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Volume II

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THE USAF STABILITY AND CONTROL DIGITAL DATCOM
Volume II, Implementation of Datcom Methods

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MCDONNELL DOUGLAS ASTRONAUTICS COMPANY — ST. LOUIS DIVISION
ST. LOUIS, MISSOURI 63166

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AIR FORCE FLIGHT DYNAMICS LABORATORY
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AIR FORCE SYSTEMS COMMAND
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This report has been reviewed by the Office of Public Affairs (ASD/PA) and is releasable to the National Technical Information Service (NTIS). At NTIS, it will be available to the general public, including foreign nations.

This technical report has been reviewed and is approved for publication.

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

This report describes a digital computer program that calculates static stability, high lift and control, and dynamic derivative characteristics using the methods contained in the USAF Stability and Control Datcom ~~(revised April 1976)~~. Configuration geometry, attitude, and Mach range capabilities are consistent with those accommodated by the Datcom. The program contains a trim option that computes control deflections and aerodynamic increments for vehicle trim at subsonic Mach numbers. Volume I is the user's manual and presents

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program capabilities, input and output characteristics, and example problems. Volume II describes program implementation of Datcom methods. Volume III discusses a separate plot module for Digital Datcom.

The program is written in ANSI Fortran IV. The primary deviations from standard Fortran are Namelist input and certain statements required by the CDC compilers. Core requirements have been minimized by data packing and the use of overlays.

User oriented features of the program include minimized input requirements, input error analysis, and various options for application flexibility.

FOREWORD

This report, "The USAF Stability and Control Digital Datcom," describes the computer program that calculates static stability, high lift and control, and dynamic derivative characteristics using the methods contained in Sections 4 through 7 of the USAF Stability and Control Datcom (revised April 1976). The report consists of the following three volumes:

- Volume I, Users Manual
- Volume II, Implementation of Datcom Methods
- Volume III, Plot Module

A complete listing of the program is provided as a microfiche supplement.

This work was performed by the McDonnell Douglas Astronautics Company, Box 516, St. Louis, MO 63166, under contract number F33615-77-C-3073 with the United States Air Force Systems Command, Wright-Patterson Air Force Base, OH. The subject contract was initiated under Air Force Flight Dynamics Laboratory Project 8219, Task 82190115 on 15 August 1977 and was effectively concluded in November 1978. This report supersedes AFFDL TR-73-23 produced under contract F33615-72-C-1067, which automated Sections 4 and 5 of the USAF Stability and Control Datcom; AFFDL TR-74-68 produced under contract F33615-73-C-3058 which extended the program to include Datcom Sections 6 and 7 and a trim option; and AFFDL-TR-76-45 that incorporated Datcom revisions and user oriented options under contract F33615-75-C-3043. The recent activity generated a plot module, updated methods to incorporate the 1976 Datcom revisions, and provide additional user oriented features. These contracts, in total, reflect a systematic approach to Datcom automation which commenced in February 1972. Mr. J. E. Jenkins, AFFDL FGC, was the Air Force Project Engineer for the previous three contracts and Mr. B. F. Niehaus acted in this capacity for the current contract. The authors wish to thank Mr. Niehaus for his assistance, particularly in the areas of computer program formulation, implementation, and verification. A list of the Digital Datcom Principal Investigators and individuals who made significant contributions to the development of this program is provided on the following page.

Requests for copies of the computer program should be directed to the Air Force Flight Dynamics Laboratory (FGC). Copies of this report can be obtained from the National Technical Information Service (NTIS).

This report was submitted in April 1979.

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INTRODUCTION

Digital Datcom calculates static stability, high-lift and control device, and dynamic-derivative characteristics using the methods contained in Sections 4 through 7 of Datcom. The computer program also offers a trim option that computes control deflections and aerodynamic data for vehicle trim.

Even though the development of Digital Datcom was pursued with the sole objective of translating the Datcom methods into an efficient, user-oriented computer program, differences between Datcom and Digital Datcom do exist. Such is the primary subject of this volume, Implementation of Datcom Methods, which contains the program formulation for those methods in variance with Datcom methods. Program implementation information regarding system resources necessary to make the program operational are presented in Sections 5 and 6.

Section 6 also lists each of the routines and references their appearance in the program listings provided as a microfiche supplement to this volume.

Users should refer to Datcom for the validity and limitations of methods involved. However, potential users are fore-warned that Datcom drag methods are not recommended for performance. Where more than one Datcom method exists, the summary in Table 1 indicates which method or methods are employed in Digital Datcom. Tables 2, 3, and 4 define the basic output data in each Mach regime and shows the overlay in which each is computed.

The computer program is written in Fortran IV for the CDC Cyber 175. Through the use of overlay and data packing techniques, core requirement is 67,000 octal words for execution with the NOS operating system using the FTN compiler. Central processor time for a case executed on the NOS system depends on the type of configuration, number of flight conditions, and program option selected. Usual requirements are on the order of one to two seconds per Mach number.

Direct all program inquires to AFFDL FGC, Wright-Patterson Air Force Base, Ohio 45433. Phone (513) 255-4315.

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
Airfoil Section Aerodynamics	Airfoils	4.1.1-4.1.2	SUBSONIC	NDM	50		*User input or calculated by the airfoil section module
α_0	Wings	4.1.3.1	SUBSONIC	1	15,16	CALCAO	
			TRANSONIC	NDM			Experimental data input required
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{L_α}	Wings	4.1.3.2	SUBSONIC	1	15,16	WTLIFT	
			TRANSONIC	1	24	TRSØNI	*Transonic fairing performed
			SUPERSONIC	1	27		
			HYPERSONIC	1	27		
C_L	Wings	4.1.3.3	SUBSONIC	1	15,16	LIFTCF	
			TRANSONIC	1	35	WINGCL	*Graphical Method Used
			SUPERSONIC	1	27	SUPLNG	
			HYPERSONIC	1	27	SUPLNG	
$C_{L_{MAX}}$	Wings	4.1.3.4	SUBSONIC	2,3	15,16	CLMXBS	Method 2 high aspect ratio, Method 3 low
			TRANSONIC	1		CLMXB1	
			SUPERSONIC	NP			
			HYPERSONIC	NP			

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

*Subject of Section 4 of this volume

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{m_o}	Wings	4.1.4.1	SUBSONIC	1	31,33	CMALPH	
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{m_α}	Wings	4.1.4.2	SUBSONIC	1	31,33	CMALPH	
			TRANSONIC	1	25	TRANCM	*
			SUPERSONIC	1	27	SUPLNG	
			HYPERSONIC	1	27	SUPLNG	
C_m	Wings	4.1.4.3	SUBSONIC	1	31,33	CMALPH	*Straight-tapered low aspect ratio
			TRANSONIC	NDM			*Compute aerodynamic center
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{D_o}	Wings	4.1.5.1	SUBSONIC	1	3,5	CDRAG	
			TRANSONIC	1	24	TRSØNI	*
			SUPERSONIC	1	18	SUPDRG	
			HYPERSONIC	1	18	SUPDRG	
C_D	Wings	4.1.5.2	SUBSONIC	1	3,5	CDRAG	
			TRANSONIC	1	35	WINGCL	*
			SUPERSONIC	1	18	SUPDRG	
			HYPERSONIC	1	18	SUPDRG	

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

*Subject of Section 4 of this volume

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{L_α}	Bodies	4.2.1.1	SUBSONIC	1	6	BØDYRT	
			TRANSONIC	1	6	BØDYRT	*Faired between subsonic and supersonic
			SUPERSONIC	1	19	SUPBØD	
			HYPERSONIC	1	26	HYPBØD	
C_L	Bodies	4.2.1.2	SUBSONIC	1	6	BØDYRT	
			TRANSONIC	NDM			
			SUPERSONIC	2	19	SUPBØD	
			HYPERSONIC	3	26	HYPBØD	
C_L	Body Asymmetric	4.2.1.3	SUBSONIC	2	4	BØDØPT	*
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{m_α}	Bodies	4.2.2.1	SUBSONIC	2	6	BØDYRT	
			TRANSONIC	1	6	BØDYRT	Faired Between Subsonic and Supersonic
			SUPERSONIC	1	19	SUPBØD	
			HYPERSONIC	1	25	HYPBØD	
C_m	Bodies	4.2.2.2	SUBSONIC	1	6	BØDYRT	
			TRANSONIC	NDM			
			SUPERSONIC	1	19	SUPBØD	
			HYPERSONIC	1	26	HYPBØD	

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*Subject of Section 4 of this volume

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{m_o}, C_m	Body Asymmetric	4.2.2.3	SUBSONIC	NDM	4	BØDØPT	*
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{D_o}	Bodies	4.2.3.1	SUBSONIC	1	6	BØDYRT	
			TRANSONIC	1	6	BØDYRT	
			SUPERSONIC	1	19	SUPBØD	
			HYPERSONIC	2	26	HYPBØD	
C_D	Bodies	4.2.3.2	SUBSONIC	1	6	BØDYRT	
			TRANSONIC	1	6	BØDYRT	
			SUPERSONIC	1	19	SUPBØD	Excludes Elliptical Cross Sections
			HYPERSONIC	1	26	HYPBØD	Excludes Spherically-Blunted Ogive Method
C_{D_o}, C_D	Body Asymmetric	-	SUBSONIC	NDM	4	BØDØPT	*
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
α_o	Wing-Body Asymmetric	4.3.1.1	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			

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AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{L_α}	Wing-Body	4.3.1.2	SUBSONIC	1,2	7	WBLIFT	Method 1 Low AR, Method 2 Hi AR
			TRANSONIC	1	25	WBTRAN	
			SUPERSONIC	1	20	SUPWB	
			HYPERSONIC	1	20	SUPWB	Uses Supersonic Method 1
C_L	Wing-Body	4.3.1.3	SUBSONIC	1	7	WBLIFT	
			TRANSONIC	NDM	35	WBCLB	Linear Slope If No Exper. Data
			SUPERSONIC	1	7	WBLIFT	Uses Subsonic Method 1
			HYPERSONIC	1	7	WBLIFT	Uses Subsonic Method 1
$C_{L_{MAX}}$	Wing-Body	4.3.1.4	SUBSONIC	2	7	WBLIFT	
			TRANSONIC	NDM			
			SUPERSONIC	1	20	SUPWB	
			HYPERSONIC	NDM			
C_{m_o}	Wing-Body	4.3.2.1	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{m_α}	Wing-Body	4.3.2.2	SUBSONIC	1	7	WBCM	
			TRANSONIC	1	25	TRANCM	
			SUPERSONIC	1	20	SUPWB	
			HYPERSONIC	1	20	SUPWB	Uses Supersonic Method

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Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_m	Wing-Body	4.3.2.3	SUBSONIC	NDM	7	WBCM	*
			TRANSONIC	NDM			* See Section 4 for formulation of $(X_{ac}/c)_{WB}$
			SUPERSONIC	NDM			*
			HYPERSONIC	NDM			*
C_{m_o}, C_m	Wing-Body Asymmetric	4.3.2.4	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{D_o}	Wing-Body	4.3.3.1	SUBSONIC	1	7	WBDRAG	
			TRANSONIC	1	7,24	WBCDL	
			SUPERSONIC	1	20	SUPWB	
			HYPERSONIC	1	20	SUPWB	Uses Supersonic Method
C_D	Wing-Body	4.3.3.2	SUBSONIC	1	7	WBDRAG	
			TRANSONIC	1	7,24	WBCDL	
			SUPERSONIC	1	20	SUPWB	
			HYPERSONIC	1	20	SUPWB	Uses Supersonic Method
$\partial\epsilon/\partial\alpha$, q/q_∞	Wing Flow Fields	4.4.1	SUBSONIC	1	9	DWASH, DYPRLS	
			TRANSONIC	1	35	TRAWBT	
			SUPERSONIC	2	21	SDWASH, DPRESR	
			HYPERSONIC	NDM			

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Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
$\partial\epsilon/\partial\alpha$ Canards	Wing Flow Fields	4.4.1	SUBSONIC	3	9	DWASH	
			TRANSONIC	NDM			
			SUPERSONIC	3	21	SDWASH	
			HYPERSONIC	NDM			
C_{L_α}	Wing-Body-Tail	4.5.1.1	SUBSONIC	1,2	10	WBTAIL	Method 1 for $b_W >> bH$
			TRANSONIC	1	35	TRAWBT	Linearized about $C_L = 0$
			SUPERSONIC	1,2	28	SUPWBT	Method 2 for Canard Config
			HYPERSONIC	NDM			
C_L	Wing-Body-Tail	4.5.1.2	SUBSONIC	1	10	WBTAIL	
			TRANSONIC	1	35	CLWBT	
			SUPERSONIC	1	28	SUPWBT	Excludes Shock Expansion Method
			HYPERSONIC	1	28	SUPWBT	Uses Supersonic Method
$C_{L_{MAX}}$	Wing-Body-Tail	4.5.1.3	SUBSONIC	NP			
			TRANSONIC	NP			
			SUPERSONIC	NP			
			HYPERSONIC	NP			
C_{m_α}	Wing-Body-Tail	4.5.2.1	SUBSONIC	1,2	10	WBTAIL	Method 2 for Canard Config
			TRANSONIC	1	35	TRAWBT	Linearized about $C_L = 0$
			SUPERSONIC	1,2	28	SUPWBT	Method 2 for Canard Config
			HYPERSONIC	1,2	28	SUPWBT	Uses Supersonic Methods

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Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_m	Wing-Body-Tail	4.5.2.2	SUBSONIC	NDM	10	WBTAIL	*Extended Datcom Method
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{D_0}	Wing-Body-Tail	4.5.3.1	SUBSONIC	1	10	WBTAIL,VTDRAG	Untrimmed
			TRANSONIC	1	35	WBTCDØ	*
			SUPERSONIC	1	28	SUPWBT	Untrimmed
			HYPERSONIC	1	28	SUPWBT	Uses Supersonic Method
C_D	Wing-Body-Tail	4.5.3.2	SUBSONIC	1	10	WBTAIL	*Same Method All Speeds Overlay 38 for Trim
			TRANSONIC	1	35	CDWBT	
			SUPERSONIC	1	28	SUPWBT	
			HYPERSONIC	1	28	SUPWBT	
$(\Delta C_L)_{POWER}$	All	4.6.1	SUBSONIC	1	13,30	PRPWEF,JETPWE	
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
$(\Delta C_L)_{POWER_{max}}$	All	4.6.2	SUBSONIC	NP			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			

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Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
$(\Delta C_m)_{POWER}$	All	4.6.3	SUBSONIC	1	13,30	PRPWEF,JETPWE	
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
$(\Delta C_D)_{POWER}$	All	4.6.4	SUBSONIC	1	13,30	PRPWEF,JETPWE	
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
$(\Delta C_L)_{GROUND}$	All	4.7.1	SUBSONIC	1,2	11	GRDEFF	See Datcom
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
$(\Delta C_{L_{MAX}})_{GROUND}$	All	4.7.2	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
$(\Delta C_m)_{GROUND}$	All	4.7.3	SUBSONIC	1	11	GRDEFF	
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			

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AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
$(\Delta C_D)_{GROUND}$	All	4.7.4	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	2 NDM NDM NDM	11	GRDEFF	
α_0	Low Aspect Ratio Wings, Wing-Bodies	4.8.1.1	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	14	LØARWB	
C_N	Low Aspect Ratio Wings, Wing-Bodies	4.8.1.2	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	14	LØARWB	
C_{A_0}	Low Aspect Ratio Wings, Wing-Bodies	4.8.2.1	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	14	LØARWB	
C_A	Low Aspect Ratio Wings, Wing-Bodies	4.8.2.2	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	14	LØARWB	

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Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{m_o}	Low Aspect Ratio Wings, Wing-Bodies	4.8.3.1	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_m	Low Aspect Ratio Wings, Wing-Bodies	4.8.3.2	SUBSONIC	1	14	LØARWB	
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{Y_β}	Wings	5.1.1.1	SUBSONIC	1	17	SUBLAT	
			TRANSONIC	NDM			
			SUPERSONIC	1	23	SUPLAT	
			HYPERSONIC	1	23	SUPLAT	Uses Supersonic Method
C_Y @ α	Wings	5.1.1.2	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{ℓ_β}	Wings	5.1.2.1	SUBSONIC	1	17	SUBLAT	
			TRANSONIC	1	35	WINGCL	*
			SUPERSONIC	1	23	SUPLAT	
			HYPERSONIC	1	23	SUPLAT	Uses Supersonic Method

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

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Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{ℓ} @ α	Wings	5.1.2.2	SUBSONIC	NDM			See Datcom for details
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{n_β}	Wings	5.1.3.1	SUBSONIC	1	17	SUBLAT	
			TRANSONIC	NDM			
			SUPERSONIC	1	23	SUPLAT	
			HYPERSONIC	1	23	SUPLAT	Uses Supersonic Method
C_n @ α	Wings	5.1.3.2	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{Y_β}	Wing-Bodies	5.2.1.1	SUBSONIC	1	17	SUBLAT	
			TRANSONIC	1	17	SUBLAT	
			SUPERSONIC	1	23	SUPLAT	
			HYPERSONIC	1	23	SUPLAT	Uses Supersonic Method
C_{ℓ} @ α	Wing-Bodies	5.2.1.2	SUBSONIC	NDM			See Datcom for Details
			TRANSONIC	NDM			
			SUPERSONIC	NP			
			HYPERSONIC	NDM			

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Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{ℓ_β}	Wing-Bodies	5.2.2.1	SUBSONIC	1	17	SUBLAT	
			TRANSONIC	1	35	WBCLB	*Use Linear C_L if No Exper Data
			SUPERSONIC	1	23	SUPLAT	
			HYPERSONIC	1	23	SUPLAT	USES SUPERSONIC METHOD
C_ℓ @ α	Wing-Bodies	5.2.2.2	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{n_β}	Wing-Bodies	5.2.3.1	SUBSONIC	1	17	SUBLAT	
			TRANSONIC	1	17	SUBLAT	
			SUPERSONIC	1	23	SUPLAT	
			HYPERSONIC	1	23	SUPLAT	Uses Supersonic Method
C_n @ α	Wing-Bodies	5.2.3.2	SUBSONIC	NDM			See Datcom for Details
			TRANSONIC	NDM			
			SUPERSONIC	NP			
			HYPERSONIC	NDM			
C_{Y_β}	Tail-Bodies	5.3.1.1	SUBSONIC	1,2	17	SUBLAT	Method 2 for Twin Vertical Panels (on wing only)
			TRANSONIC	NDM			
			SUPERSONIC	1			
			HYPERSONIC	NP	23	SUPLAT	

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED
*Subject of Section 4 of this volume

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_Y @ α	Tail-Bodies	5.3.1.2	SUBSONIC	NDM			See Datcom for Details
			TRANSONIC	NDM			
			SUPERSONIC	NP			
			HYPERSONIC	NDM			
C_{ℓ_β}	Tail-Bodies	5.3.2.1	SUBSONIC	1	17	SUBLAT	
			TRANSONIC	NDM			
			SUPERSONIC	1	23	SUPLAT	
			HYPERSONIC	1	23	SUPLAT	
C_ℓ @ α	Tail-Bodies	5.3.2.2	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{n_β}	Tail-Bodies	5.3.3.1	SUBSONIC	1	17	SUBLAT	
			TRANSONIC	NDM			
			SUPERSONIC	1	23	SUPLAT	
			HYPERSONIC	1	23	SUPLAT	
C_n @ α	Tail-Bodies	5.3.3.2	SUBSONIC	NDM			See Datcom for Details
			TRANSONIC	NDM			
			SUPERSONIC	NP			
			HYPERSONIC	NDM			

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
$(1+\frac{\partial\sigma}{\partial\beta})\frac{q_V}{q_\infty}$	Tail-Bodies	5.4.1	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	17	SUBLAT	
$K_{Y_{\beta_o}}$	Low Aspect Ratio Wing, Wing-Bodies	5.5.1.1	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	14	LØARWB	
K_{Y_β}	Low Aspect Ratio Wing, Wing-Bodies	5.5.1.2	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	14	LØARWB	
$K'_{\ell_{\beta_o}}$	Low Aspect Ratio Wing, Wing-Bodies	5.5.2.1	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	14	LØARWB	
K'_{ℓ_β}	Low Aspect Ratio Wing, Wing-Bodies	5.5.2.2	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	14	LØARWB	

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
$K'_{n_{\beta_o}}$	Low Aspect Ratio Wings, Wing-Bodies	5.5.3.1	SUBSONIC	1	14	LØARWB	
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
K'_{n_β}	Low Aspect Ratio Wings, Wing-Bodies	5.5.3.2	SUBSONIC	1	14	LØARWB	
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{Y_β}	Wing-Body-Tails	5.6.1.1	SUBSONIC	1	17	SUBLAT	
			TRANSONIC	NDM			
			SUPERSONIC	1	23	SUPLAT	
			HYPERSONIC	NDM			
C_Y @ α	Wing-Body-Tails	5.6.1.2	SUBSONIC	NDM			See Datcom for details
			TRANSONIC	NDM			
			SUPERSONIC	NP			
			HYPERSONIC	NDM			
C_{ℓ_β}	Wing-Body-Tails	5.6.2.1	SUBSONIC	1	17	SUBLAT	
			TRANSONIC	NDM			
			SUPERSONIC	1	23	SUPLAT	
			HYPERSONIC	NDM			

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_ℓ @ α	Wing-Body-Tails	5.6.2.2	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{n_β}	Wing-Body-Tails	5.6.3.1	SUBSONIC	1	17	SUBLAT	
			TRANSONIC	NDM			
			SUPERSONIC	1	23	SUPLAT	
			HYPERSONIC	NDM			
C_n @ α	Wing-Body-Tails	5.6.3.2	SUBSONIC	NDM			See Datcom for details
			TRANSONIC	NDM			
			SUPERSONIC	NP			
			HYPERSONIC	NDM			
α_δ, C_{ℓ_δ}	Section characteristics with control devices	6.1.1.1	SUBSONIC	1	36	LIFTFP	Jet Flaps in "JETFP" overlay 55
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{ℓ_α}	Section characteristics with control devices	6.1.1.2	SUBSONIC	1	36	LIFTFP	Jet Flaps in "JETFP" overlay 55
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
$c_{\ell_{max}}$	Section characteristics with control devices	6.1.1.3	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	36	LIFTFP	
Δc_m	Section characteristics with control devices	6.1.2.1	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	2	37, 55	FLAPCM	Jet Flaps in "JETFP" overlay 55
c_{m_α}	Section characteristics with control devices	6.1.2.2	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	37, 55	FLAPCM	Jet Flaps in "JETFP" overlay 55
c_m (near $c_{\ell_{max}}$)	Section characteristics with control devices	6.1.2.3	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	37	FLAPCM	
c_{h_α}	Section characteristics with control devices	6.1.3.1	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM 1 NDM	36 41	HINGE SSHING	

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
c_{h_δ}	Section characteristics with control devices	5.1.3.2	SUBSONIC	1	36	HINGE	
			TRANSONIC	NDM			
			SUPERSONIC	1	41	SSHING	
			HYPERSONIC	NDM			
$(c_{h_f})_{\delta_t}$	Section characteristics with control devices	6.1.3.3	SUBSONIC	NP			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
$(c_{h_t})_{\delta f}$	Section characteristics with control devices	6.1.3.4	SUBSONIC	NP			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{L_δ}	Flapped Planform	6.1.4.1	SUBSONIC	1	36, 55	LIFTFP	Jet Flaps in "JETFP" overlay 55
			TRANSONIC	1	36	LIFTFP	
			SUPERSONIC	1	41	SSSYM	
			HYPERSONIC	NDM			
C_{L_α}	Flapped Planform	6.1.4.2	SUBSONIC	1	41, 55	SSSYM	Jet Flaps in "JETFP" overlay 55
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
$C_{L_{MAX}}$	Flapped Planform	6.1.4.3	SUBSONIC	1	36, 55	LIFTFP	Jet Flaps in "JETFP" overlay 55
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
ΔC_m	Flapped Planform	6.1.5.1	SUBSONIC	2	37, 55	FLAPCM	Jet Flaps in "JETFP" overlay 55
			TRANSONIC	1	37	FLAPCM	
			SUPERSONIC	1	41	SSSYM	
			HYPERSONIC	NDM			
C_{m_α}	Flapped Planform	6.1.5.2	SUBSONIC	1	37, 55	FLAPCM	Jet Flaps in "JETFP" overlay 55
			TRANSONIC	1	37	FLAPCM	
			SUPERSONIC	1	37	FLAPCM	
			HYPERSONIC	1	37	FLAPCM	
C_{h_α}	Flapped Planform	6.1.6.1	SUBSONIC	1	36	HINGE	
			TRANSONIC	NDM			
			SUPERSONIC	1	41	SSHING	
			HYPERSONIC	NDM			
C_{h_δ}	Flapped Planform	6.1.6.2	SUBSONIC	1	36	HINGE	
			TRANSONIC	NDM			
			SUPERSONIC	1	41	SSHING	
			HYPERSONIC	NDM			

NDM-NO DATCOM METHOD   NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_D	Flapped Planform	6.1.7	SUBSONIC	1	38	DRAGFP	
			TRANSONIC	NDM			
			SUPERSONIC	NP			
			HYPERSONIC	NDM			
C_{ℓ_δ}	Flapped Planform	6.2.1.1	SUBSONIC	1	52	LATFLP	
			TRANSONIC	1	40	TRNYRL	
			SUPERSONIC	1	53	SPRYAW	
			HYPERSONIC	NDM			
$C_{\ell_{\delta_H}}$	Flapped Planform	6.2.1.2	SUBSONIC	1	52	LATFLP	
			TRANSONIC	NP			
			SUPERSONIC	NP			
			HYPERSONIC	NDM			
C_{n_δ}	Flapped Planform	6.2.2.1	SUBSONIC	1	52	LATFLP	
			TRANSONIC	1	40	TRNYRL	
			SUPERSONIC	1	53	SPRYAW	
			HYPERSONIC	NDM			
C_{Y_δ}	Flapped Planform	6.2.3	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
Hypersonic Control Effective-ness	Tail-Bodies	6.3.1	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	1	42	HYPFLP	
Transverse-Jet Control Effective-ness	All	6.3.2	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	1	47	TRANJT	
Inertial Controls		6.3.3	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
Aerodyna-mically Boosted Tabs	Tabbed Planform	6.3.4	SUBSONIC	1	36	CTABS	
			TRANSONIC	1	36	CTABS	Below Mach 0.9 (See Datcom)
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{L_q}	Wings	7.1.1.1	SUBSONIC	1	43	SUBPAW	
			TRANSONIC	1	43	SUBPAW	Uses subsonic method
			SUPERSONIC	1	43	SUPPAW	
			HYPERSONIC	NDM			

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Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{m_q}	Wings	7.1.1.2	SUBSONIC	1	43	SUBPAW	
			TRANSONIC	1	43	SUBPAW	
			SUPERSONIC	1	43	SUPCMQ	
			HYPERSONIC	NDM			
C_{Y_p}	Wings	7.1.2.1	SUBSONIC	1	45	SUBRYW	
			TRANSONIC	NDM			
			SUPERSONIC	1	45	SUPRYW	
			HYPERSONIC	NDM			
C_{ℓ_p}	Wings	7.1.2.2	SUBSONIC	1	45	SUBRYW	
			TRANSONIC	NDM			
			SUPERSONIC	1	45	SUPRYW	
			HYPERSONIC	NDM			
C_{n_p}	Wings	7.1.2.3	SUBSONIC	1	45	SUBRYW	
			TRANSONIC	NDM			
			SUPERSONIC	1	45	SUPRYW	
			HYPERSONIC	NDM			
C_{Y_r}	Wings	7.1.3.1	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			

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Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{ℓ_r}	Wings	7.1.3.2	SUBSONIC	1	45	SUBRYW	
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{n_r}	Wings	7.1.3.3	SUBSONIC	1	45	SUBRYW	
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
$C_{L_{\dot{\alpha}}}$	Wings	7.1.4.1	SUBSONIC	1	43	SUBPAW	
			TRANSONIC	1	43	SUBPAW	Triangular wings only
			SUPERSONIC	1	44	SUPCLD	
			HYPERSONIC	NDM			
$C_{m_{\dot{\alpha}}}$	Wings	7.1.4.2	SUBSONIC	1	43	SUBPAW	
			TRANSONIC	1	43	SUBPAW	Straight tapered wings only
			SUPERSONIC	1	54	SUPCMD	
			HYPERSONIC	NDM			
C_{L_q}	Bodies	7.2.1.1	SUBSONIC	1	43	SUBPAW	
			TRANSONIC	1	43	SUBPAW	Uses subsonic method
			SUPERSONIC	1	43	SUPPAW	
			HYPERSONIC	1	43	SUPPAW	

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{m_q}	Bodies	7.2.1.2	SUBSONIC	1	46	DYNBØD	
			TRANSONIC	1	46	DYNBØD	Uses subsonic method
			SUPERSONIC	1	46	DYNBØD	
			HYPERSONIC	1	46	DYNBØD	
$C_{L_{\dot{\alpha}}}$	Bodies	7.2.2.1	SUBSONIC	1	46	DYNBØD	
			TRANSONIC	1	46	DYNBØD	Uses subsonic method
			SUPERSONIC	1	46	DYNBØD	
			HYPERSONIC	1	46	DYNBØD	
$C_{m_{\dot{\alpha}}}$	Bodies	7.2.2.2	SUBSONIC	1	46	DYNBØD	
			TRANSONIC	1	46	DYNBØD	Uses subsonic method
			SUPERSONIC	1	46	DYNBØD	
			HYPERSONIC	1	46	DYNBØD	
C_{L_q}	Wing-Bodies	7.3.1.1	SUBSONIC	1	46	DNPAWB	
			TRANSONIC	1	46	DNAPWB	Uses subsonic method
			SUPERSONIC	1	46	DNPAWB	
			HYPERSONIC	NDM			
C_{m_q}	Wing-Bodies	7.3.1.2	SUBSONIC	1	46	DNPAWB	
			TRANSONIC	1	46	DNPAWB	Uses subsonic method
			SUPERSONIC	1	46	DNPAWB	
			HYPERSONIC	NDM			

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{Y_p}	Wing-Bodies	7.3.2.1	SUBSONIC	1	45	SUBRYW	Uses wing method (7.1.2.1)
			TRANSONIC	NDM			
			SUPERSONIC	1	45	SUPRYW	Uses wing method (7.1.2.1)
			HYPERSONIC	NDM			
C_{ℓ_p}	Wing-Bodies	7.3.2.2	SUBSONIC	1	45	SUBRYW	Uses wing method (7.1.2.2)
			TRANSONIC	NDM			
			SUPERSONIC	1	45	SUPRYW	Uses wing method (7.1.2.2)
			HYPERSONIC	NDM			
C_{n_p}	Wing-Bodies	7.3.2.3	SUBSONIC	1	45	SUBRYW	Uses wing method (7.1.2.3)
			TRANSONIC	NDM			
			SUPERSONIC	1	45	SUPRYW	Uses wing method (7.1.2.3)
			HYPERSONIC	NDM			
C_{Y_r}	Wing-Bodies	7.3.3.1	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
C_{ℓ_r}	Wing-Bodies	7.3.3.2	SUBSONIC	1	45	SUBRYW	Uses wing method (7.1,3.2)
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			

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Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{n_r}	Wing-Bodies	7.3.3.3	SUBSONIC	1	45	SUBRYW	Uses wing method (7.1.3.3)
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
$C_{L_{\dot{\alpha}}}$	Wing-Bodies	7.3.4.1	SUBSONIC	1	46	DNPAWB	Uses subsonic method
			TRANSONIC	1	46	DNPAWB	
			SUPERSONIC	1	46	DNPAWB	
			HYPERSONIC	NDM			
$C_{m_{\dot{\alpha}}}$	Wing-Bodies	7.3.4.2	SUBSONIC	1	46	DNPAWB	Uses subsonic method
			TRANSONIC	1	46	DNPAWB	
			SUPERSONIC	1	46	DNPAWB	
			HYPERSONIC	NDM			
C_{L_q}	Wing-Body-Tails	7.4.1.1	SUBSONIC	1, 2	46	DNPWBT	All use subsonic methods. Method 2 for canard config.
			TRANSONIC	1, 2	46	DNPWBT	
			SUPERSONIC	1, 2	46	DNPWBT	
			HYPERSONIC	NDM			
C_{m_q}	Wing-Body-Tails	7.4.1.2	SUBSONIC	1, 2	46	DNPWBT	All use subsonic methods. Method 2 for canard config.
			TRANSONIC	1, 2	46	DNPWBT	
			SUPERSONIC	1, 2	46	DNPWBT	
			HYPERSONIC	NDM			

NDM-NO DATCOM METHOD   NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{Y_p}	Wing-Body-Tails	7.4.2.1	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	2 NDM NDM NDM	46	SUBWBT	
C_{ℓ_p}	Wing-Body-Tails	7.4.2.2	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	46	SUBWBT	
C_{n_p}	Wing-Body-Tails	7.4.2.3	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	2 NDM NDM NDM	46	SUBWBT	
C_{Y_r}	Wing-Body-Tails	7.4.3.1	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	NP NDM NDM NDM			
C_{ℓ_r}	Wing-Body-Tails	7.4.3.2	SUBSONIC TRANSONIC SUPERSONIC HYPERSONIC	1 NDM NDM NDM	46	SUBWBT	

