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AN ANALYSIS AND CORRELATION OF AIRCRAFT WAVE DRAG

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AN ANALYSIS AND CORRELATION OF AIRCRAFT WAVE DRAG[1] [2]

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SUMMARY

A computer program, developed by the Boeing Company for use on the IBM 7090 electronic data processing system, has been studied at the Langley Research Center. The results of this study indicate that, in addition to providing reasonably accurate supersonic wave-drag estimates, the computer program provides a useful tool which can be used in design studies and for configuration optimization. A detail description of the program is given in the appendix.

INTRODUCTION

Since the rule was formulated, and verified experimentally, that the transonic wave drag of an aircraft is essentially the same as the wave drag of an equivalent body of revolution having the same cross-sectional area distribution as the aircraft (ref. 1), attempts have been made to estimate aircraft wave drag by examining the equivalent-body area distributions. It has been found that reasonably good wave-drag estimates can be made near a Mach number of 1 if the slender-body theory (ref. 2) is applied to the aircraft area distribution. This procedure can be extended to higher Mach numbers with good results by using the supersonic area rule (refs. 3 and 4) to determine the equivalent-body area distributions.

For most practical applications, however, the complexity of the supersonic area rule requires that this procedure be adapted to the high-speed electronic computer. As a result, several digital-computer programs which apply this theoretical approach to the solution of aircraft wave drag have been developed. One such program, which was developed by the Boeing Company, is presented with the permission of the Boeing Company in the appendix to this paper. It is the purpose of this paper to present, in addition to the wave-drag computer program, a review of the theoretical approach used in the program, and some experimental correlations which may serve as an indication of the accuracy of the wave-drag estimates obtained from the program.

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[1]An abbreviated version of this report was presented in "Proceedings of NASA Conference on Supersonic Transport Feasibility Studies and Supporting Research - September 17-19, 1963." NASA TM X-905, Dec. 1963, pp. 153-163.

[2]Title, Unclassified.

[redacted]

SYMBOLS

A	cross-sectional area
$C_{D,WAVE}$	wave-drag coefficient
D	wave drag
l	overall length
M	Mach number
N_X	the number of equal intervals into which the portion of the X-axis, X_A to X_B, is to be divided
N_θ	the number of equal intervals into which the domain of θ (-90° to $+90^\circ$) is to be divided
n,i,j	integers
q	dynamic pressure
r	radius
V	velocity
v	volume
x,y,z	coordinates along X, Y, and Z axes
X,Y,Z	axis system of airplane
X_A,X_B	end points of the interval along the X-axis outside of which no Mach plane intercepts the aircraft
β	$\sqrt{M^2 - 1}$
θ	angle between the Y-axis and a projection onto the Y-Z plane of a normal to the Mach plane. (θ positive in the positive Y-Z quadrant.)
μ	Mach angle
ρ	density

[redacted]

Subscripts:

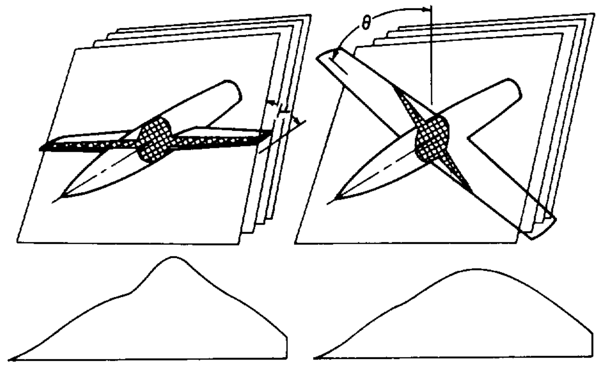
MAX maximum

BASE base

THEORETICAL APPROACH

Supersonic Area Rule

A review of the supersonic-area-rule wave-drag computing procedure is illustrated in figure 1. Each equivalent body of revolution is determined by passing a series of parallel cutting planes through the configuration. The cutting planes are inclined with respect to the aircraft axis at the Mach angle μ. The area of the equivalent body at each station is the projection onto a plane normal to the aircraft axis of the area intercepted by the cutting plane. It is evident that the series of parallel cutting planes can be oriented at various angles θ, around the aircraft axis, and in order to determine the drag accurately, a family of equivalent bodies, each corresponding to a particular value of θ, must be considered. Thus, at each Mach number, a series of equivalent bodies of revolution are generated. The wave drag of each equivalent body is determined by the von Kármán slender-body formula (ref. 2) which gives the drag as a function of the free-stream conditions and the equivalent-body area distribution. The wave drag of the aircraft at the given Mach number is then taken to be the integrated average of the equivalent-body wave drags.



$$D(\theta) = -\frac{\rho V^2}{4\pi}\int_0^l\int_0^l A''(x_1)\,A''(x_2)\,\mathrm{LOG}\left|x_1 - x_2\right| dx_1\,dx_2$$

$$D = \frac{1}{2\pi}\int_0^{2\pi} D(\theta)\,d\theta$$

Figure 1.- Illustration of wave-drag computing procedure.

It should be noted, however, as discussed in reference 5, that the supersonic area rule is not an exact theory. In addition to the slender-body-theory assumptions, the supersonic area rule assumes that an aircraft, which usually departs considerably from a body of revolution, can be represented by a series of equivalent bodies of revolution. The theory, therefore, does not account for wave reflections which may occur due to the presence of the fuselage, wing, or tail surfaces. Also, the theory does not account for the induced drag at zero lift of configurations with highly twisted and cambered lifting surfaces. Nevertheless, for most configurations, the supersonic area rule does account for the major part of the wave drag and provides a useful procedure for the analysis of aircraft wave drag.

Machine Program

A major problem in adapting this procedure to machine computation is that of describing a rather complex aircraft to the computer in sufficient detail. The manner in which an aircraft is mathematically described to the computer for the program presented herein is illustrated in figure 2. The lower right portion of the figure shows a typical aircraft for which the supersonic wave drag is to be computed. The upper left portion of the figure shows the aircraft as it is described to the computer.

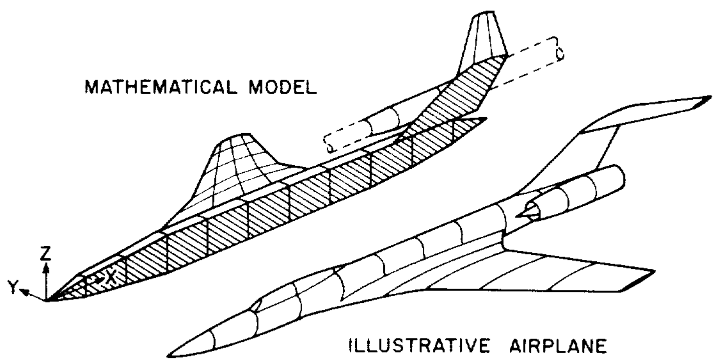


Figure 2.- Mathematical representation of illustrative airplane for machine-computing procedure.

The locations of all the aircraft components are referred to an X-Y-Z-axis system with its origin at the nose of the fuselage. The fuselage is assumed to be sufficiently close to a body of revolution that it can be described in terms of the radii of equivalent circles which have the same area as the fuselage at each station. The variation in fuselage radius along the axis between stations is assumed to be linear.

The wing is described as a sequence of streamwise airfoils distributed along the span. The contour of the wing is assumed to be linear between successive ordinates. The horizontal and vertical tails are described in a manner similar to that of the wing.

The engine nacelles are located by specifying the x, y, and z coordinates of the nacelle center line at the inlet face and are described in a manner similar to that of the fuselage by giving the radii at successive stations. The discontinuities caused by the inlet and exit faces are eliminated by assuming that infinitely long cylinders extend in both directions from the inlet and the exit. The effects of inlet spillage on the wave drag can be included by properly contouring the cylindrical extension near the inlet face.

Once the aircraft description has been stored in the memory unit of the computer, the equivalent-body area distributions are determined by solving for the normal projection of the areas intercepted by the cutting planes.

In addition to the aircraft wave drag, which is evaluated by applying the method of references 6 and 7 to the solution of the von Kármán integral (ref. 2), the program lists the wave drags of the aircraft equivalent bodies at each Mach number, as well as selected equivalent-body area distributions. This additional information is particularly useful in tailoring a configuration for minimum wave drag because, in order for a configuration to be optimized at some supersonic Mach number, it is necessary to examine the series of equivalent bodies corresponding to the particular Mach number. It should also be noted that the area distributions required in the computation of sonic-boom overpressures ($\theta = -90^{\circ}$) are provided.

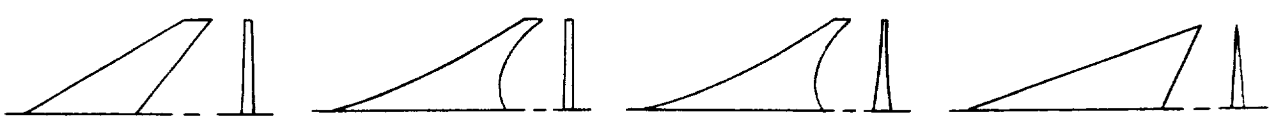
EXPERIMENTS

Optimum Bodies of Revolution

A series of bodies of revolution which have minimum wave drag for a given length, volume, and base area (ref. 8) and which have a base-to-maximum-area ratio of 0.532 have been tested over the Mach number range from 0.60 to 3.95. The variations in wave-drag coefficient with Mach number were determined by integrating the measured surface-pressure coefficients for three optimum bodies which had fineness ratios of 7, 10, and 13, respectively. The experimental results for Mach numbers from 0.60 to 1.20 were obtained in the Langley 8-foot transonic pressure tunnel, and those for Mach numbers of 1.61 and 2.01 were obtained in the Langley 4- by 4-foot supersonic pressure tunnel. The data for Mach numbers from 2.50 to 3.95 were obtained in the Langley Unitary Plan wind tunnel.

Semispan Wings

The series of semispan wings was tested in the Langley 4- by 4-foot supersonic pressure tunnel over the Mach number range from about 1.4 to 2.2. Detailed descriptions of the wings and the test setup are given in references 9 and 10. Sketches of the wings are shown below.



Wing 1 Wing 2 Wing 3 Wing 4

Wing 1 had a trapezoidal planform and a linear spanwise thickness distribution. Wing 2 had a complex planform with a linear spanwise thickness distribution. Wing 3 had a complex planform as well as a complex spanwise thickness distribution. Wing 4 had an arrow planform with a linear thickness distribution. All of the wings in the series had circular-arc airfoil sections.

Transition of the boundary layer was fixed near the wing leading edges by narrow strips of distributed roughness particles, and the drags were measured at the zero-lift condition. The wave-drag coefficients were determined by subtracting the equivalent flat-plate turbulent skin-friction drag coefficients from the measured total-drag coefficients.

Airplane Configurations

Tests were made over the Mach number range from 1.4 to 3.2 for several of the proposed supersonic transport configurations and a typical supersonic fighter. Sketches of the configurations are shown in figure 3. Detail descriptions of the models and the tests are given in references 11 to 16. Boundary-layer transition was fixed near the leading edges of all of the models by narrow strips of distributed roughness particles. The experimental wave-drag coefficients were determined for each configuration by subtracting the equivalent flat-plate turbulent skin-friction drag and an estimated camber drag from the measured total drag at zero lift.

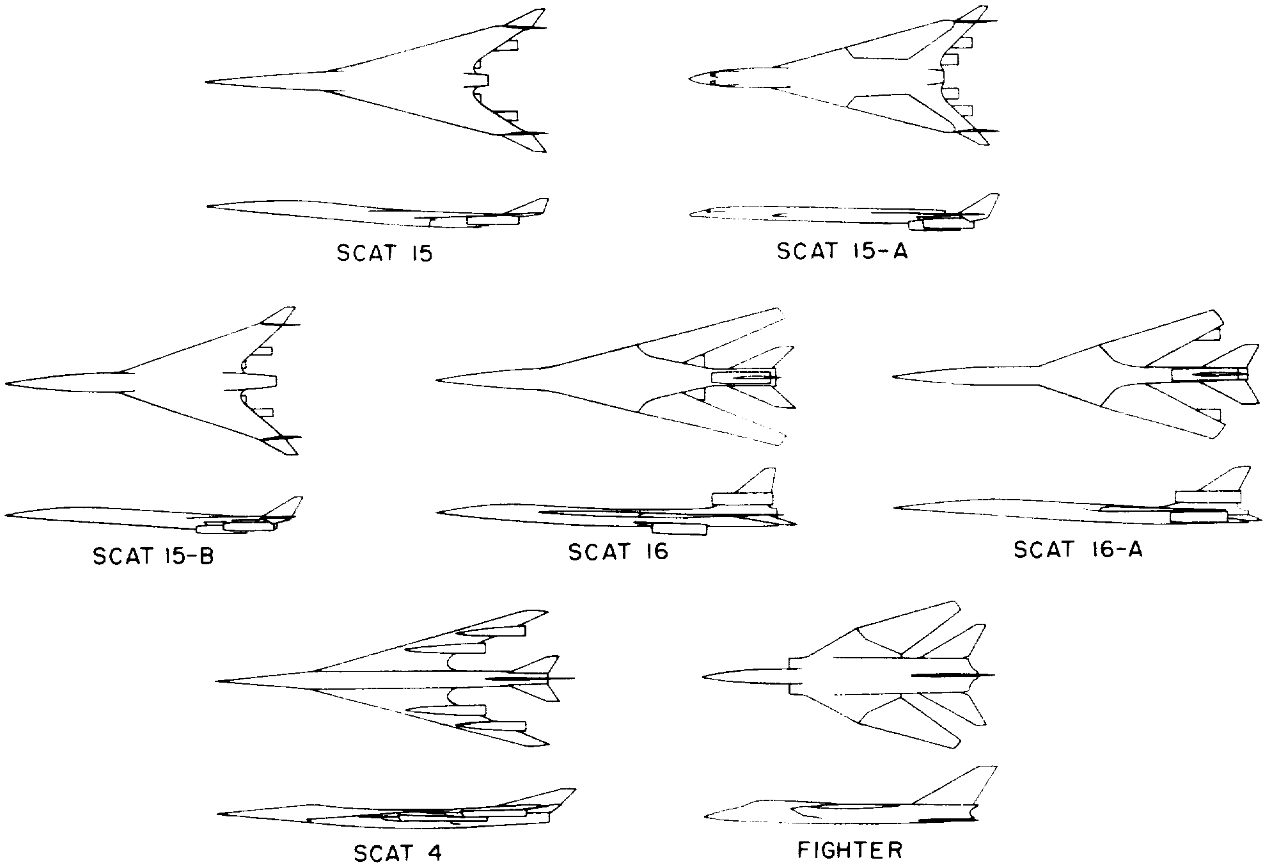


Figure 3.- Description of airplane configurations.

DISCUSSION

In order to indicate the accuracy of the wave-drag estimates obtained from the program, the machine-computed wave-drag values are compared with experimental results for the optimum bodies of revolution, semispan wings, and airplane configurations in figures 4, 5, and 6, respectively.

Optimum Bodies of Revolution

Figure 4 shows a comparison of the machine-computed wave-drag coefficients with experimental results and the more precise characteristics theory. Also shown are the drag levels indicated by the slender-body theory which is based on the body normal-area distributions. The characteristics theory, indicated by the solid line, shows excellent agreement with the experimental results. The slender-body theory which uses the normal area distribution, shown as a short-dashed line, gives good agreement near a Mach number of 1. However, as the Mach number is increased, the slender-body theory overestimates the optimum-body wave drag. It should also be noted that the effects of Mach number are greater at the lower fineness ratios than at the higher fineness ratios.

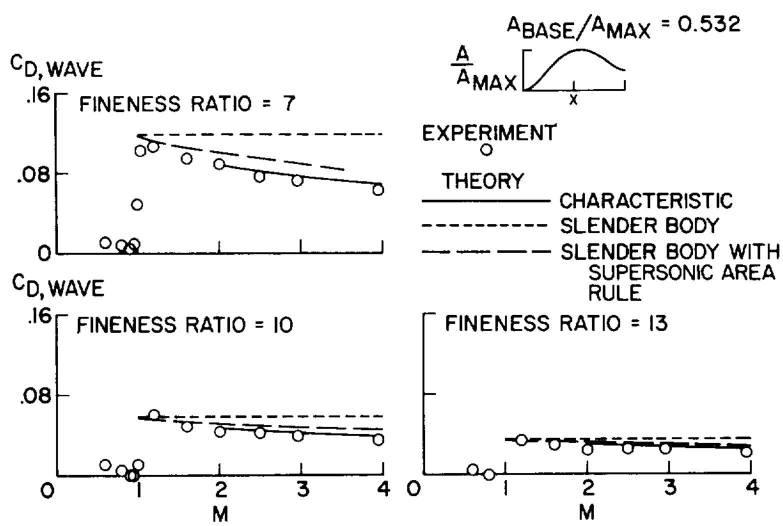


Figure 4.- Comparison of computed wave drag with experimental results for optimum bodies of revolution.

