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AN ANALYSIS OF PERFORMANCE ESTIMATION METHODS FOR AIRCRAFT

by Clinton E. Brown and Chuan Fang Chen

Prepared by
HYDRONAUTICS, INC.
Laurel, Md.
for Langley Research Center

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SUMMARY

A study has been made of the various measurements and analytical extrapolations which affect the prediction of full scale flight performance from wind tunnel tests of geometrically scaled down models. The effects considered include wind tunnel measurements and calibrations, measurements of viscous shear forces at widely varying Reynold's numbers, interference forces between engine installations and airframe, and various sources of drag not usually found in model testing. The general result of the analysis is that the rms accuracy of prediction of drag at full scale can be better than 3 percent at design cruise conditions when currently available techniques for testing are used. These estimates are for both the subsonic and supersonic cruising design conditions with the assumption that predictions made for full scale include all factors which are known to introduce drag; omission or neglect of any such factors can lead to larger and usually unconservative errors. In certain flight conditions particularly at very high subsonic speeds and in any case in which large areas of separated flow exist the errors may easily exceed those stated above. Preliminary study of methods of in-flight thrust prediction and measurement lead to the conclusion that flight test drag results at design conditions should be capable of ±5 percent accuracy. Under such conditions flight measurements and wind tunnel extrapolations should show agreement to within 5 percent. More correctly stated it is anticipated that a 68 percent probability exists that properly executed wind tunnel drag predictions and calculated drag values from well instrumented flight tests should agree within 5 percent.

INTRODUCTION

The estimation of full-scale aircraft performance from data obtained on both engines and airframes in ground laboratories has been and continues to be a major problem for design engineers and research scientists. As aircraft have become more refined to meet requirements of high-speed flight, the problems of engine and airframe interaction, friction estimation, and others even more subtle have assumed great importance, partially because of the economic penalty associated with faulty performance estimation and partly because the speeds, Mach numbers, Reynolds numbers, temperatures and performance demands are higher than ever before. The meticulous care which goes into today's highly sophisticated wind tunnel model testing can be nullified if accuracy cannot be obtained in the complex computation involved in extrapolation from models at test conditions to real aircraft at true flight conditions. It is therefore of importance to review the process, periodically introducing new information obtained from the development of theory as well as that obtained experimentally.

Methods of arriving at a predicted performance have been evolving rapidly in the last few years and have reached a high state of sophistication with the advent of the supersonic transport project. It is important to understand that the care and effort going into current national programs provide a much better base for performance estimation than has heretofore been possible and it is particularly pertinent to note that such predictions concurred in by teams from government and private industry can be significantly more reliable than performance estimates made in the early stages of aircraft procurement when the usual contractor optimism is often not warranted because of the meager data available at that stage of development. Other important considerations are (1) the fact that "design" performance can be predicted more precisely than off design, (2) that in assessing level of confidence in predictions a check should be made on the purpose of the wind tunnel tests to acertain that the maximum levels of precision have indeed been used, and (3) that often the lift-drag ratio or other performance parameter is not defined in that certain drag components may be charged to either the airframe or the powerplant.

The present method of arriving at a predicted performance starts with wind tunnel model tests. In the early stages there are frequently basic though small geometric differences between the model and the final airplane. Often the geometry of the aircraft is not "frozen" at model test time, usually details of engine installation and operationally required appurtenances are not modeled, and modification of the model must sometimes be made to accommodate the supporting stings or struts. These geometrical differences require corrections to be calculated on some rational basis (theory or past experiment) but some residual error will always exist unless tests are conducted using models of the production airplane incorporating all of the changes made during development. In order to assure a complete run of turbulent flow over the model surface, the normal partial coverage of laminar flow must be induced to become turbulent so that more accurate estimates can be made of the level of viscous drag. This transition-fixing is usually accomplished through the use of sand grains or other particles placed near the forward edges of all surfaces; under certain conditions the additional form drag associated with the particles must be subtracted from the model drag data. The wind tunnel itself can be considered an instrument having a certain error which is, of course, variable from one setting to another and from one facility to another. The sources of error in the wind tunnel come from (1) force balance errors, (2) wind stream angle errors and spatial deviations of the flow angles, and (3) Mach number or speed calibration errors, and errors introduced in correcting for the presence of tunnel walls, stings, and supports.

Once a set of wind tunnel data corrected for all factors is at hand a considerable number of computations must still be made to arrive at the final trimmed lift-drag polar for the full-scale airplane.

The basic assumption made is that the turbulent friction drag and associated form drag are the only drag components affected by the change in Reynolds numbers from model conditions to full-scale flight. Thus, the major correction is the difference in the turbulent friction drag coefficients at model and flight conditions multiplied by the appropriate surface areas and the dynamic pressure. A second large correction usually occurs in adding on the internal flow drag contributions of the engine installation. Errors from this source can be large since wind tunnel modeling is difficult and often impractical and because interference drags are possible when jets exhaust in the vicinity of blunt base areas. One human error, which can be large, may arise in defining airframe drag, for example the inlet spillage drag, boundary layer air bleed drag, by-pass drag, and boat-tail drag is assigned to the airframe by the NASA's Langley Research Center but other groups elect to assign some of these drag components to the engine thereby changing the estimated airframe lift-drag ratio. Unless these definitions are clearly understood by all concerned disagreements in predicted flight characteristics will arise. Errors of this type, while important, fall outside the realm of science or statistics! Additional secondary corrections are required for bringing the airplane to proper trim, to correct for model-prototype geometry alterations, to correct for jet plume interference effects, particularly at Mach numbers near one and to account for airplane air leaks and unplanned surface irregularities at seams, doors, flaps, and control surfaces.

The purpose of the investigation reported herein is to assess the above described process by carefully considering each of the many factors which enter the computations and attempting to determine the magnitude of possible errors. Since the errors will be related to the particular airplane as well as to a particular wind tunnel or tunnels, it is not possible to fix precisely on the uncertainties, however, an attempt has been made to generalize as much as possible. Conditions giving rise to large uncertainties are also discussed and suggestions for their control are made.

The final check of the predicted performance of an airplane is the flight test; however, the measurements made in flight are subject to error and the final flight results involve an imperfect extrapolation process, hence a brief investigation was conducted concerning the uncertainties which exist in the flight prediction methods.

Finally comments are given as to the overall agreement which should exist between wind tunnel predictions and flight results when each process is carried out with proper consideration for all known measurement inadequacies. It cannot be overemphasized that scientific methods applied in this work are meaningless without complete consideration of all drag factors; failure to make required corrections or omission of drag producing factors can introduce large bias errors which may be much greater than the "uncertainty" in the corrections.

In carrying out this investigation the authors visited the Ames Research Center, Edwards Flight Research Center and the Langley Research Center. The authors are greatly indebted to the management and staffs of these facilities for their cooperation and invaluable assistance in describing the merits as well as the deficiencies of the many test procedures involved in the process of aircraft performance prediction.

FACTORS AFFECTING THE ACCURACY OF THE FINAL WIND TUNNEL LIFT-DRAG POLARS

Force Balance

Wind tunnels for testing aircraft models at transonic and supersonic speeds generally use electrical strain-gage balances mounted inside the model and supported from the rear on a slender sting. The angle of attack and yaw of the model are varied by suitably rotating the sting. Force and torque measurements of interest in performance estimation are most often measured in components of chord-wise force, normal force, and moment about some defined point on the balance system. The selection of a suitable force balance is governed by the maximum loads expected in the test. Data obtained with a balance which is operated to only fraction of its design load may introduce significant uncertainties in the final results. Unfortunately, statistical data on the accuracy of wind tunnel balances is rarely obtained; instead calibrations runs and re-runs are made and maximum observed errors are usually stated. Calibration of the force balances installed in the wind tunnel are usually conducted to check "bench" calibrations but are seldom capable of the same precision as the bench tests where maximum errors less than 1/4 percent of full range are quoted. Balances which provide for three components of force and three of moment often lose precision when combined loads are present. In good practice however, the performance tests should be capable of measuring chord force and normal force to within 1/4 percent of design load. For programs in which performance data is secondary or time and hence money is limiting it is often necessary to accept a somewhat increased balance error. The important task, however, is to properly assess the precision of a given set of measurements so that misconceptions are avoided.

Wind Stream Uniformity

The test region of a wind tunnel is never exactly uniform with respect to flow inclination and velocity. Obviously, the quality (uniformity) of the flow field varies from one wind tunnel to another and may vary from one operation condition to another, as when Mach number is changed or when the pressure level is altered to provide for Reynolds number variation. In a given facility the errors in stream alignment over the test region may vary from point to point by as much as 0.2 degrees, in a typical case; but the integrating effect of the model (which is difficult to evaluate) tends to reduce these errors considerably. It should be noted that such localized alignment errors may tend to affect pitching moment results to a greater extent than lift and drag values. A common technique to reduce the alignment and air flow errors is to take data with model inverted. This technique should be very effective when the model is rotated so that it does not move to another location in the test space. Averaging the forces in the original and the inverted positions thus reduces the airflow errors considerably and can also provide an indication of the magnitude of the average local flow angle error. When allowances are included for model deflections due to weight and aerodynamic loads, it is apparently possible to set or measure the geometrical angles to a precision ±.02 degrees. Hence, the major errors in alignment arise from the air stream variations.

