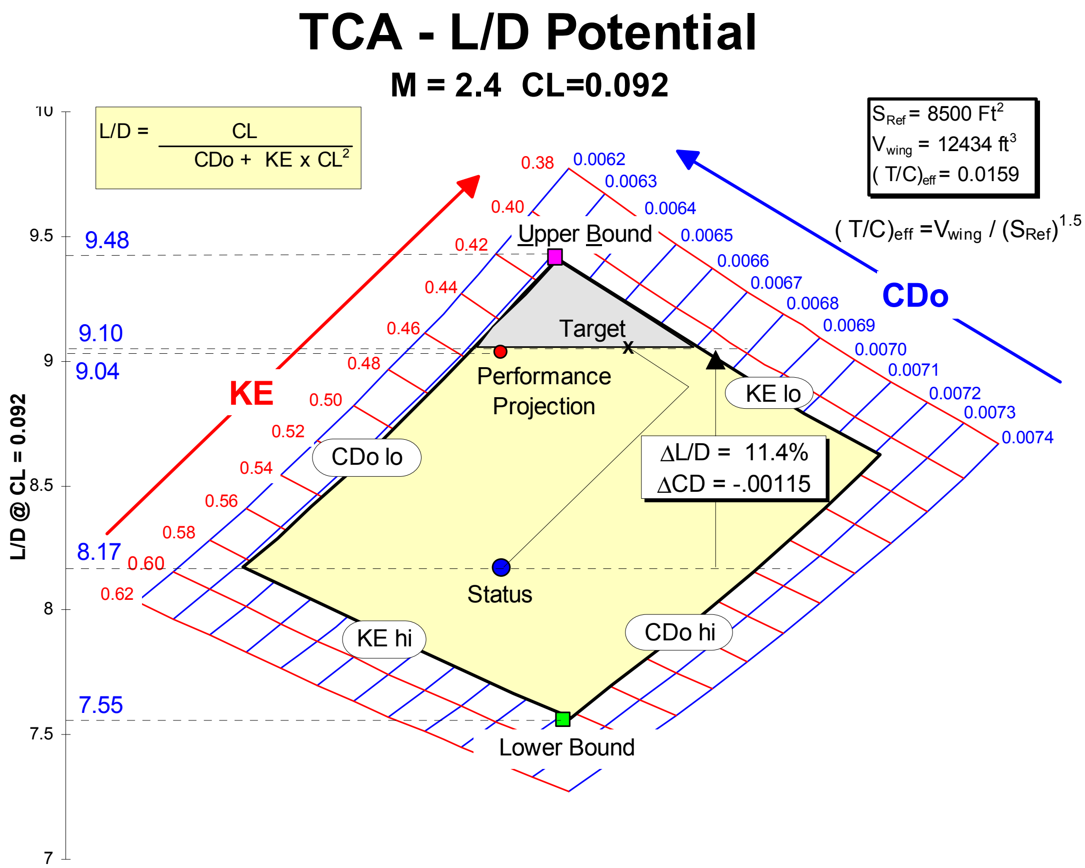
The performance projections should, therefore, be evaluated and tracked for the design CL.