This greater departure from slender-body theory should be expected as the bodies become less slender. The long-dashed line shows the results obtained from the machine program which uses the slender-body theory in combination with the supersonic area rule. As can be seen, when the slender-body theory is applied to the proper equivalent bodies, as in the machine program, the Mach number effects on the optimum-body wave drag are predicted with a fair degree of accuracy.

Semispan Wings

The most severe test of the theoretical approach used in this machine program lies in its application to the calculation of the drag of wings. Figure 5 shows a comparison of the machine-computed wave drags with experimental results for the series of semispan wings over the Mach number range from about 1.4 to 2.2. As can be seen from the figure, the program tends to underestimate the wave drag of the semispan wings. This result for wings alone is not surprising, since a wing departs considerably from the equivalent body of revolution assumed by the theory.

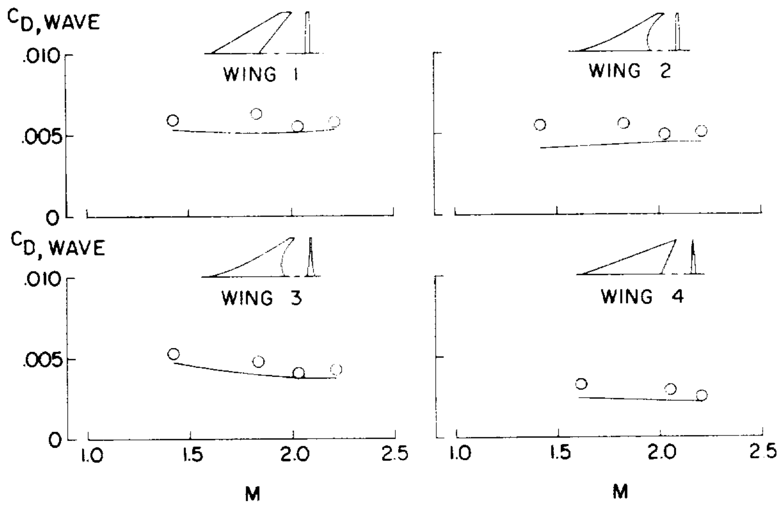


Figure 5.- Comparison of machine-computed wave drag with experimental results for semispan wings.

Airplane Configurations

A comparison of the machine-computed wave-drag coefficients with experimental results for the complete airplane configurations is shown in figure 6. The experimentally determined wave-drag coefficients are plotted against the machine-computed values. The solid line is the locus of perfect agreement between theory and experiment. The arrows shown on the figure indicate the wave-drag trends with increasing Mach number. This comparison indicates that the machine program, which uses slender-body theory in combination with the supersonic area rule, can produce good estimates of the wave drag of complex airplane configurations at supersonic speeds. The major departures from perfect agreement between theory and experiment shown in the figure are believed to be due to the difficulties in adequately

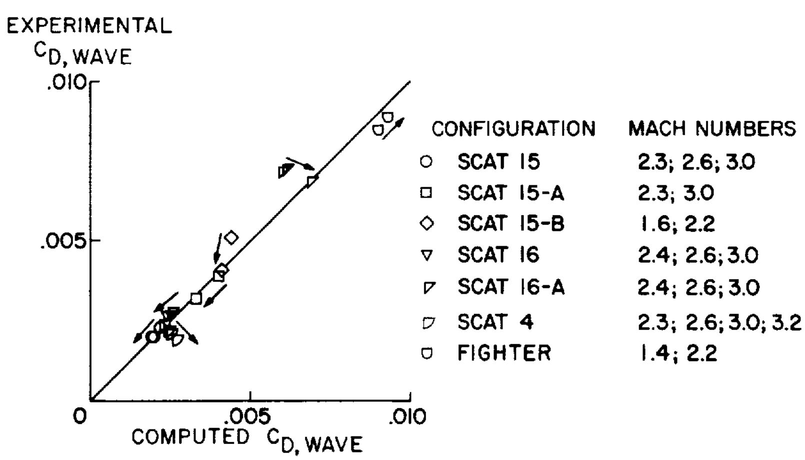


Figure 6.- Comparison of machine-computed wave drag with experimental results for airplane configurations. (Arrows indicate the direction of increasing Mach number.)

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describing the extremely complex configurations to the computer, and to regions of separated flow which may have existed on some of the models at the off-design Mach numbers.

CONCLUDING REMARKS

A computer program, developed by the Boeing Company, which applies the slender-body theory in combination with the supersonic area rule to the solution of aircraft wave drag has been studied at the Langley Research Center. The results of this study indicate that, in addition to providing reasonably accurate supersonic wave-drag estimates, the computer program provides a useful tool which can be used in design studies and for configuration optimization.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., December 13, 1963.

APPENDIX

COMPUTER PROGRAM FOR THE DETERMINATION OF AIRCRAFT WAVE DRAG AT ZERO LIFT

The computer program developed by the Boeing Company applies the slender-body theory in combination with the supersonic area rule to determine aircraft wave drag. For programing purposes, an aircraft is assumed to consist of a wing, a fuselage, up to eight pod pairs (or nacelles), either one or two fins (vertical tails), and a canard surface (or horizontal tail). Except for the single vertical fin which may be asymmetrically located, the aircraft is assumed to be symmetrical about the X-Z plane. The program was written in the FORTRAN language (ref. 17) for use on the IBM 7090 electronic data processing system. The purpose of this appendix is to present a detailed description of the program, describe the manner in which the input data must be prepared, and give a FORTRAN listing of the source program and the subroutines which are not included on the standard FORTRAN II library tape. The machine tabulated output for three sample cases is also presented in tables I, II, and III.

DESCRIPTION OF PROGRAM

The program first reads in a number of integers which specify the absence or presence of various components, the amount of detail to be used to describe each component, and the number of equal intervals into which the domain of each of the two independent variables, X and θ, is to be divided. Various program options are also indicated at this point. The program then reads a title. Finally, the program reads in the geometric parameters which define each component of the configuration as described in the text. A detailed description of the input and the input format required by the program are given in the section of this appendix entitled "Preparation of Input Data."

Wing Volume

The segment of a wing between two successive airfoils (fig. 7) is considered to be composed of a number of blocks, each extending from an X-value at which an upper and lower ordinate are specified (input data) to the next such X-value. The contour definition of each airfoil is assumed to be linear between successive ordinates. Further, corresponding points on successive airfoils are assumed to be joined by straight lines. The wing is therefore treated as a polyhedron. This polyhedral fit to the wing simplifies the calculation of the area intercepted by a cutting plane through the wing.

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The shape of each block can thus be seen (fig. 7) to consist of a pair of parallel trapezoidal faces each in a wing airfoil plane (including the possibility of one edge of the trapezoid being degenerate) with a straight-line edge from each vertex on the inboard trapezoid joining the corresponding vertex on the outboard trapezoid. The volume of such a solid is found to be:

$$V = \frac{\Delta y}{6}\left[\Delta x_1\left(2\ \Delta z_1 + \Delta z_2\right) + \Delta x_2\left(\Delta z_1 + 2\ \Delta z_2\right)\right]$$

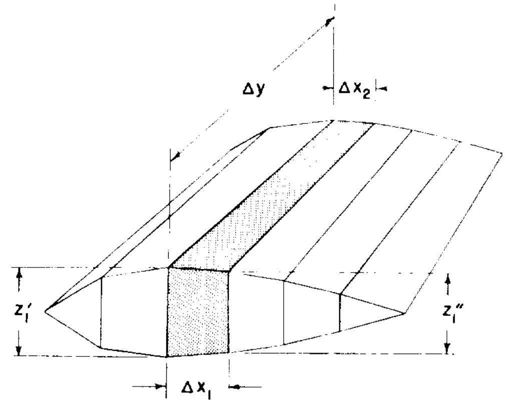

Figure 7.- Mathematical representation of a wing segment between successive airfoils.

where

$$\Delta z = \frac{1}{2}\left(z_1' + z_1''\right)$$

If there is a wing given in the input for the case being considered, the volume of the wing is computed by summing the volumes of the individual blocks. Later in the program the volume of the wing average equivalent body is computed. If a sufficient number of cutting planes have been used to define the wing equivalent bodies of revolution, then the two values of wing volume should be essentially the same. Thus, a check on the accuracy of the equivalent body area distributions is provided.

Transformation of Wing, Fin, and Canard Coordinates

The wing is described to the machine program by up to 10 airfoils, each being specified by the x, y, z coordinates of the leading edge, by the chord length, and by an array of up to 30 upper ordinates. The airfoil ordinates are expressed as a percentage of the chord length and are given at an array of percent-chord locations. The same array of percent-chord locations must serve for all wing airfoils in any one case. If the input which specifies the number of wing airfoil ordinates is negative, then the program will expect to read in lower ordinates also. Otherwise, the airfoil is assumed to be symmetrical and the program constructs the lower ordinates.

Each coordinate of a point on the wing surface is transformed by the machine program from percent-chord data into units of length and then referred to the origin of the reference axis, which is the nose of the fuselage. If there is no fuselage, the wing apex is taken to be the origin of the reference axis. The maximum upper ordinate and the maximum lower ordinate on each airfoil are noted for future reference.

The fins (vertical tails) and canard (or horizontal tail) are treated by the program similarly, except that they are defined in less detail and there is no possibility of describing a nonsymmetric fin airfoil. The fins and canard are defined by locating the root and tip airfoils in the same manner as the wing and by giving the airfoil ordinates. However, the fins and the canard must each consist of a constant airfoil section, with each of the sections being described by a maximum of 10 airfoil ordinates.

Optimum Area-Distribution Fit to Fuselage

The fuselage is defined by up to 30 cross-sectional areas, given at any longitudinal spacing. The method of references 6 and 7 is then used in the subroutine EMLORD to determine the optimum area distribution which contains the given body areas. This method of estimating the wave drag of a slender body whose cross-sectional areas are given at arbitrarily spaced stations involves the determination of an area distribution which matches the given one at the specified stations, which otherwise has minimum wave drag, and which appears as a continuous analytic expression. The optimum area distribution so found is evaluated at every integral percentage of the fuselage length. This "enriched" area distribution is used as the definition of the fuselage during the remainder of the computation. An array of 101 radii, each corresponding to a station on the enriched fuselage area distribution, is computed by assuming the cross-sections to be everywhere circular. The variation in fuselage radius along the axis between the enriched stations is assumed to be linear.

Determination of the Intercepted Areas

The program selects a value of θ (-90° plus some multiple of n times $\Delta\theta$ where $n = 0,1,2,3,\ldots,N_\theta$) so that the domain of θ (-90° to $+90^\circ$) is divided into N_θ equal subintervals. Associated with each value of θ is an interval on the X-axis outside of which no Mach plane of this family will intersect any component of the aircraft. Let X_A and X_B denote the end points of this interval. If there is no fuselage, X_A and X_B are initially set equal to zero. If there is a fuselage X_A is initially set equal to the first fuselage x-station and X_B is initially set equal to the last fuselage station.

For each airfoil of the wing, the x-intercept of the Mach plane through the leading edge of that airfoil is compared with the previous X_A, and the algebraic lesser of the two is selected as the new value of X_A. Similarly, the Mach plane through the trailing edge of each airfoil is examined to determine if its x-intercept is greater than the previous X_B. The fins and the canard are each analyzed in the same manner to determine if they cause further shifting in X_A and X_B.

For all of the wing, tail, and canard surfaces, the assumption is made that the first and last ordinate of each airfoil is zero. A further assumption is that a Mach plane through the nose of the airfoil will intersect that airfoil nowhere else.

To determine the most forward Mach plane which touches a pod is more difficult, because the forward end of the pod can be a circle in a plane parallel to the y-z plane. The x-intercept, X, of the Mach plane which is tangent to the outer edge of that circle is given by

$$X = x - (\beta \cos \theta)(y + r \cos \theta) - (\beta \sin \theta)(z + r \sin \theta)$$

where x, y, and z are the coordinates of the pod center line at the leading edge and r is the radius. The same equation represents the aftermost Mach plane touching a pod when x, y, z, and r refer to the aft end of the pod. This equation is used to examine each pod to determine whether the pods cause further shifting in X_A and X_B.

The interval X_A to X_B associated with each value of θ is now divided into N_X equal subintervals, ΔX. The Mach planes are then defined by the successive values of X associated with each value of θ. Thus,

$$X = x - (\beta \cos \theta)y - (\beta \sin \theta)z$$

where

$$X = X_A + n\,\Delta X \qquad (n = 0,1,2,3,\ldots,N_X)$$

The program then proceeds to find the projection onto the y-z plane of the area of each component of the aircraft intercepted by the Mach planes.

The wing has been shown in figure 7 to consist of a number of blocks. Given the coordinates of the vertices of each wing block and the equation for each Mach plane, the subroutine SWING computes the y-z projection of the area of intersection of each Mach plane with each block.

First, a block which has the planform of the entire right wing, and which has a constant inboard thickness equal to the maximum thickness of the first airfoil and a constant outboard thickness equal to the maximum thickness of the last airfoil is examined. If the return from SWING is zero and the wing leading and trailing edges are not convex, then the wing is not intersected by the Mach plane being considered and the following procedure is bypassed. Otherwise, a block which has the planform of the segment between the first and second airfoils is examined. The procedure is repeated with the successive segments. For any segment which is intersected by the Mach plane being considered, the blocks comprising that segment are examined. The sum of the projected areas is accumulated until the last block in the right wing has been examined. After the last block has been examined, the entire procedure is repeated for the left wing. The final total result is an array of wing equivalent body areas corresponding to the particular values of X and θ.

The pods (or nacelles) are defined by up to 30 radii, given at arbitrarily spaced stations along the pod axis. The variation in pod radius between stations is assumed to be linear. The pods are located by specifying the x, y, z coordinates of the pod center line at the leading edge. Any external appendage which occurs in pairs located symmetrically about the x-y plane, and which can be described as a body of revolution is treated as a pod. Also, for the purpose of determining the intercepted areas, the fuselage is treated as a single pod located on the aircraft reference axis. The fuselage and the left and right members of each pair of pods are separately treated by the subroutine SPOD, which determines the projection onto the y-z plane of the areas intercepted by the Mach planes. If either the first or last cross-sectional area of a pod or fuselage is not zero, the program assumes that the body continues with constant area in the appropriate direction to infinity.

The process used to determine the fin and canard equivalent body areas is the same as that used on the wing. The process is simplified, however, because the fins and canard are each defined by only two airfoils.

Computation of Wave Drag

After the total equivalent-body area distribution for each value of θ has been determined, the wave drag of each equivalent body is computed by the subroutine EMLORD which applies the method of references 6 and 7. The values of $D(\theta)/q$ thus obtained are then used in the numerical integration of

$$\frac{D}{q} = \frac{1}{\pi} \int_{-\pi/2}^{\pi/2} \frac{D(\theta)}{q}\, d\theta$$

to yield the aircraft wave drag.

Computation of the Wing Average Equivalent-Body Volume

If there is a wing in the case being considered, the volume of the wing average equivalent body is found by:

$$v = \frac{1}{\pi} \int_{-\pi/2}^{\pi/2} \int_{X_A}^{X_B} A(x,\theta)\, dx\, d\theta$$

This volume is determined for the purpose of comparison with the exact wing volume which was determined earlier in the program. If a sufficient number of

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Mach planes (N_X) have been used to define the equivalent bodies, then the two values of wing volume should be essentially the same.

Computation of the Wing Average Equivalent-Body Area Distribution

The program can be used to compute the area distribution of the wing average equivalent body when the input data are arranged as indicated below. If the configuration of any case consists only of a wing and a fuselage, and the fuselage cross-sectional areas are set everywhere equal to zero, the program then branches into a routine which computes the wing average equivalent-body area distribution. The fuselage length and the wing location must be specified so that none of the Mach planes which pass through the first and last fuselage stations intercept the wing. This condition produces an identical range of X values $(X_A$ to $X_B)$ for each value of θ and therefore simplifies the computation. The wing average equivalent-body areas at each X station are then found by evaluating the integral

$$A(X) = \frac{1}{2\pi} \int_0^{2\pi} A(X,\theta)d\theta$$

at each value of X.

