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(T. Galloway)

GASP- GENERAL AVIATION SYNTHESIS PROGRAM

VOLUME V - WEIGHTS

PART 1 - THEORETICAL DEVELOPMENT

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V.1 WEIGHTS

There are several subroutines in GASP that are used to determine the component weights of the aircraft configuration. These can be grouped into the propulsion related components and the airframe components. The airframe group also includes subroutines that deal with design load conditions, aircraft balance, and tail sizing. Figure V.1.1 lists the weight related subroutines.

There are several options available for determining propulsion related weights and the options depend on whether it is a turbine or internal combustion type of engine. All turbine engines' weights are determined in subroutine ENGWGT. Internal combustion type engine weights may be determined from trend equations in ENGWGT or from a more detailed formulation as found in HOPWSZ and RCWSZ subroutines. Propeller weight may be determined by ENGWGT or the weight methodology of Hamilton-Standard which is contained in subroutine WAIT. If a known propulsion system is being studied, the component weights of the propulsion system may be input directly.

The weight of each aircraft structural component is estimated by a regression analysis equation derived from similar aircraft, and a large number of parameters is used to describe the aircraft. These include linear dimensions, dimensionless parameters such as aspect ratio and fineness ratio, and cargo data such as number of passengers. Their use is based on the assumption that aircraft component weights tend to be in proportion to certain of its dimensions. The tail sizing subroutine is an option which develops horizontal and vertical tail dimensions which satisfy specified stability and control criteria.

PROPULSION:	
ENGWGT (160)	CONTROLS AND COMPUTES ENGINE WEIGHT COMPUTATIONS
HOPWSZ	HORIZONTAL OPPOSED PISTON ENGINE WEIGHT AND SIZE
RCWSZ	ROTARY COMBUSTION ENGINE WEIGHT AND SIZE
WAIT	PROPELLER WEIGHT
AIRFRAME:	
WGHT (510)	STRUCTURAL WEIGHT AND WING POSITIONING
DLOAD	DESIGN LOAD FACTORS
TAIL	TAIL SIZING FOR LONGITUDINAL AND DIRECTIONAL STABILITY

FIGURE V.1.1 WEIGHT SUBROUTINES

V.1.1 Propulsion System Weights

V.1.1.1 Subroutine ENGWGT

This 160 card subroutine is concerned with the weight of the propulsion system, and some 30 input parameters are used to calculate any of about 20 output quantities. The steps followed depend strongly upon the type of engine and propeller being used, and frequently the weights are expressed as statistical functions of other parameters of the aircraft.

The call integers of the subroutine are NTYEX and NTYPX which correspond to engine and propeller types as follows:

NTYEX =
- 1, 2, 3, 11, 12, 13: reciprocating engines
- 4, 14: rotating combustion engines
- 5, 6: turboshaft and turboprop engines
- 7: turbojet or turbofan engine (V.1.1)

NTYPX =
- 1, 11: fixed-pitch propeller
- 2, 12: constant speed propeller
- 3, 13: constant speed, full-feathering propeller
- 4, 14: constant speed, full-feathering, deice propeller
- 5, 15: constant speed, full feathering, reverse deice propeller
- 6, 16: Q-fan propulsor (V.1.2)

The subroutine's generality is possible only because of the large number of inner paths depending on these two integers, and it is not feasible to describe in detail the flow of logic for every pair NTYEX and NTYPX.

The integers NTYEX = 11 to 14 provide the option to use subroutines HOPWSZ and RCWSZ to develop parametric values of weight and size of horizontally opposed piston engines or rotary combustion type of engines. Similarly, values of NTYPX greater than 10 utilize subroutine WAIT through a call to ENGINE to return values of propeller weights. Otherwise, the flow of logic is parallel for arbitrary values of the call integers.

Following the initial zero specification of component weights, the subroutine defines a value for sea level static engine specific weight. If this has not been input, it is computed by

$$SW_{SLS} = \begin{cases} 1.5(1 + .15\ K_{SPCHG}) \ , & NTYEX = 1,\ 2,\ 3 \\ 1 + .2\ K_{SPCHG} & NTYEX = 4 \\ .5 & NTYEX = 5,\ 6 \\ .13 & NTYEX = 7 \end{cases} \qquad (V.1.3)$$

This parameter has units of lb per HP except for the turbojet or turbofan engines, NTYEX = 7, in which case the units are lb per lb of thrust. The same parameter is returned by subroutines HOPWSZ (NTYEX = 11, 12, 13) and RCWSZ (NTYEX = 14), or is equal to .5 (NTYEX = 15, 16).

In terms of this parameter, the engine weight is

$$W_{ENG} = SW_{SLS}\ HP_{MSLS} \qquad (V.1.4)$$

or

$$W_{ENG} = SW_{SLS}\ FN_{SLS} \qquad (V.1.5)$$

where

HP_{MSLS} = maximum sea level static horsepower

FN_{SLS} = sea level static thrust

The nacelle weight is then developed as the product of two parameters

$$W_{NAC} = UW_{NAC} \, A_{NAC} \tag{V.1.6}$$

where

UW_{NAC} = nacelle weight per unit surface area, lb per sq ft

A_{NAC} = total nacelle area, sq ft

In this equation UW_{NAC} is an input parameter and A_{NAC} is computed internally (KNAC = 0, 1) or input (KNAC = 2).

Next, if the number of engines is two or more ($EN_p \neq 1$), the pylon weight is expressed as

$$W_{PYLON} = F_{PYL} (W_{ENG} + W_{NAC})^{.736} \tag{V.1.7}$$

where

F_{PYL} = input scale factor

and W_{ENG}, W_{NAC} are obtained from equations (V.1.4) or (V.1.5) and (V.1.6).