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

Table 1 SUMMARY OF DIGITAL DATCOM METHODS

AERODYNAMIC PARAMETER	CONFIGURATION	DATCOM SECTION	MACH REGIME	METHOD NUMBER	OVERLAY	SUBROUTINE	REMARKS
C_{n_r}	Wing-Body-Tails	7.4.3.3	SUBSONIC	1	46	SUBWBT	
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			
$C_{L_{\dot{\alpha}}}$	Wing-Body-Tails	7.4.4.1	SUBSONIC	1, 2	46	DNPWBT	All use subsonic methods. Method 2 for canard config.
			TRANSONIC	1, 2	46	DNPWBT	
			SUPERSONIC	1, 2	46	DNPWBT	
			HYPERSONIC	NDM			
$C_{m_{\dot{\alpha}}}$	Wing-Body-Tails	7.4.4.2	SUBSONIC	1, 2	46	DNPWBT	All use subsonic methods. Method 2 for canard config.
			TRANSONIC	1, 2	46	DNPWBT	
			SUPERSONIC	1, 2	46	DNPWBT	
			HYPERSONIC	NDM			
Control surface angular velocity derivatives		7.5	SUBSONIC	NDM			
			TRANSONIC	NDM			
			SUPERSONIC	NDM			
			HYPERSONIC	NDM			

NDM-NO DATCOM METHOD NP-NOT PROGRAMMED

TABLE 2 OVERLAYS DEFINING EACH OF THE BASIC OUTPUT SUBSONIC PARAMETERS

CONFIGURATION	STATIC STABILITY					STATIC STABILITY DERIV					DYNAMIC STABILITY DERIVATIVES									
	C_D	C_L	C_m	C_N	C_A	C_{L_α}	C_{m_α}	C_{Y_β}	C_{n_β}	C_{ℓ_β}	C_{L_q}	C_{m_q}	$C_{L_{\dot\alpha}}$	$C_{m_{\dot\alpha}}$	C_{ℓ_p}	C_{Y_p}	C_{n_p}	C_{n_r}	C_{ℓ_r}	–
BODY - B ASY. SYM.	4 6	4 6	4 6	4 6	4 6	4 6	4 6	4 6	4 6	4 6	46	46	46	46					46	
WING - W	3	15	31	15 31	31	15 31	31	17	17	17	43	43	43	43	45	45	45	45	46	
HORIZONTAL TAIL—HT	5	16	33	16 33	33	16 33	33	17	17	17	46	46	46	46	46	46	46	46		
VERTICAL TAIL-VT OR VENTRAL FIN-F	8	8	8	8	8	8	8	17	17	17	46	46	46	46	46	46	46	46	46	
B + W OR LOW AR WING-BODY	7,11 14	7,11 14	7,11 14	7,11 14	7,11 14	7,11 14	7,11 14	17 14	17 14	17 14	46	46	46	46	46	46	46	46	46	
B+H	7	7	7	7	7	7	7	17	17	17	46	46	46	46	45	46	46	46		
B+V OR B+V+F	7	7	7	7	7	7	7	17	17	17	46	46	46	46	46	46	46	46		
B+W+H	10 11	10 11	10 11	10 11	10 11	10 11	10 11	17	17	17	46	46	46	46	46	46	46	46		
B+W+V OR B+W+V+F	10 11	10 11	10 11	10 11	10 11	10 11	10 11	17	17	17	46	46	46	46	46	46	46	46		
B+W+H+V OR B+W+H+V+F	10 11	10 11	10 11	10 11	10 11	10 11	10 11	17	17	17	46	46	46	46	45	46	46	46	46	
POWER INCREMENTS	ΔC_D 13,30	ΔC_L 13,30	ΔC_m 13,30	ΔC_n 13,30	ΔC_A 13,30	ΔC_{L_α} 13,30	ΔC_{M_α} 13,30	ΔC_{Y_β}	ΔC_{n_β}	ΔC_{ℓ_β}										
DOWNWASH DATA	q/q_∞ 9	ϵ 9	$\partial\epsilon/\partial x$ 9																	

TABLE 3 OVERLAYS DEFINING EACH OF THE BASIC TRANSONIC OUTPUT PARAMETERS

CONFIGURATION	STATIC STABILITY					STATIC STABILITY DERIV					DYNAMIC STABILITY DERIVATIVES									
	C_D	C_L	C_m	C_N	C_A	C_{L_α}	C_{m_α}	C_{Y_β}	C_{n_β}	C_{ℓ_β}	C_{L_q}	C_{m_q}	$C_{L_{\dot\alpha}}$	$C_{m_{\dot\alpha}}$	C_{ℓ_p}	C_{Y_p}	C_{n_p}	C_{n_r}	C_{ℓ_r}	-
BODY - B	24			L2	L2	24	24	24	24	24	46	46	46	46					46	
WING - W	24 L2	L2		L2	L2	24	25			L2	43	43	43	43						
HORIZONTAL TAIL—HT	24 L2	L2		L2	L2	24	25			L2	46	46	46	46						
VERTICAL TAIL-VT OR VENTRAL FIN-F	L2	35	35	35	35	35	35				46	46	46	46						
B+W	24 L2	35		L2	L2	25	25	17	17	35	46	46	46	46						
B+H	24 L2	35		L2	L2	25	25	17	17	35	46	46	46	46						
B+V OR B+V+F	L2	L2		L2	L2	35	35				46	46	46	46						
B+W+H	L2	L2		L2	L2	35	35				46	46	46	46						
B+W+V OR B+W+V+F	L2	L2		L2	L2	35	35				46	46	46	46						
B+W+H+V OR B+W+H+V+F	L2	L2		L2	L2	35	35				46	46	46	46						
POWER INCREMENTS	ΔC_D	ΔC_L	ΔC_m	ΔC_N	ΔC_A	ΔC_{L_α}	ΔC_{m_α}	ΔC_{Y_β}	ΔC_{n_β}	ΔC_{ℓ_β}										
DOWNWASH DATA	q/q_∞ 35	ϵ 35	$\partial\epsilon/\partial\alpha$ 35																	

L2 - SECOND LEVEL METHODS, OVERLAY 35

TABLE 4 OVERLAYS DEFINING EACH OF THE BASIC SUPERSONIC–HYPERSONIC OUTPUT PARAMETERS

CONFIGURATION	STATIC STABILITY					STATIC STABILITY DERIV.					DYNAMIC STABILITY DERIVATIVES									
	C_D	C_L	C_m	C_N	C_A	C_{L_α}	C_{m_α}	C_{Y_β}	C_{n_β}	C_{ℓ_β}	C_{L_q}	C_{m_q}	$C_{L_{\dot\alpha}}$	$C_{m_{\dot\alpha}}$	C_{ℓ_p}	C_{Y_p}	C_{n_p}	C_{n_r}	C_{ℓ_r}	–
BODY B SUPERSONIC	19	19	19	19	19	19	19	19	19	19	46	46	46	46					46	
BODY B HYPERSONIC	26	26	26	26	26	26	26	26	26	26										
WING W	27	27		27	27	27	27	23	23	23	43	43	44	54	45	45	45			
HORIZONTAL TAIL–HT	22	22		22	22	22	22	23	23	23	46	46	46	46	45	45	45			
VERTICAL TAIL VT OR VENTRAL FIN F	20	20	20	20	20	20	20	23	23	23	46	46	46	46						
B+W	20	20		20	20	20	20	23	23	23	46	46	46	46	45	45	45			
B+H	20	20		20	20	20	20	23	23	23	46	46	46	46	45	45	45			
B+V OR B+V+F	20	20		20	20	20	20	23	23	23	46	46	46	46						
B+W+H	28	28		28	28	28	28	23	23	23	46	46	46	46						
B+W+V OR B+W+V+F	20	20		20	20	20	20	23	23	23	46	46	46	46						
B+W+H+V OR B+W+H+V+F	28	28		28	28	28	28	23	23	23	46	46	46	46						
POWER INCREMENTS	ΔC_D	ΔC_L	ΔC_m	ΔC_N	ΔC_A	ΔC_{L_α}	ΔC_{m_α}	ΔC_{Y_β}	ΔC_{n_β}	ΔC_{ℓ_β}										
DOWNWASH DATA	q/q_∞ 21	ϵ 21	$\partial\epsilon/\partial\alpha$ 21																	

SECTION 2

PROGRAM ORGANIZATION

The Digital Datcom program consists of a MAIN program, EXECUTIVE subroutines, METHOD subroutines and UTILITY subroutines. The organization and interfaces between these program components are shown in Figure 1. The MAIN program performs executive functions that control and direct all computations; the EXECUTIVE subroutines perform noncomputational tasks, which include input data manipulation and selection of output formats; UTILITY subroutines perform standard mathematical computations; and METHOD subroutines implement the Datcom stability methods.

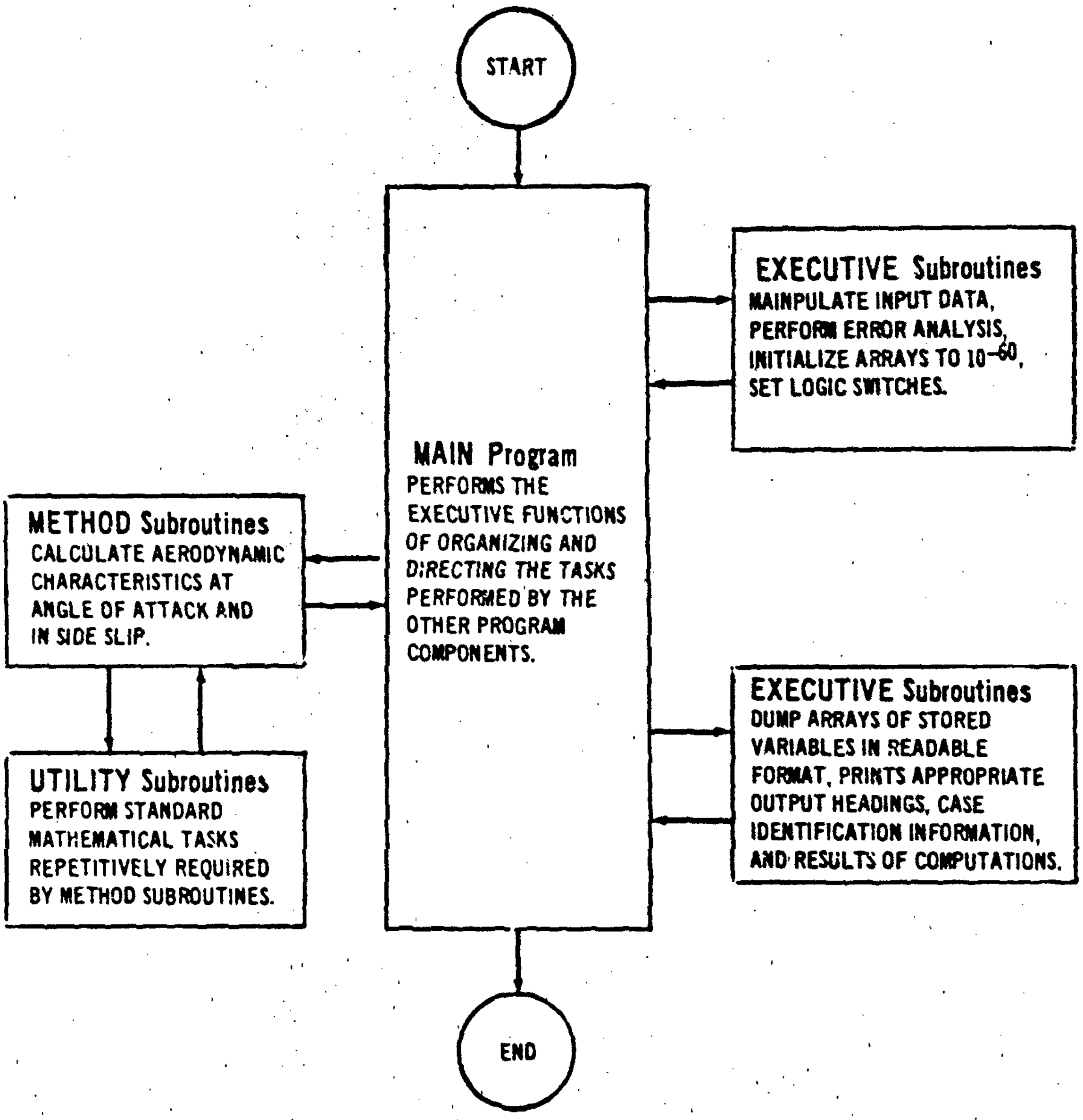


FIGURE 1 OVERLAY PROGRAM STRUCTURE

SECTION 3

EQUATIONS FOR GEOMETRIC PARAMETERS

One of the main features of the Digital Datcom program is that a minimum of input data are required. Minimal inputs require the program to calculate basic geometric parameters required by the Datcom methods. Equations for pertinent geometric parameters are defined in this section.

3.1 PLANFORM PARAMETERS

The nomenclature used in the equations for calculating theoretical and exposed planform areas, taper ratios and aspect ratios are shown in Figure 2. Equations for these parameters are presented below for a double delta or cranked planform. Straight-tapered planform parameters are obtained by setting $b^*_o/2 = 0.0$, $C_b = C_t$, $A^*_o = 1.0$ in the following equations:

$$b_b/2 = b/2 - b_o^*/2$$

$$b_b^*/2 = b^*/2 - b_o^*/2$$

$$r_b^* = (b_b^*/2)/(b_b/2)$$

$$\lambda_I = C_b/C_r$$

$$C_r^* = C_r[\lambda_I + (1 - \lambda_I)\, r_b^*]$$

$$\lambda_I^* = C_b/C_r^*$$

$$\lambda_o^* = C_t/C_b$$

$$\lambda_w^* = \lambda_I^* \lambda_o^*$$

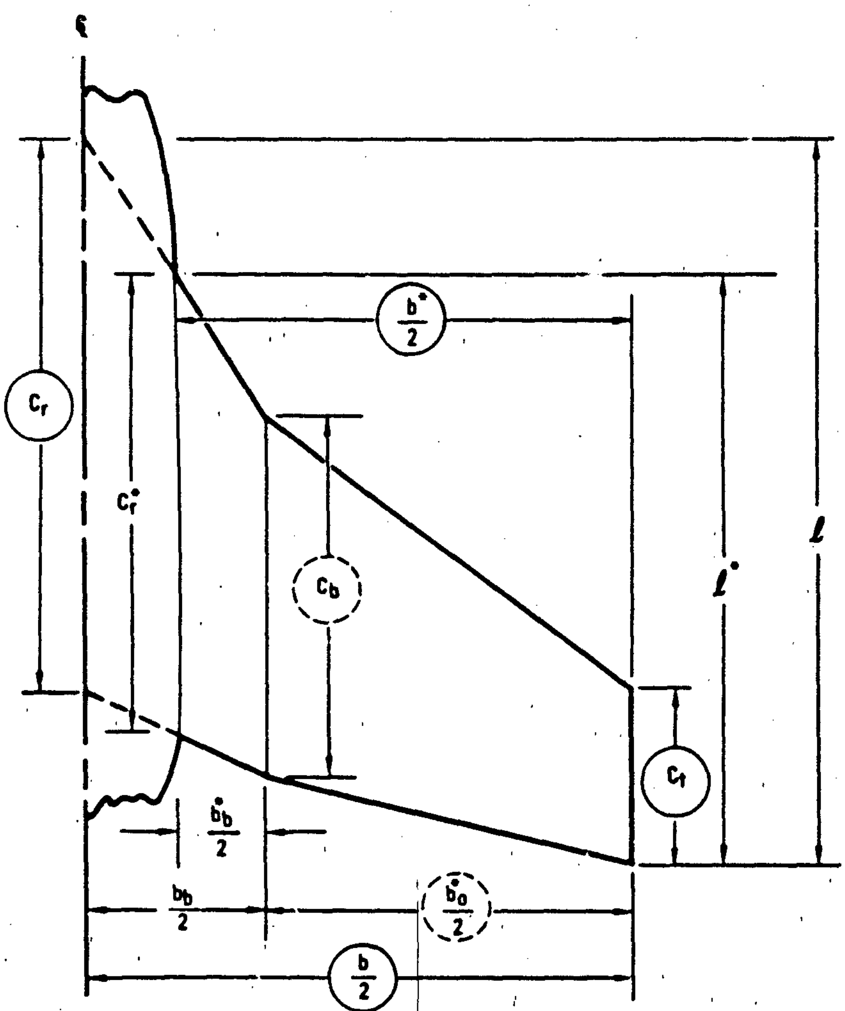
$$\lambda_w = C_t/C_r$$

$$S_I^* = (C_r^* + C_b)\, b_b^*/2$$

$$S_I = (C_r + C_b)\, b_b/2$$

$$S_o^* = (C_b + C_t)\, b_o^*/2$$

$$S_W^* = S_I^* + S_o^*$$



REQUIRED INPUTS ALL PLANFORMS

REQUIRED INPUTS DOUBLE DELTA AND CRANKED PLANFORMS

* INDICATES EXPOSED PARAMETER

λ = TAPER RATIO

A = ASPECT RATIO

S = AREA

$\frac{b}{2}$ = SEMI-SPAN

SUBSCRIPTS

I = INBOARD PANEL

O = OUTBOARD PANEL

w = TOTAL WING

FIGURE 2 PLANFORM NOMENCLATURE

$$S_W = (C_r + C_b)\ b_b/2 + S_o^*$$

$$A_I^* = 4(b_b^*/2)^2/S_I^*$$

$$A_o^* = 4(b_o^*/2)^2/S_o^*$$

$$A_W^* = 4(b^*/2)^2/S_W^*$$

$$A_W = 4(b/2)^2/S_W$$

Datcom methods use correlations that are based on wing sweep angles measured at various chordlines. The nomenclature used to calculate sweep angles is presented in Figure 3. Sweep angle equations are presented below for a double delta or cranked wing. To obtain straight taper wing sweep angles set C_o and $\Lambda_{n_o} = 0$ in the following equations:

$$C_I = 4(1 - \lambda^*_I)/[A^*_I(1 + \lambda^*_I)]$$

$$C_0 = 4(1 - \lambda^*_0)/[A^*_0(1 + \lambda^*_0)]$$

$$\Lambda n_I = \tan^{-1}[C_I(m-n) + \tan\Lambda m_I]$$

$$\Lambda n_o = \tan^{-1}[C_o(m-n) + \tan\Lambda m_o]$$

$$(\Lambda_n^*)_{eff} = \cos^{-1}](S_I^* \cos \Lambda n_I + S_o^* \cos \Lambda n_o)/S^*]$$

The nomenclature used to calculate the exposed mean aerodynamic chord (MAC) for a double delta or cranked wing is shown in Figure 4. The parameters necessary to define the lateral and longitudinal location of the exposed MAC are included. Equations to calculate and locate the MAC are presented below. To obtain values for a straight-tapered wing set $C^*_o = 0$, $Y^*_o = 0$, $S^*_o = 0$ in the equations below:

$$\bar{C}_I^* = 2C_r^*(1 + \lambda_I^* + \lambda_I^{*2})/3(1 + \lambda_I^*)$$

$$\bar{C}_o^* = 2C_b(1 + \lambda_o^* + \lambda_o^{*2})/3(1 + \lambda_o^*)$$

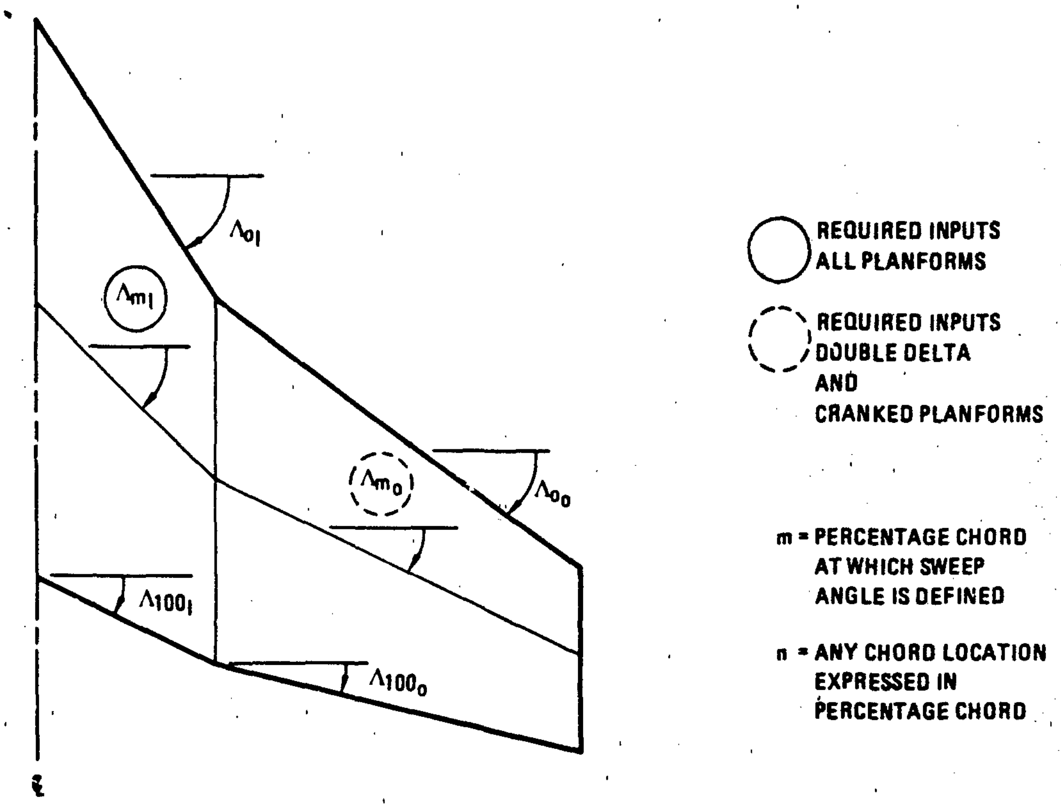


FIGURE 3 SWEEP ANGLE NOMENCLATURE

$$\overline{C}^*_W = (S^*_I \, \overline{C}^*_I + S^*_o \, \overline{C}^*_o)/S^*$$

$$\overline{Y}^*_I = (b^*_b/2)(1 + 2\lambda^*_I)/3(1 + \lambda^*_I)$$

$$\overline{Y}^*_o = (b^*_o/2)(1 + 2\lambda^*_o)/3(1 + \lambda^*_o) + b^*_b/2$$

$$\overline{Y}^* = (S^*_I \, \overline{Y}^*_I + S^*_o \, \overline{Y}^*_o)/S^*$$

$$X^*_r = [S^*_I \, \overline{Y}^*_I \tan\Lambda o_I + S^*_o (b^*_b/2 \tan\Lambda o_I + (\overline{Y}^*_o - b^*_b/2) \tan\Lambda o_o)]/S^*$$

$$\overline{X}^* = \overline{C}^*_w/2 + X^*_r$$

$$\overline{X}^*_r = \overline{C}^*_w/4 + X^*_r$$

The theoretical or reference mean aerodynamic chord is calculated with nomenclature of Figure 5 as follows:

$$\overline{C}_I = 2C_r(1 + \lambda_I + \lambda^2_I)/3(1 + \lambda_I)$$

$$\overline{C}_r = (S_I \, \overline{C}_I + S_o \, \overline{C}_o)/S_r$$

$$\overline{X}_r = \overline{C}_r/4 + X_r$$

Special geometric parameters are required to calculate wing pitching moments. The nomenclature used to define these parameters is presented in Figure 6. Equations for these parameters are presented below:

$$\varepsilon^* = (b^*_b/2 \tan\Lambda o_I + b^*_o/2 \tan\Lambda o_o)/C^*{}_r$$

$$A_I = 4(b_b/2)^2/S_I$$

$$\Delta Y' = b^*_b/4$$

$$(b^*_o/2)' = b^*_b/4 + b^*_o/2$$

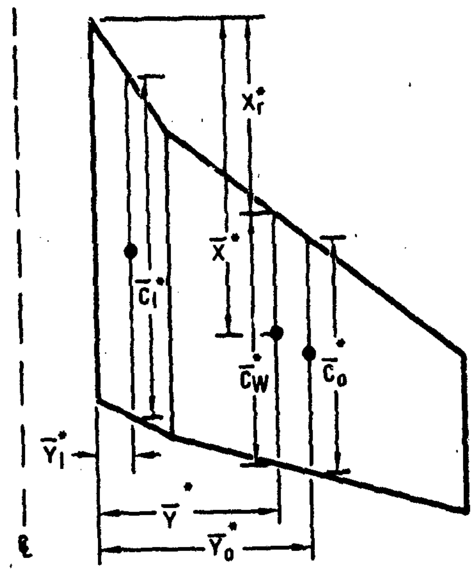


FIGURE 4 EXPOSED MEAN AERODYNAMIC CHORD NOMENCLATURE

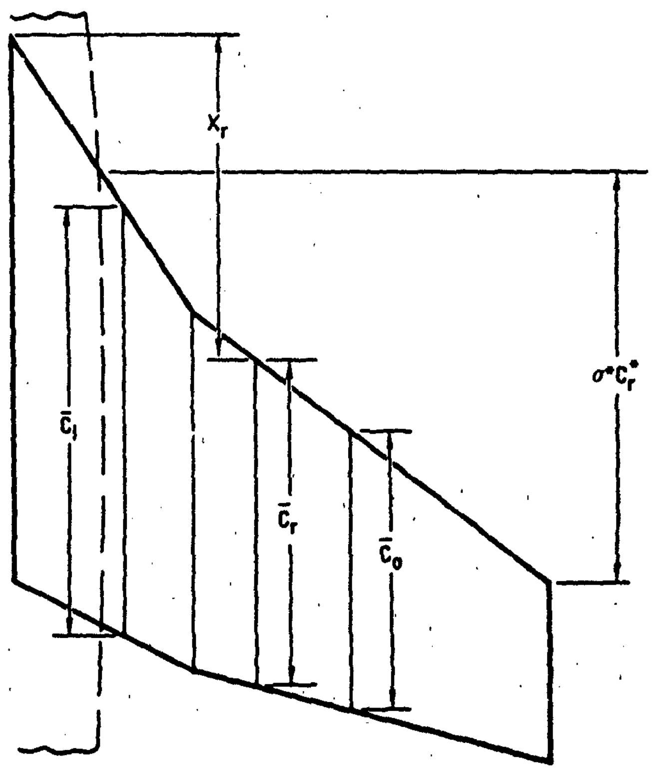


FIGURE 5 THEORETICAL OR REFERENCE MEAN AERODYNAMIC CHORD NOMENCLATURE

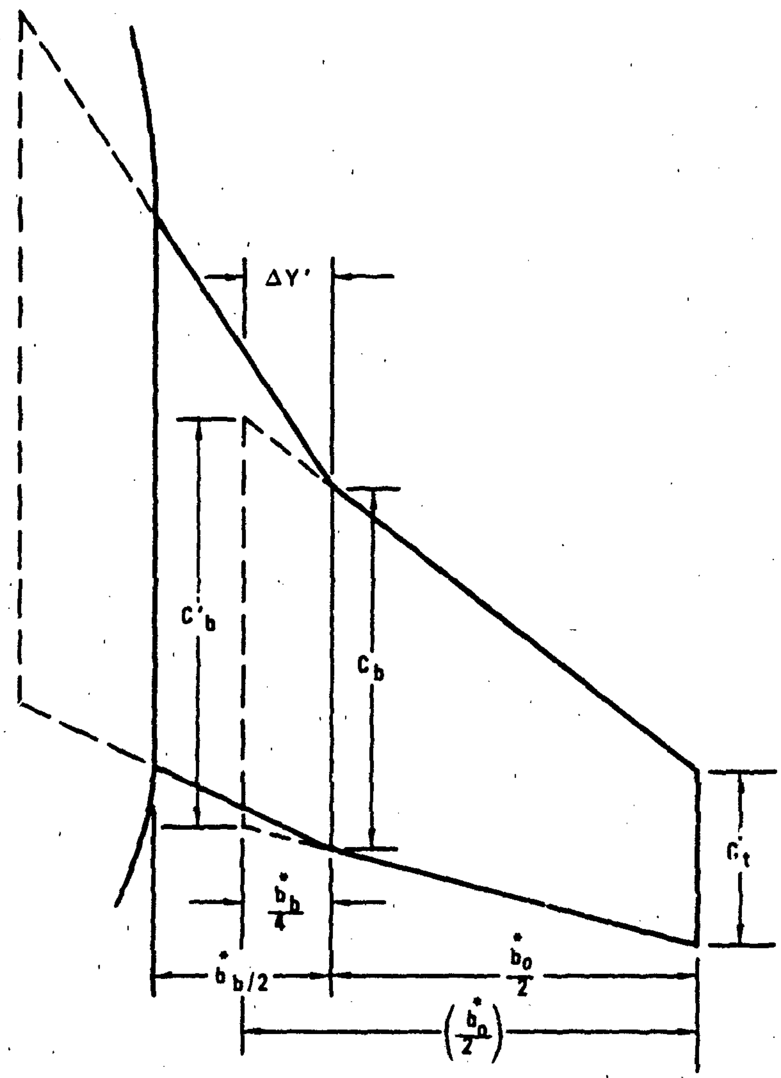


FIGURE 6 SPECIAL WING PITCHING MOMENT GEOMETRY

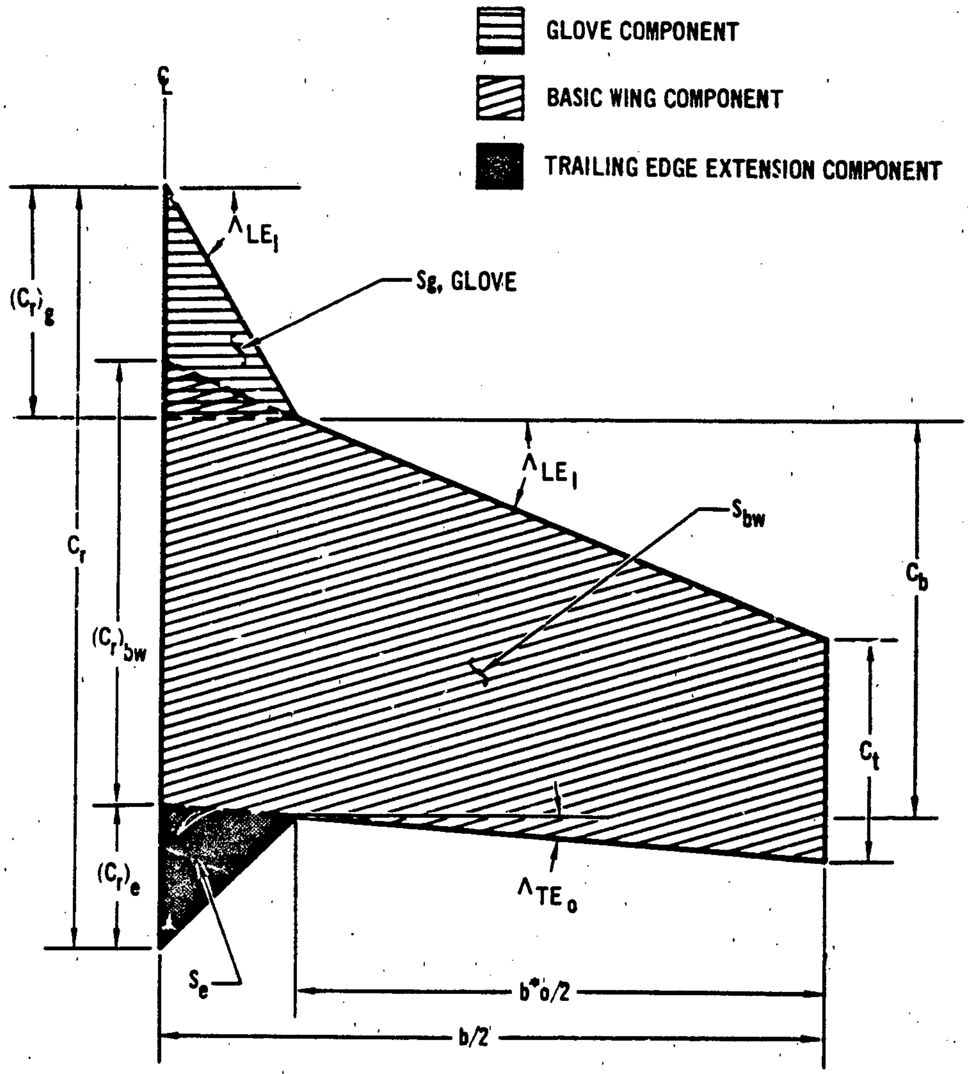


FIGURE 7 SUPERSONIC NON-STRAIGHT WING PLANFORM ($\Lambda_{LE_0} < \Lambda_{LE_I}$)

$$C_b' = C_t + (b_o^*/2)' \left[\frac{C_b - C_t}{b_o^*/2}\right]$$

$$(S_o^*)' = (C_b' + C_t)\ (b_o^*/2)'$$

$$(A_o)' = 4[(b_o^*/2)^2]'/(S_o^*)'$$

$$(\lambda_o^*)' = C_t/C_b'$$

Supersonic nonstraight wing analyses require the wing to be synthesized from basic wing, glove, and trailing edge extension components as shown on Figure 7. When the leading edge outboard sweep angle is greater than the leading edge inboard sweep angle, an additional geometric parameter, S_2, is required and is shown in Figure 8. Equations for calculating geometric parameters for the various wing components as required by the stability methods are presented below:

<u>All Planforms</u>

$$(C_r^*)_{bw} = C_b + \left[\frac{b^*}{2} - \frac{b_o^*}{2}\right][\text{Tan}\ \Lambda_{LE_o} - \text{Tan}\ \Lambda_{TE_o}]$$

basic wing component

$$S_{bw}^* = \frac{[(C_r^*)_{bw} + C_t]\ b^*}{2}$$

$$A_{bw}^* = \frac{b^{*2}}{S^*_{bw}}$$

$$\lambda^*_{bw} = \frac{C_t}{(C_r^*)_{bw}}$$

glove component

$$(C_r^*)_g = (\text{Tan}\ \Lambda_{LE_I}) \left(\frac{b^*}{2} - \frac{b_o^*}{2}\right)$$

$$S_g^* = (C_r^*)_g \left(\frac{b^*}{2} - \frac{b_o^*}{2}\right)$$

$$A_g^* = \frac{4\left[\frac{b^*}{2} - \frac{b_o^*}{2}\right]^2}{S_g^*}$$

$$\lambda_g = 0$$

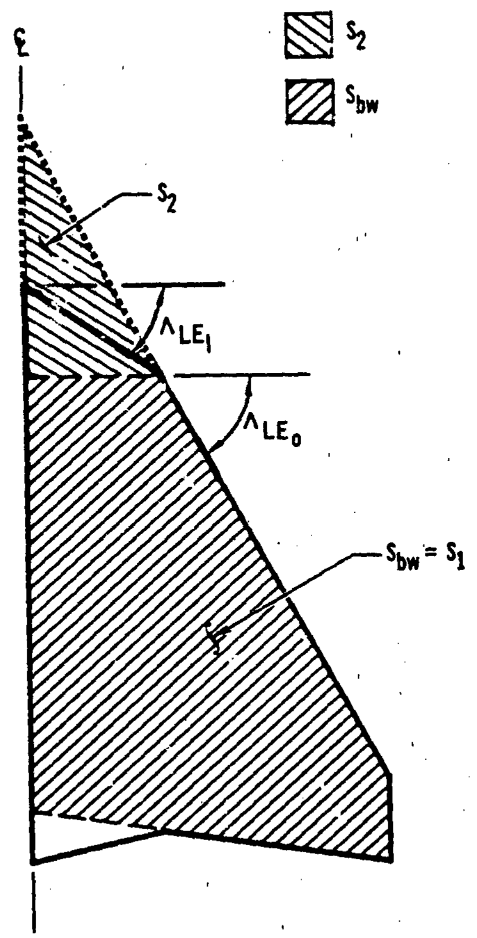


FIGURE 8 SUPERSONIC NON-STRAIGHT WING PLANFORM ($\Lambda_{LE_0} > \Lambda_{LE_I}$)

trailing edge extension span $$b_e^* = 2\left(\frac{b^*}{2} - \frac{b_o^*}{2}\right)$$

If $\Lambda_{LE_o} > \Lambda_{LE_I}$ $$S^*_2 = \left[\frac{b^*}{2} - \frac{b_o^*}{2}\right] (\tan \Lambda_{LE_o})$$

$$S_1 = S_{bw}$$

Geometric parameters required for horizontal and vertical tail analyses are identical to those for wings. Tail parameters can be calculated by substituting tail geometry for wing geometry in the wing equations. Vertical tail lateral stability calculations require additional geometry parameters as shown in Figures 9a and 9b. Equations are listed below:

<u>Straight Tapered Vertical Tail</u>

$$C_v = C_r - (C_r - C_t)(Z_H)/(b_v/2)$$

$$X = X_H + (\overline{X}_R)_v - X_v - Z_H (\text{Tan } \Lambda_{LE_I})$$

<u>Non-Straight Vertical Tail</u>

If $Z_H > \frac{b_v}{2} - \frac{b_o^*}{2}$

$$X = X_H + (\overline{X}_R)_v - X_v - \left(\frac{b_v}{2} - \frac{b_o^*}{2}\right) (\text{TAN } \Lambda_{LE_I}) - \left(Z_H + \frac{b_o^*}{2} - \frac{b_v}{2}\right) \text{TAN } \Lambda_{LE_o}$$

$$C_v = C_t + (C_b - C_t)\left(\frac{b_v}{2} - Z_H\right)/\left(\frac{b_o^*}{2}\right)$$

If $ZH \leq \frac{b_v}{2} - \frac{b_o^*}{2}$

$$X = X_H + \overline{X}_R - X_v - Z_H (\text{TAN } \Lambda_{LE_o})$$

$$C_v = C_r - (C_r - C_b)(Z_H)/\left(\frac{b_v}{2} - \frac{b_o^*}{2}\right)$$

For a horizontal lifting surface, an equivalent dihedral is defined as follows:

$$\Gamma_{eq} = \frac{\Gamma_i \left(\frac{b_i^*}{2}\right) + \Gamma_o \left(\frac{b_o^*}{2}\right)_{\Gamma_o}}{\frac{b^*}{2}}$$

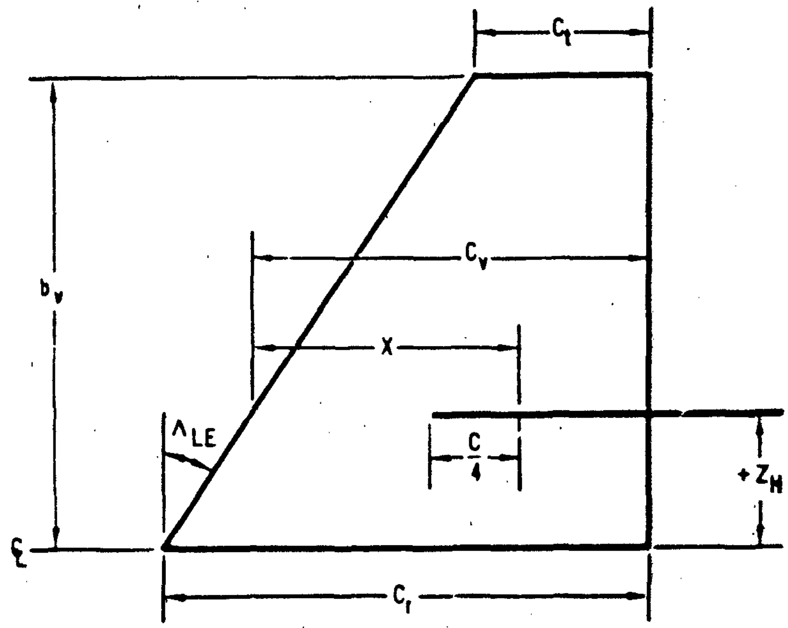


FIGURE 9(a) STRAIGHT TAPERED VERTICAL TAIL GEOMETRY

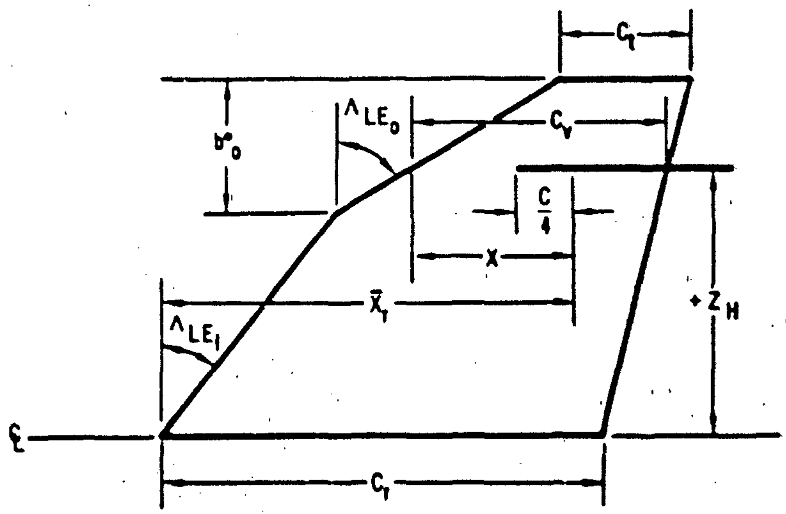


FIGURE 9(b) NON-STRAIGHT TAPERED VERTICAL TAIL GEOMETRY

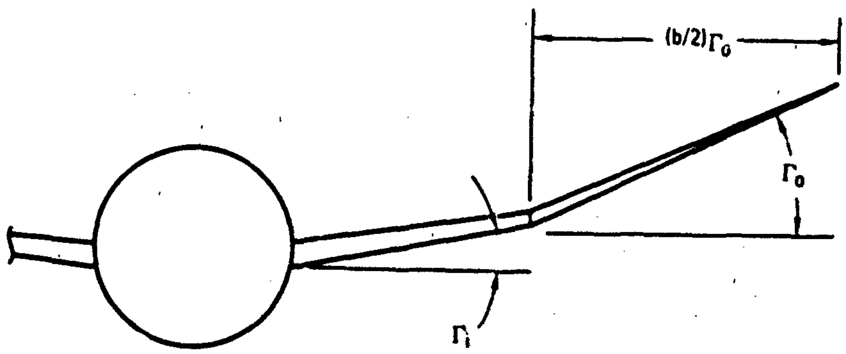


FIGURE 10 EQUIVALENT DIHEDRAL ANGLE NOMENCLATURE

3.2 BODY PARAMETERS

Longitudinal stability analyses for bodies in the supersonic and hypersonic speed regimes require the body to be synthesized in nose, afterbody, and tail segment components as defined in Figure 11. Geometry parameters for the various body segments analyses are defined below:

$$\ell'_B = \ell_N + \ell_A$$

$$\ell_{BT} = \ell_B - \ell'_B$$

$$d_{cyl} = \frac{d_1 + d_N}{2}$$

$$S_p = 2\int_o^{\ell_B} r_x \,(dx) \quad \text{Body planform area}$$

$$S_b = \frac{\pi d_2^2}{4} \quad \text{Body base area}$$

$$X_c = \frac{2\int_o^{\ell_B} r_x \, x \,(dx)}{Sp} \quad \text{Distance from nose of body to centroid of planform area}$$

$$V_B = \int_o^{\ell b} S_x \,(dx) \quad \text{Volume of body}$$

If $d_2 > d_1$, calculate flare angle $\theta_f = TAN^{-1}\left[\frac{.5(d_2-d_1)}{\ell BT}\right]$

If $d_2 < d_1$, calculate boattail angle $\theta_B = TAN^{-1}\left[\frac{.5(d_1-d_2)}{\ell BT}\right]$

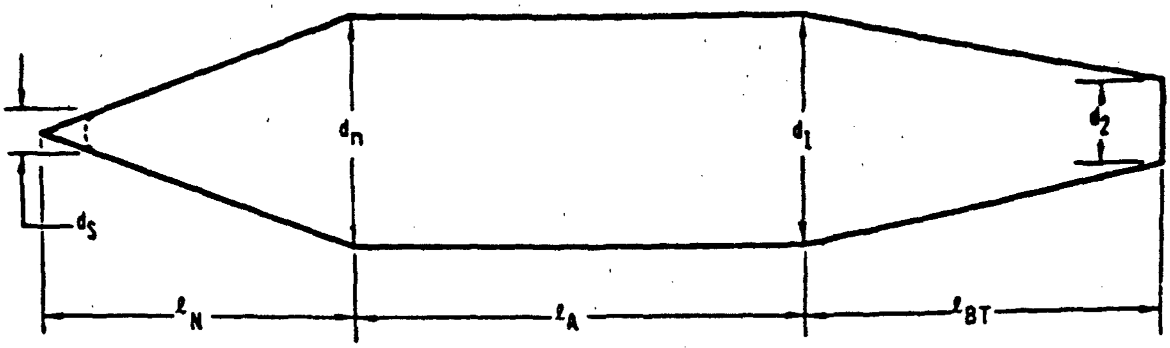
3.3 GENERAL SYNTHESIS PARAMETERS

Synthesizing and interference nomenclature for longitudinal and lateral stability calculations are defined in Figure 12. The geometric parameters are presented in equation format below:

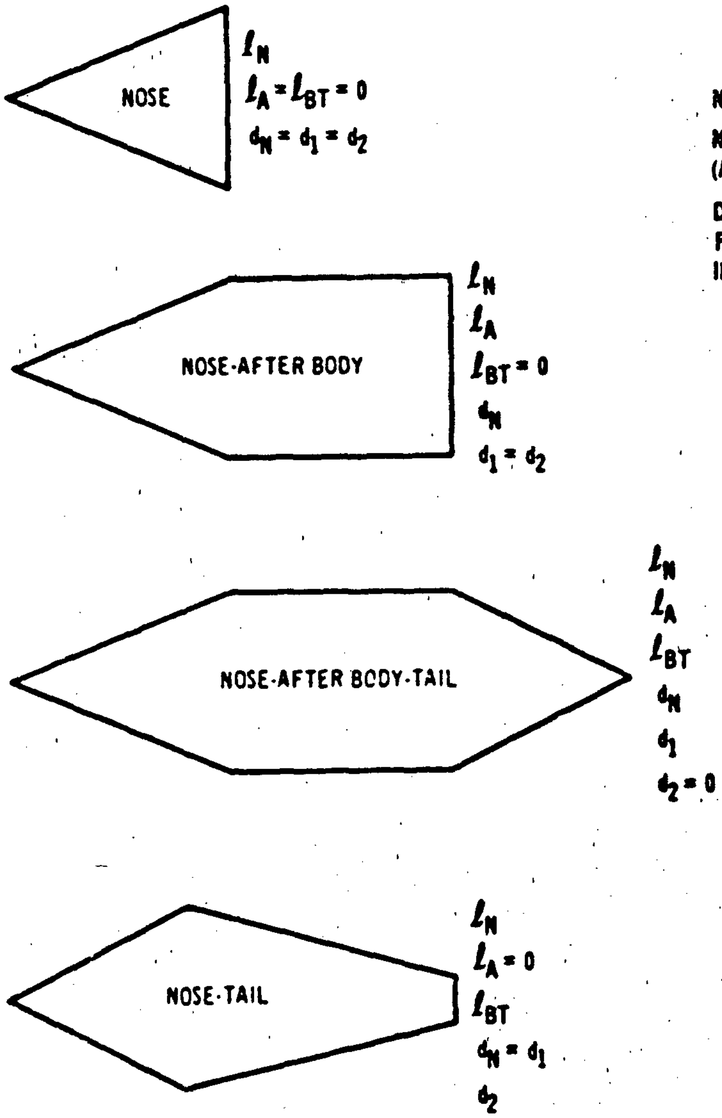
$$\Delta X_w = (b/2 - b^*/2)\ TAN\ \Lambda o_I\ COS\ (\alpha_i)_w$$

$$\Delta X_{cg} = X_{cg} - (X_w + \Delta X_w)$$

$(X_{ac})_w = (X_{ac}/C_r^*)_w\ C_r^*$; where (X_{ac}/C_r^*) is calculated in wing pitching moment subroutine



POSSIBLE SUPERSONIC AND HYPERSONIC BODY CONFIGURATIONS



NOTES:

NOSE AND TAIL SEGMENTS MAY BE CONICAL (AS SHOWN) OR OGIVAL.

DIAMETERS d_N, d_1, AND d_2 ARE COMPUTED FROM LINEAR INTERPOLATION OF INPUTS x_i VS R

FIGURE 11 SUPERSONIC AND HYPERSONIC BODY GEOMETRY

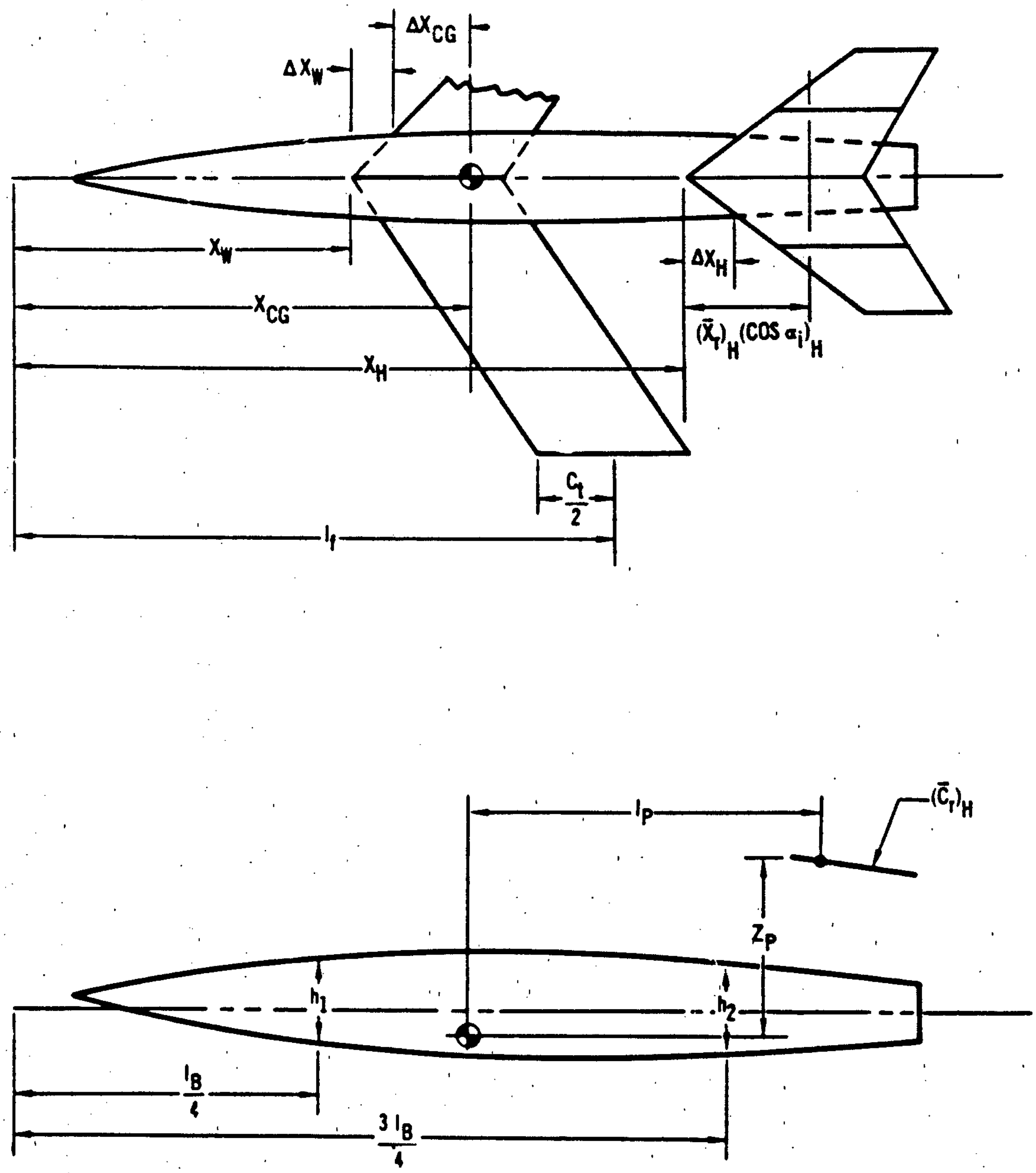


FIGURE 12 GENERAL SYNTHESIS NOMENCLATURE

$$(\Delta X_{ac})_w = \Delta X_{cg} - (X_{ac})_w \text{ COS } (\alpha_i)_w$$

$$\Delta X_H = (b/2 - b^*/2)_H \text{ TAN } \Lambda o_{I_H} \text{ COS } (\alpha_i)_H$$

$$(\Delta X_{cg})_H = X_{cg} - (X_H + \Delta X_H)$$

$$Z_H^* = Z_H - \Delta X_H \text{ TAN } (\alpha_i)_H$$

$$(X_{ac})_H = (X_{ac}/C_r^*)_H C_r^*$$

$$(Z_{ac})_H = Z_H^* - (X_{ac})_H \text{ SIN } (\alpha_i)_H - Z_{cg}$$

$$\Delta(X_{ac})_H = (\Delta X_{cg})_H - (X_{ac})_H \text{ COS } (\alpha_i)_H$$

$$(X_{\bar{C}/4})_H = X_H - (\bar{X}_r)_H \text{ COS } (\alpha_i)_H$$

$$Z_w' = -Z_w + (C_r/4) \text{ SIN } \alpha_i$$

$$\ell_f = X_w + \Delta X_w + \left(\frac{b_o^*}{2}\right) \text{ TAN } \Lambda_{LE_o} + \left(\frac{b_b^*}{2}\right) \text{ TAN } \Lambda_{LE_I} + \frac{C_t}{2}$$

$$\ell_p = X_v - X_{cg} + (X_r)_w + \left(\frac{\bar{C}_r}{4}\right)_v$$

$$Zp = Z_{cg} + (\bar{Y}_R)_v$$

3.4 DOWNWASH PARAMETERS

Downwash geometric nomenclature is defined in Figure 13. The equations presented below are used primarily in the subsonic speed regime:

$$Z_H' = Z_H - \bar{X}_{r_H} \text{ SIN } (\alpha i)_H - Zw + C_{r_w} \text{ SIN } (\alpha i)_w$$

$$\ell_H = X_H + \bar{X}_{r_H} \text{ COS } (\alpha i)_H - (X_w + C_{r_w} \text{ COS } (\alpha i)_w)$$

$$\Delta L_H = Z_H' \text{ TAN } (\alpha i)_w$$

$$L_T = L_H - \Delta L_H$$

$$\Delta h_{H_1} = Z_H'/\text{COS } (\alpha i)_w$$

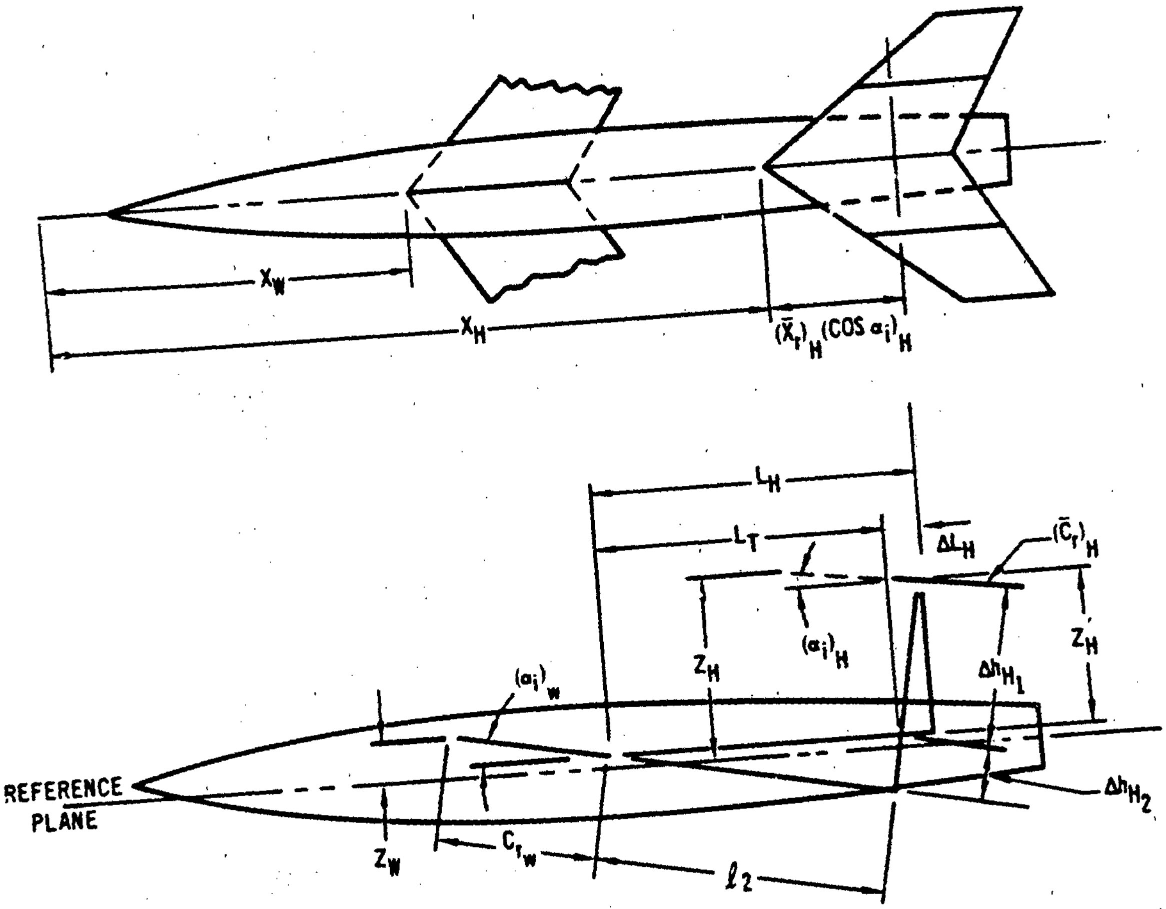


FIGURE 13 DOWNWASH NOMENCLATURE

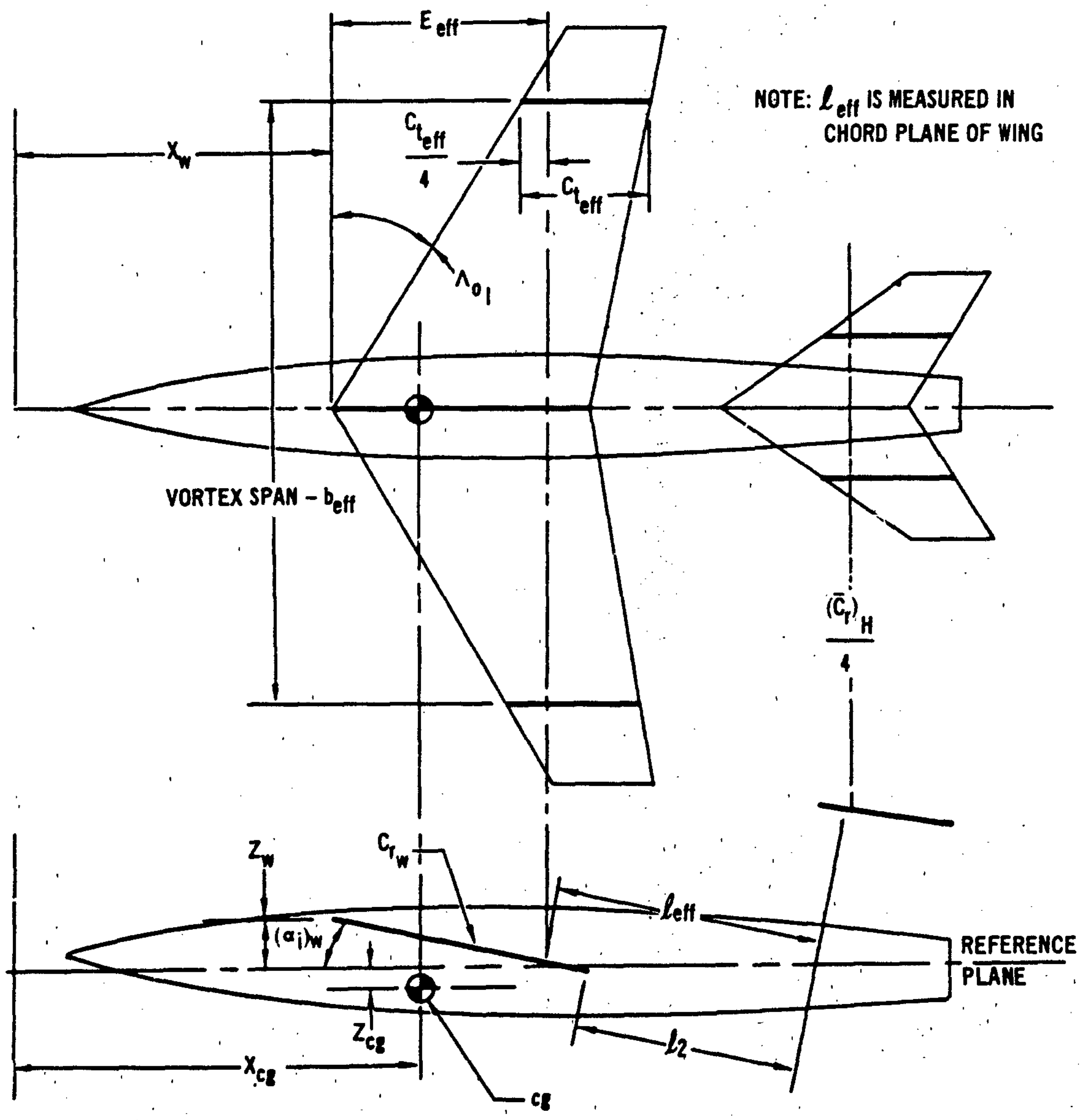


FIGURE 13 DOWNWASH NOMENCLATURE (CONCLUDED)

$$\Delta h_2 = L_T \text{ SIN } (\alpha i)_w$$

$$h_H = \Delta h_{H_1} + \Delta h_{H_2}$$

$$\ell_2 = L_T \text{ COS } (\alpha i)_w$$

$$\gamma = \text{ARCTAN } (h_H/\ell_2)$$

$$\ell_3 = (C_r)_w - (X_r)_w$$

If $b_{eff}/2 \leq (b/2 - b_o^*/2)_w$

$$C_{t_{eff}} = C_{r_w} - \left[\frac{C_r - C_b}{b/2 - b_o^*/2}\right]_w (b_{eff}/2)$$

$$E_{eff} = (b_{eff}/2) \text{ TAN } \Lambda o_I + C_{t_{eff}}/4$$

$$\ell_{eff} = \ell_2 - (E_{eff} - C_{r_w})$$

If $b_{eff}/2 > (b/2 - b_o^*/2)$

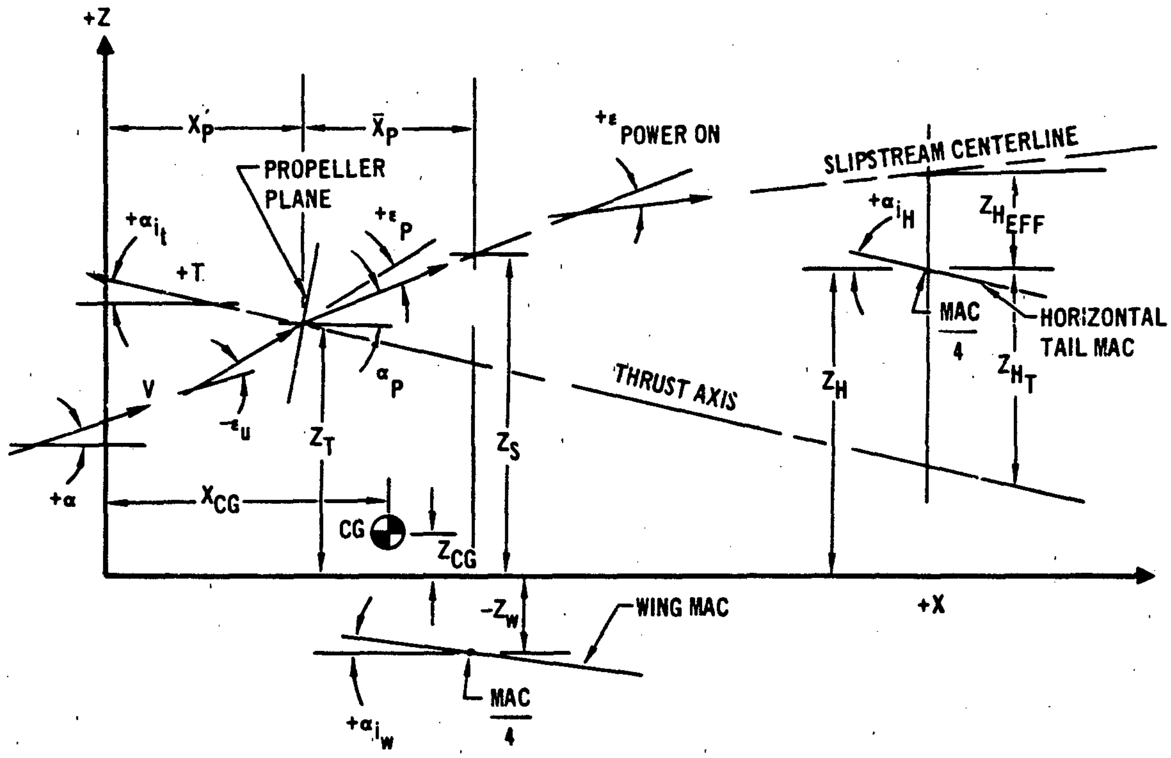
$$C_{t_{eff}} = C_{b_w} - \left[\frac{C_b - C_t}{b_o^*/2}\right]_w [b_{eff}/2 - (b/2 - b_o^*/2)]$$

$$E_{eff} = (b/2 - b_o^*/])_w \text{ TAN } \Lambda o_I + [b_{eff}/2 - (b/2 - b_o^*/2)_w] \text{ TAN } \Lambda o_o + C_{t_{eff}}/4$$

$$\ell_{eff} = \ell_2 - (E_{eff} - C_{r_w})$$

3.5 POWER EFFECTS PARAMETERS

Geometric parameters required to calculate propeller and jet power effects are defined in Figures 14 through 18. Power effects are only calculated for longitudinal stability results in the subsonic speed regime.



$$\bar{X}_P = X_w + \bar{X}_{r_w} \cos \alpha_{i_w} - X'_P$$

$$\bar{Z}_w = Z_w - \bar{X}_{r_w} \sin \alpha_{i_w}$$

$$\alpha'_P = \alpha_{SCH} + \alpha_{i_t} + \epsilon_u - \epsilon_P$$

$$Z_s = Z_T + \bar{X}_P \tan \alpha'_P$$

$$Z_{h_t} = Z_h - Z_T + \left[(X_h + \bar{X}_{r_h} \cos \alpha_{i_h} - X'_P) \tan \alpha_{i_T} \right]$$