Tabulated Output

The full 80-column card image of each input data card is first printed to identify the results which will follow, and to provide an easy check on the input data. (See tables I, II, and III.) The enriched fuselage area distribution is then printed, together with the wave drag (expressed as D/q) of the fuselage alone. The wave-drag values (D/q) associated with each value of θ are then tabulated, along with the wave drag (D/q) of the entire aircraft. A check on the accuracy of the equivalent-body area distributions is next provided by printing a comparison of the exact wing volume with the volume of the wing average equivalent body. Finally, the program prints the equivalent-body area distributions for five values of θ from -90° to $+90^\circ$ for configurations which are not symmetrical with respect to the x-y plane, and from -90° to 0° for configurations which are symmetrical.

If the input data have been arranged for computation of the wing average equivalent-body area distribution, this result is printed in addition to the equivalent-body area distributions corresponding to each value of θ.

PREPARATION OF INPUT DATA

Since the aircraft is assumed to be symmetrical about the x-z plane, only half of the aircraft need be described to the computer. The convention used in presenting all input data is that the half of the aircraft on the positive y side of the x-z plane is presented. The computer then uses this information to construct the complete aircraft.

A single case consists of the wave-drag computation for a single configuration at a single Mach number. The input data for each case are presented on at least two punched cards. In addition to the first two input data cards, the number of remaining cards depends on the number of components used to describe the configuration, whether or not a component has been described in the preceding case, and the amount of detail used to describe each component.

First Two Data Input Cards

The first data input card for each case contains 18 integers, each punched to the right of a 4-column field. (See tables I(a), II(a), and III.) An identification of the card columns, the name used by the source program, and a description of each integer is as follows:

Columns	Name	Description
01-04	MACH	Mach number × 1000 (If the input is 1000, the program assumes that $M = 1.000001$ to avoid the singularity which occurs at $M = 1.0$) $M \geqq 1$.
05-08	NX	The number of equal intervals into which the portion of the X-axis, X_A to X_B, is to be divided. NX $\leqq 50$ and must be an even number.
09-12	NTHETA	The number of equal intervals into which the domain of θ (-90° to $+90^{\circ}$) is to be divided. NTHETA $\leqq 36$ and must be a multiple of four.
13-16	NWAF	The number of airfoils used to describe the wing. $2 \leqq$ NWAF $\leqq 10$.
17-20	NWAFOR	The number of upper ordinates used to define each wing airfoil section. $3 \leqq$ NWAFOR $\leqq 30$. If NWAFOR is given a negative sign, the program will expect to read the lower ordinates also. Otherwise, the airfoil is assumed to be symmetrical.
21-24	NFUSOR	The number of stations at which the fuselage cross-sectional areas are to be specified. $4 \leqq$ NFUSOR $\leqq 30$.
25-28	NPOD	The number of pairs of pods (or nacelles) on the configuration. NPOD $\leqq 8$.

Columns	Name	Description
29-32	NPODOR	The number of stations at which the pod radii are to be specified. $4 \leqq \text{NPODOR} \leqq 30$.
33-36	NFIN	The number of vertical tails. $\text{NFIN} \leqq 2$.
37-40	NFINOR	The number of upper ordinates used to define each fin (vertical tail) airfoil section. $3 \leqq \text{NFINOR} \leqq 10$. The fin airfoil is assumed to be symmetrical.
41-44	NCANOR	The number of upper ordinates used to define each canard (or horizontal tail) airfoil section. $3 \leqq \text{NCANOR} \leqq 10$. If NCANOR is given a negative sign, the program will expect to read the lower ordinates also. Otherwise, the airfoil is assumed to be symmetrical.
45-48	J1	J1 = 0 if there is no wing. J1 = 1 if the wing description is to be provided for this case. J1 = 2 if the wing description is identical with that of the previous wing description.
49-52	J2	J2 = 0 if there is no fuselage. J2 = 1 if the fuselage description is to be provided for this case. J2 = 2 if the fuselage description is identical with that of the previous fuselage description.
53-56	J3	J3 = 0 if there are no pods. J3 = 1 if the pod description is to be provided for this case. J3 = 2 if the pod description is identical with that of the previous pod description.
57-60	J4	J4 = 0 if there are no fins. J4 = 1 if the fin description is to be provided for this case. J4 = 2 if the fin description is identical with that of the previous fin description.
61-64	J5	J5 = 0 if there is no canard. J5 = 1 if the canard description is to be provided for this case. J5 = 2 if the canard description is identical with that of the previous canard description.
65-68	J6	J6 = 1 if the entire configuration is symmetrical with respect to the x-y plane. J6 = 0 if the entire configuration is not symmetrical with respect to the x-y plane. J6 = -1 if the wing volume only is to be computed for this case.
69-72	J7	J7 = 9999
73-80		Case number

[redacted]

The second data input card for each case contains any desired title in columns 1 through 72. (See table I(a).)

Remaining Data Input Cards

The remaining data input cards for each case contain a detailed description of each component of the aircraft. Each card contains up to 10 numbers, each punched to the left of a 7-column field with decimals and is identified in columns 73-80. The cards are arranged in the order: wing data cards, fuselage data cards, pod (or nacelle) data cards, fin (vertical tail) data cards, and canard (or horizontal tail) data cards (table I(a)).

Wing data cards.- The first wing data card (or cards) contains the percent-chord locations at which the ordinates of all the wing airfoils are to be specified. There will be exactly NWAFOR percent-chord locations given. Each card is identified in column 73-80 (table 1(a)) by the symbol XAF j where j denotes the number of the last percent-chord location given on that card. For example, if NWAFOR = 16, there are 16 ordinates to be specified for every airfoil, and two data cards will be required. The first XAF card is identified as XAF 10 and the second as XAF 16.

The next wing data cards (there will be NWAF of them) each contain four numbers which give the location and chord length of each of the wing airfoils that is to be specified. The cards representing the most inboard airfoil are given first, followed by the cards for successive airfoils. The information is arranged on each card as follows:

Columns	Description
1-7	x-ordinate of the airfoil leading edge
8-14	y-ordinate of the airfoil leading edge
15-21	z-ordinate of the airfoil leading edge
22-28	the airfoil streamwise chord length
73-80	the card identification, WAFORG j where j denotes the particular airfoil. For example, WAFORG 1 denotes the first (most inboard) airfoil.

Following the WAFORG cards are the wing airfoil ordinate (WAFORD) cards. The first card contains up to 10 of the upper ordinates of the first airfoil expressed as a percent of the chord length. If more than 10 ordinates are to be specified for each airfoil (there will be NWAFOR of them) the remaining upper ordinates are continued on successive cards. If the airfoil is not symmetrical (indicated by a negative value of NWAFOR on the first data input card for this case), the lower ordinates of the first airfoil are presented in the same manner on the next cards. The program expects both upper and lower ordinates to be punched as positive percent-of-chord values. The remaining airfoils are each described in the same manner, and the cards are arranged in the order which begins with the most inboard airfoil and proceeds outboard. Each card is identified in columns 73-80 as WAFORD j, where j denotes the particular airfoil.

Fuselage data cards.- The first card (or cards) specifies the array of fuselage stations at which the values of the fuselage cross-sectional area are to be specified (table I(a)). There will be NFUSOR stations given and the first fuselage station must be zero. This card (or cards) is identified in columns 73-80 by the symbol XFUS j where j denotes the number of the last fuselage station given on that card. The XFUS cards are followed by a card (or cards) which gives the fuselage cross-sectional areas, identified by the symbol FUSARD j in columns 73-80.

Pod data cards.- The first pod or nacelle data card (or cards) specifies the location of the origin of each pair of pods. The information is arranged on each card as follows (table I(a)):

Columns	Description
1-7	x-ordinate of the origin of the first (most inboard) pod pair
8-14	y-ordinate of the origin of the first pod pair
15-21	z-ordinate of the origin of the first pod pair
22-28	x-ordinate of the origin of the second pod pair
29-35	y-ordinate of the origin of the second pod pair
36-42	z-ordinate of the origin of the second pod pair
43-49	x-ordinate of the origin of the third pod pair
50-56	y-ordinate of the origin of the third pod pair
57-63	z-ordinate of the origin of the third pod pair
64-70	x-ordinate of the origin of the fourth pod pair
73-80	the card identification, PODORG

The PODORG data are continued on successive cards until all of the pod origins (NPOD of them) have been specified.

The next pod input data card (or cards) contains the x-ordinates, referenced to the pod origin, at which the pod radii (there will be NPODR of them) are to be specified. The first x-value must be zero, and the last x-value is the length of the pod. These cards are identified in columns 73-80 by the symbol XPOD j where j denotes the pod number. For example, XPOD 1 represents the first (most inboard) pod.

The next pod input data cards give the pod radii corresponding to the pod stations that have been specified. These cards are identified in columns 73-80 as PODR j.

For each additional pair of pods, new XPOD and PODR cards must be provided.

Fin data cards.- If there is a single vertical fin (NFIN = 1), it may be located anywhere on the configuration. If NFIN = 2, the program will expect data for a single fin, but assumes that an exact duplicate is located symmetrically with respect to the x-z plane. Exactly three data input cards (table I(a)) are used to describe a fin. The information presented on the first fin data input card is as follows:

Columns	Description
1-7	x-ordinate of lower airfoil leading edge
8-14	y-ordinate of lower airfoil leading edge
15-21	z-ordinate of lower airfoil leading edge
22-28	chord length of lower airfoil
29-35	x-ordinate of upper airfoil leading edge
36-42	y-ordinate of upper airfoil leading edge
43-49	z-ordinate of upper airfoil leading edge
50-56	chord length of upper airfoil
73-80	the card identification, FINORG

The second fin data input card (table I(a)) contains up to 10 percent-chord locations (exactly NFINOR of them) at which the fin airfoil ordinates are to be specified. The card is identified in columns 73-80 as XFIN.

The third fin data input card contains the fin airfoil ordinates expressed as a percent of the chord length. Since the fin airfoil must be symmetrical, only the ordinates on the positive y side of the fin chord plane are specified. The card identification, FINORD, is given in columns 73-80.

Canard data cards.- If the canard (or horizontal tail) airfoil is symmetrical, exactly three cards are used to describe the canard, and the input is given in the same manner as for the fin (table I(a)). If, however, the canard airfoil is not symmetrical (indicated by a negative value of NCANOR on the first data input card for this case), a fourth canard data input card will be required to give the lower ordinates. The information presented on the first canard data input card is as follows:

[redacted]

Columns	Description
1-7	x-ordinate of the inboard airfoil leading edge
8-14	y-ordinate of the inboard airfoil leading edge
15-21	z-ordinate of the inboard airfoil leading edge
22-28	chord length of the inboard airfoil
29-35	x-ordinate of the outboard airfoil leading edge
36-42	y-ordinate of the outboard airfoil leading edge
43-49	z-ordinate of the outboard airfoil leading edge
50-56	chord length of the outboard airfoil
73-80	the card identification, CANORG

The second canard data input card (table I(a)) contains up to 10 percent-chord locations (exactly NCANOR of them) at which the canard airfoil ordinates are to be specified. The card is identified in columns 73-80 as XCAN.

The third canard data input card contains the upper ordinates of the canard airfoil, expressed as a percent of the chord length. This card is identified in columns 73-80 as CANORD. If the canard airfoil is not symmetrical, the lower ordinates are presented on a second CANORD card. As in the case for the wing, the program expects both upper and lower ordinates to be punched as positive percent-of-chord values.

PROGRAM AND SUBROUTINE LISTING

The IBM 7090 electronic data processing system main frame, the input tape unit logical 5, and the output tape unit logical 6 are the on-line components used. A very minor use is also made of the on-line printer. The input data must be transferred from the punched data cards onto tape by an off-line card-to-tape machine.

The program, as initially developed by the Boeing Company, has been slightly modified in order to achieve compatibility with the Langley IBM 7090 data processing system. A complete FORTRAN listing of the source program and the subroutines which are not included on the standard FORTRAN II library tape as they have been used at the Langley Research Center follows.

```
CP7120                       ZERO-LIFT WAVE DRAG, ENTIRE CONFIGURATION
C
      COMMON S
C
      DIMENSION ABC(12),ABCD(14)
C
      DIMENSION CANMAX(2,2),CANORD(2,3,10),CANORG(2,4),DRAGTH(37),
     1FINORD(2,3,10),FINORG(2,4),FUSARD(30),FUSRAD(30),JJ(7),
     2ORDMAX(10,2),P(8,3),PODORD(8,30),PODORG(8,3),R(49),RP(101),
     3RX(101),S(6,51,37),SF(49),SI(101),WAFORD(10,3,30),WAFORG(10,4),
     4XAF(30),XCAN(10),XF(49),XFIN(10),XFUS(30),XI(101),XP(101),
     5XPOD(8,30),XXA(37),XXB(37)
C
      DIMENSION W(10,4)
      EQUIVALENCE (W,WAFORG)
C
      ACOSF(X)=ARTNQF(SQRTF(1.-X**2),X)
      IF ACCUMULATOR OVERFLOW 1,1
B   1 PI=202622077325
      KEY=0
      NCASE=0
C
C                  DATA INPUT SECTION
C
    5 READ INPUT TAPE 5,10,MACH,NX,NTHETA,NWAF,NWAFOR,NFUSOR,NPOD,NPODOR
     X,NFIN,NFINOR,NCANOR,J1,J2,J3,J4,J5,J6,J7
   10 FORMAT(18I4)
      IF (MACH)990,990,15
   15 IF (NTHETA) 990,990,20
   20 IF (J7-9999) 25,35,25
   25 WRITE OUTPUT TAPE 6,30
   30 FORMAT(48H1DECK-STACKING ERROR -- I CANNOT GO ON LIKE THIS)
      GO TO 990
   35 READ INPUT TAPE 5,40,ABC
   40 FORMAT(12A6)
      JJ(1)=J1
      JJ(2)=J2
      JJ(3)=J3
      JJ(4)=J4
      JJ(5)=J5
      JJ(6)=J6
      JJ(7)=J7
      NCASE=NCASE+1
      WRITE OUTPUT TAPE 6,41,NCASE
   41 FORMAT(1H124X19HINPUT DATA FOR CASEI3)
      XMACH=FLOATF(MACH)/1000.
   45 FORMAT(10 F7.0)
```

```
      NREC=2
      IF (J1-1) 68,50,68
 50   N=XABSF(NWAFOR)
      READ INPUT TAPE 5,45,(XAF(I),I=1,N)
      NREC=NREC+(N+9)/10
      DO 56 I=1,NWAF
 56   READ INPUT TAPE 5,45,(WAFORG(I,J),J=1,4)
      NREC=NREC+NWAF
      L=1
      IF (NWAFOR) 58,58,60
 58   L=2
 60   DO 65 I=1,NWAF
      DO 65 J=1,L
      READ INPUT TAPE 5,45,(WAFORD(I,J,K),K=1,N)
      IF (NWAFOR) 65,65,62
 62   DO 64 K=1,N
 64   WAFORD(I,2,K)=WAFORD(I,1,K)
 65   CONTINUE
      NREC=NREC+NWAF*L*((N+9)/10)
 68   IF (J2-1) 85,71,85
 71   N=NFUSOR
      READ INPUT TAPE 5,45,(XFUS(I),I=1,N)
      READ INPUT TAPE 5,45,(FUSARD(I),I=1,N)
      NREC=NREC+2*((N+9)/10)
      DO 80 I=1,N
 80   FUSRAD(I)=SQRTF(FUSARD(I)/PI)
 85   IF (J3-1) 100,90,100
 90   READ INPUT TAPE 5,45,((PODORG(I,J),J=1,3),I=1,NPOD)
      NREC=NREC+(NPOD+2)/3
      N=NPODOR
      DO 97 I=1,NPOD
      READ INPUT TAPE 5,45,(XPOD(I,J),J=1,N)
      READ INPUT TAPE 5,45,(PODORD(I,J),J=1,N)
 97   NREC=NREC+2*((N+9)/10)
100   IF (J4-1) 110,105,110
105   READ INPUT TAPE 5,45,((FINORG(I,J),J=1,4),I=1,2)
      N=NFINOR
      READ INPUT TAPE 5,45,(XFIN(I),I=1,N)
      READ INPUT TAPE 5,45,(FINORD(1,1,J),J=1,N)
      NREC=NREC+3
110   IF (J5-1) 124,115,124
115   READ INPUT TAPE 5,45,((CANORG(I,J),J=1,4),I=1,2)
      N=XABSF(NCANOR)
      READ INPUT TAPE 5,45,(XCAN(I),I=1,N)
      NREC=NREC+2
      L=1
      IF (NCANOR) 116,116,118
116   L=2
118   DO 120 I=1,L
```