For propeller types NTYP = 1 to 6, the propeller weight is given by numerically complex functions of propeller diameter D_{PROP}, number of blades B_L, propeller blade activity factor A_F, engine speed RPM and horsepower HP_{MSLS}, and cruise Mach number EM_{CRU}. These input parameters are used to define numerical values for the constants XK_1, XC_W, F_1, F_2, ..., F_6. For NTYP = 1 to 5, the propeller weight is then estimated by the function

$$W_{FAN} = SK_1 \, F_1^2 \, F_2^{.7} \, F_3^{.75} \, F_4^{15} \, F_5^{.12} \, F_6^{.5} + XC_W \tag{V.1.8}$$

and for NTYP = 6, the fan weight is similarly written as

$$W_{FAN} = X_{K1} \, F_1^{1.85} \, F_2^{.7} \, F_3^{.6} \, F_4^{.5} \, F_5^{.12} \, F_6^{.5} \tag{V.1.9}$$

and the shroud weight is estimated by

$$W_{SHR} = XK_2 \, D^2_{PROP} \qquad (V.1.10)$$

where XK_2 is a numerical factor and D_{PROP} is propeller diameter, ft.

Torque is computed by its definition in terms of horsepower and RPM,

$$TORQUE = 550 \, HP_{MSLS}/(2. \, \Pi * RPM/60.) \qquad (V.1.11)$$

From this value the gear box weight is estimated by

$$WT_{GB} = XK_3 \, TORQUE^{.84} \qquad (V.1.12)$$

where XK_3 = .085, unless input as nonzero. If the Hamilton-Standard option is specified, subroutine ENGINE returns values for the weight parameters which are computed in subroutine WAIT.

For turbojet or turbofan engines, NTYE = 7, and the engine weight parameters are very simply given between statements 500 and 600 in terms of input parameters, i.e.,

$$FN_{SLS} = WA_{SLS} \, SFN_{SLS} \qquad (V.1.13)$$

$$W_{ENG} = SW_{SLS} \, FN_{SLS} \qquad (V.1.14)$$

nacelle weight,

$$W_{NAC} = UW_{NAC} \, (\Pi \, D^2_E \, XL_{QDE}), \qquad (V.1.15)$$

pylon weight,

$$W_{PYLON} = \begin{cases} F_{PYL} (W_{ENG} + W_{NAC})^{.736} \\ 0 \text{ (if single engine configuration)} \end{cases} \qquad (V.1.16)$$

where input parameters are

WA_{SLS} = sea level static airflow

SFN_{SLS} = sea level static specific thrust

SW_{SLS} = sea level static engine specific weight

FN_{SLS} = sea level static thrust

UW_{NAC} = nacelle weight per unit area

F_{PYL} = factor for pylon weight

The remaining statements of the subroutine are concerned only with print statements for several values of NTYE and NTYP.

V.1.1.2 Subroutine HOPWSZ

This subroutine deals with numerical weight and size of horizontally opposed piston engines. The input to the subroutine is descriptive information as to engine geometry of which the following are the most important:

SKWGT = engine weight calibration factor (nominally 1)

SKWDTH = engine width calibration factor (nominally 1)

RWH = ratio of engine width to height (nominally 1.3)

HPMSLS = maximum sea level standard horsepower of engine

NCYL = number of engine cylinders where $4 \leq NCYL \leq 8$.

HPQAB = ratio of rated horsepower to piston bore area, hp per sq in

Several numerical scale factors are defined as functions of NCYL and other input parameters, after which the principal output quantities are found as

SWSLS = specific weight (sea level standard) lb per hp

ANAC = nacelle surface area, sq ft

XLN = length of nacelle, ft = 1.2 * length of engine (XL)

WN = width of nacelle = 1.08 * width of engine (W)

HN = height of nacelle = 1.2 * height of engine (H)

Using these definitions the engine specific weight is given by

$$SWSLS = SKWGT * SWN \left(\frac{1800.}{PSTSPD}\right) \left(\frac{145.}{BMEP}\right) \qquad (V.1.17)$$

where SWN is found by a table look-up. The variable PSTSPD, piston speed, is computed from

$$PSTSPD = 33000. * HPQAB/BMEP \qquad (V.1.18)$$

Engine width, W, is given by

$$W = SKWDTH * (k * HPMSLS + 30.0) \qquad (V.1.19)$$

$$k = \begin{cases} .0167; & NCYL = 4 \\ .0125 & NCYL = 6 \\ .0106 & NCYL = 8 \end{cases} \qquad (V.1.20)$$

For unsupercharged engines the engine width to length ratio, RWL, is computed from

$$RWL = B - .09 * RWH \qquad (V.1.21)$$

where depending on engine type

$$B = \begin{cases} 1.31 \text{ or } 1.11; & NCYL = 4 \\ 1.10 \text{ or } 0.92; & NCYL = 6 \\ 1.0 \text{ or } 0.83; & NCYL = 8 \end{cases} \qquad (V.1.22)$$

For supercharged engines

$$RWL = B - .156 * RWH \qquad (V.1.23)$$

where

$$B = \begin{cases} 1.53; & NCYL = 4 \\ 1.40; & NCYL = 6 \\ 1.31; & NCYL = 8 \end{cases} \qquad (V.1.24)$$

Engine length and height are then given by

$$L = W/RWL \qquad (V.1.25)$$

and

$$H = W/RWH \qquad (V.1.26)$$

Nacelle surface area is computed from

$$ANAC = 0.8 * (WN + 2.76 * HN) * XLN/144. \qquad (V.1.27)$$

V.1.1.3 Subroutine RCWSZ

This subroutine deals with weight and size characteristics of the rotating combustion engine. The subroutine begins with a tabulation of several input data parameters, after which the input/output computations begin, in much the same order as was described for the previous subroutine. Principal input parameters are listed in the call statement as

HPMSLS = maximum sea level static horsepower

ROTN = number of rotors

GR = ratio of propeller rpm to maximum engine rpm

SKWGT = engine weight calibration factor (nominally 1.)