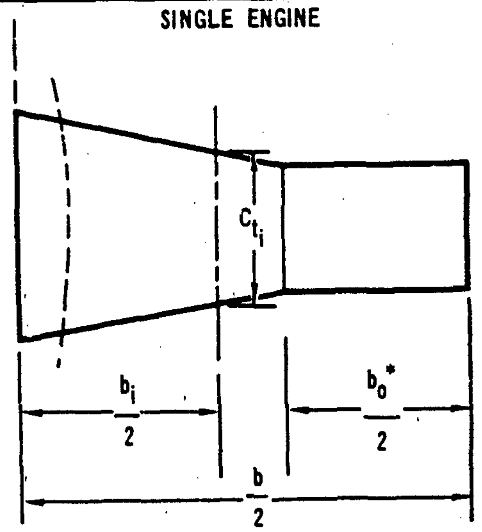
$$\ell_h = \left[X_h + \bar{X}_{r_h} \cos \alpha_{i_h} \right] - \left[X_w + \bar{X}_{r_w} \cos \alpha_{i_h} \right]$$

$$Z_{h_{EFF}} = Z_s - Z_h + l_h \tan (\alpha'_P - \epsilon_{POWER})$$

FIGURE 14 DEFINITION SKETCH FOR PROPELLER POWER EFFECT CALCULATIONS

CASE 1 $\frac{b_i}{2} \leq \left(\frac{b}{2} - \frac{b_o^*}{2}\right)$

SINGLE ENGINE



$$C_{t_i} = C_r - \left[\frac{C_r - C_b}{b/2 - b_o^*/2}\right] \left[\frac{b_i}{2}\right]$$

$$\frac{b_i^*}{2} = \frac{b_i}{2} - \left[\frac{b}{2} - \frac{b^*}{2}\right]$$

$$S_i^* = \left[C_r^* + C_{t_i}\right] \frac{b_i^*}{2}$$

$$A_i^* = \frac{4\left[\frac{b_i^*}{2}\right]^2}{S_i^*}$$

$$\lambda_i = \frac{C_{t_i}}{C_r^*}$$

$$\overline{C}_i^* = \frac{2\, C_r^* \left(1 + \lambda_{l_i}^* + \lambda_{l_i}^{*2}\right)}{3\,(1 + \lambda_{l_i}^*)}$$

$$\overline{Y}_{r_{l_i}}^* = \frac{\left[\frac{b_i^*}{2}\right](1 + 2\lambda^*_{l_i})}{3(1 + \lambda^*_{l_i})}$$

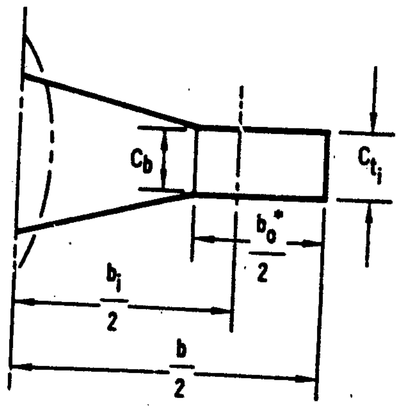
$$X^*_{l_i} = \overline{Y}^*_{l_i} \text{ TAN } \Lambda_{0_l}$$

$$\overline{X}^*_{r_{l_i}} = X^*_{l_i} + \frac{\overline{C}^*_{l_i}}{4}$$

FIGURE 15 GEOMETRY FOR DETERMINING IMMERSED WING PARAMETERS

CASE 2 $\frac{b_i}{2} \geq \left(\frac{b}{2} - \frac{b_o^*}{2}\right)$

SINGLE ENGINE



$$\frac{b_{o_i}^*}{2} = \frac{b_o^*}{2} - \left[\frac{b}{2} - \frac{b_i}{2}\right]$$

$$C_{t_i} = C_b - \left[\frac{C_b - C_t}{\frac{b_o^*}{2}}\right]\left[\frac{b_{o_i}^*}{2}\right]$$

$$S_{o_i}^* = \left[C_b + C_{t_i}\right]\left[\frac{b_{o_i}^*}{2}\right]$$

$$S_i^* = S_I^* + S_{o_i}^*$$

$$\lambda_{o_i}^* = \frac{C_{t_i}}{C_b}$$

$$\overline{C_{o_i}^*} = \frac{2\,C_b\left(1 + \lambda_{o_i}^* + (\lambda_{o_i}^*)^2\right)}{3\,(1 + \lambda_{o_i}^*)}$$

$$\bar{C}_t^* = \frac{S_I^*\ \bar{C}_I^* + S_{o_i}^*\ \bar{C}_{o_i}^*}{S_i^*}$$

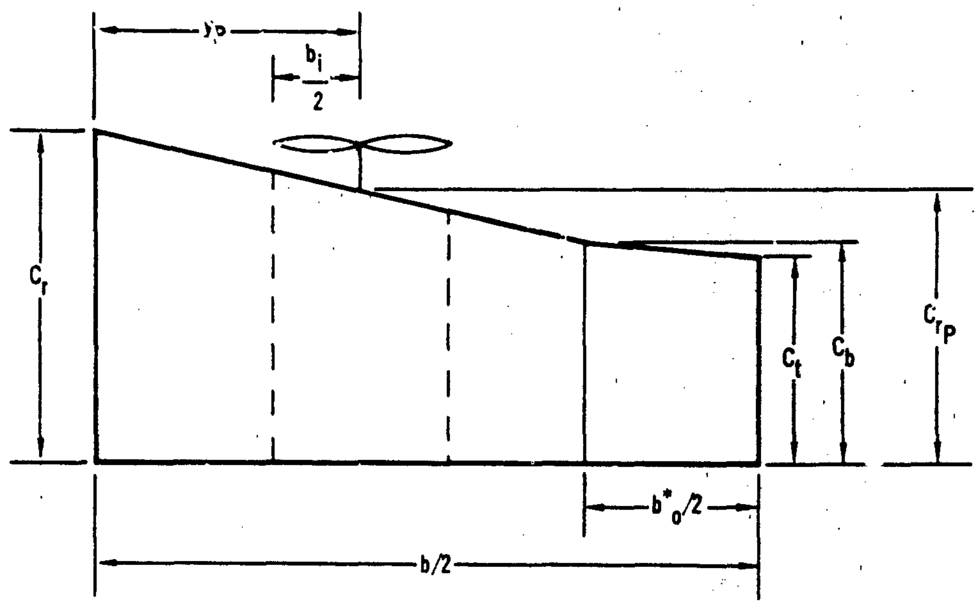
$$\bar{Y}_{o_i}^* = \frac{\left[\frac{b_o^*}{2}\right]_i \left[1 + 2\lambda_{o_i}^*\right]}{3\,(1 + \lambda_{o_i}^*) + \frac{b_b^*}{2}}$$

$$\bar{Y}_i^* = \frac{S_I^*\ \bar{Y}_I^* + S_{o_i}^*\ \bar{Y}_{o_i}^*}{S_i^*}$$

$$X_i^* = \frac{S_I^*\ \bar{Y}_I^*\ \text{TAN}\ \Lambda_{o_I} + S_{o_i}^*\left(\frac{b_b^*}{2}\ \text{TAN}\ \Lambda_{o_I} + \left(\bar{Y}_{o_i}^* - \frac{b_b^*}{2}\right)\text{TAN}\ \Lambda_{0_o}\right)}{S_i^*}$$

$$\bar{X}_{r_i}^* = \frac{\bar{C}_i^*}{4} + X_i^*$$

FIGURE 16 GEOMETRY FOR DETERMINING IMMERSED WING PARAMETERS (CONT'D)



IF $b^*_o/2 = 0.0$ $\qquad C_{r_P} = C_r - \dfrac{(C_r - C_t)\, y_P}{b/2}$

IF $y_P \leqq b/2 - b^*_o/2$ $\qquad C_{r_P} = C_r - \dfrac{(C_r - C_b)\, y_P}{b/2 - b^*_o/2}$

IF $y_P > b/2 - b^*_o/2$ $\qquad C_{r_P} = C_b - \dfrac{(C_b - C_t)\,(y_P - b/2 + b^*_o/2)}{b^*_o/2}$

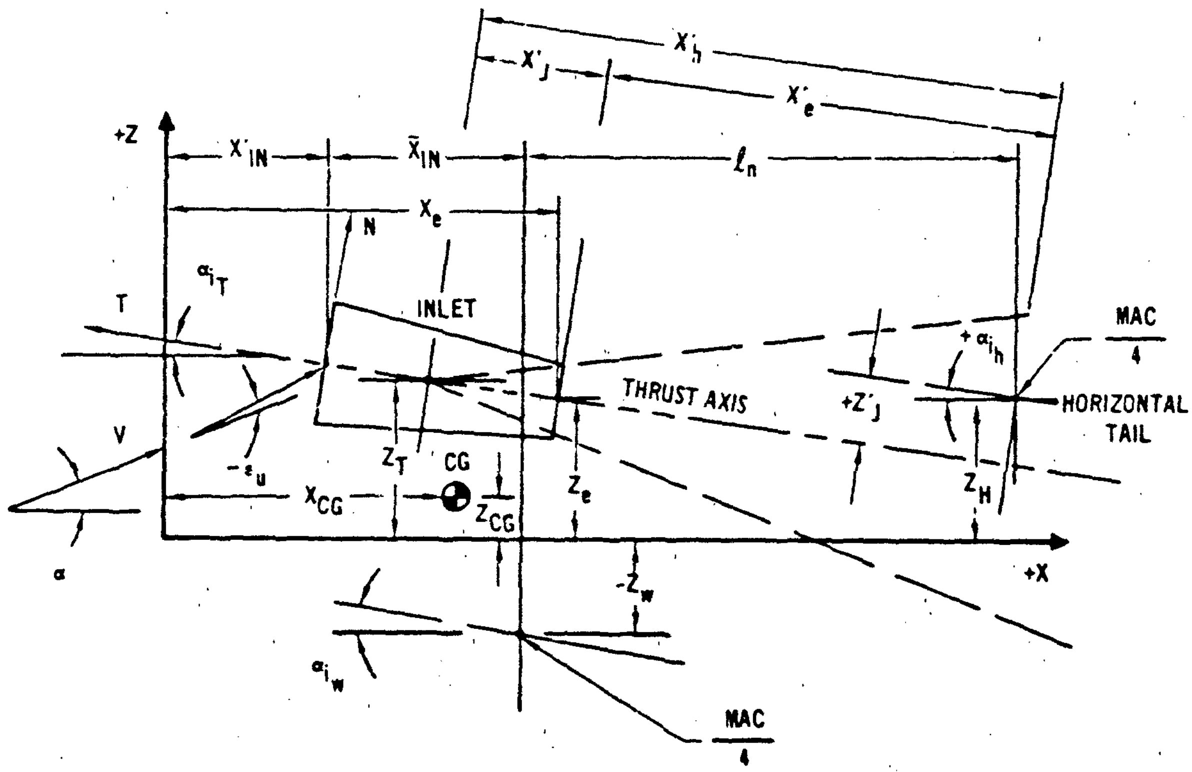
$S^*_i = 2\left[2(b_i/2)\right] C_{r_P}$ $\qquad$ IMMERSED AREA FOR TWO ENGINES

$$A^*_i = \frac{\left[2(bi/2)\right]^2}{0.50\, S^*_i}$$

$$\lambda^*_i \equiv 1.0$$

$$\overline{C}^*_{l_i} \equiv C_{r_P}$$

FIGURE 17 GEOMETRY FOR DETERMINING IMMERSED WING PARAMETERS (CONCLUDED)



$$X'_e = \frac{X_H + (\bar{X}_{T_h})(\cos \alpha_{i_h}) - X_e}{\cos \alpha_{i_t}}$$

$$Z'_J = (X_H + (\bar{X}_{T_h}) \cos \alpha_{i_h} - X_e) \sin \alpha_{i_t} + (Z_H - Z_T) \cos \alpha_{i_t}$$

$$X'_J = 4.6\, R_J$$

$$X'_h = X'_J + X'_e$$

FIGURE 18 DEFINITION SKETCH FOR JET POWER CALCULATIONS

3.6 GROUND EFFECTS PARAMETERS

Ground effects are only calculated for longitudinal stability results in the subsonic speed regime. Lifting surface heights that are required by the Datcom ground effect analyses are defined in Figure 19 and are presented in equation format as follows:

Equations for Calculating $h_{0.75b/2}$

IF $\Gamma_i = 0$ AND $(b/2)_{\Gamma_o} \leq 0.25\,(b/2)$

$$h_{0.75b/2} = h_{0.75C_r} + \Delta X \text{ TAN } (\alpha_i)_W$$

IF $\Gamma_i = 0$ AND $(b/2)_{\Gamma_o} > 0.25\,(b/2)$

$$h_{0.75b/2} = h_{0.75C_r} = \text{TAN } \Gamma_o \left[(b/2)_{\Gamma_o} - 0.25\,(b/2)\right] + \Delta X \text{ TAN } (\alpha_i)_W$$

IF $\Gamma_i \neq 0$ AND $(b/2)_{\Gamma_o} \leq 0.25\,(b/2)$

$$h_{0.75b/2} = h_{0.75C_r} + 0.75\,(b/2)\text{ TAN } \Gamma_i + \Delta X \text{ TAN } (\alpha_i)_W$$

IF $\Gamma_i \neq 0$ AND $(b/2)_{\Gamma_o} > 0.25\,(b/2)$

$$h_{0.75b/2} = h_{0.75C_r} + \left|(b/2) - (b/2)_{\Gamma_o}\right| \text{ TAN } \Gamma_i + \left|(b/2)_{\Gamma_o} - 0.25(b/2)\right| \text{ TAN } \Gamma_o + \Delta X \text{ TAN } (\alpha_i)_W$$

Equations for Calculating h

$$h = 1/2(h_{0.75C_r} + h_{0.75b/2})$$

$$h_{0.75C_r} = H_G + Z_W - 0.75\,C_r \text{ TAN } (\alpha_i)_W$$

IF $\Gamma_i = 0$ AND $(b/2)_{\Gamma_o} \leq 0.25\,(b/2)$

$$h = h_{0.75C_r} + 0.50 \text{ TAN } (\alpha_i)_W$$

IF $\Gamma_i = 0$ AND $(b/2)_{\Gamma_o} > 0.25\,(b/2)$

$$h = h_{0.75C_r} + 0.50\left|\text{TAN } \Gamma_o \left\{(b/2)_{\Gamma_o} - 0.25\,(b/2)\right\} + \Delta X \text{ TAN } (\alpha_i)_W\right|$$

IF $\Gamma_i \neq 0$ AND $(b/2)_{\Gamma_o} \leq 0.25\,(b/2)$

$$h = h_{0.75C_r} + 0.50\left|0.75\,(b/2) \text{ TAN } \Gamma_i + \Delta X \text{ TAN } (\alpha_i)_W\right|$$

IF $\Gamma_i \neq 0$ AND $(b/2)_{\Gamma_o} > 0.25\,(b/2)$

$$h = h_{0.75C_r} + 0.50\left|(b/2) - (b/2)_{\Gamma_o}\right| \text{ TAN } \Gamma_i + 0.50\left|(b/2)_{\Gamma_o} - 0.25\,(b/2)\right| \text{ TAN } \Gamma_o + 0.50\,\Delta X \text{ TAN } (\alpha_i)_W$$

Equations for Calculating H

$$\left(h_{\bar{C}_{r/4}}\right)_W = H_G + Z_W\,((\bar{x}_r)_W \text{ TAN } (\alpha_i)_W$$

IF $\Gamma_i = 0$ AND $(\bar{y}_r)_W = \left|b/2 - (b/2)_{\Gamma_o}\right|$

$$H = \left(H_{\bar{C}_{r/4}}\right)_W$$

IF $\Gamma_i = 0$ AND $(\bar{y}_r)_W \cdot \left|b/2 - (b/2)_{\Gamma_o}\right|$

$$H = \left(h_{\bar{C}_{r/4}}\right)_W + \left|(\bar{y}_r)_W + (b/2)_{\Gamma_o} - b/2\right| \text{ TAN } \Gamma_o$$

IF $\Gamma_i \neq 0$ AND $(\bar{y}_r)_W = \left|b/2 - (b/2)_{\Gamma_o}\right|$

$$H = \left(h_{\bar{C}_{r/4}}\right)_W + (\bar{y}_r)_W \text{ TAN } \Gamma_i$$

IF $\Gamma_i \neq 0$ AND $(\bar{y}_r)_W \quad \left|b/2 - (b/2)_{\Gamma_o}\right|$

$$H = \left(h_{\bar{C}_{r/4}}\right)_W + \left|b/2 - (b/2)_{\Gamma_o}\right| \text{ TAN } \Gamma_i + \left|(\bar{y}_r)_W - (b/2)_{\Gamma_o} - b/2\right| \text{ TAN } \Gamma_o$$

Equations for Calculating H_H

$$\left(h_{\bar{c}_r/4}\right)_H = H_G + Z_H - (\bar{x}_r)_H \; TAN\,(\alpha_i)_H$$

IF $\Gamma_{i_H} = 0$ AND $(\bar{y}_r)_H = \left|(b/2)_H - (b/2)_{\Gamma_{o_H}}\right|$ $\qquad H_H = \left(h_{\bar{c}_r/4}\right)_H$

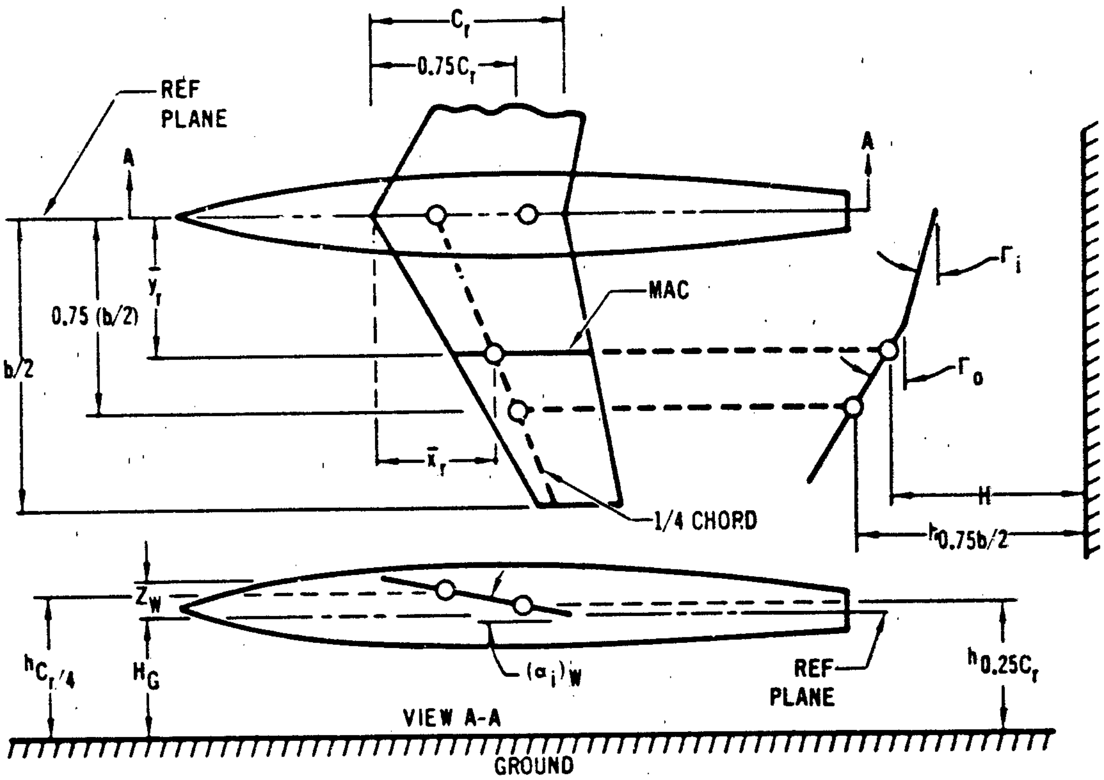
IF $\Gamma_{i_H} = 0$ AND $(\bar{y}_r)_H \quad \left|(b/2)_H - (b/2)_{\Gamma_{o_H}}\right|$ $\qquad H_H = \left(h_{\bar{c}_r/4}\right)_H + \left|(\bar{y}_r)_H + (b/2)_{\Gamma_{o_H}} - (b/2)_H\right| \; TAN\,\Gamma_{o_H}$

IF $\Gamma_i \neq 0$ AND $(\bar{y}_r)_H = \left|(b/2)_H - (b/2)_{\Gamma_{o_H}}\right|$ $\qquad H_H = \left(h_{\bar{c}_r/4}\right)_H + (\bar{y}_r)_H \; TAN\,\Gamma_{i_H}$

IF $\Gamma_i \neq 0$ AND $(\bar{y}_r)_H \quad \left|(b/2)_H - (b/2)_{\Gamma_{o_H}}\right|$ $\qquad H_H = \left(h_{\bar{c}_r/4}\right)_H + \left|(b/2)_H - (b/2)_{\Gamma_{o_H}}\right| TAN\,\Gamma_{i_H}$

$$+ \left|(\bar{y}_r)_W + (b/2)_{\Gamma_{o_H}} - (b/2)_H\right| TAN\,\Gamma_{o_H}$$

Ground effect methods require calculation of a planform parameter, ΔX, in addition to the previously defined ground heights. This parameter is shown in Figure 20.



$h_{0.75\,C_r}$ = HEIGHT OF 3/4 CHORD OF WING ROOT CHORD ABOVE GROUND

$= H_G + Z_W - 0.75\,C_t\ \mathrm{TAN}\ (\alpha_i)_W$

$h_{C_r/4}$ = HEIGHT OF 1/4 CHORD OF WING ROOT CHORD ABOVE GROUND

$= h_{0.75\,C_r} + 0.50\,C_r\ \mathrm{TAN}\ (\alpha_i)_W$

$h_{0.75\,b/2}$ = HEIGHT OF WING ABOVE GROUND AT 1/4 CHORD OF WING 75% SEMI-SPAN CHORD

h = AVERAGE HEIGHT ABOVE GROUND OF THE 1/4 CHORD POINT OF WING CHORD AT 75% SEMI-SPAN AND THE 3/4 CHORD POINT OF THE WING ROOT CHORD.

$= 0.50\,(h_{0.75\,b/2} + h_{0.75\,C_r})$

H = HEIGHT OF 1/4 CHORD POINT OF WING MEAN AERODYNAMIC CHORD ABOVE THE GROUND

H_H = HEIGHT OF 1/4 CHORD POINT OF HORIZONTAL TAIL MEAN AERODYNAMIC CHORD ABOVE THE GROUND

FIGURE 19 GROUND EFFECT WING AND TAIL HEIGHTS

Straight Tapered Wing

$\Delta X = 0.75\,C_r - 0.75\,(b/2)\,\text{TAN}\,\Lambda^*_{25}$

Cranked or Double Delta Wing

IF $b^*_{o/2} \leq 0.25\,(b/2)$ $\quad \Delta X = 0.75\,C_r - 0.75(b/2)\,\text{TAN}\,\Lambda_{25_I}$

IF $b^*_{o/2} > 0.25\,(b/2)$ $\quad \Delta X = 0.75\,C_r - \text{TAN}\,\Lambda_{25_o}\left[b^*_{o/2} - 0.25(b/2)\right] - \text{TAN}\,\Lambda_{25_I}\left[(b/2) - b^*_{o/2}\right]$

Straight Tapered Wing　　Cranked or Double Delta Wing

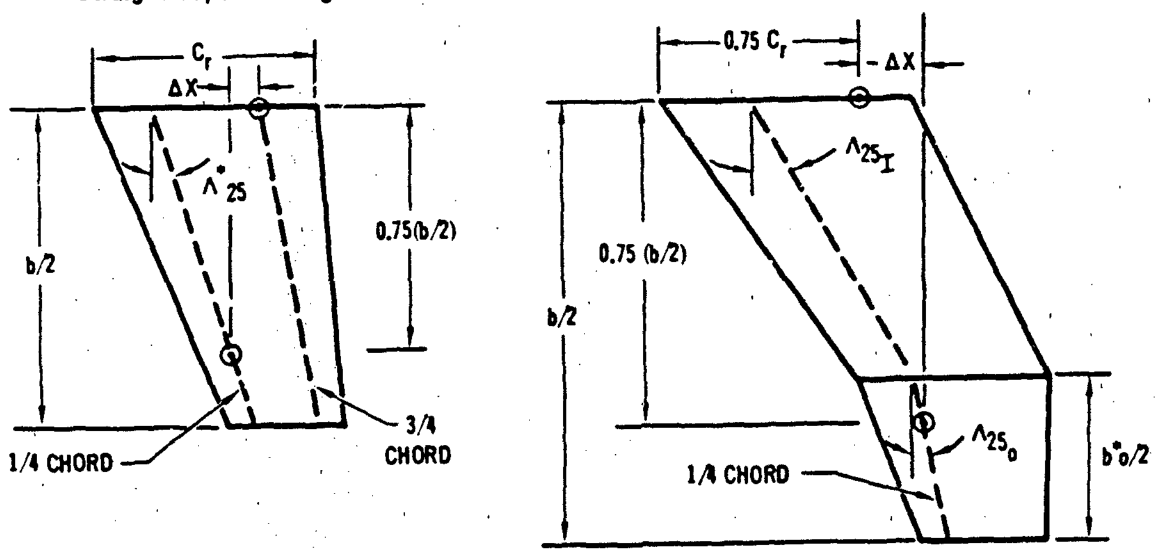


FIGURE 20 GROUND EFFECTS PLANFORM PARAMETER ΔX

SECTION 4

INCORPORATION OF METHODS

This section summarizes those methods which were incorporated into Digital Datcom but were not defined in the Datcom Handbook or involve method interpretation. Though some of the methods included are not, in, general, standard Datcom methods, they permit greater flexibility in using the program, and provide output for some parameters which can be closely approximated or are difficult to obtain experimentally. All of the methods presented in this section are referenced to Table 1 of Section 1 and the Datcom. Methods, or procedures, not outlined in this section follow the Datcom method and users should consult the Datcom for method limitations and formulation.

4.1 AIRFOIL SECTION AERODYNAMICS

This section describes a procedure that can be used to obtain the geometric and aerodynamic section characteristics of virtually any user defined airfoil section. Its incorporation into Digital Datcom frees the user from the labor of calculating those section parameters that were required inputs, yet allow him the flexibility to alter those parameters for which he has data.

The Airfoil Section Module will accept the following user inputs:

- The airfoil section designation
- Section upper and lower cartesian coordinates
- Section mean line and thickness distribution

By these three methods, many airfoil sections can be described and the section characteristics calculated.

Since the Airfoil Section Module (ASM) uses the Mach and Reynolds number inputs, they must be defined in namelist FLTCØN using MACH and RNNUB. However, the ASM uses the unit Reynolds number and by implication treats a section one foot (or meter) in length.

This module brings together the outstanding features of two separate studies. Kinsey and Bowers (AFFDL-TR-71-87) have written a program that calculates the airfoil coordinates of select NACA designations, then uses the Weber technique to calculate the section aerodynamic characteristics. Nieldling of McDonnell Aircraft has written a similar program using the Weber method, then incorporates additional methods to refine the theoretical

TABLE 5 AIRFOIL SECTION MODULE ROUTINE DESCRIPTION

PROGRAM/SUBROUTINE	PURPOSE
M50032 (OVERLAY 50,0)	MODULE EXECUTIVE PROGRAM
INIZ	INITIALIZE IOM
SECI	READ USER INPUTS
SECO	TRANSFER MODULE OUTPUTS
CSLOPE	CALCULATE VARIABLE SLOPE FOR SUPERSONIC AIRFOILS
XYCORD	CALCULATE AIRFOIL SECTION FROM USER INPUTS
DELY	CALCULATE DATCOM PARAMETER ΔY
AIRFOIL (OVERLAY (50,1))	MAIN PROGRAM FOR NACA DESIGNATION INPUTS
DECODE	READ USER INPUT NACA DESIGNATION, DECODE
COORD4	CALCULATE 4–DIGIT NACA AIRFOIL
COORD4M	CALCULATE 4–DIGIT (MODIFIED) NACA AIRFOIL
COORD5	CALCULATE 5–DIGIT NACA AIRFOIL
COORD5M	CALCULATE 5–DIGIT (MODIFIED) NACA AIRFOIL
COORD1	CALCULATE 1–SERIES NACA AIRFOIL
COORD6	CALCULATE 6–SERIES NACA AIRFOIL
CORDSP	CALCULATE SUPERSONIC AIRFOIL COORDINATES
SLEQ	SIMULTANEOUS LINEAR EQUATION SOLVER
THEORY (OVERLAY (50,2))	MAIN PROGRAM FOR AIRFOIL AERODYNAMICS
IDEAL	CALCULATE SECTION IDEAL AERODYNAMICS
SLOPE	CALCULATE LIFT AND MOMENT SLOPES
ASMINT	NON–LINEAR INTERPOLATION ROUTING
MAXCL (OVERLAY (50,3))	CALCULATE VARIABLE CLMAX FOR SECTION

predictions. A cross of the two procedures (coordinates of NACA airfoils and viscous correction from Kinsey and Bowers, and the aerodynamic methods of Nieldling) yields a program that generates fairly accurate results.

The module is incorporated into Digital Datcom as Overlay 50, and includes three secondary overlay programs. The routines use the IOM arrays for data storage so that core size will be kept to a minimum. Table 5 describes each of the 22 module routines and the logic flow of the module is presented in Figurees 21 through 24.

4.1.1 Weber's Method

The calculation of the pressure distribution over the surface of an airfoil in an incompressible inviscid flow is accomplished by use of the method of singularities. Conformal transformations are used as an intermediate step in deriving the methods for determining the distributions of singularities from which the velocity distributions are calculated. The routine inputs are the airfoil coordinates distributed in any fashion, the angle of attack, and the Mach number. The airfoil shape is defined by curve fitting the input coordinates to obtain the airfoil geometry at thirty-two required points, i.e.:

$$X = 0.5\ (\cos\theta_\nu + 1)$$

$$\theta_\nu = \nu\pi/32 \text{ for } o \le \nu \le 32$$

The chord line is obtained by joining the leading and trailing edges of the airfoil, where the leading edge is defined as the forward most point so that all points on the airfoil surface have a positive x coordinate.