[redacted]

```
      READ INPUT TAPE 5,45,(CANORD(1,I,J),J=1,N)
  120 NREC=NREC+1
      IF (NCANOR) 124,124,122
  122 DO 123 J=1,N
  123 CANORD(1,2,J)=CANORD(1,1,J)
  124 DO 125 I=1,NREC
  125 BACKSPACE 5
      DO 131 I=1,NREC
      READ INPUT TAPE 5,126,ABCD
  126 FORMAT(13A6,A2)
      IF (I-3) 127,127,129
  127 WRITE OUTPUT TAPE 6,128
  128 FORMAT(1H )
  129 WRITE OUTPUT TAPE 6,130,ABCD
  130 FORMAT(1H 13A6,A2)
  131 CONTINUE
      IF (XMACH-1.) 133,132,140
  132 XMACH=1.000001
      GO TO 140
  133 WRITE OUTPUT TAPE 6,232,ABC
      WRITE OUTPUT TAPE 6,135,NCASE,XMACH
  135 FORMAT(14H0       CASE NO.I3,13H, MACH NO. = F6.4)
      GO TO 5
  140 BETA=SQRTF(XMACH**2-1.)
      NWAFOR=XABSF(NWAFOR)
      NCANOR=XABSF(NCANOR)
      N=XMAXOF(NFUSOR,1)
      XX=XFUS(N)
C
C           TEST FOR CONVEX LEADING, TRAILING EDGES
C
  190 IF (J1-1) 800,191,800
  191 KATE=0
      IF (NWAF-2) 199,199,192
  192 N=NWAF-1
      DXA=W(NWAF,1)-W(1,1)
      DXB=DXA+W(NWAF,4)-W(1,4)
      DY=W(NWAF,2)-W(1,2)
      DO 195 I=2,N
      IF((W-W(I,1))*DY+(W(I,2)-W(1,2))*DXA) 194,194,193
  193 KATE=1
      GO TO 199
  194 IF((W(I,1)+W(I,4)-W(1,1)-W(1,4))*DY-
     1(W(I,2)-W(1,2))*DXB) 195,195,193
  195 CONTINUE
C
C           COMPUTE VOLUME OF EXTERNAL WING
```

```
C
  199 V=0.
      DO 205 I=2,NWAF
      DY=WAFORG(I,2)-WAFORG(I-1,2)
      E1=.01*WAFORG(I-1,4)
      E2=.01*WAFORG(I,4)
      DO 200 J=2,NWAFOR
      DX=       XAF(J)-XAF(J-1)
      DX1=DX*E1
      DX2=DX*E2
      DZ1=(WAFORD(I-1,1,J-1)+WAFORD(I-1,2,J-1)+WAFORD(I-1,1,J)+WAFORD(I-
     X1,2,J))*E1
      DZ2=(WAFORD(I,1,J-1)+WAFORD(I,2,J-1)+WAFORD(I,1,J)+WAFORD(I,2,J))*
     XE2
  200 V=V+DY*(DX1*(2.*DZ1+DZ2)+DX2*(DZ1+2.*DZ2))/6.
  205 CONTINUE
      IF (J6) 790,208,208
C
C           TRANSFORM WING COORDINATES FROM PCT-CHORD TO ACTUAL UNITS
C           OF LENGTH, REFERRED TO COMMON ORIGIN OF PROBLEM. COMPUTE
C           MAXIMUM ORDINATE OF EACH AIRFOIL.
C
  208 DO 215 I=1,NWAF
      E=.01*WAFORG(I,4)
      E3=WAFORG(I,3)
      DO 210 J=1,NWAFOR
      WAFORD(I,1,J)=E*WAFORD(I,1,J)+E3
      WAFORD(I,2,J)=-E*WAFORD(I,2,J)+E3
  210 WAFORD(I,3,J)=WAFORG(I,1)+E*XAF(J)
  215 CONTINUE
      DO 219 I=1,NWAF
      DO 216 J=2,NWAFOR
      K=J-1
      IF (WAFORD(I,1,K)-WAFORD(I,1,J)) 216,217,217
  216 CONTINUE
  217 ORDMAX(I,1)=WAFORD(I,1,K)
      DO 218 J=2,NWAFOR
      K=J-1
      IF (WAFORD(I,2,K)-WAFORD(I,2,J)) 219,219,218
  218 CONTINUE
  219 ORDMAX(I,2)=WAFORD(I,2,K)
  800 IF (J4-1) 825,805,825
  805 DO 815 I=1,2
      J=3-I
      E=.01*FINORG(J,4)
      E2=FINORG(J,2)
      DO 810 K=1,NFINOR
      EE=FINORD(1,1,K)*E
      FINORD(J,1,K)=E2+EE
```

[redacted]

```
      FINORD(J,2,K)=E2-EE
  810 FINORD(J,3,K)=FINORG(J,1)+E*XFIN(K)
  815 CONTINUE
      FINMX1=0.
      FINMX2=0.
      DO 820 K=1,NFINOR
      FINMX1=MAX1F(FINMX1,FINORD(1,1,K))
  820 FINMX2=MAX1F(FINMX2,FINORD(2,1,K))
      FINTH1=2.*(FINMX1-FINORG(1,2))
      FINTH2=2.*(FINMX2-FINORG(2,2))
  825 IF (J5-1) 220,830,220
  830 DO 840 K=1,2
      I=3-K
      E=.01*CANORG(I,4)
      E3=CANORG(I,3)
      DO 835 J=1,NCANOR
      CANORD(I,1,J)=E*CANORD(1,1,J)+E3
      CANORD(I,2,J)=-E*CANORD(1,2,J)+E3
  835 CANORD(I,3,J)=CANORG(I,1)+E*XCAN(J)
  840 CONTINUE
      DO 860 I=1,2
      DO 845 J=2,NCANOR
      K=J-1
      IF (CANORD(I,1,K)-CANORD(I,1,J)) 845,850,850
  845 CONTINUE
  850 CANMAX(I,1)=CANORD(I,1,K)
      DO 855 J=2,NCANOR
      K=J-1
      IF (CANORD(I,2,K)-CANORD(I,2,J)) 860,860,855
  855 CONTINUE
  860 CANMAX(I,2)=CANORD(I,2,K)
C
C            FIT EMINTON-LORD OPTIMUM AREA DISTRIBUTION OF FUSELAGE
C
  220 IF (J2-1) 290,225,290
  225 N=NFUSOR
      ELL=XX
      SN=FUSARD(1)
      SB=FUSARD(N)
      NN=N-2
      DO 230 I=1,NN
      XF(I)=XFUS(I+1)/ELL
  230 SF(I)=FUSARD(I+1)
      K=1
      CALL EMLORD(ELL,SN,SB,NN,XF,SF,FDRAG,R,K,L)
      WRITE OUTPUT TAPE 6,232,ABC
  232 FORMAT(1H15X12A6)
```

```
      GO TO (245,235,235),L
  235 WRITE OUTPUT TAPE 6,240,NCASE,L
  240 FORMAT(15H0      CASE NO,I3,17H ERROR RETURN NO,I2,28H FROM EMLORD
     X FIT TO FUSELAGE)
      GO TO 5
  245 WRITE OUTPUT TAPE 6,250,FDRAG
  250 FORMAT(47H0                 FUSELAGE AREA DISTRIBUTION (D/Q =F9.5,1H)/
     X/)
      XI(1)=0.
      XI(101)=XFUS(N)
      SI(1)=SN
      SI(101)=SB
      DO 275 I=2,100
      Z=I-1
      EX=.01*Z
      XI(I)=EX*ELL
      SUM=0.
      DO 270 J=1,NN
      Y=XF(J)
      E=(EX-Y)**2
      E1=EX+Y-2.*EX*Y
      E2=2.*SQRTF(EX*Y*(1.-EX)*(1.-Y))
      IF (E-1.E-8) 265,265,260
  260 E3=.5*E*LOGF((E1-E2)/(E1+E2))+E1*E2
      GO TO 270
  265 E3=E1*E2
  270 SUM=SUM+E3*R(J)
      E4=(ACOSF(1.-2.*EX)-(2.-4.*EX)*SQRTF(EX-EX**2))/PI
      IF DIVIDE CHECK 275,275
  275 SI(I)=SN+(SB-SN)*E4+SUM
      DO 282 M=1,51
      N1=M-1
      N2=N1+50
      WRITE OUTPUT TAPE 6,280,N1,XI(M),SI(M),N2,XI(M+50),SI(M+50)
  280 FORMAT(I9,2F11.4,I13,2F11.4)
  282 CONTINUE
      DO 285 I=1,101
  285 RX(I)=SQRTF(SI(I)/PI)
C
C           SELECT X AND THETA
C
  290 XN=NX
      NN=NX+1
      XL=NTHETA
      LL=NTHETA+1
      DELTH=PI/XL
      A=1.
C
      DO 685 K=1,LL
```

[redacted]

```
      IF (J6) 1700,1490,1700
 1700 IF (K-(LL+1)/2) 1490,1490,1710
 1710 N=LL+1-K
      XXA(K)=XXA(N)
      XXB(K)=XXB(N)
      DO 1720 J=1,NN
      DO 1720 I=1,6
 1720 S(I,J,K)=S(I,J,N)
      GO TO 685
 1490 E=K-1
      THETA=-.5*PI+E*DELTH
      COSTH=COSF(THETA)
      SINTH=SINF(THETA)
      B=-BETA*COSTH
      C=-BETA*SINTH
C
C           COMPUTE END-POINTS OF SEGMENT OF X-AXIS OUTSIDE OF WHICH
C           S(X,THETA) IS ZERO FOR CURRENT VALUE OF THETA
C
      XA=0.
      XB=0.
      IF (J2) 1505,1505,1500
 1500 XB=XX
 1505 IF (J1) 1535,1535,1510
 1510 DO 1530 I=1,NWAF
      IF (I-1) 1525,1515,1525
 1515 IF (J2) 1520,1520,1525
 1520 XA=WAFORG(1,1)+B*WAFORG(1,2)+C*WAFORG(1,3)
      XB=WAFORG(1,1)+WAFORG(1,4)-B*WAFORG(1,2)+C*WAFORG(1,3)
      GO TO 1530
 1525 XA=MIN1F(XA,WAFORG(I,1)+B*WAFORG(I,2)+C*WAFORG(I,3))
      XB=MAX1F(XB,WAFORG(I,1)+WAFORG(I,4)-B*WAFORG(I,2)+C*WAFORG(I,3))
 1530 CONTINUE
 1535 IF (J3) 1570,1570,1540
 1540 DO 1565 I=1,NPOD
      DO 1560 J=1,NPODOR
      XA=MIN1F(XA,PODORG(I,1)+XPOD(I,J)+B*(PODORG(I,2)+COSTH*PODORD(I,J)
     X)+C*(PODORG(I,3)+SINTH*PODORD(I,J)))
      XB=MAX1F(XB,PODORG(I,1)+XPOD(I,J)-B*(PODORG(I,2)+COSTH*PODORD(I,J)
     X)+C*(PODORG(I,3)-SINTH*PODORD(I,J)))
 1560 CONTINUE
 1565 CONTINUE
 1570 IF (J4) 1610,1610,1575
 1575 DO 1605 I=1,2
      IF (I-1) 1600,1580,1600
 1580 IF (J1) 1585,1585,1600
 1585 IF (J2) 1590,1590,1600
```

 [redacted]

```
 1590 IF (J3) 1595,1595,1600
 1595 XA=FINORG(1,1)+B*FINORG(1,2)+C*FINORG(1,3)
      XB=FINORG(1,1)+FINORG(1,4)-B*FINORG(1,2)+C*FINORG(1,3)
      GO TO 1605
 1600 XA=MIN1F(XA,FINORG(I,1)+B*FINORG(I,2)+C*FINORG(I,3))
      XB=MAX1F(XB,FINORG(I,1)+FINORG(I,4)-B*FINORG(I,2)+C*FINORG(I,3))
 1605 CONTINUE
 1610 IF (J5) 1655,1655,1615
 1615 DO 1650 I=1,2
      IF (I-1) 1645,1620,1645
 1620 IF (J1) 1625,1625,1645
 1625 IF (J2) 1630,1630,1645
 1630 IF (J3) 1635,1635,1645
 1635 IF (J4) 1640,1640,1645
 1640 XA=CANORG(1,1)+B*CANORG(1,2)+C*CANORG(1,3)
      XB=CANORG(1,1)+CANORG(1,4)-B*CANORG(1,2)+C*CANORG(1,3)
      GO TO 1650
 1645 XA=MIN1F(XA,CANORG(I,1)+B*CANORG(I,2)+C*CANORG(I,3))
      XB=MAX1F(XB,CANORG(I,1)+CANORG(I,4)-B*CANORG(I,2)+C*CANORG(I,3))
 1650 CONTINUE
 1655 XXA(K)=XA
      XXB(K)=XB
      DELX=(XB-XA)/XN
      DDELX=.0001*DELX
C
      DO 680   J=1,NN
      E=J-1
      X=XA+E*DELX
      IF (J-1) 294,292,294
  292 X=X+DDELX
      GO TO 298
  294 IF (J-NN) 298,296,298
  296 X=X-DDELX
  298 SUM=0.
      IF (J1) 410,410,300
C
C            COMPUTE S(X,THETA) FOR WING
C
  300 SUM=0.
      DO 405 M=1,2
      EE=(-1.)**(M-1)
      N=NWAF
      P(1,1)=WAFORG(1,1)
      P(2,1)=WAFORG(N,1)
      P(3,1)=WAFORG(1,4)+P(1,1)
      P(4,1)=WAFORG(N,4)+P(2,1)
      P(1,2)=WAFORG(1,2)*EE
      P(2,2)=WAFORG(N,2)*EE
      P(3,2)=P(1,2).
```

```
      P(4,2)=P(2,2)
      P(1,3)=ORDMAX(1,2)
      P(2,3)=ORDMAX(N,2)
      P(3,3)=P(1,3)
      P(4,3)=P(2,3)
      P(5,3)=ORDMAX(1,1)
      P(6,3)=ORDMAX(N,1)
      P(7,3)=P(5,3)
      P(8,3)=P(6,3)
      DO 305 L=1,4
      P(L+4,1)=P(L,1)
  305 P(L+4,2)=P(L,2)
      IF (SWING(A,B,C,X,P)) 306,306,310
  306 IF (KATE) 405,405,310
  310 DO 400 L=2,NWAF
      IF (NWAF-2) 312,340,312
  312 IF (L-2) 315,330,315
  315 P(1,1)=WAFORG(L-1,1)
      P(3,1)=WAFORG(L-1,4)+P(1,1)
      P(5,1)=P(1,1)
      P(7,1)=P(3,1)
      DO 320 I=1,7,2
  320 P(I,2)=P(I+1,2)
      P(1,3)=ORDMAX(L-1,2)
      P(3,3)=P(1,3)
      P(5,3)=ORDMAX(L-1,1)
      P(7,3)=P(5,3)
  330 P(2,1)=WAFORG(L,1)
      P(4,1)=WAFORG(L,4)+P(2,1)
      P(2,2)=WAFORG(L,2)*EE
      P(4,2)=P(2,2)
      P(2,3)=ORDMAX(L,2)
      P(4,3)=P(2,3)
      P(6,3)=ORDMAX(L,1)
      P(8,3)=P(6,3)
      DO 335 I=2,4,2
      P(I+4,1)=P(I,1)
  335 P(I+4,2)=P(I,2)
      IF (SWING(A,B,C,X,P)) 400,400,340
  340 NU=0
      DO 395 N=2,NWAFOR
      IF (N-2)345,370,345
  345 DO 365 I=1,6
      IF (I-3) 355,365,350
  350 IF (I-4) 365,365,355
  355 DO 360 INK=1,3
  360 P(I,INK)=P(I+2,INK)
```

```
 365 CONTINUE
 370 P(3,1)=WAFORD(L-1,3,N)
     P(4,1)=WAFORD(L,3,N)
     P(7,1)=P(3,1)
     P(8,1)=P(4,1)
     P(3,2)=P(1,2)
     P(4,2)=P(2,2)
     P(7,2)=P(5,2)
     P(8,2)=P(6,2)
     P(3,3)=WAFORD(L-1,2,N)
     P(4,3)=WAFORD(L,2,N)
     P(7,3)=WAFORD(L-1,1,N)
     P(8,3)=WAFORD(L,1,N)
     IF (N-2) 380,375,380
 375 P(1,3)=WAFORD(L-1,2,1)
     P(2,3)=WAFORD(L,2,1)
     P(5,3)=WAFORD(L-1,1,1)
     P(6,3)=WAFORD(L,1,1)
 380 E=SWING(A,B,C,X,P)
     IF (J6-40) 384,384,381
 381 IF (ABSF(THETA)-.01) 382,382,384
 382 WRITE OUTPUT TAPE 6,383,X,THETA,M,L,N,E
 383 FORMAT(4H0 X=F8.2,7H THETA=F6.2,3H M=I1,3H L=I2,3H N=I2,3H E=F8.2/
    X/)
     WRITE OUTPUT TAPE 6,1383,((P(I,INK),I=1,8),INK=1,3)
1383 FORMAT(20X,8F10.2)
 384 SUM=E+SUM
     IF (E) 385,390,385
 385 NU=1
     GO TO 395
 390 IF (NU) 400,395,400
 395 CONTINUE
 400 CONTINUE
 405 CONTINUE
     S(1,J,K)=SUM
 410 IF (J2) 435,435,415
C
C           COMPUTE S(X,THETA) FOR FUSELAGE
C
 415 N=101
     MU=0
     E=0.
     CALL SPOD(N,BETA,X,THETA,XI,RX,E,E,E,AREA,MU)
     IF (MU-1) 420,430,420
 420 WRITE OUTPUT TAPE 6,425,NCASE
 425 FORMAT(15H0        CASE NO.I3,45H ERROR RETURN FROM SPOD SUBROUTINE
    X(FUSELAGE))
     GO TO 5
 430 S(2,J,K)=AREA
```

```
  435 IF (J3) 470,470,440
C
C          COMPUTE S(X,THETA) FOR NACELLES
C
  440 SUM=0.
      DO 465 L=1,NPOD
      XZERO=PODORG(L,1)
      ZZERO=PODORG(L,3)
      DO 445 N=1,NPODOR
      XP(N)=XPOD(L,N)
  445 RP(N)=PODORD(L,N)
      DO 460 M=1,2
      EE=(-1.)**(M-1)
      YZERO=PODORG(L,2)*EE
      MU=0
      CALL SPOD(NPODOR,BETA,X,THETA,XP,RP,XZERO,YZERO,ZZERO,AREA,MU)
      IF (MU-1) 450,460,450
  450 WRITE OUTPUT TAPE 6,455,NCASE,L
  455 FORMAT(15H0       CASE NO.I3,22H ERROR RETURN, POD NO.I2)
      GO TO 5
  460 SUM=SUM+AREA
  465 CONTINUE
      S(3,J,K)=SUM
  470 IF (J4) 575,575,475
C
C          COMPUTE S(X,THETA) FOR FINS
C
  475 IF (NFIN-1) 575,480,480
  480 SUM=0.
      DO 570 L=1,NFIN
      EE=(-1.)**(L-1)
      P(1,1)=FINORG(1,1)
      P(3,1)=FINORG(1,4)+P(1,1)
      P(5,1)=FINORG(2,1)
      P(7,1)=FINORG(2,4)+P(5,1)
      P(1,2)=FINMX1*EE
      P(2,2)=(FINMX1-FINTH1)*EE
      P(5,2)=FINMX2*EE
      P(6,2)=(FINMX2-FINTH2)*EE
      P(1,3)=FINORG(1,3)
      P(5,3)=FINORG(2,3)
      DO 485 M=1,7,2
  485 P(M+1,1)=P(M,1)
      P(3,2)=P(1,2)
      P(4,2)=P(2,2)
      P(7,2)=P(5,2)
      P(8,2)=P(6,2)
```