Variations in Mach number in the test space occur in all wind tunnels and are generally quoted to be not more than ±.006 in transonic tunnels and somewhat larger at supersonic speeds, typical values being ±.015. Naturally some facilities may be better, some worse.

Wind Tunnel Interference Between Model, Wall and Sting

At high subsonic and transonic speeds, wall interference is usually taken to be negligible as a result of the effects of the slots or porous walls used in such tunnels. Calibrations made using different ratios of model frontal area to tunnel passage area (blockage ratio) have shown only small effects to blockage ratios as high as 1.2 percent (see Reference 1). However, it is possible to find drag variations greater than would be expected in the data of Reference 1 which appear to be associated with angle of attack. Theoretical analyses (Reference 2) would seem to bear out the result that only minor corrections are involved since for a typical case of a large wing in a tunnel (span equal to .7 the tunnel width) the mean downwash correction at $C_L = .6$ was about .05 degrees. In a closed tunnel the upwash correction would have been roughly .2 degrees thus indicating at worst an uncertainty of the same magnitude. In properly designed tunnels of polygonal cross-section having slots at the corners or in tunnels having continuous porosity it appears that the assumption of zero correction is justified. It is essential to recognize, however, that wind tunnels having poorly designed porous sections have been built and care must be taken to insure the favorable situation discussed above.

Wall corrections at supersonic speeds can be justifiably omitted as long as the wall reflection of the leading shock wave or the tunnel normal shock wave does not intersect the model or support system near the tail. Data taken at supersonic speeds where such interactions occur are known to produce sizable errors and should not be used (see Reference 3, page 4).

At transonic speeds tests have shown that some corrections are needed to properly account for sting effects on the aircraft model afterbodies. In Reference 4 afterbody drag coefficient alterations with various stings and with jets were found to be as much as .02 based on the fuselage frontal area. In cases where the normal fuselage closure is altered to provide space for the support sting the correction from the wind tunnel condition to the proper geometry is difficult to make and it is considered that errors as large or possibly larger than those measured in Reference 4 might be incurred. Basing the drag increments (±.01)on wing area for a typical subsonic transport leads to error possibilities of ±.0010 from this source. This error may be greatly reduced if a separate test is conducted using dummy stings and alternate supports. When the sting is inserted in a jet exhaust hole the problem becomes one involving engine-airframe integration which will be discussed in a subsequent section. Sting interference at supersonic speeds can be completely eliminated by proper techniques in the selection of sting geometry, see Reference 5.

Drag Due to Grit Particles Used in Fixing Transition

The fixing of transition to turbulent flow is one of the more difficult problems facing the wind-tunnel operator. Great care is needed to properly size, locate and distribute grit particles in such a way that turbulent flow is obtained over the model under all test conditions. This is necessary to allow an accurate estimate of the change in drag due to friction from test to flight Reynolds numbers. If unnoticed laminar flow occurs on a portion of the test model the actual model friction drag will be smaller than the assumed turbulent drag. Hence, the remaining drag which is assumed invariant with Reynolds number shift will be underestimated and the performance predicted for full scale conditions of flight will be too high. Braslow, Hicks and Harris have given an excellent summary of the problem in Reference 6; it is demonstrated that up through the transonic range of Mach numbers, transition may be fixed without measurable grit drag. At higher Mach numbers, beyond 2, it is usually necessary to correct for grit drag and various techniques have been devised and tested. Recent unpublished results at the Langley Research Center show the possibility of reducing the uncertainty in the grit drag to the level of plus or minus one count at $M = 2.7$. For this technique the boundary layer transition point for a clean model is found from sublimation studies; calculation of the incremental drag which would occur if transition were at the grit then makes possible the estimation of grit drag. It is considered sufficient to make the sublimation studies at the design lift coefficient.

Internal Flow Measurement Accuracy

Aircraft having externally mounted nacelles are generally less susceptible to drag errors than are nested engine designs. Nevertheless, sizable uncertainties can arise if care is not taken in modeling the nacelle and its air flow. It is common practice to compute engine thrust using a ram-drag term based on stream conditions ahead of the aircraft. The drag associated with the inlet called spillage drag is essentially the pressure drag acting on the stream tube entering the inlet up to the inlet entrance (additive drag) plus the additional drag on the engine nacelle caused by varying the mass flow through the engine from some particular reference setting. In overall testing for performance, the airplane total drag must include the spillage drag and it is important to correctly model the air flow and to measure the overall effect of variations in engine air flow.

For subsonic cruise aircraft the inlet should be designed to produce very small spillage drag. Note that the spillage drag in inviscid subsonic flow should be closely zero by the D'Alembert paradox, however when poorly selected cowl shapes are used, especially at low entering mass flow ratios, lip separation occurs and drag is produced. This occurrence is inevitable if sharp cowl leading edges are used as in supersonic inlets, hence it is essential that tests be conducted to account for the variation of spillage drag with mass flow ratio. At cruise (design) conditions for both the subsonic transport and the supersonic transport the uncertainties associated with spillage drag will be essentially negligible, however in off design conditions the spillage drag can become quite large and in these cases Reynolds number variations and measurement accuracy can introduce sizable uncertainties. For transonic speeds there are three distinct effects to be considered; first, is the interference between the engine air stream-tube and the airframe, second is the local effect of the jet on the nacelle afterbody drag and third is the drag variation with nacelle position, incidence, or cant angle. Patterson (Reference 3, page 259) has shown that the latter effects can produce changes in drag coefficient of ±.0003 for typical small transports at Mach numbers near .75. The second effect illustrated by Cahn's work (Reference 4) with jets indicated changes in C_D from jet-on to jet-off of as much as ±.01 based on nacelle diameter. Correcting to drag coefficient based on wing area would provide possible uncertainties of ±.0001 to ±.0003 due to jet effects on a transport like the DC-9, the variation depending on the afterbody bluntness; in this case the uncertainties arise from the difficulty in being sure that model test results such as Cahn's are not seriously affected when applied to very high Reynold's number conditions of flight. The first mentioned engine air stream-tube effects include the spillage drag plus any interference drag between the stream tube and airframe such as might occur from changes in overall displacement area distribution (area rule); this effect would be expected to be most important in the transonic speed range where shock waves may be standing on wings and fuselage. For nested engine installations with inlets close by the fuselage, the first two effects mentioned above may merge and the difficulties are accentuated when the exhaust jets interact with the fuselage and tail. Another problem to be faced is that of correcting for oversized boundary layer scoops necessary at the low test Reynolds numbers. When such corrections are made the remaining uncertainty is usually negligible. Rünckel (Reference 3, page 229) has measured a base or tail-section wave drag roughly 25 percent of the total drag for a configuration like the F-111 at a Mach number of 1.2. Clearly interaction between jet and aft end of such a design will be difficult to estimate over the entire Mach number and engine mass flow range unless specific test results are available. Even when such tests are available, Reynold's number scaling reduces the precision of estimation.

At supersonic speeds, the inlets may have very small spillage drag especially at the design conditions and little uncertainty in the measured values exists. However, at off-design conditions there may be large amounts of spillage drag as well as drag due to interaction of the engine air stream tube and the wings and fuselage. Jet plume effects may be present when the fuselage and tail extend into the pressure field of the jet plume. Estimates have been made of the magnitude of interaction drag of simulated nacelle wakes for a typical case in which the nacelle exits are forward of the fuselage trailing edge. These computations are given in Appendix 1 and indicate possible drag effects of ±.0002 at M = 3, and ±.0003 at M = 1.5. In this case a drag exists on the model which would be absent in the presence of a fully expanded exhaust jet. At Mach numbers lower than cruise, a jet plume interaction with the tail would be expected and would easily be of the same magnitude and sign as the drag increment computed for the collapsing wake. Certainly at the design conditions these effects can be corrected with insignificant residual error.

SUMMATION OF EFFECTS

In estimating errors involved in wind tunnel results, there are two distinct types of uncertainties: the first involves the random errors which are associated with measurements on the given model and which according to statistical methods could be reduced by repeat runs and smoothing techniques; the second type of uncertainty involves the wind tunnel calibration itself and the resulting errors would generally not be reduced by repeating runs or smoothing. Improved accuracy (if wanted) for these latter errors can only be obtained by averaging properly with data of other equally precise wind tunnels or by increasing the quality of the wind tunnel itself. Calculative corrections to the data are also of the latter type since repetition would only involve a repeated computation procedure. In summing the errors for this report however, both types will be treated as one time measurements since repetition of measurements often would not make significant improvement in the total uncertainties and it is usually true that smooth curves obtained in tests involve zero shifts which may be slowly varying in time or varying as a function of the load history. In such cases much time is required to obtain repeat runs and runs with various load histories and in view of the small increase in accuracy so obtained it is seldom worthwhile.