The airfoil is placed in a uniform stream V_o at an angle of attack relative to the chord line. The velocity V_o is resolved into components parallel and normal to the chord line.

$$V_{xo} = V_o \cos\alpha$$

$$V_{zo} = V_o \sin\alpha$$

Combining the results for the parallel and normal flows, the velocity distribution equation for a symmetrical airfoil at angle of attack is

$$V(x,z) = \frac{V_o}{\sqrt{1 + (dz/dx)^2}} \left\{ \cos\alpha \left[1 + \frac{1}{\pi} \int_o^1 \frac{dz}{dx'} \frac{dx'}{x - x'} \right] \right.$$

$$\left. \pm \sin\alpha \sqrt{\frac{1 - x}{x}} \left[1 + \frac{1}{\pi} \int_o^1 \left(\frac{dz}{dx'} - \frac{2z(x')}{1 - (1 - 2x')^2} \right) \frac{dx'}{x - x'} \right] \right\}$$

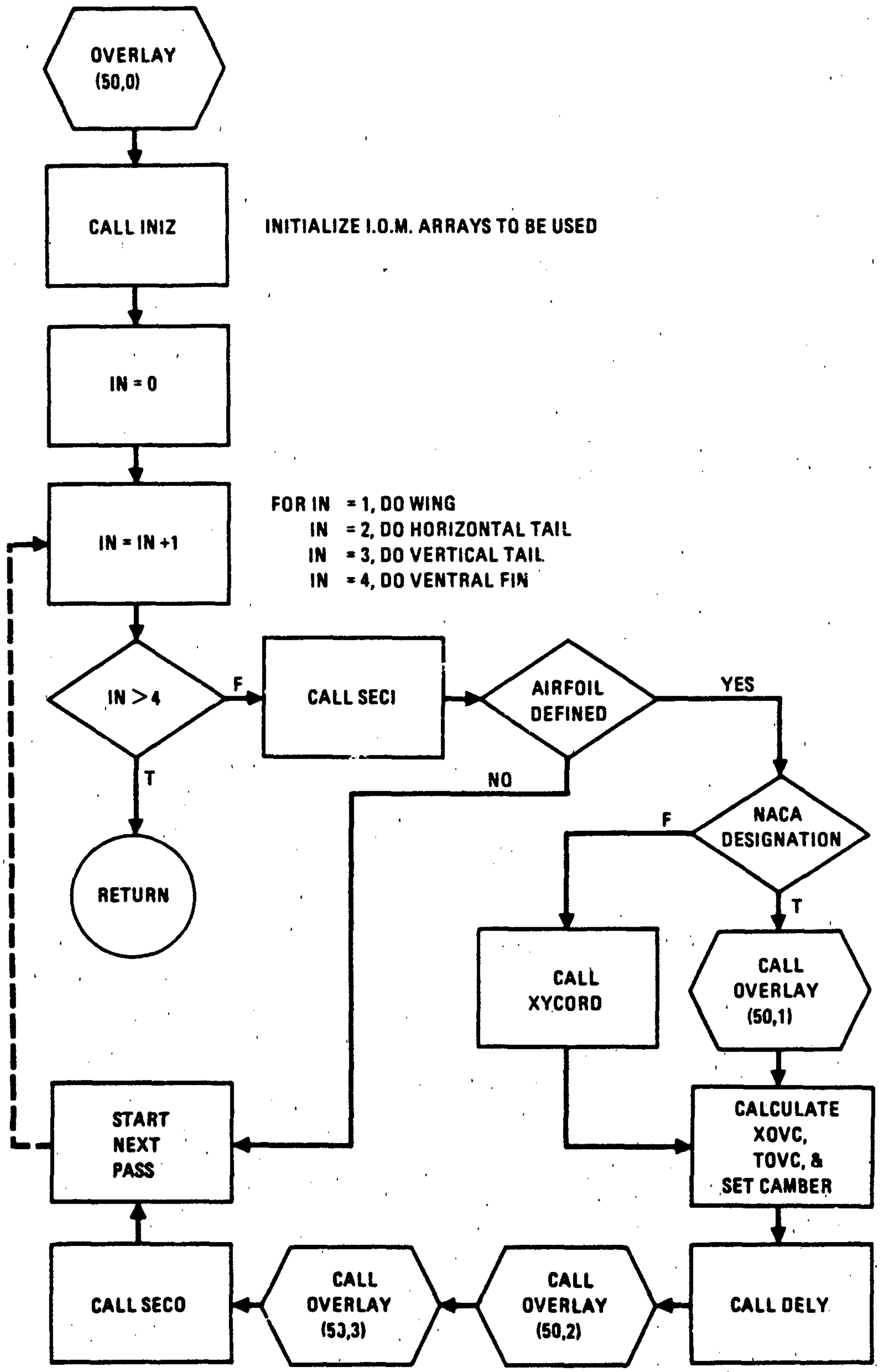


FIGURE 21 AIRFOIL SECTION MODULE – EXECUTIVE ROUTING

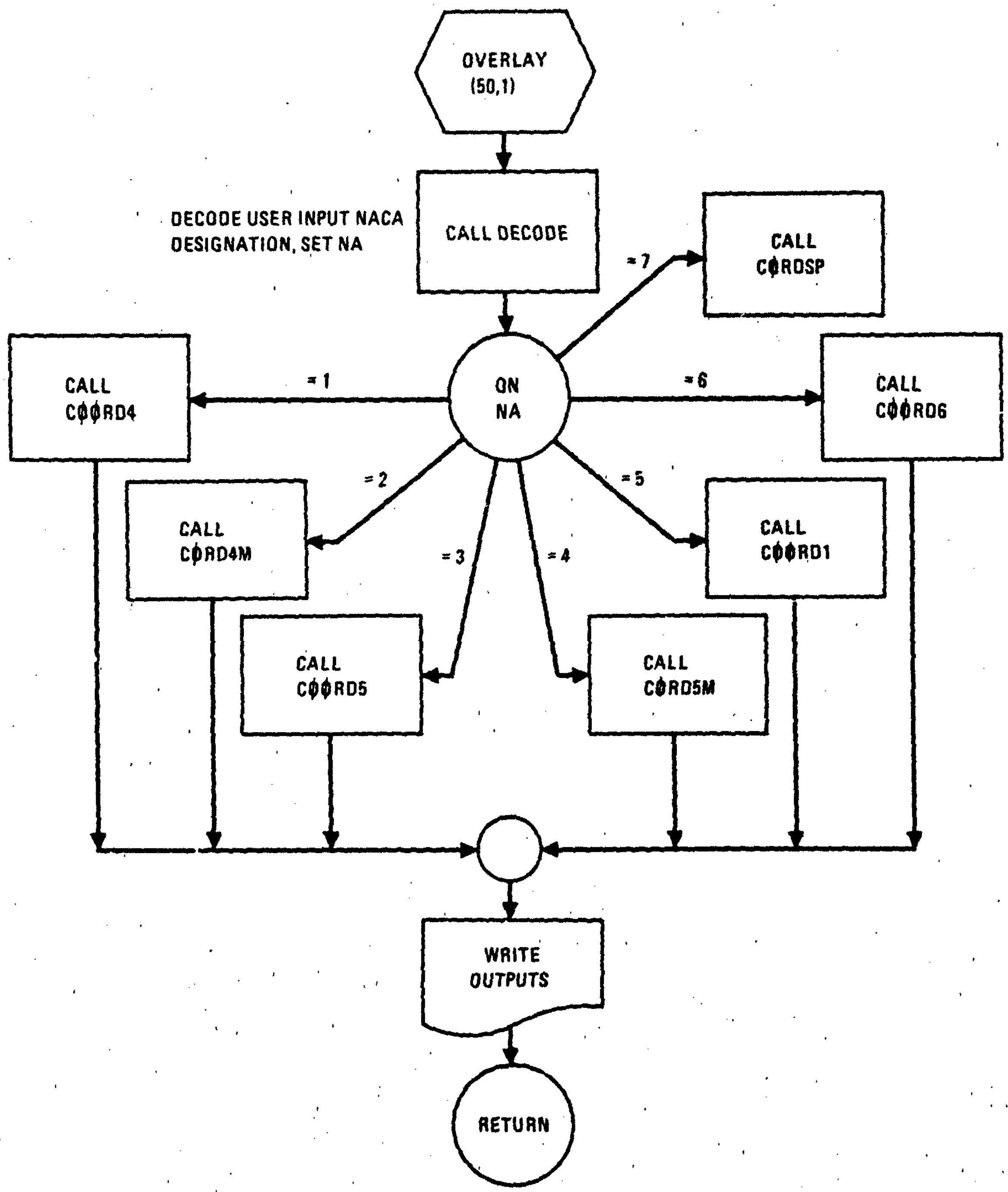


FIGURE 22 AIRFOIL SECTION MODULE – NACA DESIGNATION ROUTINE

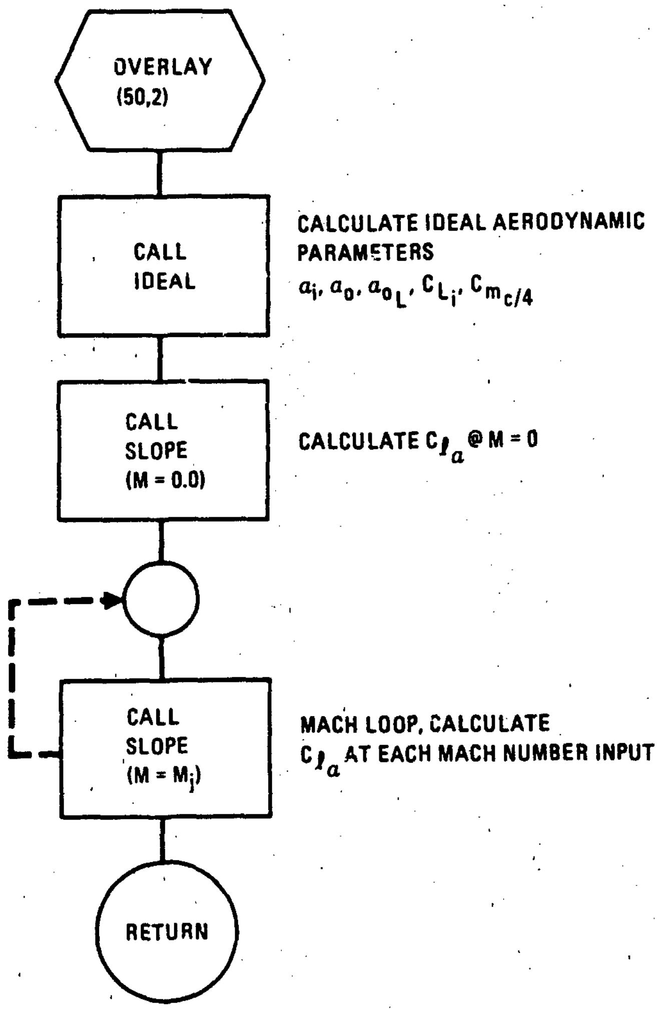


FIGURE 23 AIRFOIL SECTION MODULE – SECTION AERODYNAMICS ROUTINE

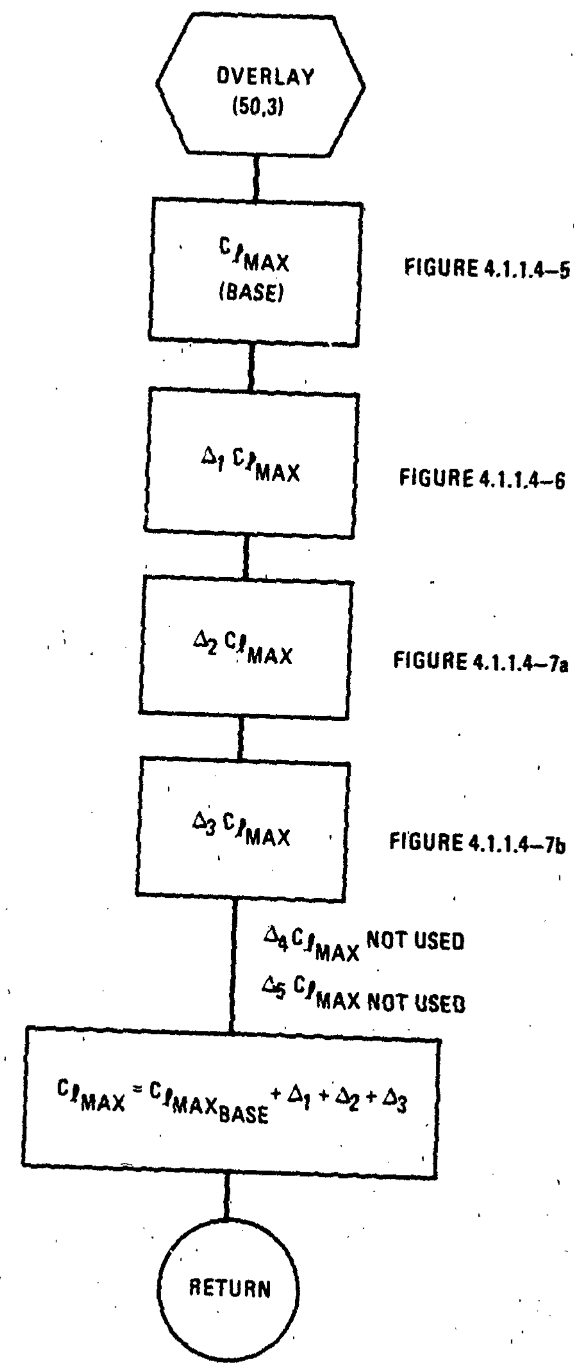


FIGURE 24 AIRFOIL SECTION MODULE – SECTION MAXIMUM LIFT ROUTINE

In the Weber Method certain combinations of the above terms have been redefined as follows:

$$s^{(1)}(x) = \frac{1}{\pi} \int_0^1 \frac{dz}{dx'} \frac{dx'}{x - x'}$$ (Function for Source Distribution in Parallel Flow)

$$s^{(2)}(x) = \frac{dz}{dx}$$ (Slope of Thickness Distribution)

$$s^{(3)}(x) = \frac{1}{\pi} \int_0^1 \left(\frac{dz}{dx'} - \frac{2z(x')}{1 - (1 - 2x')^2} \right) \frac{dx'}{x - x'}$$ (Function for Vortex Distribution in Normal Flow, to Account for α)

These functions are approximated by sums and products of the airfoil ordinates and certain coefficients which are independent of the section shape by

$$s^{(1)}(x\ ) = \sum_{\nu=1}^{N-1} s_{\upsilon\nu}{}^{(1)} z_\upsilon \qquad s^{(2)}(x\ ) = \sum_{\nu=1}^{N-1} s_{\upsilon\nu}{}^{(2)} z_\upsilon$$

$$s^{(3)}(x\ ) = \sum_{\upsilon=1}^{N-1} s_{\upsilon\nu}{}^{(3)} z_\upsilon + s_{N\nu}{}^{(3)} \sqrt{\frac{\rho}{2C}}$$

The effects of camber on the resulting velocity distribution are obtained by assuming the camber to be small compared with the chord. This results in the camber effect being accounted for in the parallel flow $V_{xo} = V_o \cos \alpha$ only.

The Vortex Distribution, $\gamma(X)$, on the chord line which produces a given velocity normal to the chord line and which is zero at the trailing edge is

$$\frac{\gamma(x_\nu)}{2V_{xo}} = \sum_{\upsilon=1}^{N-1} s_{\upsilon\nu}{}^{(4)} z_{s\upsilon} = s^{(4)}(x_\upsilon)$$ (Vortex Distribution due to Camber)

The total velocity $V_x(x,o)$ on the chord line for an airfoil with camber and incidence is

$$V_x(x,0) = V_o \cos\alpha \left[1 + S^{(1)}(x) \pm S^{(4)}(x) \right]$$

$$\pm V_o \sin\alpha \sqrt{\frac{1-x}{x}} \left[1 + S^{(3)}(x) \right]$$

with the + sign being for the upper surface and the - sign for the lower surface.

The resulting velocity distribution at the airfoil surface is computed using

$$S^{(5)}(x) = \frac{dz_s(x)}{dx} \qquad \text{(Slope of Camber Line)}$$

where $$\frac{V(x)}{V_o} = \frac{\cos\alpha \left[1 + S^{(1)}(x) \pm S^{(4)}(x) \right] \pm \sin\alpha \sqrt{\frac{1-x}{x}} \left[1 + S^{(3)}(x) \right]}{\sqrt{1 + \left[S^{(2)}(x) \pm S^{(5)}(x) \right]^2}}$$

which is the complete expression for an arbitrary airfoil at angle of attack in an ideal flow. The $S^{(4)}(X)$ and $S^{(5)}(X)$ terms are computed by approximation. The pressure coefficient is obtained by

$$C_p = 1 - \frac{\left\{ \cos\alpha \left[1 + S^{(1)}(x) \pm S^{(4)}(x) \right] \pm \sin\alpha \sqrt{\frac{1-x}{X}} \left[1 + S^{(3)}(x) \right] \right\}^2}{1 + \left[S^{(2)}(x) \pm S^{(5)}(x) \right]^2}$$

The term $1 + S^{(1)}(x) \pm S^{(4)}(x)$ accounts for the vorticles being put into a flow with velocity $V_o\ (1 + S^{(1)}(x) + S^{(4)}(x))$ instead of V_o. The term $(1 + S^{(3)}(x))$ accounts for the differences in the vortex distribution between the thick and thin wing. The term $1/\ [1 + [S^{(2)}(x) \pm S^{(5)}(x)]^2]$ is the correction between velocities on the chord line and on the surface.

4.1.2 Compressibility Correction and Integration

The effects of compressibility are accounted for in Weber's Method by the application of compressibility factors to the velocity distribution contributions due to thickness and camber, respectively.

$$\beta = \sqrt{(1 - M_o^2)} \qquad C_{p_i} = 1 - \frac{(1 + S^{(1)})^2}{1 + (S^{(2)})^2}$$

$$B = \sqrt{(1 - M_o^2\ (1 - M_o\ C_{p_i}))}$$

The velocity distribution in compressible flow is then given by

$$\left(\frac{V}{V_o}\right)^2 = \frac{\left\{\cos\alpha\left[1 + \frac{S^{(1)}}{B} \pm \frac{S^{(4)}}{\beta}\right] \pm \frac{\sin\alpha}{\beta}\left[1 + \frac{S^{(3)}}{B}\right]\sqrt{\frac{1 - x}{x}}\right\}^2}{1 + \left[\frac{S^{(2)} \pm S^{(5)}}{B}\right]^2}$$

The compressible pressure coefficient from the compressible form of Bernoulli's equation is

$$C_p = \frac{1}{0.7\ M_o^2}\left\{\left[1 + 0.2\ M_o^2\ \left[1 - \left(\frac{V}{V_o}\right)^2\right]\right]^{3.5} - 1\right\}$$

The airfoil lift, axial force and pitching moment are computed from the compressible and incompressible solutions in the following manner

Set $\quad l_x = C_{p_u}(M) - C_{p_l}(M)$

$$C_l = \int_0^1 l_x\ dx$$

or $\quad C_l = \int_0^{\pi} l_x \frac{\sin\theta}{2}\ d\theta \quad$ where $\theta = \frac{\nu\pi}{N}$, $\nu = 0 \rightarrow N$

Therefore trapezoidal rule

$$CL\ (M) = \frac{\pi}{N}\left\{\frac{1}{2}\left[l_x \left.\frac{\sin\theta}{2}\right|_0 + l_x \left.\frac{\sin\theta}{2}\right|_N\right] + \sum_{\nu=1}^{N-1} l_x \left.\frac{\sin\theta}{2}\right|_\nu\right\}$$

$$= \frac{\pi}{\cos\alpha_N} \sum_{\nu=1}^{N-1} \left[l_x \left.\frac{\sin\theta}{2}\right|_\nu \right]$$

Similarly

$$CA\ (M) = \frac{\pi}{N} \sum_{\nu=1}^{N-1} \left[C_{p_u}\ (M)\ (S^{(2)}(x) + S^{(5)}(x)) - C_{p_l}\ (M)\ (S^{(2)}(x) - \right.$$

$$\left. \ldots\ldots\ldots\ S^{(5)}(x))\right]\left.\frac{\sin\theta}{2}\right|_\nu \qquad \ldots\ldots\ldots + 1/2\ C_{p_u}\ (M)\ \sqrt{2\rho}$$

and

$$CM\ (M) = \frac{\pi}{N} \sum_{\nu=1}^{N-1} \left[l_x\ (x-.25)\ \frac{\sin\theta}{2} \right]_\nu$$

4.1.3 <u>Ideal Parameters</u>

The ideal parameters are obtained from thin airfoil theory, which in effect means results are obtained for the meanline characteristics in an incompressible inviscid flow. The ideal angle of attack α_i is obtained from

$$\alpha_i = \int_0^1 z_c \frac{1-2x}{\pi\left[x(1-x)\right]^{3/2}}\ dx$$

However, at the leading and trailing edges the equation is undefined and increments in the vicinity of the leading and trailing edges must be determined, in addition to the integration over the interior portion of the chord.

$$\Delta\alpha \Big|_{\substack{x = 0 \text{ to} \\ x = .0381}} = .3739\ z_u \Big|_{x = .0381} + .04745\ \frac{dz}{dx}\Big|_{x=0}$$

$$\Delta\alpha_i \Big|_{x=.9619 \text{ to } x=1.0} = -.3739\, z_s\Big|_{x=.9619} + .04745 \frac{dz}{dx}\Big|_{x=1}$$

resulting in

$$\Delta\alpha_i = 57.3 \left[\Delta\alpha_{i_{x=0 \text{ to } x=.0381}} + \Delta\alpha_{i_{x=.0381 \text{ to } x=.9619}} + \Delta\alpha_{i_{x=.9619 \text{ to } x=1.0}} \right]$$

The angle of attack for zero lift is obtained in a similar manner

$$\alpha_{OL} = -\int_o^1 z_s \left[\frac{1}{(1-x)\sqrt{x[1-x]}} \right] dx$$

with

$$\alpha_{OL_{x=.9619 \text{ to } x=1.0}} = -.7834\, z_s\Big|_{x=.9619} + .09518 \frac{dz}{dx}\Big|_{x=1.0}$$

The total value is given by

$$\alpha_{OL} = 57.3 \left[\alpha_{OL_{x=.9619 \text{ to } x=1.0}} + \alpha_{OL_{x=0 \text{ to } x=.9619}} \right]$$

The ideal lift coefficient is now simply

$$c_{l_i} = \frac{2\pi}{57.3} \left[\alpha_i + \alpha_{OL} \right]$$

The pitching moment about the quarter chord is

$$C_{m_{C/4}} = \frac{2\pi}{N} \sum_{\nu} z_s \cos\theta_\nu + \frac{\pi}{57.3} \frac{\alpha_{OL}}{2}$$

4.1.4 Crest Critical Mach Number

The crest critical Mach number is precisely defined as that free stream Mach number for which local sonic flow is first reached at the airfoil surface crest on the assumption of shock free flow. Its significance is founded on its relation to the drag rise Mach number. Various empirical studies have been aimed at finding the critical pressure ratio at the crest which corresponds to a drag rise in the test data. Nitzberg (NACA RMA9G20) proposed a critical pressure ratio for drag rise of

$$P_{CREST}/P_{TOTAL} = 0.5283$$

which corresponds to a crest Mach number of M = 1.0. Sinnot (RAS TDM-6407) proposed the ratio

$$P_{CREST}/P_{TOTAL} = 0.515$$

which corresponds to a Mach number at the crest of M = 1.02 and which correlates better with drag-rise data. Sinnot's value is used in the Airfoil Section Module, thus the crest critical Mach number corresponds to a local flow at Mach 1.02 at the crest rather than sonic conditions. The relationship between the crest pressure and crest critical Mach number is

$$C_{P_{CREST}} = \frac{0.515(1 + 0.2\, M_{CC}^2)^{3.5} - 1}{0.7\, F\, M_{CC}^2}$$

where

$$F = \left[\beta_{CC} + 1/2\,(1 - \beta_{CC})C_{P_{CREST}}\right]^{-1}$$

M_{CC} = CREST CRITICAL MACH

$C_{P_{CREST}}$ = INCOMPRESSIBLE VALUE

$\beta_{CC} = \sqrt{1-M_{CC}^2}$

Rewritten so that M_{CC} is a function of $C_{P_{CREST}}$, the relation is approximated by

$$M_{CC} = \left[1.023 - .9507\, C_{P_{CREST}} - .414\, C_{P_{CREST}}^2 - .1506\, C_{P_{CREST}}^3 - .0212\, C_{P_{CREST}}^4\right]^{-1}$$

The crest location for each angle of attack is determined by comparing the airfoil surface slope for each x location to tangent α. The final location is obtained by interpolating between the two given x locations whose airfoil slopes bracket the tangent α value. The $C_{P_{CREST}}$ value is obtained by interpolation of the Weber incompressible pressure distribution between the two x values surrounding X_{CREST}. The crest critical lift coefficient is obtained using the Karman-Tsien compressibility rule on the M = 0 integrated Weber lift coefficient.

$$C_{L_{CC}} = \left| \frac{CL(M)}{\beta_{CC} - \dfrac{M_{CC}^2}{1 + \beta_{CC}} \left| \dfrac{CL(M)}{2} \right|} \right|$$

where, $CL(M) = C_L$ for $M = 0$.

No specific boundary layer correction is used. However, the Datcom recommends a 5% correction factor to bring the results in line with experimental data, and the viscous correction of section lift curve slope proposed by Kinsey and Bowers (Appendix B, Volume I) has been incorporated.

4.2 TRANSONIC WING C_L FAIRING, TRANSONIC WING X_{ac} FAIRING, and TRANSONIC WING C_{D_w} FAIRING

Datcom wing methods in the transonic Mach regime calculate aerodynamic parameters only at specific Mach numbers. Data at the requested Mach number is then determined by interpolation. This approach is used for the wing lift curve slope (C_{L_α}), wave drag (C_{D_w}), and aerodynamic center (X_{ac}). Nonlinear fairings for each of these parameters are discussed in the following paragraphs.

4.2.1 Transonic Fairings of Wing C_{L_α}

Wing lift curve slope, C_{L_α}, is calculated in subroutine TRSØNI, overlay 24. The same methods are used for the horizontal tail in subroutine TRSØNJ, also in overlay 24.

Datcom section 4.1.3.2 defines the methods for calculation of C_{L_α} at five discrete Mach numbers from 0.6 to 1.4. Values at Mach 0.6 and 1.4 use the subsonic and supersonic methods, respectively. The routine used to fair this curve is a modified version of subroutine ASMINT used in the Airfoil Section Module, overlay 50. To ensure a smooth continuous interpolation, a curve is constructed by fitting the points by a left-hand parabola joined to a series of cubic curves, and finally a right-hand parabola. This technique yields a function which has continuous derivatives everywhere. The slope of the curve at subsonic Mach numbers is obtained by differentiating the equation on Datcom page 4.1.3.2-49 with respect to Mach number. At Mach 1.4 the slope is found by calculating values at Mach 1.3, 1.4 and 1.5 and assuming a curve of the form:

$$C_{L_\alpha} = A + B/\beta + C/\beta^2$$

Subsonic methods are used to Mach 0.75, or 0.1 less than the force break Mach number (M_{fb}), whichever is smaller, and transonic fairings are initiated at that point.

Subroutines TRANWG and TRANHT are used to calculate C_{L_α} at Mach 1.3, 1.4, and 1.5 and return C_{L_α} and its slope at Mach 1.4. Subroutines TRSØNI and TRSØNJ calculate C_{L_α} using the subsonic equation if the Mach number is less than 0.75 (or M_{fb} - 0.1), calculate the slope of the subsonic C_{L_α} curve at Mach 0.75, and call the new fairing routine if the Mach number if greater than 0.75.

4.2.2 Transonic Fairing of Wing C_{D_W}

The wing wave drag, C_{D_W}, is calculated in subroutines TRSØNI and TRSØNJ, overlay 24, for the wing and horizontal tail, respectively. The method is given in Datcom section 4.1.5.1.

Digital Datcom performs a linear interpolation of Datcom Figure 4.1.5.1-29 at fifteen discrete Mach numbers to determine the variation of C_{D_W}. Non-linear interpolations of this curve are performed as required at the user defined Mach numbers using the fairing routine developed for wing C_{L_α}. Two additional constraints were applied to perform this fairing.

a. If the linear slope to the left or right of a given point, except the end points, is less than UNUSED, (10^{-60} on CDC computers), the slope at that point is set to zero.

b. Any computed value less than zero is set to zero.

Within the fairing routine, the number of points in the curve is used to discriminate between a fairing of C_{D_W} and C_{L_α}.

4.2.3 Transonic Fairings of Wing Aerodynamic Center

Aerodynamic center, X_{ac}, is calculated in subroutines TRANCM and TRHTCM, overlay 25, for the wing and horizontal tail, respectively.

Datcom section 4.1.4.2 defines the method for calculation of X_{ac} at six discrete Mach numbers from 0.6 to 1.4. Values at 0.6 and 1.4 are determined using the subsonic and supersonic methods, respectively; the remaining four points are obtained from Datcom Figure 4.1.4.2-30 corresponding to $\overline{V}$ = -2, -1, 0 and +1. If the thickness ratio is less than or equal to 7%, these data are interpolated for the aerodynamic center. If the thickness ratio is greater than 7%, the curve is defined using points which are a function of the force break Mach number, M_{fb}. An increment to the aerodynamic center is found from Datcom Figure 4.1.4.2-33 and applied at the fifth point (M_{fb} +0.07) and the resulting curve is then interpolated for the aerodynamic center. The following table summarizes the interpolation table:

	Using Six Points $t/c < 7\%$	Using Eight Points $t/c > 7\%$
M_1	0.60	0.60
M_2	M for $\overline{V} = -2$	$(0.60+M_{fb})/2$
M_3	M for $\overline{V} = -1$	M_{fb}
M_4	M for $\overline{V} = 0$	$M_{fb} + 0.03$
M_5	M for $\overline{V} = +1$	$M_{fb} + 0.07$
M_6	1.40	$M_{fb} + 0.14$
M_7	-	M for $\overline{V} = +1$
M_8	-	1.4

The interpolation routine used is similar to the routine used for C_{L_α} and C_{D_W} (Sections 4.2.1 and 4.2.2).

4.3 TRANSONIC WING C_L , TRANSONIC WING C_{D_L}, TRANSONIC WING-BODY-TAIL C_{D_0}, TRANSONIC WING-BODY-TAIL C_D, TRANSONIC WING C_{ℓ_β} , and TRANSONIC WING-BODY C_{ℓ_β}

This section describes those methods used to compute the transonic configuration aerodynamics using Second Level Methods, and are summarized in Table 6. Additionally, the partial output is described.

4.3.1 Transonic Wing Lift Coefficient, C_L

The wing lift curve versus angle of attack is programmed in subroutine WINGCL. The method described in Datcom section 4.1.3.3 is used as a guide to produce trends and is not construed to be an exact method of solution. Since the method is an approximate one, the following procedure was employed to produce the wing lift characteristics applicable to thin, low aspect ratio wings:

1. The required experimental data inputs by the user are α_o (zero lift angle of attack) and α_* (the angle of attack where the lift becomes nonlinear).
2. The lift variation is assumed to be linear up to α_*, and nonlinear to $\alpha_{C_{L_{max}}}$ (maximum lift angle of attack).

TABLE 6 PROGRAMMED TRANSONIC SECOND LEVEL METHODS SUMMARY

DATCOM SECTION	AERODYNAMIC PARAMETER	CONFIGURATION	SUBROUTINE PROGRAMMED	EXPERIMENTAL DATA INPUT REQUIRED	PARTIAL OUTPUT AVAILABLE
4.1.3.3	C_L	WINGS	WINGCL	a_0, a_*	a_0, a_*
4.1.5.2	C_{D_L}	WINGS	WINGCL	C_L OR a_0, a_*	C_{D_L}/C_L^2
5.1.2.1	C_{l_β}	WINGS	WINGCL	C_L OR a_0, a_*	C_{l_β}/C_L
5.2.2.1	C_{l_β}	WING–BODY	WBCLB	C_L	C_{l_β}/C_L
4.5.3.2	C_D	WING–BODY–TAIL	CDWBT	$C_{D_{WB}}$ C_{D_H} C_{L_H} q/q_∞ ϵ	(NONE)
4.5.3.1	C_{D_0}	WING–BODY–TAIL	WBTCDØ	$C_{D_{0V}}$ OR $C_{D_{0WBT}}$* ITYPE (TYPE OF GENERAL CONFIGURATION)	M_D

*$C_{D_{0WBT}}$ IS AVAILABLE FROM THE SECOND LEVEL ROUTINE OF DATCOM, SECTION 4.5.3.1, SUBROUTINE WBTCDØ.

3. The nonlinear lift region is modeled by a mathematical relationship that satisfies the following conditions:

$$C_L = C_{L_{max}} \quad \text{at} \quad \alpha = \alpha_{C_{L_{max}}}$$

$$C_L = C_{L_\alpha} (\alpha_* - \alpha_o) \quad \text{at} \quad \alpha = \alpha_*$$

$$\frac{dC_L}{d\alpha} = C_{L_\alpha} \quad \text{at} \quad \alpha = \alpha_*$$

$$\frac{dC_L}{d\alpha} = 0 \quad \text{at} \quad \alpha = \alpha_{C_{L_{max}}}$$

A modified polynomial of the form

$$y = A + B(X-X_o) + C(X-X_o)^N$$

is utilized to satisfy each of the boundary conditions and yield a curve somewhat parabolic in shape. This relationship has provided excellent results in modeling the nonlinear lift range. Derivation of the unknowns A, B, C and N is described in Section 4.3.7.

Two other user options are available from the routine; (a) the user may input only α_o, or (b) the user inputs only α_*. Since both α_o and α_* are required to estimate the lift variation by the preceding technique, the subroutine will provide an estimate for the missing parameter from a quadratic expression. Specifically, a quadratic polynomial can be faired through the nonlinear lift region if α_* is an unknown. Applying the generalized boundary conditions to a polynomial of order two, and solving for α_* will yield an estimate for this unknown. Conversely, if α_o is not input, it can be determined in a similar manner.

The relationships used are as follows:

1. α_* not input

$$\alpha_* = \alpha_{C_{L_{max}}} + 2\left[\alpha_o - \alpha_{C_{L_{max}}} + \frac{C_{L_{max}}}{C_{L_\alpha}}\right]$$

2. α_o not input

$$\alpha_o = \alpha_{C_{L_{max}}} - \frac{C_{L_{max}}}{C_{L_\alpha}} + \frac{\alpha_* - \alpha_{C_{L_{max}}}}{2}$$

If neither α_o nor α_* are user inputs, no solution is possible, but the program calculated values for C_{L_α}, $C_{L_{max}}$ and $\alpha_{C_{L_{max}}}$ are available as partial output.

4.3.2 <u>Transonic Wing Drag due to Lift, C_{D_L}</u>

The programmed procedure for computing the ratio C_{D_L}/C_L^2 is exactly as described in Datcom section 4.1.5.2. The method does a three dimensional table lookup for Figure 4.1.5.2-55a (A tan (Λ_{LE}) = 0) and for Figure 4.1.5.2-55b (A tan(Λ_{LE}) = 3). Figure 4.1.5.2-55c shows a linear relationship of the dependent variable $(t/c)^{-1/3}$ C_{D_L}/C_L^2 as a function of the transonic similarity parameter A tan (Λ_{LE}) for each value of the ratio $(M^2 - 1)/(t/C)^{2/3}$; it was assumed that this linear relationship would hold for all other taper ratios other than 0.50. Therefore, linear extrapolations on all varibles would be performed if required.

This method was programmed in subroutine WINGCL with the calculation for wing C_L. Since C_L is required to calculate C_{D_L}, the calculation of wing C_L would enable the calculation of this parameter if C_L is not input as experimental data. The routine will not overwrite experimental data input, and thus the user oriented features are retained.

The ratio C_{D_L}/C_L^2 is available from the routine and will be output for user reference if C_{D_L} cannot be calculated.

4.3.3 <u>Transonic Wing Roll Derivative, C_{ℓ_β}</u>

Like the wing C_{D_L} calculation described, the method of Datcom Section 5.1.2.1 requires wing lift to calculate $_\beta$ from the relationship C_{ℓ_β}/C_L, equation 5.1.2.1-c. Thus, this method is also programmed in subroutine WINGCL. The calculated value for C_{ℓ_β} will not overwrite any experimental data input. The ratio C_{ℓ_β}/C_L is provided if the calculation for C_{ℓ_β} cannot be completed. No exceptions are taken for the Datcom method. The ratio C_{ℓ_β}/C_L at Mach numbers 0.6 and 1.4 are obtained by calling the subsonic and supersonic aerodynamic modules.