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      DO 505 M=2,4
      P(M,3)=P(1,3)
  505 P(M+4,3)=P(5,3)
      IF (SWING(A,B,C,X,P)) 570,570,510
  510 NU=0
      DO 565 M=2,NFINOR
      IF (M-2) 515,540,515
  515 DO 535 N=1,6
      IF (N-3) 525,535,520
  520 IF (N-4) 525,535,525
  525 DO 530 I=1,3
  530 P(N,I)=P(N+2,I)
  535 CONTINUE
  540 P(3,1)=FINORD(1,3,M)
      P(4,1)=P(3,1)
      P(7,1)=FINORD(2,3,M)
      P(8,1)=P(7,1)
      P(3,2)=FINORD(1,1,M)*EE
      P(4,2)=FINORD(1,2,M)*EE
      P(7,2)=FINORD(2,1,M)*EE
      P(8,2)=FINORD(2,2,M)*EE
      P(3,3)=FINORG(1,3)
      P(4,3)=P(3,3)
      P(7,3)=FINORG(2,3)
      P(8,3)=P(7,3)
      IF (M-2) 550,545,550
  545 P(1,2)=FINORD(1,1,1)*EE
      P(2,2)=FINORD(1,2,1)*EE
      P(5,2)=FINORD(2,1,1)*EE
      P(6,2)=FINORD(2,2,1)*EE
  550 E=SWING(A,B,C,X,P)
      SUM=E+SUM
      IF (E) 555,560,555
  555 NU=1
      GO TO 565
  560 IF (NU) 570,565,570
  565 CONTINUE
  570 CONTINUE
      S(4,J,K)=SUM
  575 IF (J5) 672,672,580
C
C           COMPUTE S(X,THETA) FOR CANARDS
C
  580 SUM=0.
      DO 670 L=1,2
      EE=(-1.)**(L-1)
      P(1,1)=CANORG(1,1)
      P(2,1)=CANORG(2,1)
      P(3,1)=CANORG(1,4)+P(1,1)
```

```
      P(4,1)=CANORG(2,4)+P(2,1)
      P(1,2)=CANORG(1,2)*EE
      P(2,2)=CANORG(2,2)*EE
      P(1,3)=CANMAX(1,2)
      P(2,3)=CANMAX(2,2)
      P(5,3)=CANMAX(1,1)
      P(6,3)=CANMAX(2,1)
      DO 585 M=1,4
  585 P(M+4,1)=P(M,1)
      DO 590 N=1,2
      DO 590 M=2,6,2
      I=M+N
  590 P(I,2)=P(N,2)
      DO 605 N=1,6
      IF (N-3) 600,605,595
  595 IF (N-4) 600,605,600
  600 P(N+2,3)=P(N,3)
  605 CONTINUE
      IF (SWING(A,B,C,X,P)) 670,670,610
  610 NU=0
      DO 665 M=2,NCANOR
      IF (M-2) 615,640,615
  615 DO 635 N=1,6
      IF (N-3) 625,635,620
  620 IF (N-4) 625,635,625
  625 DO 630 I=1,3
  630 P(N,I)=P(N+2,I)
  635 CONTINUE
  640 P(3,1)=CANORD(1,3,M)
      P(4,1)=CANORD(2,3,M)
      P(7,1)=P(3,1)
      P(8,1)=P(4,1)
      P(3,2)=P(1,2)
      P(4,2)=P(2,2)
      P(7,2)=P(5,2)
      P(8,2)=P(6,2)
      P(3,3)=CANORD(1,2,M)
      P(4,3)=CANORD(2,2,M)
      P(7,3)=CANORD(1,1,M)
      P(8,3)=CANORD(2,1,M)
      IF (M-2) 650,645,650
  645 P(1,3)=CANORD(1,2,1)
      P(2,3)=CANORD(2,2,1)
      P(5,3)=CANORD(1,1,1)
      P(6,3)=CANORD(2,1,1)
  650 E=SWING(A,B,C,X,P)
      SUM=E+SUM
```

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```
      IF (E) 655,660,655
  655 NU=1
      GO TO 665
  660 IF (NU) 670,665,670
  665 CONTINUE
  670 CONTINUE
      S(5,J,K)=SUM
  672 SUM=0.
      DO 675 I=1,5
      IF (JJ(I)) 673,675,673
  673 SUM=SUM+S(I,J,K)
  675 CONTINUE
      S(6,J,K)=SUM
  680 CONTINUE
  685 CONTINUE
C
C           COMPUTE DRAG OF AREA DISTRIBUTION CORRESPONDING TO
C           A PARTICULAR VALUE OF THETA
C
      NU=NX-1
      DO 690 J=1,NU
      E=J
  690 XF(J)=E/XN
      SUM=0.
      KK=LL
      DO 745 K=1,LL
      IF (J6) 691,694,691
  691 IF (K-(LL+1)/2) 694,694,692
  692 N=LL+1-K
      DRAGTH(K)=DRAGTH(N)
      GO TO 712
  694 E=K-1
      THETA=-.5*PI+E*DELTH
      SN=S(6,1,K)
      SB=S(6,NN,K)
      DO 695 J=1,NU
  695 SF(J)=S(6,J+1,K)
      ELL=XXB(K)-XXA(K)
      CALL EMLORD(ELL,SN,SB,NU,XF,SF,E,R,K,L)
      GO TO (710,700,700),L
  700 WRITE OUTPUT TAPE 6,705,NCASE,L,THETA
  705 FORMAT(15H0       CASE NO.I3,17H ERROR RETURN NO.I2,31H FROM EMLORD
     X SUBROUTINE, THETA=F7.4)
      KK=K
      GO TO 748
  710 DRAGTH(K)=E
C
C           COMPUTE DRAG OF ENTIRE AIRCRAFT
C
```

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```
712 IF(XMODF((K-1),NTHETA)) 720,715,720
715 E=14.
    GO TO 740
720 IF (XMODF((K-1),4)) 730,725,730
725 E=28.
    GO TO 740
730 E=64.
    IF(XMODF(K,2)) 735,740,735
735 E=24.
740 SUM=SUM+E*DRAGTH(K)
745 CONTINUE
    DRAG=SUM/(45.*XL)
748 WRITE OUTPUT TAPE 6,232,ABC
    WRITE OUTPUT TAPE 6,750
750 FORMAT(57H0                    D/Q ASSOCIATED WITH VARIOUS VALUES OF THET
   XA//)
    WRITE OUTPUT TAPE 6,755
755 FORMAT(55H0                    N                    THETA                    D/Q/
   X/)
    J=XMINOF(KK,LL)
    DO 765 K=1,J
    N=K-1
    E=K-1
    THETA=(E*DELTH-.5*PI)*180./PI
    WRITE OUTPUT TAPE 6,760,N,THETA,DRAGTH(K)
760 FORMAT(I15,F20.3,F23.5)
765 CONTINUE
    IF (KK-LL) 780,770,780
770 WRITE OUTPUT TAPE 6,775,DRAG
775 FORMAT(1H0 17X 26HD/Q FOR ENTIRE AIRCRAFT = F12.5)
C
C           COMPUTE VOLUME OF WING EQUIVALENT BODY
C
780 IF (J1) 781,795,781
781 SUM1=0.
    DO 789 K=1,LL
    SUM2=0.
    DO 783 J=1,NN
    E=FLOATF(2*XMODF(J-1,2)+2)
    IF (XMODF(J-1,NX)) 783,782,783
782 E=1.
783 SUM2=SUM2+E*S(1,J,K)
    E2=SUM2*(XXB(K)-XXA(K))
    IF (XMODF(K-1,NTHETA)) 785,784,785
784 E=14.
    GO TO 789
785 IF (XMODF(K-1,4)) 787,786,787
```


[redacted]

```
786 E=28.
    GO TO 789
787 E=64.
    IF (XMODF(K,2)) 788,789,788
788 E=24.
789 SUM1=SUM1+E*E2
    VWING=SUM1/(135.*XN*XL)
790 WRITE OUTPUT TAPE 6,791,V
791 FORMAT(44H0                          VOLUME OF ENTIRE WING = 1PE12.5)
    IF (J6) 5,792,792
792 WRITE OUTPUT TAPE 6,793,VWING
793 FORMAT(44H0                    VOLUME OF EQUIVALENT BODY = 1PE12.5)
795 WRITE OUTPUT TAPE 6,796,ABC
796 FORMAT(1H123X12A6)
    IF (J1) 900,900,910
900 CALL TABOUT(JJ,S,XXA,XXB,NX,NTHETA)
    GO TO 5
910 IF (J2) 900,900,920
920 IF (J3+J4+J5) 900,930,900
930 DO 940 I=1,NFUSOR
    IF (FUSARD(I)) 940,940,900
940 CONTINUE
    CALL WEBOUT(JJ,S,XXA,XXB,NX,NTHETA)
    GO TO 5
990 IF DIVIDE CHECK 991,993
991 WRITE OUTPUT TAPE 6,992
992 FORMAT(1H1/42H0DIVIDE CHECK ON WHEN SUCCESS STOP REACHED)
    GO TO 995
993 WRITE OUTPUT TAPE 6,994
994 FORMAT(1H1/21H0SUCCESS STOP REACHED)
995 CALL EXIT
    GO TO 995
    END
```

[redacted]

```
      SUBROUTINE EMLORD(ELL,SN,SB,NN,XX,SS,DRAG,R,K,L)
C
      DIMENSION AA(49),B(49,49),C(49),P(49,49),Q(49),R(49),SS(49),XX(49)
C
      ACOSF(X)=ARTNQF(SQRTF(1.-X**2),X)
      PI=3.141592654
      DRAG=0.
      IF (K-1) 390,328,390
  328 DO 341 N=1,NN
      X=XX(N)
      Q(N)=(ACOSF(1.-2.*X)-(2.-4.*X)*SQRTF(X-X**2))/PI
      DO 338 M=N,NN
      Y=XX(M)
      IF (M-N) 330,331,330
  330 B(M,N)=0.
      GO TO 332
  331 B(M,N)=1.
  332 E=(X-Y)**2
      E1=X+Y-2.*X*Y
      E2=2.*SQRTF(X*Y*(1.-X)*(1.-Y))
      IF (E) 336,337,336
  336 P(M,N)=.5*E*LOGF((E1-F2)/(E1+E2))+E1*E2
      GO TO 338
  337 P(M,N)=E1*E2
  338 CONTINUE
      NK=N-1
      IF (NK) 341,341,339
  339 DO 340 M=1,NK
      E=P(N,M)
      P(M,N)=E
  340 B(M,N)=0.
  341 CONTINUE
      D=0.
      L=XSIMEQF(49,NN,NN,P,B,D,AA)
      GO TO (390,399,399),L
  390 DO 392 N=1,NN
  392 C(N)=SS(N)-SN-(SB-SN)*Q(N)
      DO 394 M=1,NN
      SUM=0.
      DO 393 N=1,NN
  393 SUM = SUM+P(M,N)*C(N)
  394 R(M)=SUM
      SUM=0.
      DO 395 M= 1,NN
  395 SUM=SUM+R(M)*C(M)
      DRAG=(4.*(SB-SN)**2/PI+SUM*PI)/ELL**2
  399 RETURN
      END
```


[redacted]

```
      FUNCTION INLAP(A,B,C,D,P,P1,P2)
C
      DIMENSION P(3),P1(3),P2(3)
C
      EPS=1.E-6
      L=1
      E1=A*P1(1)+B*P1(2)+C*P1(3)-D
      IF (ABSF(E1)-EPS) 10,10,20
   10 L=2
   20 E2=A*P2(1)+B*P2(2)+C*P2(3)-D
      IF (ABSF(E2)-EPS) 30,30,70
   30 GO TO (40,60),L
   40 DO 50 I=1,3
   50 P(I)=P2(I)
      M=1
      GO TO 150
   60 M=2
      GO TO 150
   70 GO TO (100,80),L
   80 DO 90 I=1,3
   90 P(I)=P1(I)
      M=1
      GO TO 150
  100 DX=P2(1)-P1(1)
      DY=P2(2)-P1(2)
      DZ=P2(3)-P1(3)
      E3=A*DX+B*DY+C*DZ
      IF (ABSF(E3)-EPS) 110,110,120
  110 M=3
      GO TO 150
  120 T=-E1/E3
      P(1)=P1(1)+T*DX
      P(2)=P1(2)+T*DY
      P(3)=P1(3)+T*DZ
      M=1
  150 INLAP=M
C                   M = 1 --- NORMAL RETURN, PT. COORDS. STORED IN P-ARRAY
C                   M = 2 --- LINE LIES IN PLANE
C                   M = 3 --- LINE PARALLEL TO PLANE
      RETURN
      END
```

[redacted]