The variables which can be considered to possess independent random errors are the normal and chord force gage readings F_N, and F_C; the mean angle of attack, α, which is the angle between the force balance axis and the mean wind vector; the dynamic pressure, q; the grit drag correction, D_G, the internal flow drag corrections, D_{INT}, and the wind tunnel sting and wall drag and lift corrections, D_{WI}. There appears to be no reason why errors in these variables should not be distributed in a Gaussian way and such normal distributions are assumed in this report. The drag and lift coefficients are usually obtained by the summations:

$$C_D = \frac{F_N \sin \alpha}{qS} + \frac{F_C \cos \alpha}{qS} \qquad [1]$$

$$C_L = \frac{F_N \cos \alpha}{qS} - \frac{F_C \sin \alpha}{qS} \qquad [2]$$

where S is the wing area.

Figure 1 shows the geometric arrangement. The errors which occur in the independent variables listed above produce errors in C_D and C_L which combine to give the final wind tunnel drag polar a band of uncertainty of definite width corresponding to a related probability that the true curve would lie within the band. Since the data available usually provides "maximum estimated errors", they may be considered to be errors which are not exceeded some large percentage of the time and if we say 95.4 percent of the time this would imply that the root-mean-square deviation (standard deviation) of the data would be one-half of "maximum". Note that the standard deviation, often labeled σ, represents the half width of a band bracketing 68.3 percent of the data points scattering around the mean value of a measured quantity. (Reference 7 contains a useful discussion of the statistical treatment of data). This above assumption will be adopted for the purposes of this investigation.

The root-sum-square of the errors in a drag polar must be obtained taking into account the correlation between C_D and C_L since these quantities of interest are influenced by the same errors in the independent variables. Thus the error contribution in C_D due to a single variable -- say Mach number, M, -- may be written

$$\Delta C_D = \left[\left(\frac{\partial C_D}{\partial M} \right) + \left(\frac{\partial C_D}{\partial q} \right) \frac{\partial q}{\partial M} + \left(\frac{\partial C_D}{\partial C_L} \right) \left(\frac{\partial C_L}{\partial q} \right) \frac{\partial q}{\partial M} \right] \Delta M \qquad [3]$$

The first term in the bracket represents the variation in drag coefficient with Mach number and must be obtained from tests in which Mach number is varied. The second term represents total effect of uncertainties in dynamic pressure since it has been assumed that the temperature and pressure can be measured relatively exactly. The final term is the effect of the Mach number uncertainty on the lift coefficient which would result in a given drag value being plotted at an incorrect lift value. Value of $\partial C_D/\partial C_L$ must be obtained from the experimental drag polar. Since,

$$q = \frac{\gamma p_s M^2}{2} \, , \; \frac{\partial q}{\partial M} = \frac{2q}{M} \tag{4}$$

and from Equations [1] and [2],

$$\left(\frac{\partial C_D}{\partial q}\right) = - \left(\frac{C_D}{q}\right) \tag{5}$$

and

$$\left(\frac{\partial C_L}{\partial q}\right) = - \frac{C_L}{q} \tag{6}$$

Equation [3] becomes

$$\Delta C_D = \left[\left(\frac{\partial C_D}{\partial M}\right) - \frac{2C_D}{M} - \left(\frac{\partial C_D}{\partial C_L}\right)\left(\frac{2C_L}{M}\right)\right] \Delta M \tag{7}$$

In a similar manner, the drag error contribution due to an error $\Delta\alpha$ is

$$\Delta C_D = \left[\frac{\partial C_D}{\partial \alpha} + \frac{\partial C_D}{\partial C_L} \; \frac{\partial C_L}{\partial \alpha}\right] \Delta\alpha$$

and by differentiating Equations [1] and [2] we obtain

$$\Delta C_D = \left[C_L - \left(\frac{\partial C_D}{\partial C_L} \right) C_D \right] \Delta\alpha \qquad [8]$$

The error in C_D due to a balance error ΔF_N obtained from Equations [1] and [2] is

$$\Delta C_D = \left[\frac{\sin\alpha}{qS} + \left(\frac{\partial C_D}{\partial C_L} \right) \frac{\cos\alpha}{qS} \right] \Delta F_N \qquad [9]$$

and for errors due to an error ΔF_c

$$\Delta C_D = \left[\frac{\cos\alpha}{qS} - \left(\frac{\partial C_D}{\partial C_L} \right) \frac{\sin\alpha}{qS} \right] \Delta F_c \qquad [10]$$

Combining all the independent errors we obtain for the overall standard deviation in drag coefficient

$$\sigma_{C_D} = \Bigg\{ \left[\left(\frac{\partial C_D}{\partial M} \right) - \frac{2C_D}{M} - \left(\frac{\partial C_D}{\partial C_L} \right) \left(\frac{2C_L}{M} \right) \right]^2 \sigma_M^2 + \left[C_L - \left(\frac{\partial C_D}{\partial C_L} \right) C_D \right]^2 \sigma_\alpha^2$$

$$+ \left[\frac{\sin\alpha}{qS} + \left(\frac{\partial C_D}{\partial C_L} \right) \frac{\cos\alpha}{qS} \right]^2 \sigma_{F_N}^2 + \left[\frac{\cos\alpha}{qS} - \frac{\partial C_D}{\partial C_L} \frac{\sin\alpha}{qS} \right]^2 \sigma_{F_C}^2$$

$$+ \left[\sigma_{C_{D_G}} \right]^2 + \left[\sigma_{INT} \right]^2 + \left[\sigma_{WI} \right]^2 \Bigg\}^{\frac{1}{2}} \qquad [11]$$

If it is now assumed that the fractional standard deviation of the force balance system is ϵ with respect to the design loads, Equation [11] may be rewritten:

$$\sigma_{C_D} = \Bigg\{ \left[\frac{\partial C_D}{\partial M} - \frac{2C_D}{M} - \left(\frac{\partial C_D}{\partial C_L}\right)\left(\frac{2C_L}{M}\right) \right]^2 \sigma_M{}^2 + \left[C_L - \frac{\partial C_D}{\partial C_L} C_D \right]^2 \sigma_\alpha{}^2$$

$$+ \left[\left(\frac{\sin\alpha}{qS} + \frac{\partial C_D}{\partial C_L} \frac{\cos\alpha}{qS} \right) \left(\epsilon F_{N_{des}} \right) \right]^2$$

$$+ \left[\left(\frac{\cos\alpha}{qS} - \frac{\partial C_D}{\partial C_L} \frac{\sin\alpha}{qS} \right) \left(\epsilon F_{c_{des}} \right) \right]^2$$

$$+ \left[\sigma_{C_{D_G}} \right]^2 + \left[\sigma_{INT} \right]^2 + \left[\sigma_{WI} \right]^2 \Bigg\}^{\frac{1}{2}} \qquad [12]$$

In the relations above the various σ values represent the standard deviations of each quantity subscripted and each must be determined by experiment or estimated by use of theory. Notice that the final drag results when plotted versus lift coefficient already have the effect of lift errors included hence the errors in C_D at any C_L define the errors in L/D. The error in L/D may be written

$$\frac{L}{D} + \Delta \frac{L}{D} = \frac{C_L}{(C_D + \Delta C_D)} = \left(\frac{L}{D}\right) \left[\frac{1}{1 + \frac{\Delta C_D}{C_D}} \right] \qquad [13]$$

For small $\frac{\Delta C_D}{C_D}$ we then obtain

$$\frac{L}{D} + \Delta \frac{L}{D} = \left(\frac{L}{D}\right) \left(1 - \frac{\Delta C_D}{C_D}\right)$$

and the standard deviation in L/D is then

$$\sigma_{L/D} = \left(\frac{C_L \sigma_{C_D}}{C_D{}^2}\right) \qquad [14]$$

Taking now two cases of interest, the large subsonic transport and the M = 3 supersonic transport, representative lift-drag polars and their associated standard deviations will be discussed. It must be remembered however that the values used for the standard deviations are variable from one facility or one model to another, hence what is about to be calculated is at best a reasonable attempt to find a "ball park" number for precision possible.

Data for the two cases is obtained from Reference 3, pages 15, 16 and 17 (Figures 12, 13 and 15). The needed values for computing the root-mean-square deviation according to Equation [12] are given in the following tables:

Subsonic Transport at M = .775	$C_L = .2$	$C_L = .5$
$\frac{\partial C_D}{\partial M}$	.09	.175
C_D	.0259	.0352
$\frac{\partial C_D}{\partial C_L}$	.01	.068
α	0	3.4°
$\left(\frac{F_{N_{des}}}{qS}\right)$	.8	
$\left(\frac{F_{c_{des}}}{qS}\right)$	.05	
ϵ	.00125	
σ_M	.003	
σ_α	.00087	
$\sigma_{C_{D_G}}$	0	
σ_{INT}	.0001	
$\sigma_{(WI)}(STING)$	.0005	

For these values the computation shows the major sources of error to be due to sting interference, angle of attack uncertainty, and balance accuracy. A standard deviation of roughly 7 counts is obtained in the cruising lift coefficient range.