SKDIAM = engine diameter calibration factor (nominally 1.)

SWSLS = specific weight, standard sea level, lb per hp

Other input quantities passed through COMMON statements, deal with the technology date level (IDATE = 1970 or 1980), and the absence or presence of supercharger (KSPCHG = 0 or 1).

The computational results are straightforward and vary principally with horsepower and number of rotors. Output quantities of the subroutine are

GR = ratio of prop rpm to engine rpm (operating condition)

SWSLS = specific weight with supercharger, lb per hp

ANAC = nacelle wetted area, sq ft

XLN = nacelle length, ft.

V.1.1.4 Propeller Weights, Subroutine WAIT

Propeller weight is estimated in this 30-card subroutine as a numerical function of seven input parameters:

NTYP = IWTCON = airplane propeller type (1 to 5)

ZMWT = Design cruise Mach number

BHP = brake horsepower

DIA = propeller diameter, ft

AFT = activity factor per blade

BLADT = number of blades

TIPSPD= tip speed, ft per sec.

Then, according to several straightforward but nonlinear functions, the output parameters are simply:

WT70 = propeller weight, 1970 technology, lb

WT80 = propeller weight, 1980 technology, lb

Equations employed are

$$W_T = K_W \left[\left(\frac{D}{10}\right)^2 \left(\frac{B}{4}\right)^{0.} \left(\frac{A.F.}{100}\right)^u \left(\frac{ND}{20,000}\right)^v \left(\frac{SHP}{10D^2}\right)^{0.12} (M+1)^{0.5} \right] + C_W \qquad (V.1.28)$$

where

WT70 or WT80 = W_T = propeller wet weight, lbs. (excludes spinner, de-icing and governor)

DIA = D = propeller diameter, ft

BLADT = B = number of blades

AFT = A.F. = blade activity factor

N = propeller speed, rpm (take-off = $\frac{V_{TIP}}{\pi D}$, V_{TIP} = TIPSPD)

BHP = SHP = shaft horsepower, HP (take-off)

ZMWT = M = Mach number (design condition: maximum power cruise)

$$C_W = y\left(\frac{D}{10}\right)^2 (B) \left(\frac{A.F.}{100}\right)^2 \left(\frac{20,000}{ND}\right)^{0.3} = \text{Counterweight Wt., lbs.} \qquad (V.1.29)$$

K_W, u, v, and y values for use in the weight equation are taken from the table below.

Propeller Type (NTYP)	Technology 1970	1980
1	(1)	(1)
2	(2)	(2)
3	(3)	(3)
4	(3)	(4)
5	(3)	(5)

	K_W	u	v	y
(1)	170	0.9	0.35	0
(2)	200	0.9	0.35	0
(3)	220	0.7	0.40	5.0
(4)	190	0.7	0.40	3.5
(5)	190	0.7	0.30	0

Propeller types associated with above K_W, u, v, y are as follows:

1. all fixed-pitch props
2. McCauley non-counterweighted, non-feathering, constant speed props
3. all Hartzell, all Hamilton-Standard small props, and feathering McCauley
4. fiberglass-bladed, constant speed, counterweighted, full feathered
5. fiberglass-bladed, constant speed, double-acting (non-counterweighted), full feathered, reverse

V.1.2 Airframe Weight

The weight of the structure, flight controls, and payload components of the aircraft are determined in subroutines WGHT (Section V.1.2.2). Subroutine DLOAD (Section V.1.2.1) determines the minimum design speeds, and structural load factors while subroutine TAIL (Section V.1.2.4) is used in one of the stability and control options.

V.1.2.1 **Subroutine DLOAD - Design Speeds and Load Factors.** The purpose of this subroutine is the computation of minimum design airspeeds and load factors of an aircraft in accordance with Federal Aviation Regulations, (FAR) Part 23 or 25. These speeds and load factors are used to permit computation of structural weights by subroutine WGHT. Both maneuver load factor and gust load factor are considered.

The primary input parameter to this subroutine is the airplane structural design category indicator CAT_D, defined as

$$CAT_D = \begin{cases} 0, \text{ FAR Part 23 normal category} \\ 1, \text{ FAR Part 23 utility category} \\ 2, \text{ FAR Part 23 aerobatic category} \\ 3, \text{ FAR Part 25 transport category} \end{cases}$$

If FAR Part 23 requirements are selected, wing loading (WOS) is also an important parameter for determining the minimum design airspeeds. Other input parameters passed through the subroutine argument list are

V_{MLFSL} = Estimate of structural design equivalent airspeed, mph

C_{LALPH} = Lift curve slope, per rad.

C_{BARW} = wing mean aerodynamic chord, ft.

The subroutine begins by changing the units of the estimated structural design velocity from miles per hour to knots,

$$V_{MLFKT} = V_{MLFSL}/1.15$$

and if $CAT_D = 3$ branches to statement 50 for transport category aircraft. If one of the FAR Part 23 categories has been selected, the minimum design airspeeds are determined and checked.