```
      SUBROUTINE WEBOUT(JJ,S,XXA,XXB,NX,NTHETA)
C
      DIMENSION JJ(7),S(6,51,37),XXA(37),XXB(37),TH(5,8),E(10),ELL(37),
     XZ(4)
C
      XN=NX
      NN=NX+1
      XL=NTHETA
      LL=NTHETA+1
      DO 2 K=1,LL
    2 ELL(K)=XXB(K)-XXA(K)
      DELTH=180./XL
      M=LL/5+1
      MA=XMODF(LL,5)
      IF (MA) 10,5,10
    5 MA=5
      M=M-1
   10 DO 15 L=1,M
      E1=L-1
      DO 15 K=1,5
      E2=K-1
   15 TH(K,L)=-90.+(5.*E1+E2)*DELTH
      WRITE OUTPUT TAPE 6,18
   18 FORMAT(1H0 44X 15HS(X,THETA) FOR )
   20 FORMAT(1H1 44X 15HS(X,THETA) FOR )
      WRITE OUTPUT TAPE 6,25
   25 FORMAT(1H+ 59X 15HENTIRE AIRCRAFT)
      DO 100 L=1,M
      LU=5*(L-1)
      N=5
      IF (L-M) 40,35,40
   35 N=MA
   40 N2=2*N
      WRITE OUTPUT TAPE 6,42
   42 FORMAT(1H04X7HTHETA =4(17X7HTHETA =))
      WRITE OUTPUT TAPE 6,45,(TH(K,L),K=1,N)
   45 FORMAT(1H+F18.2,4F24.2)
      IF (NX-50) 46,48,48
   46 WRITE OUTPUT TAPE 6,47
   47 FORMAT(1H )
   48 WRITE OUTPUT TAPE 6,50
   50 FORMAT(1H 6X1HX9X1HS4(13X1HX9X1HS))
      DO 65 J=1,NN
      E1=J-1
      E2=E1/XN
      DO 55 KK=1,N
      K=LU+KK
      E(2*KK-1)=XXA(K)+E2*ELL(K)
   55 E(2*KK)=S(6,J,K)
```


[redacted]

```
      WRITE OUTPUT TAPE 6,60,(E(I),I=1,N2)
   60 FORMAT(1H 2F10.3,4X2F10.3,4X2F10.3,4X2F10.3,4X2F10.3)
   65 CONTINUE
      IF (L-M) 70,100,100
   70 IF (NX-23) 75,90,90
   75 IF (NX-13) 85,85,80
   80 IF (XMODF(L,2)) 100,90,100
   85 IF (XMODF(L,3)) 100,90,100
   90 WRITE OUTPUT TAPE 6,20
      WRITE OUTPUT TAPE 6,25
  100 CONTINUE
      WRITE OUTPUT TAPE 6,110
  110 FORMAT(1H115X41HAREA DISTRIBUTION OF WING EQUIVALENT BODY/1H022X1H
     XX26X1HS//)
      Z(1)=32.
      Z(2)=14.
      Z(3)=32.
      Z(4)=12.
      DO 170 I=1,NN
      XI=I-1
      X=XXA+XI*ELL/XN
      SUM=0.
      DO 150 J=1,LL
      IF (J-1) 130,120,130
  120 E=7.
      GO TO 150
  130 IF (J-LL) 140,120,140
  140 K=XMODF(J,4)+1
      E=Z(K)
  150 SUM=SUM+E*S(1,I,J)
      E=SUM/XL/22.5
      WRITE OUTPUT TAPE 6,160,X,E(1)
  160 FORMAT(2F27.3)
  170 CONTINUE
      RETURN
      END
```

[redacted]

```
      SUBROUTINE TABOUT(JJ,S,XXA,XXB,NX,NTHETA)
C
      DIMENSION JJ(7),S(6,51,37),XXA(37),XXB(37),TH(5),E(10),ELL(37),
     XJAY(5)
C
      XN=NX
      NX1=NX+1
      N=NTHETA+1
      DO 10 K=1,N
   10 ELL(K)=XXB(K)-XXA(K)
      M=1
      IF (NTHETA-4) 40,40,20
   20 IF (JJ(6)) 30,30,60
   30 M=NTHETA/4
   40 DO 50 K=1,5
      E=K-1
      TH(K)=-90.+E*45.
   50 JAY(K)=1+(K-1)*M
      GO TO 80
   60 JAY(1)=1+NTHETA/2
      JAY(5)=NTHETA+1
      JAY(3)=(JAY(1)+JAY(5))/2
      JAY(2)=(1+JAY(1)+JAY(3))/2
      JAY(4)=(JAY(3)+JAY(5))/2
      XL=NTHETA
      DELTH=180./XL
      DO 70 K=1,5
      E=JAY(K)-1
   70 TH(K)=-90.+E*DELTH
   80 WRITE OUTPUT TAPE 6,90
   90 FORMAT(1H044X30HS(X,THETA) FOR ENTIRE AIRCRAFT)
      WRITE OUTPUT TAPE 6,100,(TH(K),K=1,5)
  100 FORMAT(1H05(5X7HTHETA =F7.2,4X))
      WRITE OUTPUT TAPE 6,110
  110 FORMAT(1H05(6X1HX10X1HS5X))
      DO 140 J=1,NX1
      E1=J-1
      E2=E1/XN
      DO 120 K=1,5
      N=JAY(K)
      E(2*K-1)=XXA(N)+E2*ELL(N)
  120 E(2*K)=S(6,J,N)
      WRITE OUTPUT TAPE 6,130,(E(I),I=1,10)
  130 FORMAT(1H 5(F10.3,F11.3,2X))
  140 CONTINUE
      RETURN
      END
```

 [redacted]

```
      FUNCTION SWING(A,B,C,D,P)
C
      DIMENSION H(100,2),P(8,3),P1(3),P2(3),R(3)
C
      SWING=0.
      EPS=.00005
      K=0
      L=0
    5 K=K+1
      N=(K-1)/4
      KK=0
      IF (K-4) 10,10,15
   10 I=2*K-1
      GO TO 40
   15 IF (K-6) 20,20,25
   20 I=K-4
      GO TO 40
   25 IF (K-8)30,30,35
   30 I=K-2
      GO TO 40
   35 I=K-8
   40 J=I+2**N
      M=3-XMODF(N+1,3)
      DO 45 KK=1,3
      P1(KK)=P(I,KK)
   45 P2(KK)=P(J,KK)
      NU=INLAP(A,B,C,D,R,P1,P2)
      GO TO (55,50,70),NU
   50 L=L+1
      H(L,1)=P(I,2)
      H(L,2)=P(I,3)
      L=L+1
      H(L,1)=P(J,2)
      H(L,2)=P(J,3)
      KK=1
      GO TO 70
   55 IF (R(M)+EPS-MIN1F(P(I,M),P(J,M))) 70,60,60
   60 IF (R(M)-EPS-MAX1F(P(I,M),P(J,M))) 65,65,70
   65 L=L+1
      H(L,1)=R(2)
      H(L,2)=R(3)
      KK=1
   70 IF (KK) 75,110,75
   75 IF (L-1) 110,110,80
   80 L1=L-1
      DO 90 LL=1,L1
      IF (ABSF(H(L,1)-H(LL,1))-EPS) 85,85,90
```

[redacted]

```
 85 IF (ABSF(H(L,2)-H(LL,2))-EPS) 105,105,90
 90 CONTINUE
    IF (K-4) 5,95,100
 95 IF (L-4) 5,120,120
100 IF (L-6) 110,120,120
105 L=L-1
110 IF (K-12) 5,115,115
115 IF (L-3) 125,120,120
120 SWING=SNGON(L,H)
125 RETURN
    END
```


[redacted]

```
      SUBROUTINE SPOD(N,BETA,EX,THETA,X,R,XZERO,YZERO,ZZERO,S,MU)
C
      DIMENSION X(101),R(101)
C
      BC=BETA*COSF(THETA)
      BS=BETA*SINF(THETA)
      PI=3.141592654
      A=EX+YZERO*BC+ZZERO*BS-XZERO
      M=16
      XM=M
      SUM=0.
      MM=-5
      EPS=10.**MM
      DO120I=1,M
      XI=I
      PHI=2.*PI*XI/XM
      DPHI=180.*PHI/PI
      T=BETA*COSF(THETA-PHI)
      IF (ABSF(T)-EPS) 5,45,45
    5 IF(A-X(1)) 10,10,15
   10 RHO=R(1)
      GO TO 85
   15 IF (A-X(N)) 25,20,20
   20 RHO=R(N)
      GO TO 85
   25 DO 30 J=2,N
      K=J
      IF (A-X(K)) 40,35,30
   30 CONTINUE
   35 RHO=R(K)
      GO TO 85
   40 RHO=R(K-1)+(R(K)-R(K-1))/(X(K)-X(K-1))*(A-X(K-1))
      GO TO 85
   45 E=1./T
      DO 75 K=1,N
      XX=X(K)-T*R(K)
      IF (A-XX) 50,50,75
   50 IF (K-1) 55,55,60
   55 RHO=R(1)
      GO TO 85
   60 D=(R(K)-R(K-1))/(X(K)-X(K-1))
      IF (D-E) 70,65,70
   65 MU=2
      GO TO 125
   70 B1=R(K-1)-D*X(K-1)
      B2=-A*E
      RHO=(B2*D-B1*E)/(D-E)
      GO TO 85
   75 CONTINUE
```

[redacted]

```
 80 RHO=R(N)
 85 IF (MU)     90,115,90
 90 IF (I-1) 105,95,105
 95 WRITE OUTPUT TAPE 6,100
100 FORMAT(56H0                    I                       PHI                          RHO
   X//)
105 WRITE OUTPUT TAPE 6,110,I,DPHI,RHO
110 FORMAT(I16,F21.3,F21.4)
115 B=1.+MODF(XI,2.)
120 SUM=SUM+B*RHO
    S=4.*PI*SUM**2/(9.*XM**2)
    MU=1
125 RETURN
    END
```

```
      FUNCTION SNGON(N,P)
C
C           FINDS THE AREA OF THE SMALLEST CONVEX POLYGON CONTAINING
C           AN ENTIRELY ARBITRARY SET OF N POINTS, EACH SPECIFIED BY
C           TWO CARTESIAN COORDINATES.
C
      DIMENSION P(100,2),T(100)
C
      EPS=1.E-6
      N1=N
C
C                 FIND POINT WITH LEAST X -- RENAME IT P1
C
      E=P(1,1)
      DO 5 K=2,N
    5 E=MINIF(P(K,1),E)
      DO 10 K=1,N
      L=K
      IF (E-P(K,1)) 10,15,10
   10 CONTINUE
   15 IF (L-1) 20,30,20
   20 DO 25 K=1,2
      E=P(1,K)
      P(1,K)=P(L,K)
   25 P(L,K)=E
C
C                 DISCARD POINTS COINCIDENT WITH P1
C
   30 DO 85 K=2,N
   35 DX=P(K,1)-P(1,1)
      DY=P(K,2)-P(1,2)
      IF (DX-EPS) 40,40,75
   40 IF (ABSF(DY)-EPS) 45,45,60
   45 N1=N1-1
      IF (K-1-N1) 50,50,90
   50 DO 55 J=1,2
      E=P(K,J)
      P(K,J)=P(N1+1,J)
   55 P(N1+1,J)=E
      GO TO 35
C
C                 FIND SLOPE T(K) OF LINE JOINING P1 TO K-TH POINT, K=2,N
C
   60 IF (DY) 65,65,70
   65 T(K)=-1.E6
      GO TO 80
   70 T(K)=1.E6
```

[redacted]

```
      GO TO 80
   75 T(K)=DY/DX
   80 IF (K-N1) 85,90,90
   85 CONTINUE
C
C                   ARRANGE PTS. OTHER THAN P1 IN ORDER OF INCREASING T(K) .
C                   IF TWO WITH SAME T(K), DISCARD PT. NEAREST P1
C
   90 NU=N1-1
      DO 165 J=2,NU
      DO 155 L=J,NU
      K=L+1
   94 IF (ABSF(T(J)-T(K))-EPS) 95,95,135
   95 N1=N1-1
      E=P(J,1)-P(K,1)
      IF (ABSF(E)-EPS) 105,105,100
  100 IF (E) 115,115,120
  105 E=P(J,2)-P(K,2)
      IF (ABSF(E)-EPS) 120,120,110
  110 IF (E*T(J)) 115,115,120
  115 I=J
      GO TO 125
  120 I=K
  125 IF (I-N1) 128,128,160
  128 DO 130 M=I,N1
      T(M)=T(M+1)
      P(M,1)=P(M+1,1)
  130 P(M,2)=P(M+1,2)
      GO TO 94
  135 IF (T(J)-T(K))150,150,140
  140 E=T(J)
      T(J)=T(K)
      T(K)=E
      DO 145 M=1,2
      E=P(J,M)
      P(J,M)=P(K,M)
  145 P(K,M)=E
  150 IF (K-N1) 155,160,160
  155 CONTINUE
  160 IF (J+1-N1) 165,170,170
  165 CONTINUE
C
C                   DISCARD K-TH PT IF WITHIN TRIANGLE P1--P,K-1--P,K+1,
C                   K=3,N-2
C
  170 NU=N1-2
      IF (NU-2) 220,172,172
  172 DO 215 K=2,NU
  175 E=P(K,1)-P(K+2,1)
```


[redacted]