It should be noted that the variation due to tunnel Mach number is extremely small for this case, however, at a somewhat higher Mach number the drag rise term $\left(\frac{\partial C_D}{\partial M}\right)$ can become much larger and results in as much as 20 counts of uncertainty. The band of shading in Figure 2 indicates the calculated region of 68.3 percent probability which for the conditions assumed represents about ± 2 percent of the drag. We would judge then that a ± 4 percent spread would incompass 95.4 percent (2σ) of the scatter. It is clear from the calculations that an improved estimate of the sting-fuselage interference drag would shrink the one sigma band of uncertainty to 1-1/2 percent.

Supersonic Transport at M = 3	$C_L = .04$	$C_L = .10$
$\frac{\partial C_D}{\partial M}$	.002	.01
C_D	.0097	.0151
$\frac{\partial C_D}{\partial C_L}$	.072	.182
α	1.6°	4.0°

$\frac{F_N}{qS}$ des.	.2
$\frac{F_c}{qS}$ des.	.03
ϵ	.00125
σ_M	.0075
σ_α	.00175
$\sigma_{C_{D_G}}$	.0001
σ_{INT}	.0001
σ_{WI}	0

When the above values are substituted in Equation [12], a standard deviation of 1.6 and 2.5 counts are obtained at C_L values of .04 and .1 respectively. The major sources of error are the assumed values of the wind stream angle and the internal drag both of which could be reduced somewhat by smoothing and use of repeat runs. Figure 3 shows the uncertainty as a shaded band of 68.3 percent probability which for the values taken in the table above represent a two sigma accuracy of ±3.4 percent. Again it must be stressed that every case must be considered by itself and that these numbers only indicate the levels which can be reached or even bettered somewhat by meticulous care in testing, by duplication of tests in several wind tunnels and use of different test techniques.

FACTORS AFFECTING THE ACCURACY OF EXTRAPOLATION OF WIND TUNNEL DRAG RESULTS TO FULL SCALE FLIGHT CONDITIONS

Friction Drag

The largest change in drag coefficient from model to flight conditions occurs in the viscous forces which are reduced as a result of the reduced shearing gradients which accompany increased boundary layer length. The friction drag may vary with Reynolds number, pressure gradients, geometric shape, and roughness of the surface, but the latter effects are usually considered as increments to a basic drag obtained from the integrated product of wetted area, flat-plate skin friction coefficients, and the dynamic pressure. The uncertainty in the basic skin friction coefficient of flat plates is difficult to estimate, however, in one such attempt Spalding and Chi (Reference 8) have statistically analyzed many theories and sets of experimental data and have presented an empirical method for estimating skin friction on plates which provides an rms error of approximately 10 percent which is slightly better than any other available theory. Note however that the rms value does not represent the precision of the mean, but according to statistical reasoning the precision (standard deviation of the mean values) should equal the rms values divided by the square root of the number of measurements. In view of the fact that the errors may be bias errors related to a given experimental facility, it would appear most realistic to use the number of facilities rather than the actual number of test points. While a detailed study of all the data is beyond the scope of this report it would appear that a number of sets of data approximating 16 were used by Spalding and Chi. Hence one would presume that the method of Spalding and Chi (as well as the theories of Sommer and Short and Van Driest, References 9 and 10) can give skin friction to within about ±2-1/2 percent in the range of Reynolds and Mach numbers of the data. The Spalding and Chi procedure, however, would appear somewhat gross in that no weighting of the experimental data was done. In reviewing the literature it is found that only one set of "high accuracy" data was given with careful estimates of the precision. These data were presented in Reference 7 and the estimated overall accuracy at high Reynolds number ($30 - 65 \times 10^6$) was ±5 percent or for our considerations an rms value of ±2-1/2 percent. Considering the case of high Reynolds number and supersonic speeds ($M = 2$ to 3) this data of Reference 11 (Matting et al) showed values which were in close agreement with the Spalding and Chi estimate, whereas the apparently good mean data of Reference 12 (Moore and Harkness) lie 5 percent above and the data of Winter el al, Reference 13 agree very well.

The situation is therefore not clear and it would be worthwhile to make a detailed analysis of the available data segregating the considerations to areas of particular interest such as data in the low supersonic range or the hypersonic range etc. Such a task is however beyond the scope of this report. Actually the data comparison presented by Peterson and Monta, Reference 14, indicates that the collected experimental data are averaged best in the high Reynolds number range by Spalding and Chi's method but at wind tunnel Reynolds number the method of Sommer and Short gives a better fit. In view of the foregoing discussion it is clear that a more thorough analysis of the friction drag precision is needed, however, in the absence of such an analysis, the authors believe a value of 3 percent would be a reasonable percentage standard deviation for either the Spalding and Chi method or the Sommer and Short method.

At subsonic speeds the Mach number corrections to incompressible data are small hence the largest uncertainty will be that of the low speed basic data. In the very low Mach number range, Spalding and Chi's analysis indicates an rms error of about 2 percent using 16 sets of data, hence a standard deviation of 1/2 percent appears to be a reasonable value for the subsonic transport over the entire Reynolds number range. It should be noted that in estimating the change in airplane drag coefficients from wind tunnel to flight Reynolds numbers it should not be assumed that the theoretical variation of friction drag with Reynolds number is known more precisely than the data measurements at any given Reynolds number since all theories have empirical constants and are tailored to fit the data. It is therefore not at all assured that if the true value of the skin friction coefficient is low at low Reynolds number it will also be low at high Reynolds number. Because of this fact, it is necessary for the error in the wind tunnel estimates to be root-sum-squared with the corresponding error estimates at full scale!

Boundary Layer - Pressure Drag Interactions

At all speeds but most importantly for the subsonic range, thickness effects produce variations of dynamic pressure over the aircraft; associated with this increased mean "q" is an increased friction drag and an associated pressure drag. For subsonic airfoils the pressure drag and increased viscous drag has been estimated by various researchers and a summary of the work is to be found in Reference 15. The magnitude of the drag relative to the flat plate level (form factor) is determined by the overvelocity as well as its distribution. Thwaites, Reference 15, gives the first order expression for fully turbulent airfoils as:

$$C_D = .0452\ R^{1/6} \left[\int_0^1 \left(\frac{U}{U_\infty} \right)^4 d(x/c) \right]^{5/6} \qquad [15]$$

Here R is the section Reynolds number, U/U_∞ is the local to stream velocity ratio, and x/c is the percent chord. In comparison with NACA test results given in Reference 16, the theoretical results show excellent agreement as can be seen from the table below:

FORM FACTORS

Airfoil	Measured	Calculated	Hoerner's Result
63006	1.23	1.16	1.12
63009	1.25	1.23	1.184
63012	1.31	1.31	1.252
64015	1.40	1.38	1.285
65015	1.42	1.39	-

The value measured for the 63006 apparently contains some additional drag due to the roughness elements used in the tests; note that all the data have been corrected by -6 counts. This correction was found in the usual way previously described wherein a drag level plateau is found when the roughness height is reduced below a certain level; at this point the roughness drag is assumed to be zero. Application of this technique to the NACA "standard" roughness produced an estimated six counts of standard roughness drag correction. It is common practice today in estimating subsonic aircraft drag to use the experimental correlations of Hoerner, Reference 17. Such a procedure is recommended in Reference 18, and various individual adaptations are in use in American aircraft companies. Hoerner[a] gives for airfoils with their maximum thicknesses at 30 and 40 percent of the chord

$$C_{D_{30}} = C_{D_f} \left[1+2\ t/c + 60\ (t/c)^4\right] \qquad [16]$$

$$C_{D_{40}} = C_{D_f} \left[1+1.5\ t/c + 120(t/c)^4\right] \qquad [17]$$

where C_{D_f} is the flat plate drag and t/c is the thickness ratio. The table shows that the analytic technique gives better agreement than Hoerner's result but since these considerations are only for the case of two-dimensional flow and since there is some question about the grit correction applied, it would seem reasonable to

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[a] Other additions of Hoerner's book revise the formuli slightly but conclusions above are not altered.

assume that errors for complete aircraft could be 15 percent of the increment above flat plate friction drag.