For $CAT_D = 0$ or 1, the minimum design cruise speed is

$$V_{CMIN} = \begin{cases} 33.\sqrt{WOS}, & WOS < 20 \\ [33 - .055(WOS-20)]\sqrt{WOS} & WOS \geq 20 \end{cases} \quad (V.1.30)$$

and for $CAT_D = 2$,

$$V_{CMIN} = \begin{cases} 36\sqrt{WOS} & WOS < 20 \\ [36 - .0925(WOS-20)]\sqrt{WOS} & WOS \geq 20 \end{cases} \quad (V.1.31)$$

The minimum design dive speeds are also functions of the wing loading; i.e.,

Normal Category, $CAT_D = 0$:

$$V_{DMIN} = \begin{cases} 1.4\,V_{CMIN}, & WOS < 20 \\ [1.4 - .000625(WOS-20)]V_{CMIN} & WOS \geq 20 \end{cases} \quad (V.1.32)$$

Utility category, $CAT_D = 1$:

$$V_{DMIN} = \begin{cases} 1.5\,V_{CMIN} & WOS < 20 \\ [1.5 - .001875(WOS-20)]V_{CMIN} & WOS \geq 20 \end{cases} \quad (V.1.33)$$

Aerobatic category, $CAT_D = 2$:

$$V_{DMIN} = \begin{cases} 1.55\,V_{CMIN}, & WOS < 20 \\ [1.55 - .0025(WOS-20)]V_{CMIN} & WOS \geq 20 \end{cases} \quad (V.1.34)$$

For transport category ($CAT_D = 3$) there are no lower limits established for the design cruise speed as for FAR Part 23 aircraft. The minimum design dive speed for transport category aircraft is

$$V_{DMIN} = 1.2\ V_{MLFKT} = 1.2\ V_{MO} \qquad \text{(V.1.35)}$$

The limit maneuvering load factors per the FAR's are then specified as

$$EM_{LF} = \begin{cases} 3.8, & CAT_D = 0 \\ 4.4, & CAT_D = 1 \\ 6.0, & CAT_D = 2 \\ 2.5, & CAT_D = 3 \end{cases} \qquad \text{(V.1.36)}$$

The gust load is evaluated at the altitude at which maximum operating equivalent airspeed (VMO) is equal to the speed for maximum operating Mach number (M_{MO}) so long as the altitude falls in the band

$$0 \leq h \leq 20{,}000 \text{ feet (SIGMA} \geq .53281) \qquad \text{(V.1.37)}$$

For a non-transport category aircraft (CATD $\neq$ 3) this limit is 12,500 feet (SIGMA $\geq$ .6820).

The gust load factor is calculated for a 50 ft/sec gust at VCMIN and a 25 ft/sec gust at VDMIN, and the most critical condition selected for the gust load condition.

$$G_{LF} = \max\ (G_{LFC},\ G_{LFO})_D \qquad \text{(V.1.38)}$$

where the design cruise speed and design dive speed load factors are

$$G_{LFC} = 1 + 50\ G_{LFK}\ C_{LALPH}\ V_{CMIN}/(498.\ WOS) \qquad \text{(V.1.39)}$$

$$G_{LFD} = 1 + 25\ G_{LFK}\ C_{LALPH}\ V_{DMIN}/(498.\ WOS) \qquad \text{(V.1.40)}$$

In these equations,

$$G_{LFK} = .88\ U_G/(5.3 + U_G) \qquad \text{(V.1.41)}$$

where the airplane mass ratio is defined as

$$U_G = 2.\ WOS/(.00237*SIGMA*C_{BARW}\ C_{LALPH}*32.2) \qquad \text{(V.1.42)}$$

The closing computation deals with the ultimate load factor, which is simply 50 per cent greater than the most critical of the maneuver or gust load factor

$$U_{LF} = \begin{cases} 1.5\ EM_{LF}\ , & G_{LF} < EM_{LF} \\ 1.5\ G_{LF}\ , & G_{LF} \geq EM_{LF} \end{cases} \qquad \text{(V.1.43)}$$

V.1.2.2 Subroutine WGHT. This subroutine is used to compute the weight of the structural and flight controls components of the aircraft using the geometry established by subroutine SIZE. It is also possible to size tip tanks if the wing volume is inadequate for carrying the available fuel. An additional option provides for relocating the wing on the fuselage and resizing the horizontal tail to provide a given level of longitudinal stability.

The weights of aircraft structural and flight controls components are estimated using nonlinear statistical weight equations, and the inner loop is used to compute fuel weight, which may require the consideration of wing tip tanks, as shown in Figure V.1.2. A secondary loop involves the resizing of the engines, and the third major loop requires that the aircraft center of gravity location satisfy certain stability conditions. This may mean that the tail moment arm must be changed which requires that the inner loop be recomputed.