4.3.4 Transonic Wing-Body Roll Derivative, C_{ℓ_β}

The derivative C_{ℓ_β} will be calculated by Datcom equation 5.2.2.1-d if the wing-body lift coefficient variation with angle of attack is supplied, or computed as described above. The ratio C_{ℓ_β}/C_L is given as partial output if the lift variation is not specified. This method is implemented exactly as described in Datcom and is programmed in subroutine WBCLB. Since C_{ℓ_β}/C_L at M_{fb} and Mach 1.4 are required input items for this method, they are calculated by calling the appropriate aerodynamic modules.

4.3.5 Transonic Wing-Body-Tail Drag Coefficient, C_D

This method is a "method for all speeds" as described in Datcom Section 4.5.3.2, and is incorporated in exactly the same manner as presently programmed for the subsonic solution. This method, as programmed in subroutine CDWBT, require the following experimental data inputs:

1. $C_{D_{WB}}$ vs angle of attack
2. C_{D_H} vs angle of attack
3. C_{L_H} vs angle of attack
4. q/q_∞ vs angle of attack
5. ϵ vs angle of attack
6. $C_{D_{O_V}}$ or $C_{D_{O_{WBT}}}$

If $C_{D_{O_V}}$ is not an experimental data input item, the program will calculate it from the estimated $C_{D_{O_{WBT}}}$ calculated as follows:

$$C_{D_{O_V}} = C_{D_{O_{WBT}}} - C_{D_{O_{WB}}} - C_{D_{O_H}}$$

No partial output is available from this method.

4.3.6 Transonic Wing-Body-Tail Zero Lift Drag Coefficient, C_{D_O}

This method follows exactly the method of Datcom section 4.5.3.1, and is programmed as subroutine WBTCDO. This routine does not require experimental data input, although experimental data input is an optional feature for this routine.

Utilizing appropiate configuration description parameters the program computes the drag divergence Mach number, M_D, from Figure 4.5.3.1-19. The experimental data input allows the user, at his option, to select the type of general configuration to be used in computing M_D. The three options are:

- o A - Straight wing designs without area rule.
- o B - Swept wing designs without area rule.
- o C - Swept wing designs incorporating transonic area rule theory.

The program default options are as follows:

- o No wing sweep - General Configuration A
- o Swept wing, configuration type not defined - General Configuration B

The general configuration types are defined by the parameter ITYPE, where ITYPE=1 for configuration type A, ITYPE=2 for configuration type B, and ITYPE=3 for type C. In the case of configuration type C, the line for type C, in Figure 4.5.3.1-19, was linearly extrapolated and programmed. All extrapolations in this figure, with the exception of thickness ratio, are assumed to be linear; thickness ratio is extrapolated in a quadratic fashion.

With M_D calculated from Figure 4.5.3.1-19, it is necessary to fair the C_{D_o} curve across the transonic Mach regime. The following criteria was used to fair the curve:

1. $\frac{dC_{D_o}}{dM} = 0.10 \ @ \ M = M_D$
2. $C_{D_o} = C_{D_{O M=.7}} + .002 \ @ \ M = M_D$
3. $\frac{dC_{D_o}}{dM} = \frac{C_{D_{O M=.7}} - C_{D_{O M=.6}}}{.1} \ @ \ M = .7$
4. $\frac{dC_{D_o}}{dM} = \frac{C_{D_{O M=1.4}} - C_{D_{O M=1.1}}}{.3} \ @ \ M = 1.1$

A polynomial fairing of the same type as used for the wing nonlinear lift coefficient is used here and has shown acceptable results.

The values of C_{D_o} at Mach .7 and 1.1 for this method are obtained by calling the subsonic and supersonic aerodynamic modules.

4.3.7 Data Fairing Technique

The data fairing technique used for computing the nonlinear lift region of transonic wings and the transonic wing-body-tail zero lift drag coefficient was chosen for its powerful features and ease of application.

The general fairing formula is a polynomial whose form is:

$$y = A + B(X-X_o) + C(X-X_o)^N$$

where A, B, C and N are unknowns. Given the values of y and dy/dx at two points, X_0 and X_1, four simultaneous equations can be written. These equations solved simultaneously for the four unknowns yield the following results:

$$A = y_o$$

$$B = \frac{dy}{dx} \text{ @ } X=X_0$$

$$C = \frac{y_1 - y_0 - (\frac{dy}{dx})_{X_0} \quad (X_1-X_0)}{(X_1 - X_0)^N}$$

$$N = \frac{\left[(\frac{dy}{dx})_{X_1} - (\frac{dy}{dx})_{X_0}\right](X_1-X_0)}{y_1 - y_0 \; (\frac{dy}{dx})_{X_0} \quad (X_1 - X_0)}$$

The general equation reduces to

$$y = y_0 + (\frac{dy}{dx})_{X_C} \quad (X-X_0) \; + \left[y_1 = y_0 - (\frac{dy}{dx})_{X_0} \quad (X_1-X_0)\right] \left(\frac{X-X_0}{X_1-X_0}\right)^N$$

This equation is valid for $X_0 \leq X \leq X_1$ and $(dy/dx)_{X_0} \neq (dy/dx)_{X_1}$. Neither of these conditions is violated in this application. The range of values of X will always fall between X_0 and X_1 because of the program logic, and in the nonlinear lift region the slopes at X_0 and X_1 will never be equal. For the transonic wing-body-tail C_{D_o} versus Mach fairing the Datecom relation $(dC_{D_o}/dM) = 0.10$ at $M=M_D$.

4.4 SUBSONIC WING C_m, SUBSONIC AND SUPERSONIC WING AERODYNAMIC CENTER, SUBSONIC WING-BODY C_m, and SUBSONIC WING-BODY-TAIL C_m

The subsonic wing pitching moment variation with angle of attack follows Datcom Method 1 of Section 4.1.4.3, and is programmed in subroutine CMALPH. The method is applicable to those configurations whose wing aspect ratio satisfies the following criteria:

$$A \leq \frac{6}{(1+C_1)\cos\Lambda_{LE}} \quad \text{("LOW ASPECT RATIO")}$$

For "high aspect ratio" configurations, the default wing aerodynamic center is either the quarter-chord of the wing mean aerodynamic chord, or the user input value (variable name X_{AC} in the planform section characteristics namelists). This value is used in computing pitching moment for the wing up to the angle of attack where the wing lift deviates by more than 7.5% from the linear value; at this point the method is no longer valid.

There are no methods in Datcom or Digital Datcom for supersonic wing pitching moment, though the wing X_{AC} is estimated to be at the wing planform centroid for unswept leading edges, and computed using the method and design charts of Datcom section 4.1.4.2 for other surfaces. These supersonic data are computed in subroutine SUPLNG.

There is no Datcom method for computing the wing-body pitching moment in any Mach regime. Digital Datcom, however, computes the subsonic wing-body pitching moment using the following formulation (programmed in subroutines WBCMO and WBCM):

- o Compute $(C_{m_o})_{WB}$ from regression formulation of Datcom Section 4.3.2.1, programmed in WBCMO. If the method is not applicable, $(C_{m_o})_{WB}$ is computed from Method 1.
- o Compute the wing-body aerodynamic center from Datcom Section 4.3.2.2 (WBCM), where Equation 4.3.2.2-a is used at all speeds.
- o The wing-body C_m curve is then computed as

$$C_{m_{WB}} = C_{m_{O_{WB}}} + C_{m_{C_L}} + C_{m_{C_D}}$$

where $C_{m_{C_L}}$ is the pitching moment due to lift obtained by integrating the curve of X_{AC} versus C_L from $C_L = 0$ and to C_L at the desired angle of attack, and $C_{m_{C_D}}$ is the pitching moment due to wing-body drag located at Z_{AC}.

Subsonic wing-body-tail pitching moment versus angle of attack is computed by Digital Datcom in subroutine WBTAIL, though there is no Datcom method for this parameter. The method formulation used is as follows:

$$C_{L_{j\,H}} = C_{L_{j\,WBT}} - C_{L_{j\,WB}}$$

$$\left(C_{m_j}\right)_{WBT} = \left(C_{m_j}\right)_{WB} + (q/q_\infty)_j \left(C_{m_o}\right)_H + \frac{(X_{ac} - X_{cg})_H}{\bar{c}_r}\left[\left(C_{L_j}\right)_H \cos(\alpha)_j + \left(C_{D_j}\right)_H (q/q_\infty)_j \sin(\alpha)_j\right] + \frac{(Z_{ac} - Z_{cg})_H}{\bar{c}_r}\left[\left(C_{D_j}\right)_H (q/q_\infty)_j \cos(\alpha)_j - \left(C_{L_j}\right)_H \sin(\alpha)_j\right]$$

4.5 TRANSONIC BODY C_{L_α} FAIRING AND TRANSONIC BODY C_{m_α} FAIRING

The transonic C_{L_α} and C_{m_α} derivatives for the body alone configuration is interpolated linearly between the subsonic ($M = 0.60$) and supersonic ($M = 1.40$) Mach regimes in subroutine BØDYRT.

4.6 SUBSONIC ASYMMETRICAL BODY C_L, SUBSONIC ASYMMERICAL BODY C_{m_0} C_m, AND SUBSONIC ASYMMETRICAL BODY C_{D_0}, C_D

Digital Datcom body solutions generally include lift, drag, and pitching moment coefficients. In the transonic speed regime the solutions are restricted to lift and pitching moment slopes, and drag coefficients.

4.6.1 Subsonic Bodies

Subsonic body analysis computes lift, drag, and pitching moment coefficients for either axisymmetric or cambered bodies. Digital Datcom body methods are identical to Datcom except for the base drag. Digital Datcom calculates base drag using a minimum base area equal to 30% of the body maximum cross-sectional area.

The cambered body pitching moment method is not defined in Datcom and is therefore described in detail. For clarity, the lift method, which is defined in Datcom, is also described. These body methods (subroutine BØDØPT) are executed when the parameters Z_U and Z_L are user specified (namelist BØDY). The method predicts the zero lift angle of attack, zero lift pitching moment, and body lift and pitching moment versus angle of attack. The Datcom drag methods are retained.

Zero lift angle of attack and pitching moment are calculated utilizing conventional mean line theory. The equations are:

$$\alpha_0 = \frac{-57.3}{\pi} \int_0^{0.95} \frac{Z'}{L} \left[\frac{1}{(1-X/L)\left[X/L - (X/L)^2\right]^{1/2}} \right] d(X/L), \text{ degrees}$$

$$C_{m_0} = 2.0 \int_0^{1.0} \frac{Z'}{L} \left[\frac{1-2.0\ X/L}{\left[X/L - (X/L)^2\right]^{1/2}} \right] d(X/L)$$

These parameters are defined in Figure 25.

Lift and moment for asymmetric bodies are calculated by employing a modified version of Polhamus's leading-edge suction analogy (References 2 and 3). Polhamus considers two components of lift, a potential flow term, C_{Lp}, and a vortex-lift term C_{LV}. Both of these terms are a function of body aspect ratio (A) and are defined as follows:

$$C_L = C_{Lp} + C_{LV}$$

$$C_{Lp} = K_P \sin \alpha \cos^2 \alpha$$

$$C_{LV} = K_V \sin^2 \alpha \cos \alpha$$

$$\alpha = \text{angle of attack}$$

K_P and K_V are obtained from Figure 26.

The Polhamus vortex lift equation must be modified to make it applicable to thick bodies because the onset of vortex lift for such configurations is not at zero angle of attack as it is with flat plate wings. The thick body angle of attack for onset of vortex lift (α_V) can be correlated with the fineness ratio (FR) and the thickness ratio (TR) of the body as shown in Figure 27a. The body thickness parameters are shown in Figure 27b. Experimental data used in correlation are presented in References 4 through 7. The redefined lift expressions for thick bodies are as follows:

$$C_{Lp} = K_P \sin \alpha \cos^2 \alpha$$

$$C'_{LV} = K_V \sin^2 (\alpha - \alpha_V) \cos (\alpha - \alpha_V)$$

$$C'_L = C_{Lp} + C'_{LV}$$

The body pitching moment is obtained by estimating the center-of-pressure locations of both the potential and vortex lift components. The total pitching moment is equal to the sum of the moments produced by the lift forces acting at their respective center-of-pressure locations plus the zero lift pitching moment. The potential lift center-of-pressure location employed stems from slender body theory and is presented in Figure 28 as a function of n. The equation for the powerlaw planform is of the form $R = R_{max} (X/L)^n$. The program computes an exponent n that closely approximates the input planform area. The potential lift center-of-pressure location is obtained from Figure 28 or the equation,

$$X_{cp}/L = 2n/(2n+1)$$

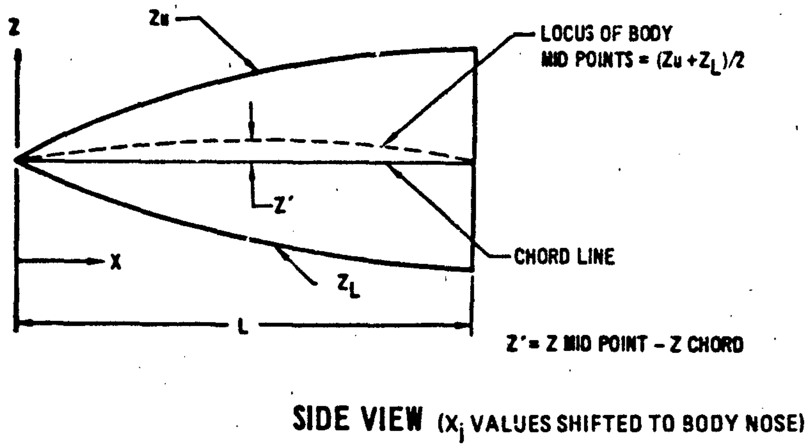
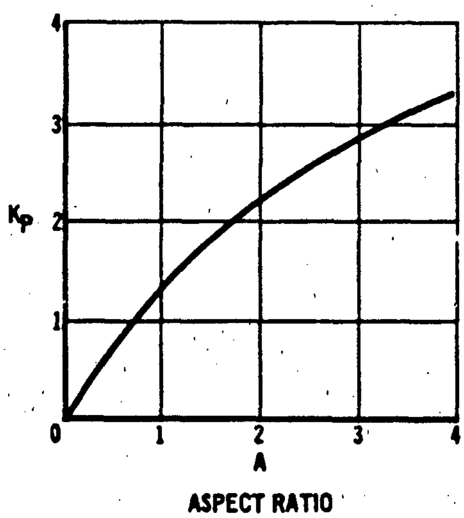
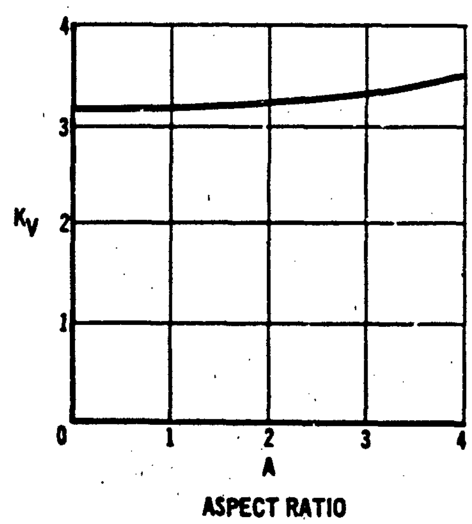


FIGURE 25 ASYMMETRIC BODY GEOMETRY INPUTS

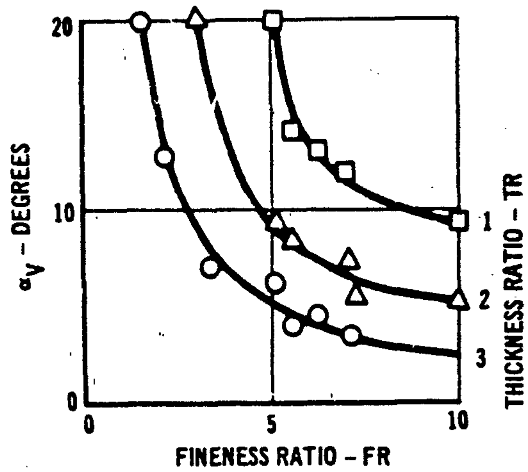


DATCOM FIGURE 4.2.1.2–36a



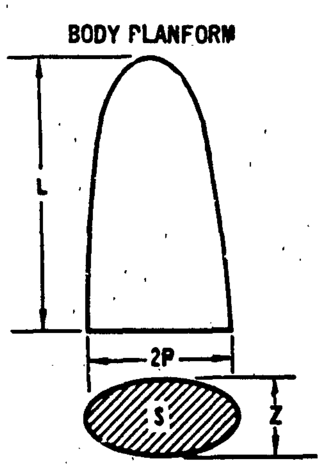
DATCOM FIGURE 4.2.1.2–36b

FIGURE 26 POTENTIAL AND VORTEX LIFT COMPONENTS



DATCOM FIGURE 4.2.1.2–37

FIGURE 27a CORRELATION OF α_V



FINENESS RATIO – FR $= \dfrac{L}{d_e}$; $d_e = \sqrt{\dfrac{4S}{\pi}}$

THICKNESS RATIO – TR $= \dfrac{2R}{Z}$

FIGURE 27b BODY THICKNESS PARAMETERS

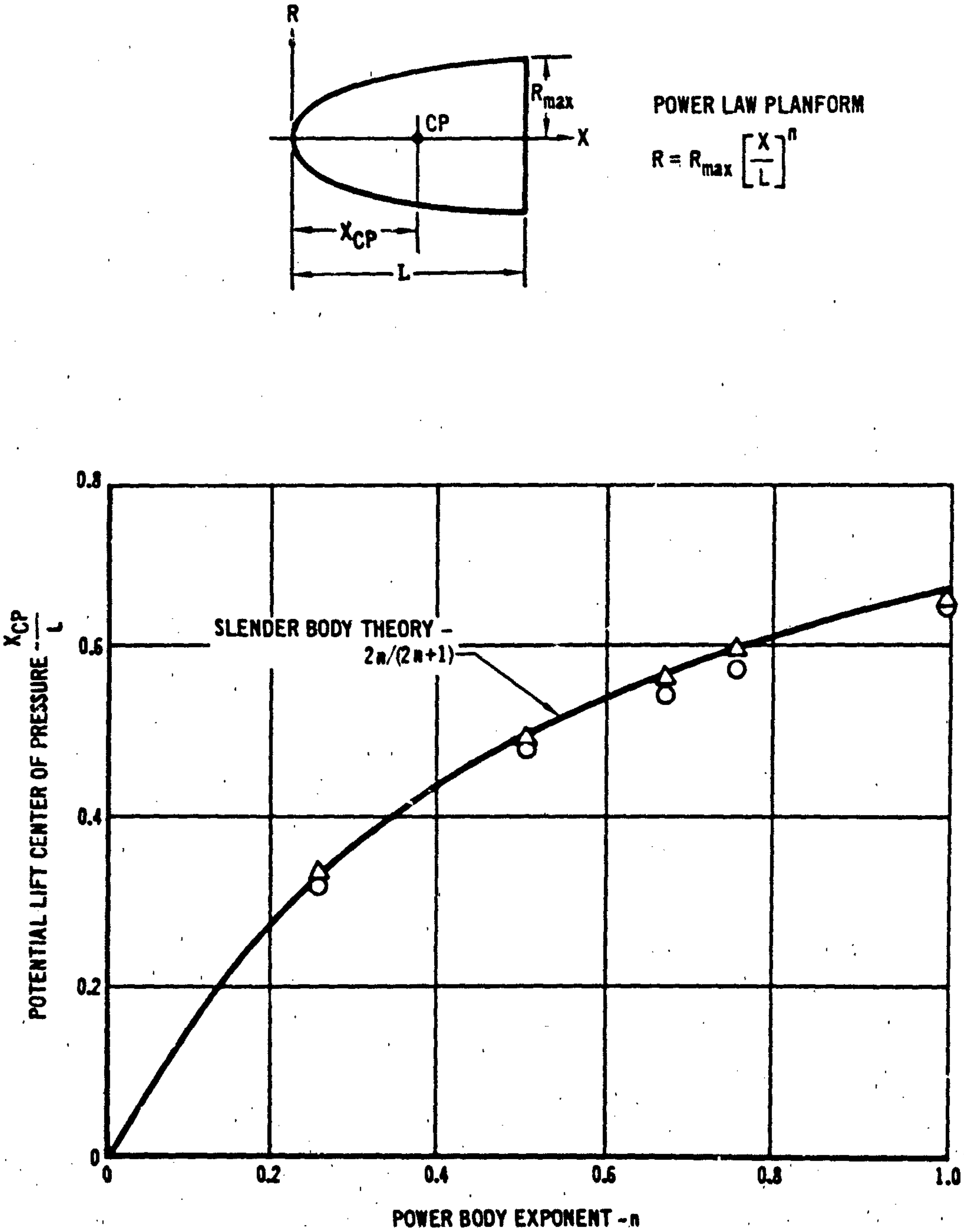


FIGURE 28 POTENTIAL LIFT CENTER OF PRESSURE

Vortex lift center of pressure is assumed to be located at the total planform centroid of area. The equation for the body pitching moment coefficient is:

$$C_m = C_{m_O} + C_{m_P} + C_{m_V}$$

$$C_{m_P} = C_{N_P} (X_{CG} - X_{CP})/L$$

$$C_{m_V} = C_{N_V} (X_{CG} - \overline{X})/L$$

where $\overline{X}$ is the location of the total planform center of area measured from the body nose. The method is applicable at angles of attack equal to or greater than the wing maximum lift angle of attack.

4.6.2 Transonic Bodies

Digital Datcom body solutions are restricted to lift and pitching moment slopes, and drag coefficients in the transonic speed regime. These data are computed by performing a linear interpolation between the subsonic ($M = 0.60$) and supersonic ($M = 1.4$) Mach regimes.

Subroutines that implement the transonic body methods are BØDYRT, SUPBØD, TRSØNI, and TRSØNJ.

4.6.3 Supersonic Bodies

Supersonic body analysis provides solutions for lift, drag and pitching moment coefficients. Datcom methods for lift, pitching moment slope, and drag coefficient require the body to be synthesized from a combination of body components comprised of a nose, after-body, and/or tail segments. Digital Datcom requires synthesized body configurations to be either nose alone, nose-after body, nose-after body-tail, or nose-tail segment combinations.

Some of the Datcom body drag methods in this speed regime have not been implemented in Digital Datcom. The effects of blunted noses on drag are not incorporated since the body lift and pitching moment methods do not reflect the influences of this parameter. Some of the Datcom interference drag methods are also deleted. In this case, the methods were omitted because of their limited range of applicability.

Calculation of wing-body, or horizontal tail-body, stability requires the lift curve slope of the body ahead of the wing or horizontal tail. Body C_N methods are executed for the portion of the body ahead of the wing, if the wing is present; the portion of the body ahead of the horizontal tail, if the horizontal tail is present; and entire body.

All methods are implemented by subroutine SUPBØD except for a portion of the drag methods contained in subroutine FIG26.

4.6.4 Hypersonic Bodies

Hypersonic body analysis is performed at user designated Mach numbers that are equal or greater than 1.4. In this speed regime, Digital Datcom stability solutions include lift, drag and pitching moment coefficients.

Hypersonic body analyses for lift and pitching moment slopes and drag coefficients, like the supersonic body methods, require the body to be synthesized from a combination of body segments. Hypersonic body analysis is unlike the other Datcom hypersonic configuration analyses since the methods are defined independent of the supersonic results. Body C_{N_α} for portions of the body ahead of the wing and/or horizontal tail are also calculated.

The methods are implemented in subroutine HYPBOD. A small portion of the drag methods are found in subroutine FIG26.

4.7 TRANSONIC WING-BODY C_L

The transonic wing-body lift coefficient, if not input using namelist EXPR--, is computed in subroutine WBCLB using the following equations:

$$C_{L_I} = (C_{L_\alpha})^*_W \ (\alpha_j)_W$$

$$(C_{L_j})_{WB} = (C_{L_\alpha})_B \ \alpha_j + \left[K_{W(B)} + K_{B(W)}\right] (C_{L_\alpha})^*_W \alpha_j$$

$$+ I_{V_{B(W)}} \left(\frac{\Gamma}{2\pi\alpha_j V r_{C_{re}}/2}\right)\left(\frac{d}{b}\right) \alpha_j \ (C_{L_\alpha})^*_W$$

$$+ \left[k_{W(B)} + k_{B(W)}\right] C_{L_I}$$

In computing the transonic wing-body pitching moment slope, the center of pressure of body-wing carryover is linearly interpolated between the values obtained at Mach 0.60 and Mach 1.40 in subroutine TRANCM.

4.8 WING-BODY-TAIL MOVEABLE HORIZONTAL TAIL TRIM

The all moveable horizontal tail incidence required to trim the vehicle ($C_{M_{C.G.}} = 0$) at angle of attack is calculated in subroutine TRIMR2. At trim, the forces on the tail are C_{L_H} and C_{D_H} (trimmed lift and drag), and are referenced to the local flow at a tail angle of attack of $(\alpha - \epsilon_H)$. Since these trimmed forces are located at the tail aerodynamic center, which is known, the total body moments can be summed as follows:

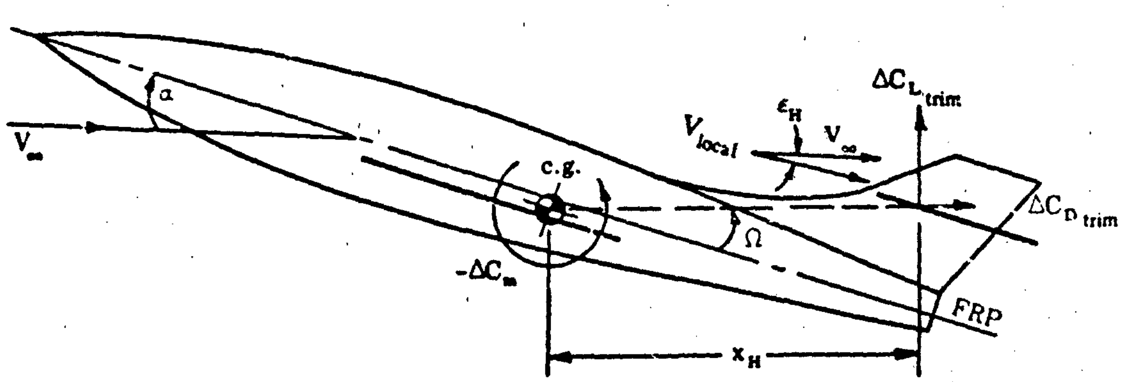
$$C_{M_{WB}} + C_{M_{OH}} \frac{q_H}{q_\infty} - C_{L_H} \frac{q_H}{q_\infty} \left[\frac{\Delta X_{AC}}{\bar{C}_W} \cos(\alpha - \epsilon_H) + \frac{\Delta Z_{AC}}{\bar{C}_W} \sin(\alpha - \epsilon_H) \right]$$

$$+ C_{D_H} \frac{q_H}{q_\infty} \left[\frac{\Delta Z_{AC}}{\bar{C}_W} \cos(\alpha - \epsilon_H) - \frac{\Delta X_{AC}}{\bar{C}_W} \sin(\alpha - \epsilon_H) \right] = 0$$

C_{D_H} can be expressed as

$$C_{DH} = C_{D_{O_H}} + \frac{(C_{L_H})^2}{\pi A_H e_H}$$

Hence, the only unknown is C_{L_H}, the tail lift at trim, which can be evaluated. From Sketch (a) note that



VIEW IN PLANE OF SYMMETRY

α = Airplane angle of attack (positive as shown)

x_H = Distance from c.g. to quarter-chord point of horizontal-stabilizer MAC

Ω = Angle defined by intersection of x_H with FRP (positive as shown with horizontal stabilizer above c.g.)

Sketch (a)

$$\frac{\Delta X_{ac}}{\bar{C}_W} = \frac{X_H}{\bar{C}_W} \cos \Omega$$

$$\frac{\Delta Z_{ac}}{\bar{C}_W} = \frac{X_H}{\bar{C}_W} \sin \Omega$$

Thus,

$$\frac{\Delta X_{ac}}{\bar{C}_W} \cos (\alpha - \varepsilon_H) + \frac{\Delta Z_{ac}}{\bar{C}_W} \sin (\alpha - \varepsilon_H)$$

$$= \frac{X_H}{\bar{C}_W} [\cos \Omega \cos (\alpha - \varepsilon_H) + \sin \Omega \sin (\alpha - \varepsilon_H)]$$

$$= \frac{X_H}{\bar{C}_W} \cos (\Omega - \alpha - \varepsilon_H)$$

$$\frac{\Delta Z_{ac}}{\bar{C}_W} \cos (\alpha - \varepsilon_H) - \frac{\Delta X_{ac}}{\bar{C}_W} \sin (\alpha - \varepsilon_H)$$

$$= \frac{X_H}{\bar{C}_W} [\sin \Omega \cos (\alpha - \varepsilon_H) - \cos \Omega \sin (\alpha - \varepsilon_H)]$$

$$= \frac{X_H}{\bar{C}_W} \sin (\Omega - \alpha + \varepsilon_H)$$

The moment equation reduces to

$$C_{M_{WB}} + C_{M_{OH}} \frac{q_H}{q_\infty} - C_{L_H} \frac{q_H}{q_\infty} \frac{X_H}{\bar{C}_W} \cos(\Omega - \alpha + \epsilon_H)$$

$$+ \left[C_{D_{OH}} + \frac{(C_{LH})^2}{\pi A_H e_H} \right] \frac{q_H}{q_\infty} \frac{X_H}{\bar{C}_W} \sin(\Omega - \alpha + \epsilon_H) = 0$$

Letting $\delta = \Omega - \alpha + \epsilon_H$ and re-arranging yields a quadratic on C_{L_H}.

$$\frac{q_H}{q_\infty} \frac{X_H}{\bar{C}_W} \sin\delta \frac{(C_{LH})^2}{\pi A_H e_H}$$

$$- \frac{q_H}{q_\infty} \frac{X_H}{\bar{C}_W} \cos\delta \, (C_{L_H})$$

$$+ C_{D_{OH}} \frac{q_H}{q_\infty} \frac{X_H}{\bar{C}_W} \sin\delta + C_{M_{WB}} + C_{M_{OH}} \frac{q_H}{q_\infty} = 0$$

Simplifying,

$$\frac{\tan\delta}{\pi A_H e_H} (C_{L_H})^2 - C_{L_H} + C_{D_{OH}} \tan\delta + \frac{C_{M_{WB}} + C_{M_{OH}} \frac{q_H}{q_\infty}}{\frac{q_H}{q_\infty} \frac{X_H}{\bar{C}_W} \cos\delta} = 0$$

From the quadratic formula,

$$C_{L_H} = \frac{1 \pm \sqrt{1 - 4 \left[\frac{\tan\delta}{\pi A_H e_H}\right] \left[C_{D_{OH}} \tan\delta + \frac{C_{M_{WB}} + C_{M_{OH}} \frac{q_H}{q_\infty}}{\frac{q_H}{q_\infty} \frac{X_H}{\bar{C}_W} \cos\delta} \right]}}{2 \left[\frac{\tan\delta}{\pi A_H e_H}\right]}$$

In this form, the equation becomes invalid for $= 0$, and can be further reduced to

$$C_{L_H} = \frac{2\left[\dfrac{C_{M_{WB}} + C_{M_{OH}}\dfrac{q_H}{q_\infty}}{\dfrac{X_H}{\bar{C}_W}\dfrac{q_H}{q_\infty}\cos\delta} + C_{D_{OH}}\tan\delta\right]}{1 + \sqrt{1-4\left[\dfrac{\tan\delta}{\pi A_H e_H}\right]\left[\dfrac{C_{M_{WB}} + C_{M_{OH}}\dfrac{q_H}{q_\infty}}{\dfrac{X_H}{\bar{C}_W}\dfrac{q_H}{q_\infty}\cos\delta} + C_{D_{OH}}\tan\delta\right]}}$$

A plus sign in front of the radical is the valid solution, otherwise at $\delta = 0$ the solution is undefined. This result is similar to Datcom equation 4.5.3.2-e, with the exception of the term "$C_{m_{OH}}\, q_H/q_\infty$."