The form factors for fuselages are generally smaller than for wings and Hoerner gives a relation

$$C_{D_{fuselage}} = C_{D_f}\ [1 + .5\ d/\ell + 6\ (c/\ell)^4] \qquad [18]$$

where C_{D_f} is the fuselage friction drag coefficient computed using the flat plate friction factors. If we consider now a subsonic transport in which the fuselage wetted area is say 2/5 of the total wetted area and the wings and fuselage are of the order of 15 percent thickness ratio, the overall form factor could vary from 1.25 to 1.35, hence the overall calculated drag coefficient arising from friction would contain an error of ±5 percent of the flat plate friction. According to the theory however, the additional friction and form drag drops off with increased Reynolds number in the same way as the basic flat plate friction drag and hence the resulting errors introduced from this source involve the difference in friction drag at wind tunnel and flight conditions multiplied by the percentage uncertainty factor. For a subsonic transport with a wetted to wing area ratio of 5, the estimated error would be ±5 percent of an estimated 80 counts of drag giving ±4 counts of drag uncertainty due to additional friction and form drag. For the supersonic transport in the subsonic flight condition the fine forms and low wing thickness ratios should reduce the uncertainty from this quarter to less than one count except perhaps for variable sweep versions in which some intermediate value would be most proper.

For most flight conditions and wind tunnel conditions aircraft surface temperature will be close to adiabatic values, hence the heat transfer corrections of almost any of the theories can be applied and the resulting uncertainty will be small and negligible compared with the uncertainty in overall friction drag.

At supersonic speeds the variations in mean dynamic pressure are generally small because of the fine forms needed to achieve low wave drags. However, the interaction of viscous effects with pressure drag can be of importance in some cases. There are two effects which have been considered: first the displacement effect of the boundary layer creates a small and essentially negligible increase in wave drag. Second and potentially greater in magnitude is the effect of shock induced boundary layer separation at the trailing edge of wings. In this case first discussed by Ferri, Reference 19, and later by Frick, Reference 20,the boundary layer separates some distance forward of the trailing edge and over this region the pressures are generally not as negative as would be obtained in inviscid flow. This effect would produce an increasing wave drag with increased Reynolds numbers and its neglect would of course result in an optimistic estimate of drag at flight conditions. An estimate of this effect has been made by use of pressure distribution data obtained in tests conducted on wings in the Langley Research Center 4' x 4' supersonic wind tunnel, Reference 21. In these tests the pressures on the wing upper surface were seen to rise sharply near the trailing edge. By extrapolating the pressure curves smoothly to the trailing edge an "assumed" correct high Reynolds number pressure distribution was obtained. The difference between the two cases appeared to be about .0001 in drag coefficient and it is estimated that in this particular case the total uncertainty would be negligible provided the effect were taken into account in the drag budget.

At transonic speeds above the design cruise condition of subsonic aircraft, normal shocks appear on the wings and fuselage often causing the boundary layer to separate. These familiar phenomena are known to produce large increases in drag, changes in lift and lift distribution, and large changes in pitching moments. Loving has pointed out in Reference 22, that changes in the boundary layer conditions entering the shocks can result in large changes in the wind tunnel measurements and therefore, one must anticipate similar changes between wind tunnel and full scale flight conditions. Although the effects are not of significance in estimates of the subsonic cruise performance, the data of Loving should be considered when predicting the drag of aircraft passing through the transonic speed range.

Roughness and Protuberance Drag

Additional drag in full scale flight will occur from sources not easily modeled in the wind tunnel. These sources of drag are the imperfections of the surfaces due to scratches, butt joints, rivets, poorly fitted doors, etc. Horton and Tetervin, Reference 23 have made a useful study of the situation using three production military aircraft. Their results show additional drag coefficients of as much as 20 to 30 counts arising from a variety of sources. However, the most severe sources of drag were found to be associated with leading edge wing gaps and control surface gaps. On the F8U aircraft the gaps were estimated to produce 20 counts of drag while the F101 appeared to have 5. The category of next highest significance appears to be butt joints and a typical calculation would require estimating the total length and elevation of butt joints, the associated drag coefficient and effective dynamic pressure ratio. Such an estimate carried out for a large transport yielded values of about 3 counts, a number in good agreement with results of Horton and Tetervin's study. Generally the drag produced by gauges, cover plates, rivets and screws, hinges and miscellaneous projections should be less than a few counts each and hence with any reasonable method the net uncertainty from these sources should be negligible; however, it is clear from Reference 23 that the sum of the individual drags is far from negligible. Scratches and holes are a source of drag that is likely to vary with age and maintenance of the aircraft however the operational aircraft investigated in Reference 23 were estimated to incur less than 3 counts of scratch drag when a realistic method was used. Czarnecki, Reference 24, has indicated that the general surface condition of manufactured panels is sufficiently smooth to prevent drag rise due to distributed surface irregularities. However, drag produced by small waves can add up to significant levels at supersonic speeds. Currently, work is in progress to improve the methods of estimation of roughness and protuberance drag; probably the methods now in use can only be assumed to possess a precision of one part in four and this value will be taken in the absence of a more detailed investigation. Of course there can be some improvement in the estimation when the aircraft is available for inspection, however, if current standards of construction are attained the drag estimates should be adequate. The problem of air leaks is an old one and it is well established that careless handling of inlets and seals has produced leaks which reduce aircraft top speed significantly. However, for the purposes of this report drag due to an unsealed leak is a human error to be corrected but not an uncertainty in the drag estimation procedure.

To arrive at an uncertainty value for the subsonic and supersonic transport roughness and protuberance drag it would seem reasonable to take four counts as the control gap drag, three counts for butt joint drag, and two counts for rivets screws and scratches. With one part in four precision the root sum square accuracy would thus be 1.3 counts. (Obviously a "ball park" value subject to improvement through additional research).

Base Drag

For sharply defined base areas not associated with jets, base drag measurements can usually be corrected for Reynolds number changes by use of existing data (see Reference 24) with little overall error, however in practical cases base areas are most often found at the jet exhaust and the estimation of the base drag becomes involved with calculation of the engine thrust. In the case of the B-70 airplane where a base area roughly 50 percent of the jet exhaust area is present the base drag could reach a value of .0005 (five counts) at M = 3 and it is doubtful that any errors larger than one count could arise provided proper tests at model conditions have been made using high speed jets and correct bleed air flow. In the absence of such tests it is clear that errors of several counts might accrue. At lower supersonic speeds the model tests of the exit on such airplanes would be very desirable since the analytical estimation of base drag with contracted nozzles and large amounts of bleed air would be difficult. For example a typical base pressure coefficient at M = 1.3 would be roughly -.25; this value times the "dead" base area of approximately 60 square feet would produce a drag - in the absence of base bleed - of 25 counts. This value would be reduced by the sizable amounts of bleed air available but the actual drag would be influenced by the jet and bleed air flow conditions. Since the effects of Reynolds number variation on base aspiration are known to be small it is doubtful if errors greater than a few counts would be expected in extrapolating test results to full scale and on aircraft having less dead base area (nacelled aircraft) the errors should be less than one count. For a twin-jet engine-in-fuselage tested at M = 1.2 Runckel, Reference 3, has measured a tail section drag of 46 percent of the total drag whereas the tail portion contains only 39 percent of the wetted area (tests with jet simulation). If it is assumed that the drag is roughly half friction and half wave drag, the tail portion would be carrying more than 25 percent of the total drag as wave drag and form drag. Clearly this amount of wave drag cannot be accurately estimated by theory and hence the possibility exists for substantial amounts of base, form, or separation drag which would have an unknown variation with Reynolds number. Nevertheless, on an aircraft of this type the measured magnitude of tail section wave drag is certainly to be anticipated and Reynolds number changes could hardly be expected to add to or eliminate more than say 20 percent of it. Thus a rather crude guess at the uncertainty in this Mach number range would be ±5 percent of the total minimum drag. At higher supersonic speeds this would be reduced considerably and at $M = 2$ and beyond the uncertainty should be of the order of one count. It is clear that detailed tests of the engine installation with simulated jets and by-pass flows are required if good drag estimates for this complicated system are to be made.

Effect of Boundary Layers on Drag Due to Lift

The difference between the model conditions and flight conditions at a given lift coefficient is mainly a change in boundary layer conditions. That is, the boundary layer becomes relatively thinner at high Reynold's numbers and regions of leading edge separation (when existent) became smaller; as a consequence small changes in pressure distribution occur which can alter the profile drag and/or alter the load distribution with attendent changes in wake energy. The key problem here is to discover the existence of leading edge separation in the wind tunnel and to estimate the variation in drag as the Reynold's number increases to that of flight. Henderson, Reference 3 page 327 has approached the problem by presenting data for the percent of expected leading edge section force obtained on various symmetrical models. Unfortunately the expected suction is a function of span loading and camber and hence the true boundary layer conditions are only very crudely indicated by the calculations. It is certainly true for symmetrical thin-winged aircraft that as much as 50 counts of drag coefficient can be regained if in passing to full scale, the Reynolds number based on leading edge radius reaches 20,000 ($M < 0.3$). A warning must be given that in such cases estimation of the uncertainties in drag becomes very difficult and must take individual models, test conditions, and available supporting data into account. However, for the specific cases of the large subsonic transport and the supersonic transport, proper design should ensure the absence of leading edge separation bubbles at the design wind tunnel condition. Under these restricted conditions then, the extrapolation to high Reynolds number flight should not produce drag changes from this quarter.