```
      IF (ABSF(E)-EPS) 180,180,195
  180 IF (P(K+1,1)-P(K,1)) 185,185,210
  185 J=N1-2
      DO 190 M=K,J
      T(M+1)=T(M+2)
      P(M+1,1)=P(M+2,1)
  190 P(M+1,2)=P(M+2,2)
      N1=N1-1
      IF (K+1-N1) 175,220,220
  195 E=P(K,2)-P(K+2,2)
      IF (ABSF(E)-EPS) 200,200,205
  200 IF((P(K+1,2)-P(K,2))*T(K+1)) 185,185,210
  205 E1=(P(K+1,2)-P(1,2))/(P(K+1,1)-P(1,1))
      E2=(P(K+2,2)-P(K,2))/(P(K+2,1)-P(K,1))
      E=(E2*P(K,1)-E1*P(1,1)+P(1,2)-P(K,2))/(E2-E1)
      IF (P(K+1,1)-E) 185,185,210
  210 IF (K+2-N1) 215,220,220
  215 CONTINUE
C
C                FIND AREA OF CONVEX POLYGON FORMED BY REMAINING POINTS
C
  220 E=0.
      NU=N1-1
      DO 225 K=2,NU
  225 E=E+ABSF(P(K,1)*P(K+1,2)-P(K+1,1)*P(K,2)-P(1,1)*P(K+1,2)
     $+P(K+1,1)*P(1,2)+P(1,1)*P(K,2)-P(K,1)*P(1,2))
      SNGON=.5*E
      N=N1
      RETURN
      END
```

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TABLE I.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 1. CALCULATION OF THE WAVE DRAG OF A COMPLETE AIRCRAFT CONFIGURATION AT M = 1.50

(a) Input Data

INPUT DATA FOR CASE 1

1500 50 12 4 10 18 2 5 1 10 10 1 1 1 1 1 9999CASE 1

SAMPLE CASE 1 CALCULATION OF COMPLETE AIRCRAFT WAVE DRAG AT M=1.50

[redacted]

0.0	10.0	20.0	30.0	40.0	50.0	60.0	70.0	80.0	100.0	XAF	10
42.8	5.2	0.0	89.2							WAFORG	1
56.2	8.0	0.0	66.0							WAFORG	2
141.5	31.7	0.0	19.7							WAFORG	3
156.4	36.0	0.0	0.0							WAFORG	4
0.0	1.66	2.19	2.45	2.49	2.33	2.00	1.56	1.05	0.0	WAFORD	1
0.0	1.62	2.14	2.39	2.43	2.27	1.96	1.53	1.02	0.0	WAFORD	2
0.0	1.17	1.54	1.73	1.75	1.64	1.42	1.10	0.96	0.0	WAFORD	3
0.0	1.17	1.54	1.73	1.75	1.64	1.42	1.10	0.96	0.0	WAFORD	4
0.0	10.0	20.0	30.0	40.0	50.0	60.0	70.0	80.0	90.0	XFUS	10
100.0	110.0	121.0	131.0	141.0	151.0	161.0	170.0			XFUS	18
0.0	18.1	44.0	59.3	70.5	75.2	75.2	75.2	75.2	75.2	FUSARD	10
75.2	75.2	74.0	64.4	50.2	28.4	10.7	0.0			FUSARD	18
141.0	4.6	8.0	94.0	9.4	-6.3					PODORG	
0.0	4.0	8.0	10.75	29.4						XPOD	1
1.890	2.050	2.205	2.325	2.325						PODR	1
0.0	4.0	8.0	10.75	29.4						XPOD	2
1.890	2.050	2.205	2.325	2.325						PODR	2
134.2	0.0	3.2	28.2	160.8	0.0	23.6	6.6			FINORG	
0.0	10.0	20.0	30.0	40.0	50.0	60.0	70.0	80.0	100.0	XFIN	
0.0	0.558	0.992	1.302	1.488	1.550	1.488	1.302	0.992	0.0	FINORD	
147.6	2.4	0.0	19.6	167.8	14.3	0.0	6.1			CANORG	
0.0	10.0	20.0	30.0	40.0	50.0	60.0	70.0	80.0	100.0	XCAN	
0.0	0.54	0.96	1.26	1.44	1.50	1.44	1.26	0.96	0.0	CANORD	

TABLE I.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 1. CALCULATION OF THE WAVE DRAG OF A COMPLETE AIRCRAFT CONFIGURATION AT $M = 1.50$ - Continued

(b) Enriched Fuselage Area Distribution

SAMPLE CASE 1 CALCULATION OF COMPLETE AIRCRAFT WAVE DRAG AT M=1.50

FUSELAGE AREA DISTRIBUTION (D/Q = 7.36421)

0	0.	0.	50	85.0000	75.2069
1	1.7000	1.3796	51	86.7000	75.2056
2	3.4000	3.8384	52	88.4000	75.2022
3	5.1000	6.9330	53	90.1000	75.2001
4	6.8000	10.4899	54	91.8000	75.2056
5	8.5000	14.4033	55	93.5000	75.2108
6	10.2000	18.6104	56	95.2000	75.2126
7	11.9000	23.0703	57	96.9000	75.2098
8	13.6000	27.6491	58	98.6000	75.2038
9	15.3000	32.2398	59	100.3000	75.2005
10	17.0000	36.7366	60	102.0000	75.2069
11	18.7000	41.0115	61	103.7000	75.2106
12	20.4000	44.8269	62	105.4000	75.2075
13	22.1000	47.9900	63	107.1000	75.1982
14	23.8000	50.7833	64	108.8000	75.1903
15	25.5000	53.3221	65	110.5000	75.2178
16	27.2000	55.6749	66	112.2000	75.2937
17	28.9000	57.8978	67	113.9000	75.3346
18	30.6000	60.0704	68	115.6000	75.2907
19	32.3000	62.2286	69	117.3000	75.1172
20	34.0000	64.3032	70	119.0000	74.7622
21	35.7000	66.2629	71	120.7000	74.1477
22	37.4000	68.0807	72	122.4000	73.1125
23	39.1000	69.7228	73	124.1000	71.7559
24	40.8000	71.1155	74	125.8000	70.1760
25	42.5000	72.2497	75	127.5000	68.4190
26	44.2000	73.1934	76	129.2000	66.5204
27	45.9000	73.9705	77	130.9000	64.5194
28	47.6000	74.5921	78	132.6000	62.4755
29	49.3000	75.0582	79	134.3000	60.3386
30	51.0000	75.3409	80	136.0000	58.0681
31	52.7000	75.4644	81	137.7000	55.6300
32	54.4000	75.4825	82	139.4000	52.9774
33	56.1000	75.4275	83	141.1000	50.0115
34	57.8000	75.3293	84	142.8000	46.5770
35	59.5000	75.2237	85	144.5000	42.8900
36	61.2000	75.1759	86	146.2000	39.0761
37	62.9000	75.1713	87	147.9000	35.2233
38	64.6000	75.1780	88	149.6000	31.4172
39	66.3000	75.1867	89	151.3000	27.7832
40	68.0000	75.1938	90	153.0000	24.4443
41	69.7000	75.1990	91	154.7000	21.2878
42	71.4000	75.2063	92	156.4000	18.2741
43	73.1000	75.2123	93	158.1000	15.3860
44	74.8000	75.2141	94	159.8000	12.6095
45	76.5000	75.2115	95	161.5000	9.9050
46	78.2000	75.2057	96	163.2000	7.2677
47	79.9000	75.2002	97	164.9000	4.8253
48	81.6000	75.2021	98	166.6000	2.6797
49	83.3000	75.2055	99	168.3000	0.9652
50	85.0000	75.2069	100	170.0000	0.

[redacted]

TABLE I.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 1. CALCULATION OF THE WAVE DRAG OF A COMPLETE AIRCRAFT CONFIGURATION AT M = 1.50 - Continued

(c) Wave Drag and Volume Check

SAMPLE CASE 1 CALCULATION OF COMPLETE AIRCRAFT WAVE DRAG AT M=1.50

D/Q ASSOCIATED WITH VARIOUS VALUES OF THETA

N	THETA	D/Q
0	-90.000	13.75306
1	-75.000	15.65638
2	-60.000	13.38225
3	-45.000	12.43245
4	-30.000	11.39625
5	-15.000	11.43813
6	0.	12.87852
7	15.000	15.19173
8	30.000	12.74069
9	45.000	14.27913
10	60.000	14.40237
11	75.000	17.99104
12	90.000	17.86331

D/Q FOR ENTIRE AIRCRAFT = 14.18827

VOLUME OF ENTIRE WING = 4.01752E 03

VOLUME OF EQUIVALENT BODY = 4.01751E 03

[redacted]

TABLE I.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 1. CALCULATION OF THE WAVE DRAG OF A COMPLETE AIRCRAFT CONFIGURATION AT M = 1.50 - Concluded

(d) Selected Equivalent-Body Area Distributions

SAMPLE CASE 1 CALCULATION OF COMPLETE AIRCRAFT WAVE DRAG AT M=1.50

S(X,THETA) FOR ENTIRE AIRCRAFT

THETA = -90.00		THETA = -45.00		THETA = 0.		THETA = 45.00		THETA = 90.00	
X	S	X	S	X	S	X	S	X	S
0.	44.888	0.	44.888	0.	44.888	0.	44.888	0.	44.888
3.876	49.755	3.725	49.483	3.933	49.859	3.725	49.483	3.478	49.047
7.751	57.915	7.450	57.211	7.866	58.182	7.450	57.211	6.956	56.075
11.627	67.644	11.176	66.467	11.799	68.098	11.176	66.467	10.434	64.551
15.503	77.819	14.901	76.280	15.732	78.400	14.901	76.280	13.912	73.679
19.379	86.947	18.626	85.329	19.665	87.559	18.626	85.329	17.390	82.474
23.254	94.387	22.351	92.825	23.598	94.972	22.351	92.825	20.868	90.015
27.130	100.316	26.077	98.811	27.531	100.873	26.077	98.811	24.346	96.192
31.006	105.431	29.802	103.905	31.464	105.996	29.802	103.905	27.824	101.277
34.881	109.973	33.527	108.456	35.397	110.539	33.527	108.456	31.302	105.796
38.757	113.810	37.252	112.426	39.330	114.547	37.252	112.426	34.780	109.862
42.633	116.734	40.977	115.820	43.263	118.893	40.977	115.820	38.258	113.356
46.509	119.665	44.703	119.423	47.196	123.567	44.703	119.423	41.736	116.132
50.384	123.868	48.428	123.287	51.129	128.251	48.423	123.287	45.214	118.555
54.260	128.919	52.153	127.675	55.062	133.937	52.153	127.675	48.692	121.863
58.136	134.206	55.878	132.988	58.995	140.450	55.878	132.988	52.170	126.139
62.011	140.213	59.604	138.929	62.928	147.283	59.604	138.929	55.648	130.771
65.887	146.812	63.329	145.120	66.861	154.269	63.329	145.120	59.126	135.667
69.763	153.645	67.054	151.635	70.794	161.182	67.054	151.635	62.604	141.196
73.639	160.411	70.779	158.220	74.727	167.823	70.779	158.220	66.082	147.153
77.514	166.879	74.504	164.635	78.660	173.722	74.504	164.635	69.560	153.288
81.390	172.819	78.230	170.568	82.593	178.857	78.230	170.568	73.038	159.379
85.266	178.075	81.955	176.297	86.526	184.019	81.955	175.865	76.516	165.257
89.141	184.427	85.680	182.325	90.459	188.452	85.680	180.284	79.994	170.743
93.017	191.498	89.405	187.701	94.392	190.973	89.405	183.667	83.472	175.717
96.893	197.258	93.131	191.335	98.325	190.387	93.131	186.685	86.950	179.983
100.769	199.252	96.856	192.873	102.258	187.541	96.856	189.334	90.428	183.453
104.644	198.265	100.581	193.884	106.191	183.791	100.581	190.701	93.906	185.935
108.520	195.695	104.306	193.716	110.124	179.397	104.306	189.605	97.384	187.424
112.396	191.603	108.031	191.593	114.057	172.973	108.031	186.947	100.862	188.310
116.271	185.730	111.757	186.742	117.990	164.658	111.757	183.665	104.340	190.158
120.147	177.575	115.482	179.847	121.923	156.377	115.482	178.740	107.818	191.649
124.023	167.185	119.207	171.635	125.856	149.623	119.207	171.585	111.296	191.592
127.898	156.604	122.932	162.941	129.788	143.140	122.932	162.941	114.774	188.242
131.774	146.521	126.658	153.486	133.721	135.677	126.653	153.486	118.252	181.873
135.650	136.720	130.383	143.229	137.654	129.549	130.383	143.414	121.730	173.514
139.526	126.522	134.108	134.082	141.587	124.802	134.108	136.255	125.208	163.912
143.401	116.104	137.833	126.657	145.520	119.816	137.833	134.088	128.686	154.496
147.277	105.587	141.558	119.186	149.453	116.073	141.558	133.620	132.164	146.985
151.153	97.602	145.284	112.105	153.386	113.055	145.284	129.822	135.642	146.722
155.028	93.762	149.009	106.374	157.319	106.507	149.009	120.533	139.120	144.756
158.904	90.868	152.734	103.456	161.252	94.554	152.734	109.499	142.598	137.762
162.780	87.678	156.459	101.713	165.185	82.615	156.459	100.279	146.076	126.324
166.656	80.385	160.185	97.210	169.118	76.394	160.185	92.208	149.554	113.323
170.531	73.283	163.910	88.136	173.051	74.647	163.910	82.638	153.032	102.212
174.407	70.792	167.635	80.813	176.984	73.336	167.635	76.593	156.510	93.233
178.283	70.051	171.360	76.532	180.917	72.185	171.360	73.481	159.988	86.233
182.158	69.421	175.085	74.164	184.850	71.150	175.085	72.093	163.466	81.384
186.034	68.902	178.811	72.076	188.783	70.008	178.811	70.796	166.944	75.133
189.910	68.359	182.536	69.828	192.716	69.153	182.535	69.335	170.422	69.659
193.786	67.929	186.261	67.929	196.649	67.929	186.261	67.929	173.900	67.929

[redacted]

TABLE II.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 2. CALCULATION OF THE WING AVERAGE EQUIVALENT BODY AT $M = 1.50$

(a) Input Data

INPUT DATA FOR CASE 2

1500	50	12	4	10	4		1	1		9999CASE 2

SAMPLE CASE 2 CALCULATION OF WING AVERAGE EQUIVALENT BODY AT M=1.50

0.0	10.0	20.0	30.0	40.0	50.0	60.0	70.0	80.0	100.0	XAF 10
8.8	5.2	0.0	89.2							WAFORG 1
22.2	8.0	0.0	66.0							WAFORG 2
107.5	31.7	0.0	19.7							WAFORG 3
122.4	36.0	0.0	0.0							WAFORG 4
0.0	1.66	2.19	2.45	2.49	2.33	2.00	1.56	1.05	0.0	WAFORD 1
0.0	1.62	2.14	2.39	2.43	2.27	1.96	1.53	1.02	0.0	WAFORD 2
0.0	1.17	1.54	1.73	1.75	1.64	1.42	1.10	0.96	0.0	WAFORD 3
0.0	1.17	1.54	1.73	1.75	1.64	1.42	1.10	0.96	0.0	WAFORD 4
0.0	55.0	110.0	166.0							XFUS 4
0.0	0.0	0.0	0.0							FUSARD 4

[redacted]

TABLE II.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 2. CALCULATION OF THE WING AVERAGE EQUIVALENT BODY AT M = 1.50 - Continued

(b) Enriched Fuselage Area Distribution

SAMPLE CASE 2 CALCULATION OF WING AVERAGE EQUIVALENT BODY AT M=1.50

FUSELAGE AREA DISTRIBUTION (D/Q = 0.)

0	0.	0.	50	83.0000	0.
1	1.6600	0.	51	84.6600	0.
2	3.3200	0.	52	86.3200	0.
3	4.9800	0.	53	87.9800	0.
4	6.6400	0.	54	89.6400	0.
5	8.3000	0.	55	91.3000	0.
6	9.9600	0.	56	92.9600	0.
7	11.6200	0.	57	94.6200	0.
8	13.2800	0.	58	96.2800	0.
9	14.9400	0.	59	97.9400	0.
10	16.6000	0.	60	99.6000	0.
11	18.2600	0.	61	101.2600	0.
12	19.9200	0.	62	102.9200	0.
13	21.5800	0.	63	104.5800	0.
14	23.2400	0.	64	106.2400	0.
15	24.9000	0.	65	107.9000	0.
16	26.5600	0.	66	109.5600	0.
17	28.2200	0.	67	111.2200	0.
18	29.8800	0.	68	112.8800	0.
19	31.5400	0.	69	114.5400	0.
20	33.2000	0.	70	116.2000	0.
21	34.8600	0.	71	117.8600	0.
22	36.5200	0.	72	119.5200	0.
23	38.1800	0.	73	121.1800	0.
24	39.8400	0.	74	122.8400	0.
25	41.5000	0.	75	124.5000	0.
26	43.1600	0.	76	126.1600	0.
27	44.8200	0.	77	127.8200	0.
28	46.4800	0.	78	129.4800	0.
29	48.1400	0.	79	131.1400	0.
30	49.8000	0.	80	132.8000	0.
31	51.4600	0.	81	134.4600	0.
32	53.1200	0.	82	136.1200	0.
33	54.7800	0.	83	137.7800	0.
34	56.4400	0.	84	139.4400	0.
35	58.1000	0.	85	141.1000	0.
36	59.7600	0.	86	142.7600	0.
37	61.4200	0.	87	144.4200	0.
38	63.0800	0.	88	146.0800	0.
39	64.7400	0.	89	147.7400	0.
40	66.4000	0.	90	149.4000	0.
41	68.0600	0.	91	151.0600	0.
42	69.7200	0.	92	152.7200	0.
43	71.3800	0.	93	154.3800	0.
44	73.0400	0.	94	156.0400	0.
45	74.7000	0.	95	157.7000	0.
46	76.3600	0.	96	159.3600	0.
47	78.0200	0.	97	161.0200	0.
48	79.6800	0.	98	162.6800	0.
49	81.3400	0.	99	164.3400	0.
50	83.0000	0.	100	166.0000	0.

[redacted]

TABLE II.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 2. CALCULATION OF THE WING AVERAGE EQUIVALENT BODY AT M = 1.50 - Continued

(c) Wave Drag and Volume Check

SAMPLE CASE 2 CALCULATION OF WING AVERAGE EQUIVALENT BODY AT M=1.50