Once the tail lift at trim (C_{L_H}) has been determined, a variation of Datcom equation 4.5.1.2-a can be used to calculate the tail incidence α_{i_H}.

$$C_{L_H} = C'_{L_H}\left(K_{H(B)} + K_{B(H)}\right)$$

$$+ C'_{L_{\alpha_H}}(\alpha_{i_H})\,[k_{H(B)} + k_{B(H)}]$$

$$+ I_{V_{B(H)}}\left(\frac{\Gamma}{2\pi\alpha V r}\right)_H \frac{(b/_2 - b^*/_2)}{(b/_2)}\, C_{L^*_{\alpha_H}}\, \alpha_{eff}$$

where $C_{L_H}{}^*$ is the pseudo lift-curve-slope of the horizontal tail in the presence of the body,

$$C_{L^*_{\alpha_H}} = C_{L_{\alpha_H}}\left(K_{H(B)} + K_{B(H)}\right)$$

C_{L_H}' and C_{L_H}' are the horizontal tail lift and lift curve slope at

$(\alpha - \varepsilon_H + \alpha_{OH})$

and α_{eff} is the effective angle of attack of the horizontal tail in the presence of the body

$$\alpha_{eff} = \alpha - \varepsilon_H + \alpha_{OH} + \alpha_{i_H} \left(\frac{k_{H(B)} + k_{B(H)}}{K_{H(B)} + K_{B(H)}} \right)$$

The incidence angle to trim can then be solved directly, and becomes

$$\alpha_{i_H} = \frac{C_{L_H} - \left(K_{H(B)} + K_{B(H)}\right)\left[C_{L_H}' + C_{L_{\alpha H}}'\left(\alpha - \varepsilon_H + \alpha_{OH}\right) I_{V_{E(H)}}\left(\frac{\Gamma}{2\pi V r}\right)_H \left(\frac{b/2 - b^*/2}{b/2}\right)\right]}{\left(k_{B(H)} + k_{H(B)}\right)\left[C_{L'_{\alpha_H}} + I_{V_{B(H)}}\left(\frac{\Gamma}{2\pi V r}\right)_H \left(\frac{b/2 - b^*/2}{b/2}\right) C_{L_{\alpha H}}\right]}$$

Once the tail lift and drag at trim has been computed the panel hinge moment about the pivot point can also be computed. Since C_{L_H} and C_{D_H} are are referenced to the local flow, they must be computed relative to the freestream flow. Relative to V_∞, trim lift and drag are

$$C_{L_{H_{TRIM}}} = (C_{L_{H_T}} \cos \varepsilon - C_{D_{H_T}} \sin \varepsilon) \frac{q_H}{q_\infty}$$

$$C_{D_{H_{TRIM}}} = C_{D_{0_H}} + \frac{\left(C_{L_{H_{TRIM}}}\right)^2}{\pi A_H e_H}$$

The pitching moment trimmed is

$$C_{M_{H_{TRIM}}} = C_{L_{H_{TRIM}}} \left[\frac{X_H}{\bar{C}_W} \cos \delta\right] + C_{D_{H_{TRIM}}} \left[\frac{X_H}{\bar{C}_W} \sin \delta\right]$$

The hinge moment about the pivot point is

$$C_{HM} = \left[C_{L_{H_{TRIM}}} \cos \alpha + C_{D_{H_{TRIM}}} \sin \alpha\right]$$

4.9 WING-BODY-TAIL TRIM WITH CONTROL DEVICES

Configuration trim with wing or horizontal tail control devices is performed in subroutine TRIMRT. The method programmed, which is not a Datcom method, essentially does a table look-up of the control device incremental pitching moment coefficient versus control deflection for the deflection required to trim. The incremental lift coefficient and drag coefficient are then obtained by performing table look-ups for these variables (which are a function of control deflection angle) at the trimmed control deflection.

4.10 STANDARD ATMOSPHERE MODEL

Incorporation of a standard atmosphere model (subroutine ATMOS) into Digital Datcom provides input and output flexibility for the user. The program can operate on Mach number and altitude as separate independent variables. The addition of vehicle weight and flight path angle permit calculation of equilibrium flight conditions.

The program allows the user to input either Mach number or velocity as an independent variable for speed reference. If velocity is input, the free stream static temperature must be available so that Mach number can be calculated. The user will also have the option to specify a flight altitude, or static pressure and temperature, as an independent variable defining the atmospheric conditions. If altitude is specified, pressure and temperature will be found using the "U.S. Standard Atmosphere, 1962."

The user may input up to 20 Mach or velocity points. If Mach number is input, the velocity will be calculated for each point where atmospheric data are input. When velocity is input the Mach number will be calculated using atmospheric conditions. If velocity is input instead of Mach numbers and atmospheric conditions are not defined, an error message will be written and Mach numbers will be calculated using a speed of sound of 1000 ft/sec.

The user may also input up to 20 atmospheric conditions. The atmosphere may be defined by altitude, pressure and temperature, or Reynolds number. If the altitude is given, pressure and temperature will be determined using the atmosphere model developed in Reference 9. The Reynolds number will be calculated using the following equation (in the foot-pound-second system of units):

$$RN/L = \rho V/\mu = 1.2527 \times 10^6 \, PM \, (T + 198.6)/T^2$$

This equation was derived using the following relationships:

$$\rho = P/RT$$

$$V = M \sqrt{\gamma RT}$$

$$\mu = 2.270 \times 10^{-8} \, T^{1.5}/(T + 198.6)$$

If the Reynolds number is not input and cannot be calculated, an error message will be written and the Reynolds number will be set to 5×10^6/ft.

Given the vehicle weight, flight path angle, and atmospheric conditions, the equilibrium flight aerodynamic data can be determined. Equilibrium flight is achieved when the following relationship is satisfied.

$$WT = (C_L \cos\delta - C_D \sin\delta)\, qS$$

Along with the untrimmed aerodynamic output, the level flight ($\delta = 0$) lift coefficient will be output. Trim data output will provide an additional line of output at the equilibrium flight conditions (subroutine FLTCL).

SECTION 5

SYSTEM RESOURCE REQUIREMENTS

Digital Datcom is a large and rather complex computer program which requires specific computer resources to execute within a fixed core requirement. The program is written to conform to the American National Standards Institute (ANSI) Standard Fortran IV. Certain computer resources must be available to make the program operational without modifications These resources are:

- o Six disk files or scratch tapes are required for manipulation and retrieval of input data. The logical I/O units used are 8, 9, 10, 11, 12 and 13. These logical units are in addition to logical unit 5 (read) and unit 6 (write).
- o The system must have capability for primary and secondary overlay structures.
- o The system must have a Fortran compiler which provides for NAMELIST input and output, and statement transfer when an end of file is detected.

Each logical unit referenced by the program is reserved for a specific purpose. The units referenced and their use in the program are listed below:

Unit	Program Usage
5	Standard system input (card reader)
6	Standard system output (printer)
8	Storage of experimental data namelists for the case being executed
9	Storage of input namelists, except experimental data, for the case being executed
10	Storage of experimental data namelists for a single Mach number
11	Storage of all input data after processing by the input diagnostic analysis module (CØNERR)
12	Storage of extrapolation messages for processing by overlay 57
13	Storage of output data for use with the Plot Module as a post-processing option

SECTION 6

PROGRAM CONVERSION MODIFICATIONS

6.1 GENERAL REMARKS

The program was written in Fortran IV for the CDC Cyber 175 computer system. Several program modifications may be required to run under other Fortran compilers or computer systems. It is recommended that users implementing the program for their computer system become familiar with their installation operating system and Fortran compiler requirements. Users are forewarned that program core requirements and run times discussed in this report may no longer be valid.

6.2 PROGRAM STRUCTURE

The program is composed of a root segment overlay (overlay 0), fifty-seven primary overlays and twenty-eight secondary overlays. Table 7 shows the overall program structure and lists those routines that are contained in each overlay. In the CDC system, the first routine in an overlay is called a "program" and subsequent routines "subroutines." Several subroutines appear in more than one overlay. These subroutines are called "common decks" and are listed in Table 8.

6.2.1 Calls to Overlay

All primary overlays are called by the root segment overlay, and secondary overlays called by their respective primary overlay using the calling sequence

CALL OVERLAY (4LDATC, XX, YY, 6HRECALL)

where: DATC is the disc file where the overlay is located,
XX is the primary overlay number in decimal, and
YY is the secondary overlay number in decimal.

Hence, each overlay is written to a disk file with the name "DATC." Users should refer to the Fortran reference manual for their system and determine the correct overlay calling procedure.

6.2.2 Common Decks

Several subroutines are used in more than one overlay. The most commonly used routines are located in the root segment for access by all overlay programs and subroutines. However, several decks are used by only a few routines and placing them in the root segment would require an increase in overall program core size. In order to maintain a low core requirement, these common decks are located in each overlay in which it is referenced.

Warning - Not all systems allow two routines to have the same name even though they are identical. If the user's system does not allow this option, three alternatives are available as follows:

- o Rename each deck that is common, and change the calling sequence to it.

  natives are available as follows:

- o Place all common decks in the root segment (overlay 0) and remove the deck from each associated overlay. The user will increase the overall program core requirement by using this technique, however, it is easier than the procedure outlined above.
- o On some systems that have multiple region capability, these common decks can be placed in a separate overlay region.

6.2.3 "OVERLAY" and "PROGRAM" Cards

Each primary and secondary overlay main program contains these two cards. The CDC Fortran compiler requires all overlays to begin with an "OVERLAY" card followed by a main program which begins with a "PROGRAM" card. These must be replaced by corresponding code required by the operating system and compiler being employed.

6.2.4 End of File Tests

Routines INPUT, CØNERR and XPERNM utilize a transfer on end of file. This statement must be modified for the Fortran compiler being used.

6.2.5 Use of "UNUSED" and "KAND"

These constants are set in BLØCK DATA. The value for "UNUSED" is set in the program as 10^{-60}. It is sometimes used as a program flag and is used for initializing all variable arrays to some number other than zero. The value for "UNUSED" can be changed if desired and must be defined in BLØCK DATA as a small positive number. The variable "KAND" defines the alphabetic character used by the NAMELIST inputs. It is set to '$' for CDC systems.

SECTION 7

PROGRAM DECK DESCRIPTION

This section contains a description of all routines in Digital Datcom. Table 7 lists the decks by overlay, Table 8 lists those "common decks" in the program, and Table 9 describes the purpose of each deck and the overlays referenced. For convenience, Table 9 lists the routines in alphabetical order. Table 10 discusses the use of each of the variables in the Digital Datcom control data blocks. The description of the plot module routines is provided in Volume III of this report.

A complete program listing, which includes Digital Datcom and the Plot Module, is provided as a microfiche supplement to this report.

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
00	DATCOM	TOP LEVEL PROGRAM CONTROL - COMMONLY USED ROUTINES
	MAIN00	
	MAIN01	
	MAIN02	
	MAIN03	
	MAIN04	
	MAIN05	
	MAIN06	
	MAIN07	
	BLØCK DATA	
	TBFUNX	
	QUAD	
	INTERX	
	TLIN3X	
	TLINEX	
	TLIN1X	
	GLØØK	
	SWITCH	
	MESSGE	
	FIG26	
	CLMCHO	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
	EXSUBT	
01	M01ØØ1	INITIALIZE PROGRAM AND PROCESS INPUT DATA
01,1	INITZE	INITIALIZES DATA BLOCKS AND PRINT FLAGS
	ZERANG	
01,2	INPUT	READ AND WRITE INPUTS
	TEST	
	WRLØIP	
	WRHTIP	
	WRVTIP	
	WRVFIP	
	INPUTL	
	INPUT2	
	INPUT3	
	INPUT4	
01,3	CHECK	CHECK MACH REGIME LIMITS, CHECK FOR MISSING NAMELISTS
	CØNV	
	ATMØS	
	MAJERR	
01,4	CØNERR	CHECK USER INPUTS FOR SYNTAX ERRORS
	NMLIST	
	TESTØR	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
	VNAME	
	LVALUE	
	RVALUE	
	CCARD	
	NMTEST	
02	MO2Ø02	CALCULATE CASE GEOMETRIC AND SYNTHESIS DATA
	WTGEØM	
	ANGLES	
	ZERANG	
	SETUP1	
	INFTGM	
	SYNDIM	
	ARCLSS	
03	MO3Ø03	CALCULATE WING DRAG DATA
	CDRAG	
	FIG53A	
04	MO4Ø04	CALCULATE SUBSONIC ASYMMETRIC BODY AERODYNAMICS
	BØDØPT	
	TRAPZ	
	EQSPCE	
	GETMAX	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
	EQSPC1	
05	MO5Ø05	CALCULATE HORIZONTAL TAIL DRAG DATA
	CDRAG	
	FIG53A	
06	MO6Ø06	CALCULATE SUBSONIC AXISYMMETRIC BODY AERODYNAMICS
	BØDYRT	
	EQSPCE	
	EQSPC1	
	GETMAX	
	TRAPZ	
	BØDYJM	
07	MO7Ø07	CALCULATE SUBSONIC WING-BODY AERODYNAMICS
07,1	WBAERØ	CALCULATE WING-BODY C_D, C_L, C_M, C_N, C_A
	BØDØWG	
	ALI	
	TRAPZ	
	WBDRAG	
	WBLIFT	
	WBCM	
	WBCMO	
	TABLEC	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
07, 2	WBCD	CALCULATE WING-BODY C_D
	WBCDL	
	TABLES	
	TBSUB	
	TBTRN	
	TBSUP	
08	M08Ø10	CALCULATE SUBSONIC VERTICAL TAIL DRAG DATA
	VTDRAG	
	VFDRAG	
09	M09Ø11	CALCULATE SUBSONIC WING FLOW FIELD AT HORIZONTAL TAIL
	DYPRLS	
	DWASH	
	TRAPZ	
10	M10Ø12	CALCULATE SUBSONIC WING-BODY-TAIL AERODYNAMICS
	BØDØWG	
	ALI	
	WGEØTL	
	WBTAIL	
11	M11Ø13	CALCULATE GROUND EFFECTS
	DMPARY	
	GRDEFF	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
12	M12Ø14	PRINTS OUTPUTS
12, 1	ØUTPUT	PRINT CONVENTIONAL OUTPUTS
	HEADR	
	PRCSID	
	INTERM	
	SWRITE	
12, 2	AUXØUT	PRINT AUXILIARY AND PARTIAL OUTPUTS
	PRCSID	
	SWRITE	
	AXPRNT	
	ARCCØS	
	PRNSEC	
12, 3	WPLØT	WRITE PLOT DATA TO UNIT 13
13	M13Ø15	CALCULATE PROPELLER POWER EFFECTS
	PRPWEF	
	ANGLES	
	ZERANG	
14	M14Ø16	CALCULATE SUBSONIC LOW ASPECT RATIO WING AND WING-BODY AERODYNAMICS
	LØARWB	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
15	M15Ø17	CALCULATE SUBSONIC WING LIFT CHARACTERISTICS
	CALCA0	
	WTLIFT	
	LIFTCF	
	CLMXBS	
	ANGLES	
16	M16Ø20	CALCULATE SUBSONIC HORIZONTAL TAIL LIFT CHARACTERISTICS
	CALCA0	
	WTLIFT	
	LIFTCF	
	CLMXBS	
	ANGLES	
17	M17Ø21	CALCULATE SUBSONIC LATERAL STABILITY DERIVATIVES
	SUBLAT	
	TLIN4X	
18	M18Ø22	CALCULATE SUPERSONIC WING DRAG DATA
	WTGEØM	
	ANGLES	
	ZERANG	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
	SETUP1	
	SUPDRG	
19	M19Ø23	CALCULATE SUPERSONIC BODY AERODYNAMICS
	SUPBØD	
	TRAPZ	
20	M20Ø24	CALCULATE SUPERSONIC WING-BODY AERODYNAMICS AND VERTICAL TAIL C_{D_O}
	SUPWB	
	BØDØWG	
	ALI	
	SUPHB	
	VRTCDØ	
	VFCDØ	
	SUPCMO	
	WBCMO	
	TABLEC	
21	M21Ø25	CALCULATE WING SUPERSONIC FLOW FIELD AT HORIZONTAL TAIL
	SDWASH	
	INFTGM	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
	SDDVC	
	SDWA	
	SDWB	
	SDWC	
	SDWD	
	SDWE	
	DPRESR	
	FIG68	
	MACH2	
	ARCSIN	
	ARCCØS	
22	M22Ø26	CALCULATE SUPERSONIC HORIZONTAL TAIL AERODYNAMICS
	SUPLTG	
23	M23Ø27	CALCULATE SUPERSONIC LATERAL STABILITY DERIVATIVES
	SUPLAT	
	TRAPZ	
	SUPLAH	
	MASRAT	
	SUPLAV	
	SUPLAF	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
24	M24Ø30	CALCULATE TRANSONIC WING AERODYNAMICS AND BODY STABILITY DATA
24, 1	TRANWB	CALCULATE WING C_{L_α}, $C_{L_{MAX}}$, $\alpha_{C_{L_{MAX}}}$, C_{D_0}; BODY C_{L_α}, C_{m_α}, C_D
	TRSØNI	
	CLMXB1	
	TRANF	
	TRANWG	
24, 2	TRANHB	CALCULATE HORIZONTAL TAIL C_{L_α}, $C_{L_{MAX}}$, $\alpha_{C_{L_{MAX}}}$, C_{D_0}
	TRSØNJ	
	CLMXB1	
	TRANF	
	TRNHT	
24, 3	TRANCD	CALCULATE WING-BODY, H.T.-BODY C_D
	WBCDL	
	TABLES	
	TBSUB	
	TBTRN	
	TBSUP	
25	M25Ø31	CALCULATE TRANSONIC WING AND WING-BODY C_{m_α}
	TRANAC	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
25, 1	TRANCM	CALCULATE WING, WING-BODY C_{m_α}
	TLIN4X	
	WBCM1	
	WBTRAN	
25, 2	TRHTCM	CALCULATE H.T., H.T.-BODY C_{m_α}
	TLIN4X	
	WBCM1	
	HBTRAN	
25,3	TRACMO	
	WBCMO	
	TABLEC	
26	M26Ø32	CALCULATE HYPERSONIC BODY AERODYNAMICS
	HYPBØD	
	TRAPZ	
27	M27Ø33	CALCULATE SUPERSONIC WING STABILITY DATA
	SUPLNG	
	M28Ø34	CALCULATE SUPERSONIC WING-BODY-TAIL AERODYNAMICS
	SUPWBT	
	BØDØWG	
	ALI	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
29	M29Ø35	CALCULATE LATERAL STABILITY GEOMETRY DATA
	TRAPZ	
	GETMAX	
30	M30Ø36	CALCULATE JET POWER EFFECTS
	JETPWE	
	TLINVS	
	FG6115	
31	M31Ø37	CALCULATE SUBSONIC WING C_m AND BODY AXIS C_N, C_A
	CMALPH	
	CACALC	
32	M32Ø40	CALCULATE SUPERSONIC VERTICAL TAIL LIFT DATA
	VTLIFT	
	VFLIFT	
33	M33Ø41	CALCULATE SUBSONIC HORIZONTAL TAIL C_m
	CMALPH	
	CACALC	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
34	M34Ø42	DEFINE NUMBER OF CARDS IN EACH EXPERIMENTAL DATA NAMELIST
	XPERNM	
	TEST	
35	M35Ø43	CALCULATE TRANSONIC WING-BODY-TAIL C_{L_α} AND SECOND LEVEL METHODS
35, 1	SETUP2	SET-UP FOR SECOND LEVEL METHODS
	CLBCLC	
35, 2	WBTRA	CALCULATE TRANSONIC WING-BODY-TAIL DATA
	TRAWBT	
35, 3	SECLEV	COMPUTE SECOND LEVEL DATA
	WINGCL	
	WBCLB	
	BØDØWG	
	ALI	
	WBTCDØ	
	CDWBT	
	CLWBT	
	CNCA	
36	M36Ø44	CALCULATE FLAP LIFT AND HINGE MOMENT DATA
	LIFTFP	
	HINGE	
	CTABS	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
37	M37Ø45	CALCULATE FLAP PITCHING MOMENTS
	SIMUL4	
	TRAPZ	
	FLAPCM	
	GDELTA	
	AGENR	
	DET4	
38	M38Ø46	CALCULATE SUBSONIC FLAP DRAG AND TRIM AERODYNAMICS
	TRIMR2	
	TRIMRT	
	DRAGFP	
39	M39Ø47	PRINT HIGH LIFT AND CONTROL DATA
	ØUTPT2	
	PRCSID	
	SWRITE	
	FLTCL	
	DUMP2	
	DMPARY	
40	M40Ø50	CALCULATE TRANSONIC LATERAL CONTROL/FLAP AERODYNAMICS
	TRNYRL	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
41	M41Ø51	CALCULATE SUPERSONIC HIGH LIFT AND CONTROL DEVICE AERODYNAMICS
	DFLCØN	
	ARCCØS	
	ARCSIN	
	SSHING	
	SSSYM	
	PTCP	
42	M42Ø52	CALCULATE HYPERSONIC FLAP AERODYNAMICS
	FIG68	
	ARCSIN	
	ARCCØS	
	SIMUL2	
	PRCSID	
	DMPARY	
42, 1	HYPFLP	CALCULATE HYPERSONIC FLAP DATA FOR FLOW PROPERTIES
	HYPRØP	
42, 2	ØUTPT4	PRINT HYPERSONIC FLAP DATA
	ALDLPR	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
43	M43Ø53	CALCULATE DYNAMIC DERIVATIVES-SUBSONIC, TRANSONIC, SUPER-SONIC
	TLIP3X	
	TLIP2X	
	TLIP1X	
	YUP	
	CMALPØ	
	SUBPAW	
	SUBPAH	
43, 1	SUPPAW	
43, 2	SUPCMQ	
43, 3	SUPPAH	CALCULATE H.T. DYNAMIC DERIVATIONS
43, 4	SUPHMQ	CALCULATE H.T. C_{m_q} DERIVATIONS
44	M44Ø54	CALCULATE SUPERSONIC WING "&" DERIVATIVES
	ARCSIN	
	TLIP3X	
	TLIP2X	
	TLIP1X	
	YUP	
	SUPCLD	
	SUPHLD	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
	CALCA	
45	M45Ø55	CALCULATE WING AND WING-BODY YAW AND ROLL DERIVATIVES
	TLIP3X	
	TLIP2X	
	TLIP1X	
	YUP	
	INTEP3	
45, 1	WINGYW	
	SUBRYW	
	SUPRYW	
45, 2	HØRTYW	
	SUBHYW	
	SUPHYW	
46	M46Ø56	CALCULATE WING-BODY-TAIL DYNAMIC DERIVATIVES
	TRAPZ	
	PRCSID	
	DMPARY	
	CLRDER	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
	DYNBØD	
	DNPAWB	
	DNPWBT	
	SUBWBT	
47	M47Ø57	CALCULATE HYPERSONIC TRANSVERSE JET CONTROL AERODYNAMICS
	TRANJT	
	SIMUL2	
	TRAPZ	
	INTER3	
	ØUTTRJ	
	DMPARY	
	PRCSID	
48	M48Ø60	LOAD EXPERIMENTAL DATA NAMELISTS FOR THE CURRENT MACH NUMBER ON TAPE 10
	EXPDAT	
49	M49Ø61	DUMP ARRAYS USED IN CASE EXECUTION
	DUMPARY	
	DUMPRT	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
50	M50Ø62	CALCULATE AIRFOIL SECTION GEOMETRIC AND AERODYNAMIC DATA
	INIZ	
	SECI	
	SECØ	
	CSLØPE	
	XYCØRD	
	DELY	
50, 1	AIRFØL	
	ARCCØS	
	DECØDE	
	CØØRD4	
	CØRD4M	
	CØØRD5	
	CØRD5M	
	CØØRD1	
	CØØRD6	
	CØRDSP	
	SLEQ	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
50, 2	THEØRY	
	IDEAL	
	ASMINT	
	SLØPE	
50, 3	MAXCL	
51	M51Ø63	INITIALIZE COMPUTATIONAL ARRAYS
	INITZ1	
	INITZ2	
52	M52Ø64	CALCULATE SUBSONIC LATERAL CONTROL/FLAP AERODYNAMICS
	TLIN4X	
	LATFLP	
53	M53Ø65	CALCULATE SUPERSONIC TRAILING EDGE FLAP ROLL AND YAW AERODYNAMICS
	ARCCØS	
	DFLCØN	
	SPRYAW	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
54	M54Ø66	CALCULATE SUPERSONIC WING $C_{m_{\dot{\alpha}}}$
	TLIP3X	
	TLIP2X	
	TLIP1X	
	YUP	
	SUPCMD	
	SUPHMD	
55	M55Ø67	CALCULATE JET FLAP AERODYNAMICS
	JETFP	
56	M56Ø70	CALCULATE MACH SHADOWING DATA
	VTAREA	
	PTINT1	
	AREA1	
	BDAREA	
	PTINT2	
	AREA2	
57	M57Ø71	DUMP CASE EXTRAPOLATION MESSAGES
	CLEARA	
	DECFIG	
	SØRTER	
	READXM	

TABLE 7 DIGITAL DATCOM OVERLAY DESCRIPTION

OVERLAY	PROGRAM/SUBROUTINE NAME	OVERLAY DESCRIPTION
	STØRXM WRITXM	

TABLE 8 PROGRAM COMMON DECKS

Deck Name	Overlays Referenced
ALI	7, 10, 20, 28, 35
ANGLES	2, 13, 15, 16, 18
ARCCØS	12, 21, 41, 42, 50, 53
ARCSIN	21, 41, 42, 44
BØDØWG	7, 10, 20, 28, 35
CALCALC	31, 33
CALCAO	15, 16
CDRAG	3, 5
CLMXBS	15, 16
CLMXB1	24 (Both Secondary Overlays)
CMALPH	31, 33
DFLCØN	41, 53
DMPARY	11, 39, 42, 46, 47, 49
EQSPCE	4, 6
EQSPC1	4, 6
FIG53A	3, 5
FIG68	21, 42
GETMAX	4, 6, 29
INFTGM	2, 21
LIFTCF	15, 16
PRSCID	12, 39, 42, 46, 47
SETUP1	2, 18
SIMUL2	38, 42, 47
SWRITE	12, 39
TABLEC	7, 20, 25
TABLES	7, 24
TBSUB	7, 24
TBSUP	7, 24
TBTRN	7, 24
TEST	1, 34
TLIN4X	17, 25, 26, 52
TLIP1X	43, 44, 45, 54
TLIP2X	43, 44, 45, 54
TLIP3X	43, 44, 45, 54
TRANF	24 (Both Secondary Overlays)
TRAPZ	4, 6, 9, 19, 23, 26, 29, 37, 46, 47
WBCDL	7, 24
WBCMO	7, 20, 25
WBCM1	25 (Both Secondary Overlays)
WTGEØM	2, 18
WTLIFT	15, 16
YUP	43, 44, 45, 54
ZERANG	1, 2, 13, 18

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
AGENR	37	GENERATES COEFFICIENTS FOR G/δ CALCULATIONS BY GDELTA
AIRFØL	50	CONTROLLING PROGRAM FOR CALCULATING AIRFOIL GEOMETRY FROM NACA DESIGNATION
ALDLPR	42	PRINTS BLANKS WHEN NO COMPUTED VALUES ARE PRESENT
ALI	7,10,20,28,35	COMPUTES VORTEX INTERFERENCE FACTORS
ANGLES	2,13,15,16,18	COMPUTES TRIG AND INVERSE TRIG FUNCTIONS OF AN ARGUMENT
ARCLSS	2	CLASSIFIES WING/TAIL PLANFORM AS HIGH OR LOW ASPECT RATIO
ARCCØS	12,21,41,42,50 53	COMPUTES ARC-COSINE OF AN ARGUMENT USING STANDARD FORTRAN
ARCSIN	21,41,42,44	COMPUTES ARC-SINE OF AN ARGUMENT USING STANDARD FORTRAN
AREA1	56	CALCULATES INCREMENTAL AREAS OF VERTICAL TAIL SHADOWED BY MACH LINE
AREA2	56	CALCULATES INCREMENTAL AREA OF BODY SHADOWED BY MACH LINE
ASMINT	50	NON-LINEAR INTERPOLATION ROUTINE FOR AIRFOIL SECTION MODULE
ATMØS	1	COMPUTES PROPERTIES OF 1962 U.S. STANDARD ATMOSPHERE
AUXØUT	12	PRINT AUXILIARY OUTPUTS FOR A CASE
AXPRNT	12	PRINT AUXILIARY OUTPUTS FOR WING/TAIL PLANFORMS
BDAREA	56	EXECUTIVE FOR BODY PARTS SHADOWED BY MACH LINE SHADOWING CALCULATIONS
BLØCK DATA	0	SETS PROGRAM CONSTANTS, AND VARIABLE NAMES FOR CØNERR
BØDØPT	4	COMPUTES ASYMMETRICAL BODY AERODYNAMICS
BØDØWG	7,10,20,28,35	COMPUTES BODY VORTEX EFFECTS ON WING
BØDYRT	6	COMPUTES AXISYMMETRIC BODY C_L, C_D, C_m
BODYJM	6	COMPUTE BODY AERODYNAMICS USING JOERGENSEN'S METHOD
CACALC	31, 33	COMPUTES WING C_N, C_A
CALCA	44	COMPUTES WING ACCELERATION PARAMETERS (&)

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
CALCAO	15, 16	COMPUTES LIFTING SURFACE α_{o_L}
CCARD	1	CHECK CONTROL CARD FOR LEGAL INPUT
CDRAG	3, 5	COMPUTES LIFTING SURFACE C_D
CDWBT	35	CALCULATES TRANSONIC WING-BODY-TAIL C_D
CHECK	1	CHECK MACH REGIME LIMITS AND SET PRINT FLAGS
CLBCLC	35	CALCULATES TRANSONIC WING AND WING-BODY C_{ℓ_β} AND C_{ℓ_β}/C_L
CLEARA	57	CLEAR STORAGE ARRAYS FOR EXTRAPOLATION MESSAGES
CLMCHO	0	COMPUTES LIFTING SURFACE C_L AT MACH = 0
CLMXBS	15, 16	COMPUTES LIFTING SURFACE $C_{L_{MAX}}$
CLMXB1	24	COMPUTES LIFTING SURFACE $C_{L_{MAX}}$ AT MACH = 0.6
CLRDER	46	COMPUTES THE CONFIGURATION C_{ℓ_r} DERIVATIVE
CLWBT	35	COMPUTES TRANSONIC WING-BODY-TAIL C_L
CMALPH	31, 33	COMPUTES LIFTING SURFACE C_{m_α}
CMALPØ	43	COMPUTES LIFTING SURFACE C_{m_α} AT MACH=0
CNCA	35	CALCULATES C_N AND C_A
CØNERR	1	CONTROLLING PROGRAM FOR INPUT ERROR DIAGNOSTIC ANALYSIS
CØØRD1	50	CALCULATES NACA 1-SERIES AIRFOIL COORDINATES
CØØRD4	50	CALCULATES NACA 4-DIGIT AIRFOIL COORDINATES
CØØRD5	50	CALCULATES NACA 5-DIGIT AIRFOIL COORDINATES
CØØRD6	50	CALCULATES NACA 6-SERIES AIRFOIL COORDINATES
CØRD4M	50	CALCULATES NACA 4-DIGIT MODIFIED AIRFOIL COORDINATES

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
CØRD5M	50	CALCULATES NACA 5-DIGIT MODIFIED AIRFOIL COORDINATES
CØNV	1	SET-UP FOR UNITS SPECIFICATION
CORDSP	50	CALCULATE GEOMETRY DATA FOR SUPERSONIC AIRFOILS
CSLØPE	50	COMPUTE GEOMETRIC SLOPE FOR SUPERSONIC AIRFOILS
CTABS	36	CONTROL TABS METHOD SUBROUTINE
DATCØM	0	TOP LEVEL EXECUTIVE PROGRAM
DECFIG	57	CONVERT FIGURE NUMBERS IN EXTRAPOLATION MESSAGES
DET4	37	EVALUATES A 4x4 DETERMINATE
DECØDE	50	DECODES USER INPUT NACA DESIGNATION
DELY	50	CALCULATES AIRFOIL ΔY
DFLCØN	41,53	CALCULATES SUPERSONIC LIFT, ROLL MOMENT AND HINGE MOMENT DERIVATIVES
DMPARY	11,39,42,46,47,49	DUMP SPECIFIED ARRAY IN READABLE FORMAT
DNPAWB	46	CALCULATES WING-BODY "q" AND "$\dot{\alpha}$" DERIVATIVES
DNPWBT	46	CALCULATES WING-BODY-TAIL "q" AND "$\dot{\alpha}$" DERIVATIVES
DPRESR	21	CALCULATES NON-VISCOUS DYNAMIC PRESSURE AT HORIZONTAL TAIL
DRAGFP	38	CALCULATES SUBSONIC FLAP INDUCED DRAG
DUMPRT	49	DUMPS ARRAYS USING DMPARY
DUMP2	39	CONTROL FOR PRINTING DUMPS OF INTERMEDIATE RESULTS

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
DWASH	9	CALCULATES SUBSONIC DOWNWASH AT ANGLE-OF-ATTACK
DYNBØD	46	CALCULATES BODY DYNAMIC DERIVATIVES
DYPRLS	9	COMPUTES DYNAMIC PRESSURE AT HORIZONTAL TAIL
EQSPCE	4, 6	TRANSFORMS 4-DIMENSIONAL ARRAY SO THAT THE 3 INDEPENDENT ARRAYS ARE EQUALLY SPACED
EQSPC1	4, 6	TRANSFORMS 2-DIMENSIONAL ARRAY LIKE EQSPCE
EXPDAT	48	LOADS THE EXPERIMENTAL DATA NAMELIST FOR THE CURRENT MACH NUMBER
EXSUBT	0	READS EXPERIMENTAL DATA INPUTS
FIG26	0	CALCULATES FIG. 4.1.5.1-26; TURBULENT SKIN FRICTION COEFFICIENT
FIG53A	3, 5	CALCULATES FIG. 4.1.5.2-53A; SUBSONIC LEADING EDGE SUCTION
FIG68	21, 42	CALCULATES OBLIQUE SHOCK WAVE ANGLE (TR-1135, EQN. 150)
FG6115	30	CALCULATES FIG. 4.6.1-15; DOWNWASH INCREMENT DUE TO A SUBSONIC JET IN A SUBSONIC STREAM
FLAPCM	37	COMPUTES WING C_m DUE TO FLAPS
FLTCL	39	PRINT DATA FOR TRIM CONDITIONS
GDELTA	37	CALCULATES FLAP SPANWISE LOADING COEFFICIENT, G/δ
GETMAX	4, 6, 29	FOR Y=f(X), FIND Y_{MAX} AND X_{YMAX}
GLØØK	0	TABLE LOOKUP LOGIC FOR TLIN_X ROUTINES
GRDEFF	11	COMPUTES GROUND EFFECTS ON AERODYNAMICS
HBTRAN	25	CALCULATES $(C_{L_\alpha})_{B(H)}$ AND $(X_{ac}/\bar{c}_r)$ AT MACH=1.4 FOR TRANSONIC ANALYSIS
HEADR	12	WRITE HEADINGS FOR CASE OUTPUTS