There is however a sizeable amount of form and additional friction drag associated with lift and for two dimensional airfoils the effect may be grossly estimated by use of Equation 15 applied to upper and lower surfaces. Since the additional friction and form drag due to lift scales down with increasing Reynolds number, it is important to include this component in the estimates discussed on page 30 of this report. Note that there appears to be an inadequate amount of information on which to base these form and additional friction drag estimates.

Taking again the special cases of the subsonic and supersonic transports at their design conditions, the possible effect of Reynolds number on local lift curve slope was investigated. Using available two dimensional data in Reference 26 the effect of Reynolds number is apparently less than 2 percent and if it is assumed that such a change occurs over 50 percent of the span less than 1/2 count in drag coefficient change would occur. Considerations of the influence of the tiny changes in lift and lift distribution on the section profile drag using Equation [15] show that any profile drag changes with Reynolds number shifts will be small compared to the general uncertainty of the form drag. This conclusion is based on the probability that the lift changes are the result of trailing edge pressure changes which have the least effect on the integral of Equation [15]. It is easily shown that any reasonable alteration of the trailing edge pressures results in a negligible additional profile drag.

At supersonic speeds, the effects of separation at the trailing edges predominate, and some consideration of this problem is given in the discussion of boundary layer - pressure drag, interactions. There it is estimated that less than one count of interaction drag should normally occur.

Power Plant and Inlet Drag Factors

Nichols, Reference 27 has presented a good summary of the power plant drag and inlet drag factors and has shown that the drag associated with off design operation (reduced Mach number) can be a sizable and important item in the drag budget whether charged to the engine or the airframe. At cruise conditions for the supersonic transport, the only sizable amount of auxiliary air is that used as boundary bleed, cabin cooling, etc. and is estimated to represent about 8 percent of the air captured by the inlet. The drag of this air is probably variable and not easily estimated or measured. Nichols' estimate of the associated drag is approximately 5 percent of the airplane drag and corresponds to a drag coefficient based on the area of the entering auxiliary air stream tube of 0.8. As affected by variable cabin air - or cooling demands the drag coefficient associated with air of this sort might well vary from .6 to 1.0 and a reasonable guess would place the uncertainty at ±1 percent of the aircraft drag or 1-1/2 counts. At speeds below the cruise, the by-pass air increases until at M = 1.3 approximately 20 percent of the capture area must be handled. At a $C_D = .45$ based on stream tube frontal area the drag contribution of this air is roughly 6 percent of the airplane drag, however, after suitable tests have been made it would seem reasonable to be able to predict the drag of this air to within ±10 percent, hence the estimation should provide uncertainties of approximately ±1/2 percent or ±1 drag count. If higher efficiency exit nozzles are used for this air, the associated drag can be reduced by 90 percent and the uncertainties would become negligible. In inlet designs which ingest boundary layer air from the fuselage or the wings, proper inlet tests require larger boundary layer scoops or diverters than needed for the full scale flight conditions. It is therefore necessary to allow for this geometric change in estimating the drag variation from wind tunnel models to production aircraft. Properly done this correction should entail negligible residual error. Another candidate for error production is the drag associated with internal ducting of air taken on board; a change during production in the duct areas may easily be overlooked but may contribute to increased backpressure on the inlet system with consequent degradation in engine performance. Effects such as these can only be accounted for by constant upgrading of the performance estimates as the aircraft goes into production! At the low supersonic speeds, the inlet spillage drag should be insensitive to Reynolds number variation since except for the cowl-lip suction forces the flow is not influenced by viscosity and therefore the model test results should be directly applicable. The by-pass to the ejector should become an item properly included in the engine thrust computations and directly chargable to that system since the use of by-pass air in the ejector has a first order effect on the net thrust of the system.

On supersonic aircraft having blow-in doors on the nozzle for off design use, the drag associated with the air stream blown in or the equivalent drag of the doors and exit shroud must be obtained from tests of the engine system with a supersonic outer flow. Since the drag involved is not usually more than 10 percent of the airplane drag, variations in Reynolds number from model to flight conditions are not expected to produce significant errors; however, the interaction of the wave system originating at the blow-in doors with the airframe should be estimated and where possible modeled. It is probable that theoretical area rule computations of the drag of the jet exit system can provide adequate estimates of this drag.

Geometric Factors

Because the final version of an airplane is rarely exactly like the model which has been tested, corrections for all geometrical changes should be estimated. This process should entail no large uncertainties, however it must be carried out and must include variations in wing area, tail areas, angular positions of components, scoops, outlets, fairings, and required instrument probes. Present best practice is to carry the wind tunnel test program along on a parallel with the aircraft development. In this way final changes to be estimated are minimal.

SUMMARY OF EFFECTS OF EXTRAPOLATION TO FLIGHT CONDITIONS

The additional errors introduced in the extrapolation of wind tunnel test data to flight conditions are all independent of the basic error variables of the wind tunnel test. Hence in assessing the overall precision of the performance prediction they may be added into the root-sum-square in the normal manner. Thus we need only the estimated standard error for each independent error source. If we again consider the cases of the subsonic and supersonic transports at their cruise conditions we may take the root-sum-square of the various extrapolation errors and root-sum-square it with the total error value determined for the wind tunnel test results. Taking first the subsonic transport the following tables present the assumed and calculated results:

SUBSONIC TRANSPORT ASSUMED DATA

Test Reynolds numbers	2.8×10^6
Flight Reynolds numbers	60×10^6
$(dC_D/dC_L{}^2)_{C_{L\ opt.}}$	.046
C_{D_o} (zero twist and camber)$_{w.t.}$	.0235
C_{D_o} (zero twist and camber)$_{Flt.}$	.0155
C_D (flat plate friction)$_{w.t.}$	.0180
C_D (flat plate friction)$_{Flt.}$	.0107
Form Factor	1.23
Wetted Area/Wing Area	5.0

Using the above assumed data, the following schedule of extrapolation errors is assumed to be typical:

Error Source	Standard Deviation in Drag Coefficient
Form and Additional Friction	.00029
Friction Drag Uncertainty at W.T. Cond.	.00009
Friction Drag Uncertainty at Flt. Cond.	.00005
Engine Additive Drag (Full Scale Tests have been assumed)	.00005
Roughness and Protuberance	.00013

Thus the largest factors in the extrapolation to full scale are form factor and roughness and protuberance drag. The root sum of the squares is thus 3.4 counts which combines with the value of 7 counts at C_L = .5 (page 23) assumed for the wind tunnel tests yielding a total estimated drag uncertainty of 7.8 counts. This one sigma value is 3 percent of the 255 total drag counts and corresponds by our reasoning to a 95 percent probability that the estimate should not err by more than 6 percent. Figure 2 shows the situation. For the supersonic transport at its cruise condition corresponding to the table on page 23 the following tables are given:

SUPERSONIC TRANSPORT ASSUMED DATA (M = 3)

Test Reynolds number	4.8×10^6
Flight Reynolds number	2×10^8
dC_D/dC_L^2	.65
C_{D_o} (No twist and camber)$_{w.t.}$	.0085
C_{D_o} (No twist and camber)$_{Flt.}$	.0060
$C_{D\ wave}$ (both conditions)	.0025
Wetted Area/Wing Area	3

Using the data tabulated above, the following schedule of extrapolation errors is assumed:

Error Source	Standard Deviation of Drag Coefficient
Friction Drag at W.T. Cond.	.00012
Friction Drag at Flight Cond.	.00007
Bleed air drag	.00015
Roughness - Joints-Leaks (Production aircraft measurements taken)	.00005
Pressure drag viscous interation	.00005
Roughness and Protuberance Drag	.00013

For the supersonic transport it can be seen that the errors are all in the order of one count or less and the root-sum-square is 2.5 counts. Taking the value of 2.5 counts at $C_L = .1$ obtained on page 24 and combining, the total standard deviation of 3.6 counts is obtained corresponding to a one sigma error of 2.9 percent of the 125 total counts of drag. It is therefore anticipated for a probability of 95 percent that the predicted value of the drag coefficient at cruise would not be in error by more than 5.8 percent. The situation is shown in Figure 3. The aeroelastic deformation of wind tunnel and flying aircraft are usually not large but must be included as a possible source of drag in the drag budget. It is certain that the precision of measured model dimensions can be high enough to preclude errors from this type of measurement.