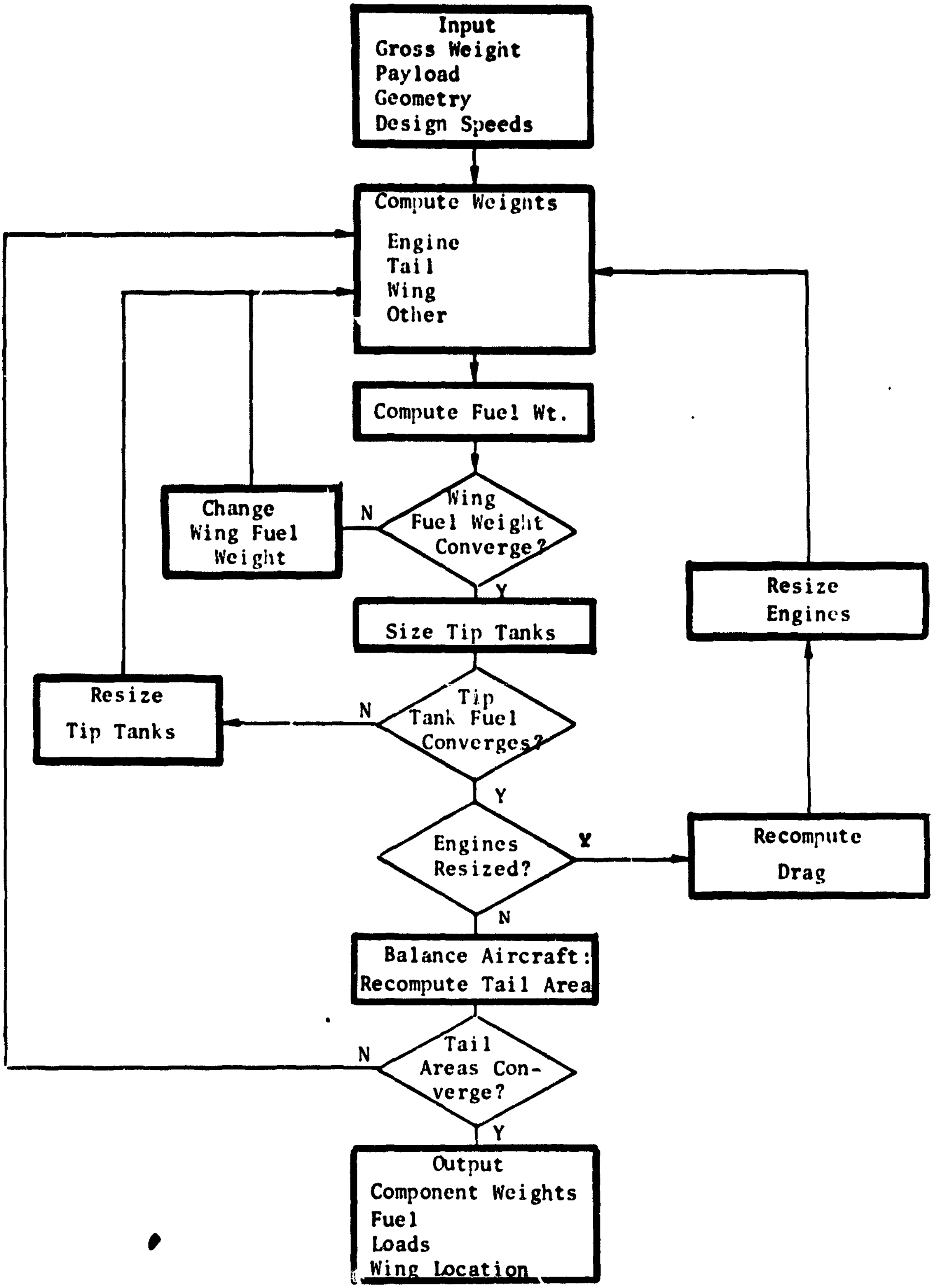


Figure V.1.2 Simplified Flow Chart for Subroutine WGHT

The complexity of the problem is increased by the fact that wing weight depends on fuel weight, and because changes may require the consideration of wing tip tanks, as shown in Figure V.1.2. A secondary loop involves the resizing of the engines, and the third major loop (IF LCWING $\neq$ 0) requires that the aircraft center of gravity location satisfy certain stability conditions. This may mean that the tail moment arm must be changed which requires that the inner loop be recomputed.

The complexity of the problem is further increased by the fact that wing weight depends on fuel weight, and because changes in the wing tip tanks may require the engines to be resized. This nonlinear dependence of the aircraft parameters also shows in the balance option, for which a change in tail area and moment arm also varies the weights of both wing,tails and fuel. As suggested by Figure V.1.2 the following four headings will be discussed here:

1. Component weights,
2. Fuel weight,
3. Tip tank weight, and
4. Center of gravity

The subroutine begins by defining a number of numerical parameters, including fuel density $FUEL_D$ and quarter chord sweep angles of wing and tail surfaces. The lift curve slope of the wing is

$$C_{LALPH} = \Pi * AR/(1. + 1 + \sqrt{C_1 C_2}) \qquad (V.1.44)$$

where C_1 and C_2 are functions of quarter chord sweep, aspect ratio, and cruise Mach number;

$$C_1 = [AR/(2.\ \cos(R_{LMC4}))]^2 \qquad (V.1.45)$$

$$C_2 = 1. - [EM_{CRU}\ \cos(R_{LMC4})]^2 \qquad (V.1.46)$$

The subroutine DLOAD is then called, which returns values for the last six call arguments

V_{DMIN} = minimum design dive speed, kts

U_{LF} = ultimate load factor, g's

G_{LF} = gust load factor, g's

EM_{LF} = maneuver load factor, g's

V_{MO} = maximum allowable operating airspeed, kts

EM_{MO} = maximum allowable operating Mach number

The input quantities to DLOAD are the first five arguments, which are maximum velocity VM_{LFSL}, wing loading W_{OS}, FAR structural design category CAT_D, lift curve slope C_{LALPH}, and wing mac, C_{BARW}, from COMMON/SIZE/.