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
HINGE	36	CALCULATES FLAP HINGE MOMENT DATA
HORTYW	45	EXECUTIVE FOR HORIZONTAL-TAIL, HORIZONTAL-TAIL-BODY YAW DERIVATIVE CALCULATIONS
HYPBØD	26	COMPUTES HYPERSONIC C_D, C_L, C_m
HYPFLP	42	COMPUTES HYPERSONIC FLAP CONTROL AERODYNAMICS
HYPRØP	42	CALCULATES EQUILIBRIUM REAL GAS FLOW PROPERTIES
IDEAL	50	CALCULATES AIRFOIL SECTION IDEAL AERODYNAMIC COEFFICIENTS
INFTGM	2, 21	CALCULATES DOWNWASH SYNTHESIZING DIMENSIONS
INITZE	1	PROGRAM INITIALIZING ROUTINE
INITZ1	51	INITIALIZE ARRAYS FOR PROGRAM USE
INITZ2	51	INITIALIZE ARRAYS FOR HIGH-LIFT AND CONTROL
INIZ	50	INITIALIZE ARRAYS FOR AIRFOIL SECTION MODULE
INPUT	1	READS INPUT NAMELISTS
INPUTL	1	READS NAMELIST "LARWB" FOR INPUT
INPUT2	1	READS HORIZONTAL TAIL NAMELISTS FOR INPUT
INPUT3	1	READS VERTICAL TAIL NAMELISTS FOR INPUT
INPUT4	1	READS VENTRAL FIN NAMELISTS FOR INPUT
INTEP3	45	TABEL LOOKUP ROUTINE FOR A SPECIFIC TABLE
INTERM	12	INTERMEDIATE LOGIC FOR OUTPUT
INTERX	0	LINEAR TABLE LOOKUP USING TLIN_X ROUTINES, 2 TO 5 DIMENSIONS
INTER3	47	TABLE LOOKUP ROUTINE FOR A SPECIFIC TABLE

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
INTERM	12	INTERMEDIATE LOGIC FOR OUTPUT
INTERX	0	LINEAR TABLE LOOKUP USING TLIN_X ROUTINES, 2 TO 5 DIMENSIONS
INTER3	47	TABLE LOOKUP ROUTINE FOR A SPECIFIC TABLE
JETFP	55	COMPUTES AERODYNAMIC INCREMENTS DUE TO JET FLAPS
JETPWE	30	COMPUTES EFFECTS OF JET POWER ON AERODYNAMICS
LATFLP	52	SUBSONIC LATERAL CONTROL/FLAP EFFECTIVENESS CALCULATIONS
LIFTCF	15, 16	COMPUTES LIFTING SURFACE C_L
LIFTFP	36	COMPUTES INCREMENTAL WING LIFT DUE TO FLAPS
LØARWB	14	COMPUTES LOW ASPECT-RATIO WING-BODY AERODYNAMICS
LVALUE	1	TEST FOR LEGAL LOGICAL CONSTANTS AND MULTIPLICATION FACTOR FOR INPUT
MACH2	21	CALCULATE PRANDTL-MEYER EXPANSION ANGLE
MAIN00	0	DATCOM PROGRAM TOP-LEVEL EXECUTIVE
MAIN01	0	PROGRAM CONTROL FOR SUBSONIC AERODYNAMICS
MAIN02	0	PROGRAM CONTROL FOR SUBSONIC GROUND EFFECTS
MAIN03	0	PROGRAM CONTROL FOR TRANSONIC AERODYNAMICS
MAIN04	0	PROGRAM CONTROL FOR SUPERSONIC AERODYNAMICS
MAIN05	0	PROGRAM CONTROL FOR SUBSONIC HIGH LIFT AND CONTROL ANALYSIS
MAIN06	0	PROGRAM CONTROL FOR TRANSONIC HIGH LIFT AND CONTROL ANALYSIS
MAIN07	0	PROGRAM CONTROL FOR SUPERSONIC HIGH LIFT AND CONTROL ANALYSIS
MAJERR	1	CHECKS FOR MISSING ESSENTIAL NAMELISTS

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
MASRAT	23	FINDS APPARENT MASS RATIO K, FIGURE 5.3.1.1-25
MAXCL	50	FINDS $C_{\ell MAX}$ FOR AIRFOIL SECTION
MESSGE	0	PRINTS TABLE LOOKUP ROUTINE EXTRAPOLATION MESSAGES
M01Ø01	1	EXECUTIVE FOR OVERLAY 1, INITIALIZE PROGRAM AND PROCESS INPUTS
M02Ø02	2	EXECUTIVE FOR OVERLAY 2, CALCULATE GEOMETRIES AND SYNTHESIS DATA
M03Ø03	3	EXECUTIVE FOR OVERLAY 3, SUBSONIC WING DRAG
M04Ø04	4	EXECUTIVE FOR OVERLAY 4, SUBSONIC ASYMMETRIC BODY AERODYNAMICS
M05Ø05	5	EXECUTIVE FOR OVERLAY 5, SUBSONIC HORIZONTAL TAIL DRAG
M06Ø06	6	EXECUTIVE FOR OVERLAY 6, SUBSONIC AXISYMMETRIC BODY AERODYNAMICS
M07Ø07	7	EXECUTIVE FOR OVERLAY 7. SUBSONIC WING, WING-BODY AERODYNAMICS
M08Ø10	8	EXECUTIVE FOR OVERLAY 8, SUBSONIC VERTICAL TAIL DRAG
M09Ø11	9	EXECUTIVE FOR OVERLAY 9, SUBSONIC WING FLOW FIELDS
M10Ø12	10	EXECUTIVE FOR OVERLAY 10, SUBSONIC WING-BODY-TAIL AERODYNAMICS
M11Ø13	11	EXECUTIVE FOR OVERLAY 11, GROUND EFFECTS
M12Ø14	12	EXECUTIVE FOR OVERLAY 12, PRINT OUTPUTS
M13Ø15	13	EXECUTIVE FOR OVERLAY 13, PROPELLER POWER EFFECTS
M14Ø16	14	EXECUTIVE FOR OVERLAY 14, LOW ASPECT RATIO AERODYNAMICS
M15Ø17	15	EXECUTIVE FOR OVERLAY 15, SUBSONIC WING LIFT
M16Ø20	16	EXECUTIVE FOR OVERLAY 16, SUBSONIC HORIZONTAL TAIL LIFT
M17Ø21	17	EXECUTIVE FOR OVERLAY 17, SUBSONIC LATERAL STABILITY

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
M18Ø22	18	EXECUTIVE FOR OVERLAY 18, SUPERSONIC WING DRAG
M19Ø23	19	EXECUTIVE FOR OVERLAY 19, SUPERSONIC BODY AERODYNAMICS
M20Ø24	20	EXECUTIVE FOR OVERLAY 20, SUPERSONIC WING-BODY AERODYNAMICS
M21Ø25	21	EXECUTIVE FOR OVERLAY 21, SUPERSONIC WING FLOW-FIELDS
M22Ø26	22	EXECUTIVE FOR OVERLAY 22, SUPERSONIC HORIZONTAL-TAIL AERODYNAMICS
M23Ø27	23	EXECUTIVE FOR OVERLAY 23, SUPERSONIC LATERAL STABILITY
M24Ø30	24	EXECUTIVE FOR OVERLAY 24, TRANSONIC WING AERODYNAMICS AND BODY STABILITY DATA
M25Ø31	25	EXECUTIVE FOR OVERLAY 25, TRANSONIC WING/WING-BODY C_{m_α}
M26Ø32	26	EXECUTIVE FOR OVERLAY 26, HYPERSONIC BODY AERODYNAMICS
M27Ø33	27	EXECUTIVE FOR OVERLAY 27, SUPERSONIC WING STABILITY
M28Ø34	28	EXECUTIVE FOR OVERLAY 28, SUPERSONIC WING-BODY-TAIL AERODYNAMICS
M29Ø35	29	EXECUTIVE FOR OVERLAY 29, LATERAL STABILITY GEOMETRY DATA
M30Ø36	30	EXECUTIVE FOR OVERLAY 30, JET POWER EFFECTS
M31Ø37	31	EXECUTIVE FOR OVERLAY 31, SUBSONIC WING C_m, BODY C_A, C_N
M32Ø40	32	EXECUTIVE FOR OVERLAY 32, SUPERSONIC VERTICAL TAIL LIFT
M33Ø41	33	EXECUTIVE FOR OVERLAY 33, SUBSONIC HORIZONTAL TAIL C_m
M34Ø42	34	EXECUTIVE FOR OVERLAY 34, DEFINE EXPERIMENTAL DATA INPUT
M35Ø43	35	EXECUTIVE FOR OVERLAY 35, TRANSONIC AERODYNAMICS
M36Ø44	36	EXECUTIVE FOR OVERLAY 36, FLAP LIFT AND HINGE MOMENTS
M37Ø45	37	EXECUTIVE FOR OVERLAY 37, FLAP PITCHING MOMENTS

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
M38Ø46	38	EXECUTIVE FOR OVERLAY 38, SUBSONIC FLAP DRAG AND TRIM AERODYNAMICS
M39Ø47	39	EXECUTIVE FOR OVERLAY 39, PRINT HIGH LIFT AND CONTROL DATA
M40Ø50	40	EXECUTIVE FOR OVERLAY 40, TRANSONIC LATERAL CONTROL/FLAP AERODYNAMICS
M41Ø51	41	EXECUTIVE FOR OVERLAY 41, SUPERSONIC HIGH LIFT AND CONTROL AERODYNAMICS
M41Ø52	42	EXECUTIVE FOR OVERLAY 42, HYPERSONIC FLAP AERODYNAMICS
M42Ø53	43	EXECUTIVE FOR OVERLAY 43, DYNAMIC DERIVATIVES
M43Ø54	44	EXECUTIVE FOR OVERLAY 44, SUPERSONIC WING "α" DERIVATIVES
M45Ø55	45	EXECUTIVE FOR OVERLAY 45, WING AND WING-BODY YAW AND ROLL DERIVATIVES
M46Ø56	46	EXECUTIVE FOR OVERLAY 46, WING-BODY-TAIL DYNAMIC DERIVATIVES
M47Ø57	47	EXECUTIVE FOR OVERLAY 47, TRANSVERSE-JET AERODYNAMICS
M48Ø60	48	EXECUTIVE FOR OVERLAY 48, LOAD EXPERIMENTAL DATA FOR MACH NUMBER
M49Ø61	49	EXECUTIVE FOR OVERLAY 49, DUMP ARRAYS
M50Ø62	50	EXECUTIVE FOR OVERLAY 50, AIRFOIL SECTION AERODYNAMICS
M51Ø63	51	EXECUTIVE FOR OVERLAY 51, INITIALIZE ARRAYS
M52Ø64	52	EXECUTIVE FOR OVERLAY 52, SUBSONIC LATERAL CONTROL/FLAP AERODYNAMICS
M53Ø65	53	EXECUTIVE FOR OVERLAY 53, SUPERSONIC TRAILING EDGE FLAP ROLL AND YAW AERODYNAMICS
M54Ø66	54	EXECUTIVE FOR OVERLAY 54, SUPERSONIC WING $C_{m_{\dot{\alpha}}}$
M55Ø67	55	EXECUTIVE FOR OVERLAY 55, JET FLAP AERODYNAMICS
M56Ø70	56	EXECUTIVE FOR OVERLAY 56, MACH SHADOWING DATA

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
M57Ø71	57	EXECUTIVE FOR OVERLAY 57, DUMP EXTRAPOLATION MESSAGES
NMLIST	1	PASS NAMELIST NAMES TO TESTOR FOR CHECKING
NMTEST	1	CHECK NAMELIST NAME AS LEGAL INPUT
ØUTPUT	12	MAIN LOGIC FOR PRINTING CASE BASIC OUTPUTS
ØUTPT2	39	PRINTS HIGH-LIFT AND CONTROL OUTPUTS
ØUTPT4	42	PRINTS HYPERSONIC CONTROL EFFECTIVENESS OUTPUTS
ØUTTRTJ	47	PRINTS TRANSVERSE JET CONTROL EFFECTIVENESS OUTPUTS
PRCSID	12,39,42,46,47	PRINTS "CASEID" CARD
PRNSEC	12	PRINTS SECOND LEVEL METHOD DATA
PRPWEF	13	CALCULATES PROPELLER POWER EFFECTS ON AERODYNAMICS
PTCP	41	CALCULATES SUBSONIC FLAP/CONTROL PRESSURE RATIO ΔD C_p
PTINT1	56	CALCULATES THE BOUNDARIES OF THE MACH LINE ON THE VERTICAL TAIL
PTINT2	56	CALCULATES THE BOUNDARIES OF THE MACH LINE ON THE BODY
QUAD	0	COMPUTES PARAMETERS FOR QUADRATIC EXTRAPOLATION
READXM	57	LEADS EXTRAPOLATION MESSAGES FROM UNIT 12
RVALUE	1	TEST IF REAL VALUE IS LEGAL INPUT
SDDVC	21	ROUTINE LOOK UP DATCOM FIGURE 4.7.1-76
SDWA	21	ROUTINE LOOK UP DATCOM FIGURE 4.7.1-76
SDWASH	21	COMPUTES $\partial\epsilon/\partial\alpha$ AND VISCOUS q/q_∞ AT THE HORIZONTAL TAIL
SDWB	21	ROUTINE LOOK-UP DATCOM FIGURE 4.7.1-76
SDWC	21	ROUTINE LOOK-UP DATCOM FIGURE 4.7.1-76
SDWD	21	ROUTINE LOOK-UP DATCOM FIGURE 4.7.1-76
SDWE	21	ROUTINE LOOK-UP DATCOM FIGURE 4.7.1-76

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
QUAD	0	COMPUTES PARAMETERS FOR QUADRATIC EXTRAPOLATION
RVALUE	1	TEST IF REAL VALUE IS LEGAL INPUT
SDWASH	21	COMPUTES $\partial\epsilon/\partial\alpha$ AND VISCOUS q/q_{∞} AT THE HORIZONTAL TAIL
SECI	50	READ AIRFOIL SECTION INPUTS
SECLEV	35	COMPUTES SECOND LEVEL METHOD MODULE DATA
SECØ	50	SET AIRFOIL SECTION MODULE OUTPUTS IN INPUT NAMELIST ARRAYS
SETUP1	2, 18	COMPUTES TRIG FUNCTIONS FOR LIFTING SURFACES
SETUP2	35	SETUP FOR TRANSONIC CONFIGURATION ANALYSIS
SIMUL2	38, 42, 47	SOLVES FOR WHERE TWO CURVES INTERSECT
SIMUL4	37	SOLVES 4 SIMULTANEOUS EQUATIONS USING DETERMINATES
SLEQ	50	SOLVES N SIMULTANEOUS EQUATIONS USING THE GAUSS-JORDAN METHOD
SLØPE	50	CALCULATES AIRFOIL SECTION C_{ℓ_α}, C_{m_0} AND $X_{a.c.}$
SØRTER	57	SORT EXTRAPOLATION MESSAGES BY FIGURE NUMBER
SPRYAW	53	CALCULATES SUPERSONIC ROLL AND YAW CHARACTERISTICS OF PLAIN T.E. FLAPS, SPOILERS AND DIFFERENTIALLY DELETED STABILIZERS
SSHING	41	CALCULATES SUPERSONIC HINGE MOMENT DERIVATIVES
SSSYM	41	CALCULATES SUPERSONIC ΔC_L AND ΔC_m FOR HIGH-LIFT AND CONTROL DEVICES
STØRXM	57	STORE EXTRAPOLATION MESSAGE DATA
SUBHYW	45	CALCULATES SUBSONIC HORIZONTAL TAIL AND HORIZONTAL TAIL-BODY "p" AND "r" DERIVATIVES
SUBLAT	17	CALCULATES SUBSONIC AND TRANSONIC LATERAL STABILITY DERIVATIVES
SUBPAH	43	CALCULATES SUBSONIC AND TRANSONIC "q" AND "$\dot{\alpha}$" DERIVATIVES FOR H.T.
SUBPAW	43	CALCULATES SUBSONIC AND TRANSONIC "q" AND "$\dot{\alpha}$" DERIVATIVES FOR WINGS
SUBRYW	45	CALCULATES SUBSONIC WING AND WONG-BODY "p" AND "r" DERIVATIVES

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
SUBWBT	46	CALCULATES SUBSONIC WING-BODY-TAIL "p" AND "r" DERIVATIVES
SUPBØD	19	CALCULATES SUPERSONIC BODY C_L, C_D. C_m, C_{L_α}, AND C_{M_α}
SUPCLD	44	CALCULATES SUPERSONIC WING $C_{L_{\dot{\alpha}}}$
SUPCMD	54	CALCULATES SUPERSONIC WING $C_{m_{\dot{\alpha}}}$
SUPCMO	20	CALCULATES SUPERSONIC CONFIGURATION C_{m_o}
SUPCMQ	43	CALCULATES SUPERSONIC WING C_{m_q}
SUPDRG	18	CALCULATES SUPERSONIC WING C_D
SUPHB	20	CALCULATES SUPERSONIC HORIZONTAL TAIL-BODY C_L, C_D, C_L AND C_{m_α}
SUPHLD	43	CALCULATE $C_{L_{\dot{\alpha}}}$ FOR SUPERSONIC HORIZONTAL TAILS
SUPHMD	54	CALCULATE $C_{M_{\dot{\alpha}}}$ FOR SUPERSONIC HORIZONTAL TAILS
SUPHMQ	43	CALCULATES SUPERSONIC H.T. C_{mq}
SUFHYW	45	CALCULATES SUPERSONIC HORIZONTAL TAIL AND HORIZONTAL-TAIL BODY "p" AND "r" DERIVATIVES
SUPLAF	23	CALCULATES SUPERSONIC VENTRAL FIN LATERAL STABILITY DERIVATIVES
SUPLAH	23	CALCULATES SUPERSONIC LATERAL STABILITY DERIVATIVES FOR HORIZONTAL TAILS
SUPLAT	23	CALCULATES SUPERSONIC LATERAL STABILITY DERIVATIVES FOR WINGS
SUPLAV	23	CALCULATES SUPERSONIC VERTICAL TAIL LATERAL STABILITY DERIVATIVES
SUPLNG	27	CALCULATES SUPERSONIC WING C_L, C_{L_α} AND C_{m_α}
SUPLTG	22	CALCULATES SUPERSONIC HORIZONTAL TAIL C_L, C_{L_α} AND C_{m_α}
SUPPAH	43	CALCULATES SUPERSONIC H.T. C_{L_q}
SUPPAW	43	CALCULATES SUPERSONIC WING C_{L_q}

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
SUPRYW	45	CALCULATES SUPERSONIC WING AND WING-BODY "p" AND "r" DERIVATIVES
SUPWB	20	CALCULATES SUPERSONIC WING-BODY C_L, C_D, C_L AND C_m
SUPWBT	28	CALCULATES SUPERSONIC WING-BODY-TAIL AERODYNAMICS
SWITCH	0	SETS LOGIC FOR ASCENDING OR DESCENDING ARRAYS FOR TLIN_X ROUTINES
SWRITE	12, 39	CONTROLS NUMERIC OUTPUTS FOR OUTPUT; WRITES BLANKS, NA OR NDM
SYNDIM	2	CALCULATES SYNTHESIS DIMENSIONS FOR BODY ANALYSIS
TABLEC	7, 20, 25	REGRESSION COEFFICIENTS FOR WBCMO
TABLES	7,24	READ MACH TABLES OF C_D EQUATION REGRESSION COEFFICIENTS
TBFUNX	0	TABLE LOOKUP FOR Y=f(X); PROVIDES dY/dX
TBSUB	7, 24	SUBSONIC C_D REGRESSION COEFFICIENT TABLES
TBSUP	7, 24	SUPERSONIC C_D REGRESSION COEFFICIENT TABLES
TBTRN	7, 24	TRANSONIC C_D REGRESSION COEFFICIENT TABLES
TEST	1, 34	NAMELIST NAME CHECKING PERFORMED IN INPUT
TESTØR	1	CHECK IF NAMELIST NAME IS LEGAL INPUT USING NMTEST
THEØRY	50	MAIN LOGIC ROUTINE FOR CALCULATING AIRFOIL SECTION AERODYNAMICS
TLINEX	0	LINEAR INTERPOLATION FOR Y=f(X1, X2)
TLINVS	30	INTERPOLATES BETWEEN TABLES FOR FG6115
TLIN1X	0	LINEAR INTERPOLATION FOR Y=f(X)
TLIN3X	0	LINEAR INTERPOLATION FOR Y=f(X1, X2, X3)
TLIN4X	17,25,26,52	LINEAR INTERPOLATION FOR Y=f(X1, X2, X3, X4)
TLIP1X	43,44,45,54	LINEAR INTERPOLATION FOR A PACKED TABLE FOR Y=f(X)
TLIP2X	43,44,45,54	LINEAR INTERPOLATION FOR A PACKED TABLE FOR Y=f(X1, X2)
TLIP3X	43,44,45,54	LINEAR INTERPOLATION FOR A PACKED TABLE FOR Y=f(X1, X2, X3)

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
TRACMO	25	EXECUTIVE TRANSONIC B-W OR B-H C_{m_0}
TRANAC	25	COMPUTES TRANSONIC PLANFORM C_L BY NON-LINEAR INTERPOLATION
TRANCD	24	CALCULATES TRANSONIC WING AND WING-BODY C_D
TRANCM	25	CALCULATES TRANSONIC WING AND WING-BODY C_m
TRANF	24	COMPUTES TRANSONIC VENTRAL FIN C_L BY NON-LINEAR INTERPOLATION
TRANHB	24	EXECUTIVE FOR TRSØNJ CALCULATIONS
TRANJT	47	HYPERSONIC TRANSVERSE JET SIZING CALCULATIONS
TRANWB	24	EXECUTIVE FOR TRSØNI CALCULATIONS
TRANWG	24	CALCULATES WING C_{L_α} AT M=1.4 FOR TRSONI
TRAPZ	4,6,7,9,19,23, 26,29,37,40,47	TRAPEZOIDAL RULE INTEGRATION ROUTINE
TRAWBT	35	CALCULATES WING-BODY-TAIL $\partial\epsilon/\partial\alpha$, q/q_∞ AND C_{L_α} TRANSONICALLY
TRHTCM	25	CALCULATES HORIZONTAL-TAIL AND HORIZONTAL-TAIL BODY C_{m_α} TRANSONICALLY
TRIMRT	38	CALCULATES SUBSONIC TRIM WITH WING OR HORIZONTAL TAIL CONTROL
TRIMR2	38	CALCULATES SUBSONIC TRIM WITH AN ALL MOVABLE HORIZONTAL TAIL
TRNHT	24	CALCULATES HORIZONTAL TAIL C_{L_α} AT MACH=1.4 FOR TRSØNJ
TRNYRL	40	TRANSONIC LATERAL CONTROL/FLAP EFFECTIVENESS CALCULATIONS
TRSØNI	24	CALCULATES TRANSONIC WING C_{L_α}, $C_{L_{MAX}}$, $\alpha C_{L_{MAX}}$; BODY C_{L_α}, C_{m_α}; WING AND WING-BODY C_{D_0}
TRSØNJ	24	USES METHOD OF TRSØNI, BUT CALCULATES USING HORIZONTAL TAIL

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
VFCDØ	20	CALCULATES VENTRAL FIN C_{D_O}
VFDRAG	8	CALCULATES VENTRAL FIN DRAG
VFLIFT	32	CALCULATES SUPERSONIC VENTRAL FIN $C_{L\alpha}$
VNAME	1	CHECK IF VARIABLE NAME IS CORRECT FOR INPUT
VRTCDØ	20	CALCULATES SUPERSONIC VERTICAL TAIL C_{D_O}
VTAREA	56	EXECUTIVE FOR VERTICAL TAIL AREA SHADOWED BY MACH LINE CALCULATIONS
VTDRAG	8	CALCULATES SUBSONIC VERTICAL TAIL C_{D_O}
VTLIFT	32	CALCULATES SUPERSONIC VERTICAL TAIL $C_{L\alpha}$
WBAERØ	7	EXECUTIVE CONTROL FOR WING-BODY AND HORIZONTAL TAIL BODY C_L, C_D AND C_m
WBCD	7	EXECUTIVE CONTROL FOR WING-BODY AND HORIZONTAL TAIL BODY C_D
WBCDL	7, 24	CALCULATES THE WING-BODY/HORIZONTAL TAIL BODY C_{D_L}
WBCLB	35	CALCULATES TRANSONIC WING-BODY C_{ℓ_β}
WBCM	7	CALCULATES SUBSONIC WING-BODY C_m
WBCMO	7, 20, 25	CALCULATES C_{m_o} FOR WING-BODIES USING REGRESSION METHOD
WBCM1	25	CALCULATES $X_{ac}/\bar{c}_r$ FOR WING-BODIES
WBDRAG	7	CALCULATES SUBSONIC WING-BODY C_D
WBLIFT	7	CALCULATES SUBSONIC WING-BODY C_L
WBTCDØ	35	CALCULATES TRANSONIC WING-BODY-TAIL C_{D_O}
WBTRA	35	CALCULATES TRANSONIC WING BODY C_{D_L}
WBTRAN	25	CALCULATES $(C_{L_\alpha})_{B(W)}$ AND $(X_{ac}/\bar{c}_r)_{B(W)}$ AT MACH=1.4 FOR TRANSONIC ANALYSIS
WBTAIL	10	CALCULATES SUBSONIC WING-BODY-TAIL AERODYNAMICS
WINGCL	35	CALCULATES TRANSONIC WING C_L
WINGYW	45	MAIN LOGIC FOR WING YAW DAMPING DERIVATIVES
WGEOTL	10	CALCULATES SUBSONIC WING VORTEX INTERFERENCE EFFECTS ON HORIZONTAL TAIL

TABLE 9 DIGITAL DATCOM ROUTINE DESCRIPTION

ROUTINE NAME	OVERLAYS REFERENCED	DESCRIPTION
WPLØT	12	WRITES DATA FOR PLOT OPTION TO UNIT 13
WRHTIP	1	PRINTS HORIZONTAL TAIL NAMELIST INPUTS
WRITXM	57	PRINTS SUMMARIZED EXTRAPOLATION MESSAGES
WRLØIP	1	PRINTS LOW ASPECT RATIO WING-BODY NAMELIST INPUTS
WRVFIP	1	PRINTS VENTRAL FIN NAMELIST INPUTS
WRVTIP	1	PRINTS VERTICAL TAIL NAMELIST INPUTS
WTGEØM	2, 18	CALCULATES WING OR TAIL GEOMETRY DATA
WTLIFT	15, 16	CALCULATE WING OR TAIL LIFT CHARACTERISTICS
XPERNM	34	DEFINE THE NUMBER OF CARDS IN THE INPUT EXPERIMENTAL DATA NAMELIST
XYCØRD	50	CALCULATES AIRFOIL SECTION X, Y COORDINATES OR THICKNESS/CAMBER DISTRIBUTION
YUP	43,44,45,54	UNPACKS DATA FOR TLIP_X ROUTINES
ZERANG	1,2,13,18	INITIALIZES ANGLES FOR ANGLES ROUTINE

TABLE 10 CONTROL DATA BLOCKS

COMMON BLOCK	VARIABLE NAME	USE/PURPOSE
ØVERLY	NLØG	NUMBER OF LOGICAL VARIABLES IN COMMON BLOCK FLØLØG TO BE INITIALIZED FALSE
	NMACH	NUMBER MACH NUMBERS
	I	MACH NUMBER INDEX
	NALPHA	NUMBER OF ANGLES OF ATTACK
	IG	HAS SEVERAL USES: (1) GROUND HEIGHTS INDEX (2) INITIALIZATION SWITCH OVERLAY 51. IF 1, INITIALIZE IOM AND COMPUTATIONAL BLOCKS, IF 2, INITIALIZE FOR FLAP ANALYSIS IF 3, INITIALIZE FOR POWER ANALYSIS
	NF	HAS SEVERAL USES: (1) FLAP DEFLECTION INDEX (2) IF NEGATIVE, "TURNS-OFF" EXTRAPOLATION MESSAGES (3) FOR TRANSONIC ANALYSIS, LOOP INDEX. IF ≥ -5, GET SUBSONIC AERO IF -6 OR -7, GET SUPERSONIC AERO (4) IF NEGATIVE BYPASS READING EXPERIMENTAL DATA INPUTS
	LF	SET TO 1 IN OVERLAY 23 TO PRINT MESSAGE THAT H.T. IS OFF BODY AND NO LAT.-STAB. PARAMETERS CALC.
	K	ALTITUDE INDEX
	NØVLY	CURRENT EXECUTING OVERLAY NUMBER
CASEID	IDCASE (74)	CHARACTERS OF CASE TITLE INPUT USING "CASEID"
	KØUNT	NUMBER OF NAMELISTS READ (MAX. 300)
	NAMSV (100)	ORDER OF NAMELISTS SAVED FROM PREVIOUS CASE

TABLE 10 CONTROL DATA BLOCKS

COMMON BLOCK	VARIABLE NAME	USE/PURPOSE
	IDIM	DIMENSIONAL SYSTEM USED 1 = FT, 2 = IN, 3 = M, or 4 = CM.
EXPER	KLIST	NUMBER OF $EXPR - NAMELISTS (100 MAX)
	NLIST (100)	NUMBER CARDS READ FOR EACH $EXPR -- AND MACH NUMBER FOR NAMELIST
	NNAMES	NUMBER $EXPR -- CARDS PRESENT
	IMACH	MACH NUMBER INDEX OF CURRENT $EXPR READ
	MDATA	TRUE IF $EXPR DATA FOR MACH NUMBER
	KBØDY	TRUE IF BODY EXPERIMENTAL INPUTS
	KWING	TRUE IF WING EXPERIMENTAL INPUTS
	KHT	TRUE IF H.T. EXPERIMENTAL INPUTS
	KVT	TRUE IF V.T. EXPERIMENTAL INPUTS
	KWB	TRUE IF WING-BODY EXPERIMENTAL INPUTS
	KDWASH (3)	TRUE IF (1) $d\epsilon/d\alpha$, OR (2) ϵ OR (3) q/q_∞
	ALPØW	TRUE IF α_{o_W} EXPERIMENTAL INPUT
	ALPLW	TRUE IF $\alpha_W{}^*$ EXPERIMENTAL INPUT
	ALPØH	TRUE IF α_{o_H} EXPERIMENTAL INPUT
	ALPLH	TRUE IF $\alpha_H{}^*$ EXPERIMENTAL INPUT
FLØLØG (LOGICAL VARIABLES)	FLTC	TRUE IF $FLTCØN PRESENT
	ØPTI	↓ $ØPTIN ↓
	BØ	↓ $BØDY ↓
	WGPL	TRUE IF $WGPLNF PRESENT

TABLE 10 CONTROL DATA BLOCKS

COMMON BLOCK	VARIABLE NAME	USE/PURPOSE
FLØLØG	WGSC	TRUE IF $WGSCHR PRESENT
	SYNT	$SNYTHS
	HTPL	$HTPLNF
	HTSC	$HTSCHR
	VTPL	$VTPLNF
	VTSC	$VTSCHR
	HEAD	CASEID
	PRPØWR	$PRØPWR
	JETPØW	$JETPWR
	LØASRT	$LARWB
	TVTPAN	TRUE IF $TVTPAN PRESENT
	SUPERS	SUPERSONIC ANALYSIS
	SUBSØN	SUBSONIC ANALYSIS
	TRANSN	TRANSONIC ANALYSIS
	HYPERS	HYPERSONIC ANALYSIS
	SYMFP	TRUE IF $SYMFLP PRESENT
	ASYFP	$ASYFLP
	TRIMC	TRIM
	TRIM	TRIM WITH FLAPS
	DAMP	TRUE IF DAMP PRESENT

TABLE 10 CONTROL DATA BLOCKS

COMMON BLOCK	VARIABLE NAME	USE/PURPOSE
FLØLØG	HYPEFF	TRUE IF $HYPEFF PRESENT
	TRAJET	$TRNJET
	BUILD	BUILD
	FIRST	FIRST ENTRY-CALL CØNERR; ALSO SUITED TO CATALOG $EXPR NAMELISTS
	DRCØNV	DERIV PRESENT
	PART	PART
	VFPL	$VFPLNF
	VFSC	$VFSCHR
	CTAB	$CØNTAB
	PLØT	TRUE IF PLØT PRESENT
ERRØR	IERR	TRUE IF MAJOR INPUT ERROR (e.g. MISSING NAMELIST)
	GØNØGØ	TRUE, EXECUTE CASE; FALSE, GO TO NEXT CASE
	IEND	TRUE IF HAVE READ ALL INPUT DATA PRESENT
	DMPALL	TRUE TO DUMP ALL ARRAYS
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