11/18/2002 12:59

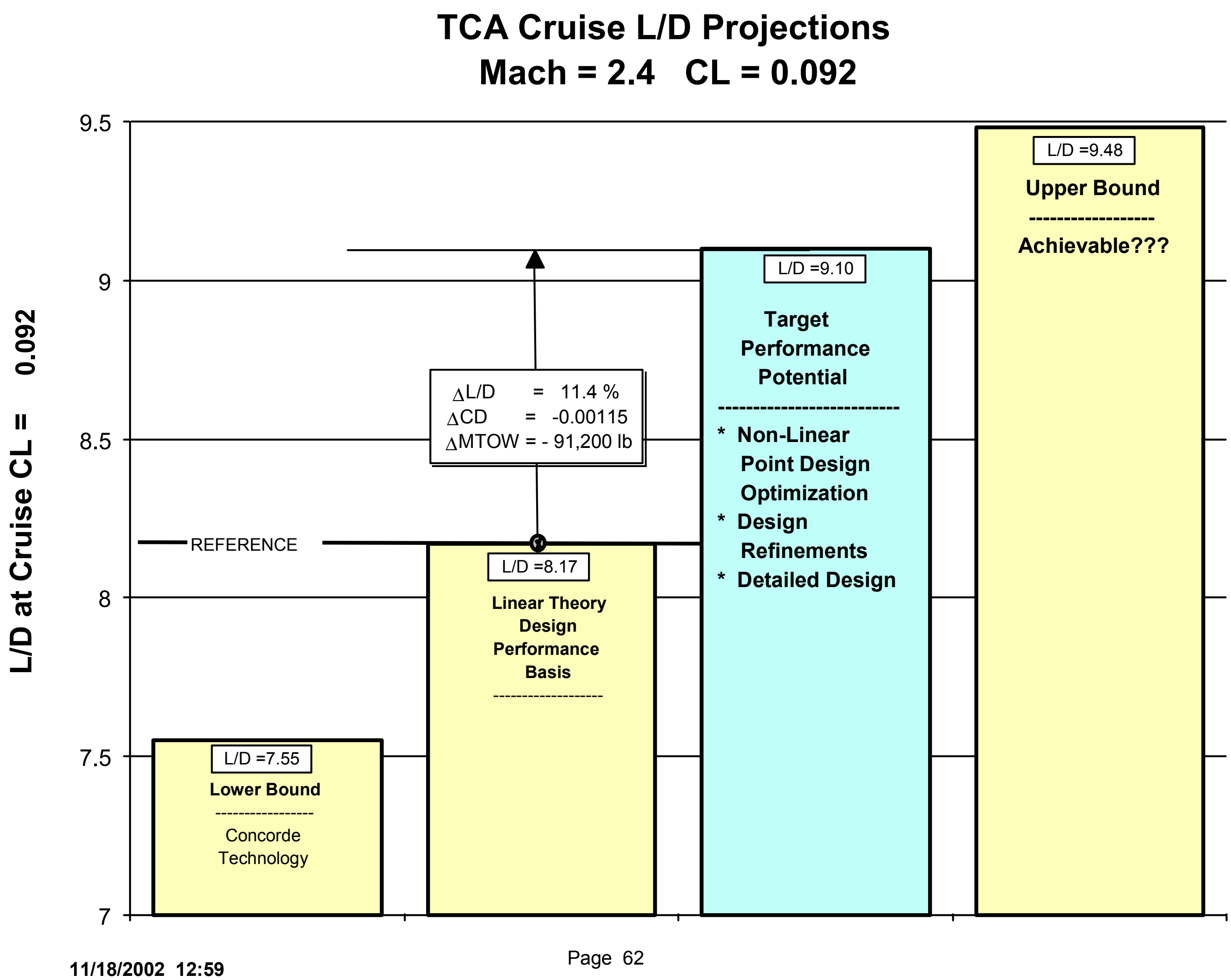
Page 60

The cruise and maximum values of L/D for the TCA are shown in this figure. The “target” cruise L/D of 9.1 is below the “target” maximum L/D of 9.35 performance limitations associated with the larger wing area.



11/18/2002 12:59 Page 61

This figures shows the aerodynamic design space for a cruise CL of 0.092.
Relative to the linear design baseline performance level, nonlinear design optimization plus detailed design improvements using the nonlinear methods is projected to increase the cruise L/Dmax by 11.4 %. This corresponds to a drag reduction of 11.5 counts, ($\Delta CD = -0.00115$).



The impact of the projected improvements in cruise L/D as shown in this figure are very significant. The net benefit to the mission sized configuration is a reduction in the maximum takeoff weight of 91,200 lb.

Being able accurately and consistently predict and to achieve these benefits will have a major effect on developing a viable HSCT.

L/D PROJECTION PROCESS FEATURES

- **DEPENDENT ON OVERALL CONFIGURATION FEATURES**
- **NOT DEPENDENT ON INITIAL OR BASELINE DESIGN STATUS PERFORMANCE**
- **CONSISTENT PROCESS**
- **BASED ON FUNDAMENTAL AERODYNAMIC PRINCIPLES - AS WE KNOW THEM TODAY**
- **EASY TO APPLY THE PROCESS**
- **ADAPTABLE - GREATER INSIGHT OR KNOWLEDGE**

11/18/2002 12:59 Page 63

The features of the process presented in this paper for predicting potential performance improvements through the application of nonlinear design optimization and detailed design integration include:

- The projections are dependent on the overall features of the configurations as well as the relative component sizes such as the area of the wing or size of the nacelles.
- The projections are not dependent on the status performance of the baseline design.
- The process is consistent and robust in the sense that the projections are not dependent on the insight or experience of any individual. This process should , therefore, be useful in guiding correct early configuration decisions.
- The prediction process is based on fundamental aerodynamic principles as we know them today.
- The prediction can be readily adapted to include modifications that are identified from greater insight or knowledge into the achievable lower limits or the various drag elements.

FURTHER DEVELOPMENTS

- **GET OTHER ITD MEMBERS TO AGREE WITH THIS PROCESS**
- **REPLACE CDw LOWER LIMIT CRITERIA BY NEW FAR FIELD VOLUME WAVE DRAG OPTIMIZATION PREDICTIONS**
- **REPLACE KE LOWER LIMIT CRITERIA BY NEW FAR FIELD LIFT WAVE DRAG PLUS INDUCED DRAG OPTIMIZATION PREDICTIONS**
- **EXTEND CONVENTIONAL TOPS DOWN L/D max PREDICTION TO HSCT TYPE CONFIGURATIONS**
- **DEVELOP LOW SUPERSONIC MINIMUM KE AND CDo CRITERIA**

11/18/2002 12:59

Page 64

We are currently working with the other members of the HSR Configuration Aerodynamics Integrated Technology Development Team members to get their concurrence with this projection process.
As previously mentioned, the process is adaptable to further enhancements. We have recently developed a new and unique method to use far field linear theory to calculate minimum wing / body volume wave drag, and minimum lift wave drag plus induced drag. These predictions will be incorporated in the projection process.
The aerodynamic performance at subsonic cruise Mach number has a significant on the overall fuel consumption. We will adapt the subsonic “Tops Down” L/D max prediction method to the HSCT configurations with optimized flap deflections.
The aerodynamic performance at low supersonic speeds may have a significant effect of the required engine size. Hence, this tends to be another critical design region . We will extend the supersonic two dimensional design space approach to establish target L/D levels achievable by off-design flap optimization.