V.1.2.2.1 Fixed Equipment Weight. Unless input in namelist INGASP as WFEX, the weight of fixed equipment is estimated in pounds as

$$W_{FE} = 61.75 * SEATS^2 - 352.5 * SEATS + 533. \qquad (V.1.47)$$

where

$$SEATS = PAX + 1 \qquad (V.1.48)$$

and PAX is the number of passengers carried by the aircraft. This trend is applicable for two-seat trainer configurations through 8 to 10 place business type aircraft. Figure V.1.3 shows the items that are considered a part of the fixed equipment as well as the breakdown for some current aircraft.

Program modified Feb. 1980 replacing equation V.1.47. If WFEX not input WFE and WFUL (next section, pg. V-1-20) are computed by new subroutine WFIXEU. Figures V.1.3 and V.1.4 are still useful guidelines.

ITEM	AIRCRAFT							
	Cessna 150	Piper Arrow	Cessna 210	Cessna 340	Citation	Learjet	DHC-6	Gulfstream I
Auxilary Propulsion Unit	-	-	-	-	-	-	-	345.5
Instrument & Navigation	5.8	15.4	20.0	61.9	85.7	80.0	70.4	211.7
Hydraulic and Pneumatic	3.3	19.8	51.1	15.3	94.2	106.0	43.3	236.7
Electrical	44.7	49.6	59.5	131.5	321.4	441.0	343.4	1073.7
Avionics	-	-	-	1.1	361.5	300.0	65.0	607.1
Furnishings	45.1	117.4	195.0	244.9	688.0	488.0	785.7	1981.9
Air Conditioning	5.4	-	7.0	142.1	187.8	200.0	243.3	497.2
Anti-Ice	-	-	2.9	3.7	63.0		190.2	284.2
Auxilary Items	-	-	2.5	2.7	2.4	-	-	6.0
Paint	-	13.3	13.6	37.5	36.0	50.0	38.2	5.9
TOTAL (W_{FE})	104.3	215.5	351.6	640.7	1840.0	1665.0	1779.5	5249.9

FIGURE V.1.3 WEIGHT OF FIXED EQUIPMENT

V.1.2.2.2 **Fixed Useful Load.** The weight of fixed useful load, WFUL, must be input in namelist INGASP. Figure V.1.4 shows the items that are usually considered a part of this category as well as the breakdown for some current aircraft.

V.1.2.2.3 **Payload Weight.** The program is set up to determine the range capability of the configuration for three payloads. The three conditions are known as the maximum payload, maximum fuel, and the design payload. These are outlined in the following paragraphs.

The maximum payload is determined from the number of passengers (PAX) and unit weight of passenger and luggage (UW_{PAX}) as input in namelist INGASP; i.e.,

$$W_{PL} = UW_{PAX}\ PAX \tag{V.1.49}$$

where UW_{PAX} = 200 pounds is the typical default value.

The maximum fuel case payload is determined by the total fuel volume capacity of the configuration, constrained by the gross weight or usable fuel volume of the aircraft. The mathematical representation is

$$W_{PL} = W_G - W_{OE} - W_{FMAX} \tag{V.1.50}$$

where W_G = design gross weight, lb

W_{OE} = aircraft operating weight empty, lbs.

W_{FMAX} = maximum fuel weight, lbs.

If W_{PL} is negative, the payload would be set to zero and W_{FMAX} reduced to stay below the gross weight limit.

The last case is the design payload case. The design payload weight is input as W_{PLX} in namelist INGASP. If no value is input, then the maximum

ITEM	AIRCRAFT							
	Cessna 150	Piper Arrow	Cessna 210	Cessna 340	Citation	Learjet	DHC-6	Gulfstream I
Crew & Crew Baggage	200.0	200.0	200.0	200.0	340.0	385.0	400.0	420.0
Trapped/Unusable Liquids	-	-	6.0	41.0	109.9	132.0	35.0	229.0
Engine 0.1	11.3	15.0	22.5	49.0	37.0	15.0	54.0	113.0
Passenger Service Items	-	-	-	-	-	10.0	-	145.0
Optional Equipment	-	30.9	-	247.0	165.3	-	-	-
Total (W_{FUL})	211.3	245.9	228.5	537.0	652.2	542.0	489.0	907.0

FIGURE V.1.4 WEIGHT OF FIXED USEFUL LOAD

payload case previously described becomes the design condition.

V.1.2.2.4 **Propulsion System Installation. Following statement 3, the propulsion system weights are accounted for as**

$$W_{EP} = \text{primary engine weight} = W_{ENG}\ EN_P \tag{V.1.51}$$

$$W_{PROP} = \text{propulsor weight} = EN_P\ W_{PROP_1} \tag{V.1.52}$$

$$W_{PES} = \text{primary engine section structural weight}$$

$$(W_{NAC} + W_{PYLON})\ EN_P \tag{V.1.53}$$

$$W_{PEI} = \text{primary engine installation weight} = SK_{PEI}\ W_{EP} \tag{V.1.54}$$

$$WP_{STAR} = CK_5\ W_{EP} + CK_7\ W_{PEI} + W_{PROP} \tag{V.1.55}$$

where W_{ENG}, W_{PROP_1}, W_{NAC}, W_{PYLON} have been determined by subroutine ENGWGT or input. SK_{PEI} is input in namelist INGASP to account for engine installation hardware. It may be input as zero if it has been accounted for elsewhere. CK_5 and CK_7 are set to 1.0 internally.

If the engine section weight W_{PES} does not have a value at this point, it will be determined by

$$W_{PES} = SK_{PES}\ W_{EP} \tag{V.1.56}$$

where SK_{PES} is input as .338, typically.