COMPARISON BETWEEN WIND TUNNEL EXTRAPOLATIONS AND FLIGHT DATA

In arriving at comparable data from flight tests the major problem is the accurate measurement of installed engine thrust. Various techniques are currently in use for estimating the thrust; of these two basic methods of comparable accuracy are the "gas generator" method and the swinging probe method. In the gas generator method a group of variables such as rpm, turbine inlet temperature, compressor pressure rise, etc. are measured and correlated with ground test data to provide an estimated thrust value. The measurements are usually redundant and so a weighting schedule is used which has been experimentally and analytically determined from the test stand results. Naturally, the precision of the prediction depends on the measurement accuracy of the many variables and error analyses have been made by others to estimate the precision. However, under the present study no evaluation of these methods and results has been conducted. The swinging probe technique and some details of the gas generator method are described in References 28 and 29 by T. W. Davidson. In the swinging probe method direct pressure and temperature measurements are made in the jet exhaust of the aircraft; from these measurements the mass flow, gross (exit nozzle) thrust, and overall thrust are calculated. Comparisons made in Reference 29 indicate that flight thrust comparisons between gas generator methods and the swinging probe method agree to within 5 percent over the Mach number range of .5 to 1.86. It is clear that the precision of the flight test methods must depend on the particular installation, the number of measurements taken, and the extent and precision of the ground laboratory tests. Factors which tend to increase the uncertainties are the variations in total pressure and flow at the compressor face, variability of leakage in the engine and ducts, and geometrical differences between test and flight engines. In this latter regard discussions with Mr. T. W. Davidson of the U. S. Naval Air Test Center have brought out the fact that engines taken from a given production line apparently yield individualistic thrust variations of as much as 5 percent under similar test conditions.

Blow in doors used in supersonic engine installations provide additional difficulties when measuring engine performance since rather extensive instrumentation is needed to measure the large amounts of secondary air with accuracy.

In the absence of a detailed evaluation of the entire flight test procedure it will be assumed that the results quoted in Reference 29 of 5 percent agreement between methods of thrust measurements is a reasonable estimate of their percent standard deviation. The estimates made in the previous sections indicates that the precision of wind tunnel extrapolations for the subsonic and supersonic cruise conditions should be as good as ±3 percent, hence it is to be expected that the results of very carefully carried out flight and wind tunnel extrapolations should agree to within 5 percent.

It is possible that improved accuracy can be obtained by exercise of special care and repetitive testing, the above values serve however as an index of what should be achieved. Greater discrepancies than those estimated may occasionally occur, and can easily occur for off-design power plant settings; however, conditions where large uncertainties are probable can usually be anticipated in advance. As discussed earlier one condition of great difficulty exists at high subsonic speeds where the position of shock waves is altered substantially by the boundary layer conditions. It is easily seen that whenever large regions of separated flow exist on a model and it is possible for the point of separation to move, large changes in the drag and lift may be incurred by changes from wind tunnel conditions to those of free flight. The uncertainties thus introduced may in these cases be difficult to determine but at least the <u>danger</u> of producing large errors can be recognized.

The history of comparisons between flight test and wind-tunnel predictions is of course full of cases having both good and poor agreement, however, the documentation in most cases is scattered or meager. In a few cases there has been extensive documentation and one of these is the X-15 research airplane. This example is particularly pertinent because it eliminates the uncertainty factors due to power plant thrust. In References 30 and 31 for Mach numbers up to 3.0 the results of wind-tunnel extrapolations were found to agree quite accurately with flight measurements, however, this one carefully documented case could easily be considered fortuitous and the broad question of the validity of wind tunnel extrapolations cannot be settled by one case history. In order to instill confidence in the orderly and scientific process for estimation of full scale flight performance it would appear desirable to insist on a complete error analysis of the predicted quantities specially tailored to the particular aircraft, the particular wind tunnels, and the estimated flignt conditions. Errors due to human carelessness or bias can best be avoided by duplication of testing and of estimation. In view of the costs of airplane development the additional cost of duplication of this sort would be negligible.

CONCLUDING REMARKS AND RECOMMENDATIONS

The science of conducting meaningful and accurate wind-tunnel tests has reached a high state of development and constant activity within those groups carrying out the work is providing improved methods for prediction of full scale flight performance. In view of the intense activity it is not surprising that the study being reported here has not uncovered any new and large factors which would introduce gross errors into the performance prediction process. The re-survey of many potential drag sources on high speed aircraft at their design cruise conditions has led to the general conclusion that modern technology should enable performance to be estimated with an accuracy better than ±3 percent. This does not ensure that predictions will be so precise since the method itself is tedious and requires highly skilled and impartial application of scientific knowledge. Certainly, incomplete analysis and lack of sufficient data can lead to performance estimates which are far from the mark, but it appears reasonable on the basis of the improved status of aeronautical knowledge that much improved forecasting of aircraft performance is to be expected.

The results of this study are almost certain not to change any currently held views on the correlation of wind tunnel and flight data. In this endeavor, however, it is clear that something can be done in the future to bring light to the matter. First, it is important that work in progress leading to improved estimation of wave drag, friction drag, roughness effects, and power plants installation losses be continued. Additional effort might easily be placed on the old problems of form drag at subsonic speed, boundary layer and shock wave interaction at supersonic speeds, and friction drag under turbulent flow conditions over typical geometric configurations involving variable dynamic pressure and lateral pressure gradients; percentage accuracy of friction data at wind tunnel Reynolds numbers is more important than it is at flight Reynolds numbers. A second step to improve the general understanding would be to request that a check list of corrections with their estimated uncertainties accompany each comparison of wind tunnel and flight data. In this way the engineers preparing the documentation will be required to face up to all the known difficulties and inadequacies, while at the same time the reviewers of the documentation will be able to apply their own experience in assessing the precision of the results presented. Another powerful way to provide added accuracy would be to require duplicate testing and estimating of flight performance. The costs of such duplication should generally be small compared to the overall development costs involved in the production of new aircraft.

APPENDIX

ESTIMATION OF NACELLE WAKE - FUSELAGE INTERFERENCE DRAG

In model tests at supersonic speeds the nacelle internal drag is sometimes handled by using a constant internal diameter tube for which the internal drag is accurately obtained as the product of surface area, dynamic pressure, and friction drag coefficient. While the internal drag is correctly evaluated there is a possible error introduced if the nacelle base flow can interfere with an adjacent fuselage or tail surface. To determine the level of such interference drag some rough estimates were made as follows:

Nacelle base radius	R
Nacelle internal radius	.8R
Distance to fuselage centerline	7R
Fuselage radius at zone of interference	3R
Corresponding fuselage slope	-.10

The pressure coefficient of the interference pressure field at a distance r laterally is:

$$\frac{\Delta p}{q} = \frac{2\delta}{\sqrt{(M^2 - 1)(r/R - 1)}}$$

where δ is the slope of mixing streamline aft of the base. Using Chapman's data in Reference 25 the angle δ for M = 1.5 and 3 are 9 degrees and 20 degrees respectively and lead to an estimated drag on the fuselage in the wind tunnel condition of:

M_∞	ΔC_D
1.5	+.0003
3.0	+.0002

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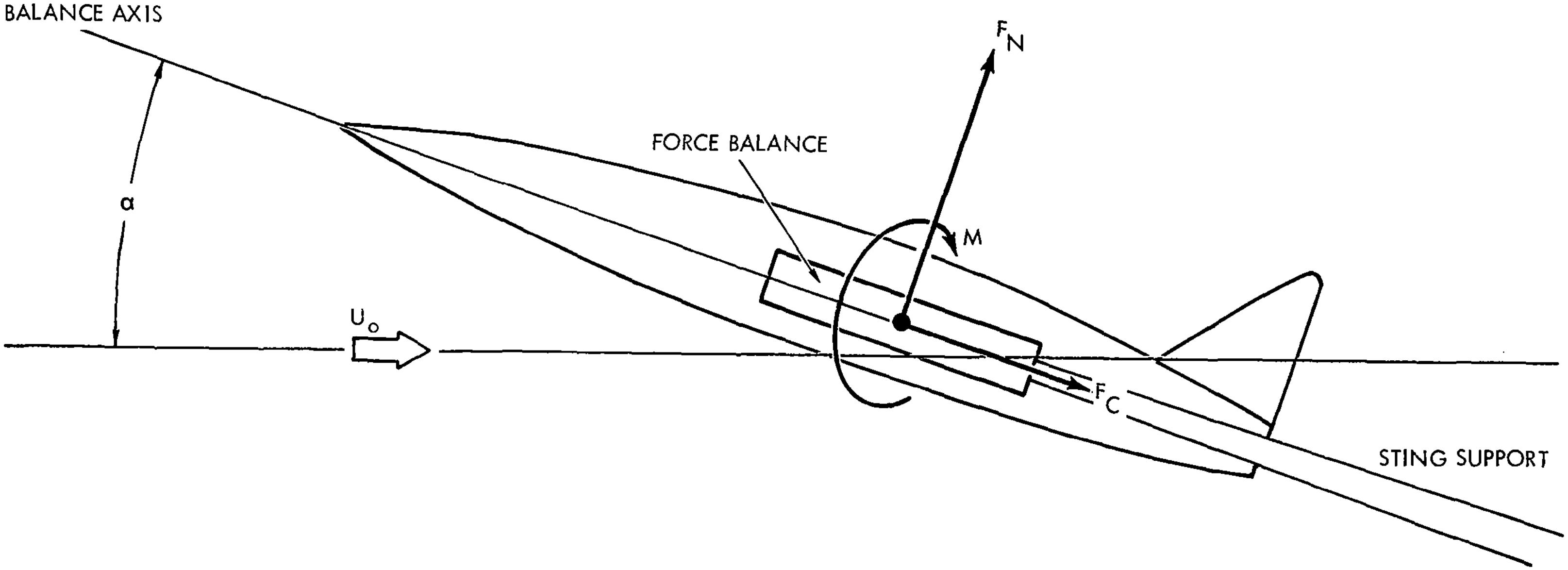