D/Q ASSOCIATED WITH VARIOUS VALUES OF THETA

N	THETA	D/Q
0	-90.000	4.82389
1	-75.000	4.61954
2	-60.000	4.56962
3	-45.000	4.75043
4	-30.000	4.72171
5	-15.000	4.85888
6	0.	4.99581
7	15.000	4.85890
8	30.000	4.72168
9	45.000	4.75045
10	60.000	4.56962
11	75.000	4.61955
12	90.000	4.82393

D/Q FOR ENTIRE AIRCRAFT = 4.74078

VOLUME OF ENTIRE WING = 4.01752E 03

VOLUME OF EQUIVALENT BODY = 4.01725E 03

TABLE II.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 2. CALCULATION OF THE WING AVERAGE EQUIVALENT BODY AT M = 1.50 - Continued

(d) Equivalent-Body Area Distributions

SAMPLE CASE 2 CALCULATION OF WING AVERAGE EQUIVALENT BODY AT M=1.50

S(X,THETA) FOR ENTIRE AIRCRAFT

THETA = -90.00		THETA = -75.00		THETA = -60.00		THETA = -45.00		THETA = -30.00	
X	S	X	S	X	S	X	S	X	S
0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
3.320	0.	3.320	0.	3.320	0.	3.320	0.	3.320	0.
6.640	0.	6.640	0.	6.640	0.022	6.640	0.161	6.640	0.363
9.960	0.097	9.960	0.271	9.960	0.667	9.960	1.175	9.960	1.683
13.280	1.442	13.280	1.665	13.280	2.279	13.280	3.107	13.280	3.923
16.600	4.359	16.600	4.561	16.600	5.161	16.600	5.977	16.600	6.849
19.920	8.318	19.920	8.488	19.920	9.040	19.920	9.924	19.920	10.929
23.240	12.716	23.240	13.015	23.240	13.800	23.240	14.909	23.240	16.064
26.560	17.713	26.560	18.070	26.560	19.037	26.560	20.332	26.560	21.640
29.880	23.256	29.880	23.601	29.880	24.584	29.880	25.973	29.880	27.406
33.200	29.044	33.200	29.401	33.200	30.399	33.200	31.817	33.200	33.286
36.520	34.889	36.520	35.260	36.520	36.272	36.520	37.677	36.520	39.121
39.840	40.663	39.840	41.026	39.840	42.027	39.840	43.425	39.840	44.856
43.160	46.213	43.160	46.553	43.160	47.512	43.160	48.846	43.160	50.148
46.480	51.379	46.480	51.709	46.480	52.570	46.480	53.746	46.480	54.932
49.800	56.059	49.800	56.337	49.800	57.081	49.800	58.087	49.800	59.038
53.120	60.077	53.120	60.300	53.120	60.887	53.120	61.615	53.120	62.291
56.440	63.368	56.440	63.484	56.440	63.875	56.440	64.318	56.440	64.631
59.760	65.753	59.760	65.812	59.760	65.957	59.760	65.998	59.760	65.876
63.080	67.237	63.080	67.180	63.080	66.985	63.080	66.581	63.080	65.963
66.400	67.652	66.400	67.506	66.400	66.980	66.400	66.033	66.400	64.791
69.720	67.034	69.720	66.715	69.720	65.763	69.720	64.247	69.720	62.417
73.040	65.224	73.040	64.777	73.040	63.439	73.040	61.311	73.040	58.793
76.360	62.295	76.360	61.707	76.360	59.960	76.360	57.212	76.360	53.983
79.680	58.334	79.680	57.573	79.680	55.334	79.680	51.868	79.680	47.879
83.000	53.313	83.000	52.371	83.000	49.614	83.000	45.465	83.000	41.110
86.320	47.244	86.320	46.116	86.320	43.195	86.320	39.184	86.320	34.806
89.640	40.320	89.640	39.513	89.640	36.979	89.640	33.294	89.640	28.674
92.960	34.061	92.960	33.248	92.960	31.019	92.960	27.607	92.960	23.702
96.280	28.801	96.280	27.983	96.280	25.780	96.280	21.972	96.280	20.147
99.600	24.465	99.600	23.620	99.600	21.314	99.600	17.749	99.600	17.915
102.920	20.555	102.920	19.813	102.920	17.175	102.920	15.549	102.920	16.208
106.240	16.981	106.240	16.323	106.240	13.629	106.240	14.069	106.240	14.808
109.560	13.755	109.560	13.024	109.560	11.370	109.560	12.663	109.560	13.471
112.880	10.879	112.880	9.720	112.880	9.968	112.880	11.331	112.880	12.197
116.200	8.229	116.200	6.896	116.200	8.657	116.200	10.071	116.200	10.985
119.520	5.576	119.520	5.454	119.520	7.437	119.520	8.886	119.520	9.837
122.840	2.678	122.840	4.362	122.840	6.309	122.840	7.774	122.840	8.751
126.160	0.157	126.160	3.330	126.160	5.272	126.160	6.735	126.160	7.729
129.480	0.	129.480	2.255	129.480	4.339	129.480	5.770	129.480	6.769
132.800	0.	132.800	1.060	132.800	3.476	132.800	4.878	132.800	5.872
136.120	0.	136.120	0.005	136.120	2.626	136.120	4.069	136.120	5.037
139.440	0.	139.440	0.	139.440	1.684	139.440	3.324	139.440	4.267
142.760	0.	142.760	0.	142.760	0.505	142.760	2.598	142.760	3.568
146.080	0.	146.080	0.	146.080	0.	146.080	1.811	146.080	2.913
149.400	0.	149.400	0.	149.400	0.	149.400	0.920	149.400	2.254
152.720	0.	152.720	0.	152.720	0.	152.720	0.	152.720	1.517
156.040	0.	156.040	0.	156.040	0.	156.040	0.	156.040	0.623
159.360	0.	159.360	0.	159.360	0.	159.360	0.	159.360	0.
162.680	0.	162.680	0.	162.680	0.	162.680	0.	162.680	0.
166.000	0.	166.000	0.	166.000	0.	166.000	0.	166.000	0.

[redacted]

TABLE II.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 2. CALCULATION OF THE WING AVERAGE EQUIVALENT BODY AT M = 1.50 - Continued

(d) Equivalent-Body Area Distributions - Continued

S(X,THETA) FOR ENTIRE AIRCRAFT

THETA = -15.00		THETA = 0.		THETA = 15.00		THETA = 30.00		THETA = 45.00	
X	S	X	S	X	S	X	S	X	S
0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
3.320	0.001	3.320	0.005	3.320	0.001	3.320	0.	3.320	0.
6.640	0.536	6.640	0.604	6.640	0.536	6.640	0.363	6.640	0.161
9.960	2.061	9.960	2.201	9.960	2.061	9.960	1.683	9.960	1.175
13.280	4.502	13.280	4.708	13.280	4.502	13.280	3.923	13.280	3.107
16.600	7.530	16.600	7.787	16.600	7.530	16.600	6.849	16.600	5.977
19.920	11.726	19.920	12.029	19.920	11.726	19.920	10.929	19.920	9.924
23.240	16.936	23.240	17.260	23.240	16.936	23.240	16.064	23.240	14.909
26.560	22.618	26.560	22.982	26.560	22.618	26.560	21.640	26.560	20.332
29.880	28.484	29.880	28.887	29.880	28.484	29.880	27.406	29.880	25.973
33.200	34.384	33.200	34.789	33.200	34.384	33.200	33.286	33.200	31.817
36.520	40.209	36.520	40.614	36.520	40.209	36.520	39.121	36.520	37.677
39.840	45.928	39.840	46.315	39.840	45.928	39.840	44.856	39.840	43.425
43.160	51.108	43.160	51.460	43.160	51.108	43.160	50.148	43.160	48.846
46.480	55.798	46.480	56.116	46.480	55.798	46.480	54.932	46.480	53.746
49.800	59.681	49.800	59.908	49.800	59.681	49.800	59.038	49.800	58.087
53.120	62.737	53.120	62.888	53.120	62.737	53.120	62.291	53.120	61.615
56.440	64.766	56.440	64.795	56.440	64.766	56.440	64.631	56.440	64.318
59.760	65.668	59.760	65.564	59.760	65.668	59.760	65.876	59.760	65.998
63.080	65.343	63.080	65.078	63.080	65.343	63.080	65.963	63.080	66.581
66.400	63.675	66.400	63.223	66.400	63.675	66.400	64.791	66.400	66.033
69.720	60.834	69.720	60.198	69.720	60.834	69.720	62.417	69.720	64.247
73.040	56.668	73.040	55.825	73.040	56.668	73.040	58.793	73.040	61.311
76.360	51.299	76.360	50.235	76.360	51.299	76.360	53.983	76.360	57.212
79.680	44.553	79.680	43.214	79.680	44.553	79.680	47.879	79.680	51.868
83.000	37.482	83.000	35.899	83.000	37.482	83.000	41.110	83.000	45.465
86.320	30.916	86.320	29.724	86.320	30.916	86.320	34.806	86.320	39.184
89.640	25.910	89.640	25.248	89.640	25.910	89.640	28.674	89.640	33.294
92.960	22.789	92.960	22.867	92.960	22.789	92.960	23.702	92.960	27.607
96.280	20.578	96.280	20.711	96.280	20.578	96.280	20.147	96.280	21.972
99.600	18.310	99.600	18.440	99.600	18.310	99.600	17.915	99.600	17.749
102.920	16.562	102.920	16.676	102.920	16.562	102.920	16.208	102.920	15.549
106.240	15.201	106.240	15.323	106.240	15.201	106.240	14.808	106.240	14.069
109.560	13.907	109.560	14.044	109.560	13.907	109.560	13.471	109.560	12.663
112.880	12.670	112.880	12.820	112.880	12.670	112.880	12.197	112.880	11.331
116.200	11.491	116.200	11.652	116.200	11.491	116.200	10.985	116.200	10.071
119.520	10.369	119.520	10.539	119.520	10.369	119.520	9.837	119.520	8.886
122.840	9.303	122.840	9.481	122.840	9.303	122.840	8.751	122.840	7.774
126.160	8.296	126.160	8.479	126.160	8.296	126.160	7.729	126.160	6.735
129.480	7.345	129.480	7.532	129.480	7.345	129.480	6.769	129.480	5.770
132.800	6.451	132.800	6.641	132.800	6.451	132.800	5.872	132.800	4.878
136.120	5.615	136.120	5.805	136.120	5.615	136.120	5.037	136.120	4.069
139.440	4.836	139.440	5.024	139.440	4.836	139.440	4.267	139.440	3.324
142.760	4.114	142.760	4.299	142.760	4.114	142.760	3.568	142.760	2.598
146.080	3.459	146.080	3.636	146.080	3.459	146.080	2.913	146.080	1.811
149.400	2.845	149.400	3.026	149.400	2.845	149.400	2.254	149.400	0.920
152.720	2.230	152.720	2.431	152.720	2.230	152.720	1.517	152.720	0.
156.040	1.544	156.040	1.793	156.040	1.544	156.040	0.623	156.040	0.
159.360	0.736	159.360	1.078	159.360	0.736	159.360	0.	159.360	0.
162.680	0.	162.680	0.	162.680	0.	162.680	0.	162.680	0.
166.000	0.	166.000	0.	166.000	0.	166.000	0.	166.000	0.

[redacted]

TABLE II.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 2. CALCULATION OF THE WING AVERAGE EQUIVALENT BODY AT M = 1.50 - Continued

(d) Equivalent-Body Area Distributions - Concluded

S(X,THETA) FOR ENTIRE AIRCRAFT

THETA = 60.00		THETA = 75.00		THETA = 90.00		THETA =		THETA =	
X	S	X	S	X	S	X	S	X	S
0.	0.	0.	0.	0.	0.				
3.320	0.	3.320	0.	3.320	0.				
6.640	0.122	6.640	0.	6.640	0.				
9.960	0.667	9.960	0.271	9.960	0.097				
13.280	2.279	13.280	1.665	13.280	1.442				
16.600	5.161	16.600	4.561	16.600	4.359				
19.920	9.040	19.920	8.488	19.920	8.318				
23.240	13.800	23.240	13.015	23.240	12.716				
26.560	19.037	26.560	18.070	26.560	17.713				
29.880	24.584	29.880	23.601	29.880	23.256				
33.200	30.399	33.200	29.401	33.200	29.044				
36.520	36.272	36.520	35.260	36.520	34.889				
39.840	42.027	39.840	41.026	39.840	40.663				
43.160	47.512	43.160	46.553	43.160	46.213				
46.480	52.570	46.480	51.709	46.480	51.379				
49.800	57.081	49.800	56.337	49.800	56.059				
53.120	60.887	53.120	60.300	53.120	60.077				
56.440	63.875	56.440	63.484	56.440	63.368				
59.760	65.957	59.760	65.812	59.760	65.753				
63.080	66.985	63.080	67.180	63.080	67.237				
66.400	66.980	66.400	67.506	66.400	67.652				
69.720	65.763	69.720	66.715	69.720	67.034				
73.040	63.439	73.040	64.777	73.040	65.224				
76.360	59.960	76.360	61.707	76.360	62.295				
79.680	55.334	79.680	57.573	79.680	58.334				
83.000	49.614	83.000	52.371	83.000	53.313				
86.320	43.195	86.320	46.116	86.320	47.244				
89.640	36.979	89.640	39.513	89.640	40.320				
92.960	31.019	92.960	33.248	92.960	34.061				
96.280	25.780	96.280	27.983	96.280	28.801				
99.600	21.314	99.600	23.620	99.600	24.465				
102.920	17.175	102.920	19.813	102.920	20.555				
106.240	13.629	106.240	16.323	106.240	16.981				
109.560	11.379	109.560	13.024	109.560	13.755				
112.880	9.968	112.880	9.720	112.880	10.879				
116.200	8.657	116.200	6.896	116.200	8.229				
119.520	7.437	119.520	5.454	119.520	5.576				
122.840	6.339	122.840	4.362	122.840	2.678				
126.160	5.272	126.160	3.330	126.160	0.157				
129.480	4.339	129.480	2.255	129.480	0.				
132.800	3.476	132.800	1.060	132.800	0.				
136.120	2.626	136.120	0.005	136.120	0.				
139.440	1.684	139.440	0.	139.440	0.				
142.760	0.525	142.760	0.	142.760	0.				
146.080	0.	146.080	0.	146.080	0.				
149.400	0.	149.400	0.	149.400	0.				
152.720	0.	152.720	0.	152.720	0.				
156.040	0.	156.040	0.	156.040	0.				
159.360	0.	159.360	0.	159.360	0.				
162.680	0.	162.680	0.	162.680	0.				
166.000	0.	166.000	0.	166.000	0.				

[redacted]

TABLE II.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 2. CALCULATION OF THE WING AVERAGE EQUIVALENT BODY AT M = 1.50 - Concluded

(e) Area Distribution of the Wing Average Equivalent Body

AREA DISTRIBUTION OF WING EQUIVALENT BODY

X	S
0.	0.
3.320	0.000
6.640	0.232
9.960	1.168
13.280	3.092
16.600	6.024
19.920	10.047
23.240	14.952
26.560	20.340
29.880	26.020
33.200	31.867
36.520	37.715
39.840	43.459
43.160	48.834
46.480	53.751
49.800	58.035
53.120	61.551
56.440	64.192
59.760	65.828
63.080	66.369
66.400	65.738
69.720	63.933
73.040	60.919
76.360	56.739
79.680	51.331
83.000	45.109
86.320	38.767
89.640	32.873
92.960	27.825
96.280	23.514
99.600	19.987
102.920	17.322
106.240	15.116
109.560	13.130
112.880	11.278
116.200	9.599
119.520	8.296
122.840	7.111
126.160	6.007
129.480	5.066
132.800	4.150
136.120	3.311
139.440	2.750
142.760	2.197
146.080	1.713
149.400	1.261
152.720	0.794
156.040	0.510
159.360	0.222
162.680	0.
166.000	0.

[redacted]

TABLE III.- MACHINE TABULATED OUTPUT FOR SAMPLE CASE 3. CALCULATION OF WING VOLUME ONLY

INPUT DATA FOR CASE 3

1500 50 12 4 10 1 -19999CASE 3

SAMPLE CASE 3 CALCULATION OF WING VOLUME ONLY

0.0	10.0	20.0	30.0	40.0	50.0	60.0	70.0	80.0	100.0	XAF	10
42.8	5.2	0.0	89.2							WAFORG	1
56.2	8.0	0.0	66.0							WAFORG	2
141.5	31.7	0.0	19.7							WAFORG	3
156.4	36.0	0.0	0.0							WAFORG	4
0.0	1.66	2.19	2.45	2.49	2.33	2.00	1.56	1.05	0.0	WAFORD	1
0.0	1.62	2.14	2.39	2.43	2.27	1.96	1.53	1.02	0.0	WAFORD	2
0.0	1.17	1.54	1.73	1.75	1.64	1.42	1.10	0.96	0.0	WAFORD	3
0.0	1.17	1.54	1.73	1.75	1.64	1.42	1.10	0.96	0.0	WAFORD	4

VOLUME OF ENTIRE WING = 4.01752E 03

[redacted]

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[redacted]

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