V.1.2.2.5 Landing Gear. For the normal tricycle gear geometry, the total landing gear weight including the running gear (wheels, tires, brakes, etc.), structure (shock struts, drag struts, support structure, etc.), and controls (retraction, steering, systems, etc.) is expressed as a percentage of the design gross weight where

$$W_{LG} = (SK_{LG})W_G \tag{V.1.57}$$

where W_{LG} = total weight of the landing gear

$$SL_{LG} = \frac{\text{landing gear}}{\text{gross weight}} \qquad (V.1.58)$$

W_G = design gross weight

The percentage will vary between 0.015 and 0.080 depending on the complexity and design loads of the system. Conventional landing gear with retracting systems, operating on improved runways, normally run between 0.025 and 0.0[illegible]. STOL-type systems operating on rough runways require longer and larger alighting gear components to accomodate the aircraft's higher rotational angle and sink speeds required to operate at the shorter field lengths. SK_{LG} for the STOL aircraft will normally vary between 0.035 and 0.08. The main gear usually weighs about 80 per cent of the total gear weight. The SK term in the weight expression above is the value that is input in namelist INGASP or a default value of 0.0318 is used. The weight fraction of the main gear is determined by inputting SK_{MG} or using the default value of 0.80.

Figures V.1.5 to V.1.7 are included as a guide in selecting SK_{LG}. It includes the total gear weight as a fraction of the gross weight for a sampling of military, commercial, and general aviation aircraft.

V.1.2.2.6 Empennage Weight. Various geometric parameters for the vertical and horizontal tail then follow. The tangent of the leading edge sweep angle of the vertical tail is

$$TSWVLE = \frac{1-SLM_V}{(1+SLM_V)AR_{VT}} + \tan(SWPQC_V) \qquad (V.1.59)$$

where SLM_V = taper ratio of vertical tail

AR_{VT} = aspect ratio of vertical tail

$SWPQC_V$ = quarter chord sweep of vertical tail

Aircraft	Gross Weight (lb.)	Total Gear Weight (lb.)	Fraction of Gross Weight (ΣK_{LG})
Bede 5J	1,000	41	0.041
Cessna 150	1,600	114	0.07125
Cessna 172	2,300	117	0.05087
Piper Arrow	2,600	98	0.0377
Cessna 182	2,800	134	0.0478
Cessna 210	3,800	182	0.0479
Cessna 340	5,975	268	0.0448
Nomad 22*	8,000	400	0.05
Citation	11,650	425	0.0365
DHC-6-330*	12,500	606	0.0485
Merlin IV	12,500	623	0.0498
Learjet 24	13,500	429	0.0318
Fregate	23,810	1,113	0.0467
Gulfstream I	35,100	1,237	0.0352
Jetstar I	40,921	1,081	0.0264

FIGURE V.1.5 GENERAL AVIATION LANDING GEAR WEIGHTS

*STOL-type aircraft

Aircraft	Gross Weight (lb.)	Total Gear Weight (lb.)	Fraction of Gross Weight (SK_{LG})
F-27	35,700	1,884	0.0528
VFW614	44,000	1,620	0.0368
Convair 440	49,100	2,158	0.0439
F-28	65,000	2,649	0.0407
737-200	100,000	4,038	0.0404
DC-9-32	108,000	4,182	0.0387
727-100	161,000	7,211	0.0448
Convair 880	184,500	6,933	0.03758
DC-8-62	335,000	11,449	0.03417
707-320	336,000	12,982	0.0386
DC-10-10	430,000	18,581	0.0432
747	775,000	32,220	0.0416

FIGURE V.1.6 COMMERCIAL LANDING GEAR WEIGHTS

Aircraft	Gross Weight (lb.)	Total Gear Weight (lb.)	Fraction of Gross Weight (SK_{LG})
DHC-4*	28,500	1,398	0.049
Breguet 941*	46,000	2,626	0.0571
DHC-5*	41,000	1,800	0.0439
C-123B	54,000	2,331	0.0432
C-118	107,000	3,895	0.0364
C-130B*	135,000	4,824	0.0357
C-130E*	155,000	5,077	0.0327
C-124C	185,000	11,700	0.0632
C-133A	275,000	10,635	0.0387
C-141A	316,000	10,529	0.0333
C-5A	728,000	37,628	0.0517

FIGURE V.1.7 MILITARY AIRCRAFT LANDING GEAR WEIGHTS

*STOL-type aircraft

The vertical tail root chord and horizontal tail root chord is then expressed as a function of area, span, and taper ratio,

$$CRCL_{VT} = 2\ S_{VT}/[B_{VT}(1 + SLM_V)]$$ (V.1.60)

$$CRCL_{HT} = 2\ S_{HT}/[B_{HT}(1 + SLM_H)]$$

where

B_{VT} = span of vertical tail

S_{VT} = the planform area of vertical tail

B_{HT} = span of horizontal tail

S_{HT} = the planform area of horizontal tail

Similar expressions follow for the chordwise distance from root chord leading edge to mean aerodynamic chord leading edge, XV_{MAC}, and for the distance from aircraft nose to wing center of gravity, which is shown as EL_{WING} in Figure V.1.8

The weights of the horizontal and vertical tails are determined from the weight trend equations developed in Reference 1, and presented below.