FIGURE 1 - USUAL BALANCE ARRANGEMENT FOR FORCE MEASUREMENTS

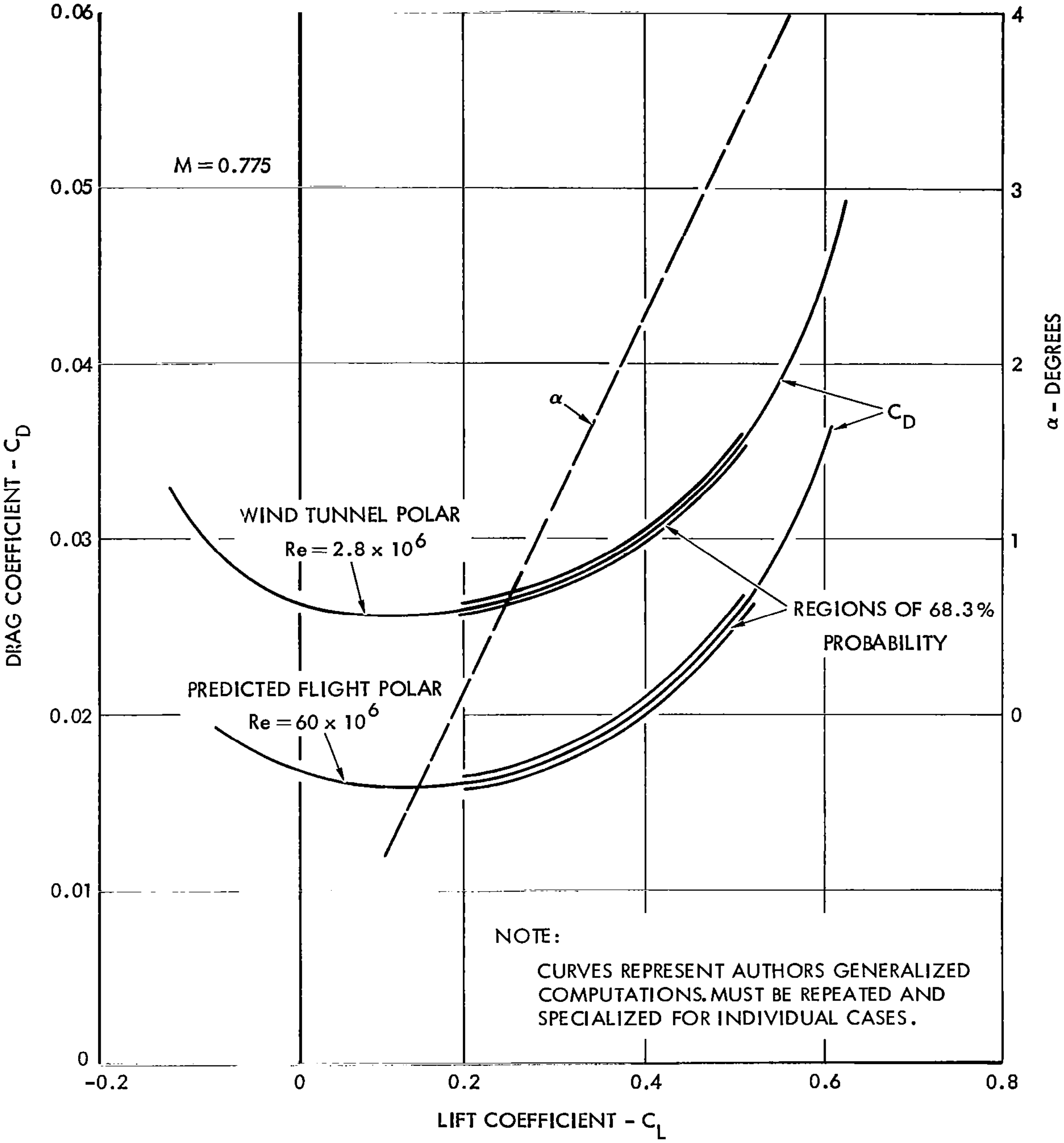

FIGURE 2 - DRAG POLAR FOR LARGE SUBSONIC TRANSPORT

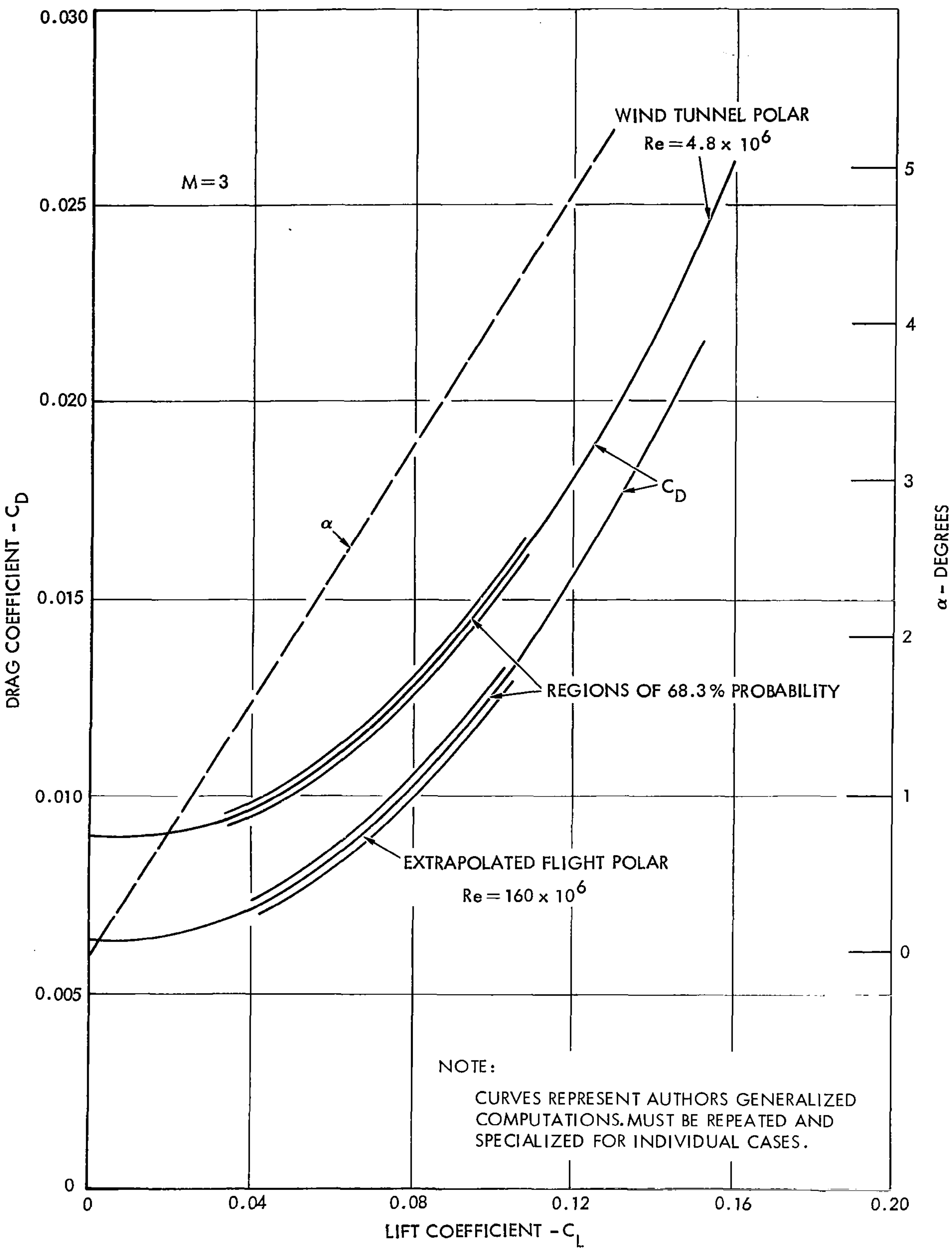

FIGURE 3 - SUPERSONIC TRANSPORT DRAG POLAR

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