Horizontal Tail

$$W_{HT} = 350\ (k)^{0.54}$$ (V.1.61)

where $$k = \frac{F_H\ S_{HT}\ \log(VD_{MIN})}{100 * ELT_H\ TC_{HT}\ CRCL_{HT}}$$ (V.1.62)

and $$F_H = 10^{-6}\ W_G\ SK_Y\ B_{HT}\ SK_{TL}\ (1 + 2 \times SLM_H)/(1 + SLM_H)$$ (V.1.63)

Vertical Tail

$$W_{VT} = 380\ (k)^{0.54}$$ (V.1.64)

where $$k = \frac{(F_V + S_{AH}\ F_H/2)\ S_{VT}\ \log(VD_{MIN})}{100\ ELT_V\ TC_{VT}\ CRCL_{VT}}$$ (V.1.65)

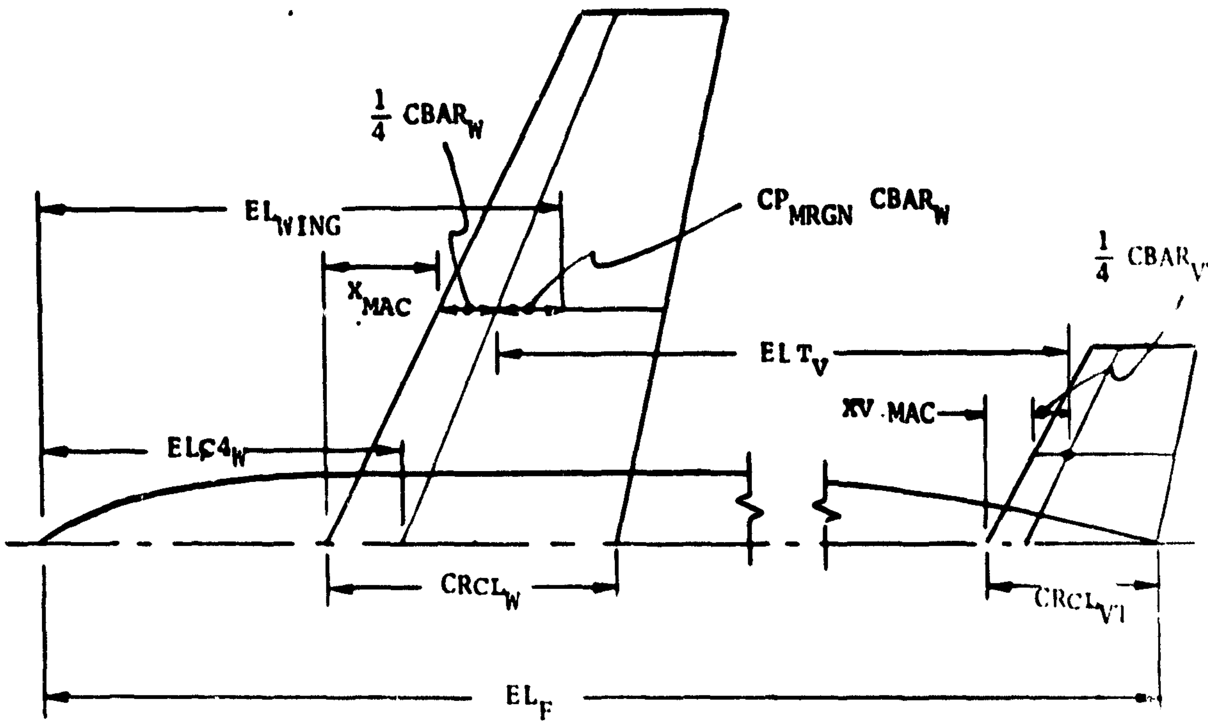


FIGURE V.1.8- Wing-Vertical Tail Geometry and $$F_V = .5 \times 10^{-6} W_G SK_z (EL_F + B) B_{VT} (1 + 2 \times SLM_V)/(1 + SLM_V)$$

(V.1.66)

where the following symbols have been used:

S_{HT} = horizontal tail planform area, sq. ft.

S_{VT} = vertical tail planform area, sq. ft.

VD_{MIN} = dive velocity, knots

S_{AH} = location of horizontal tail on vertical tail (fraction of vertical tail span from root of vertical tail)

ELT_H = horizontal tail moment arm, ft.

ELT_V = vertical tail moment arm, ft.

TC_{HT} = horizontal tail root thickness ratio

TC_{VT} = vertical tail root thickness ratio

$CRCL_{HT}$ = horizontal tail root chord, ft.

$CRCL_{VT}$ = vertical tail root chord, ft.

F_H and F_V = tail load parameters

W_G = design gross weight, lbs.

B_{HT} = horizontal tail span, ft.

B_{VT} = vertical tail span, ft.

B = wing span, ft.

EL_F = fuselage length, ft.

SLM_H = horizontal tail taper ratio

SLM_V = vertical tail taper ratio

SK_Y = horizontal tail weight trend factor (See Figure V.1.9)

SK_{TL} = horizontal tail weight penalty factor (nominally = 1.0)

SK_z = vertical tail weight trend factor (See Figure V.1.10)

The trends consider the tail loads which are a function of the gross weight, span, radii of gyration, and point of load application. The "S_{AH}" term in the vertical tail equation accounts for T- tail configurations. Figures V.1.9 and V.1.10 present the aircraft used to develop the trends. The term "SK_{TL}" is in namelist INGASP and has a default value of 1.0. It is included to provide a means for penalizing the weight of the horizontal tail when additional design loads, such as carrier landings, are a consideration. SK_{TL} would then have a value between 1.1 and 1.2 depending on the magnitude of the design loads.

The terms "SK_y" and "SK_z" are the weight trend factors for the horizontal and vertical tails respectively, as defined in reference 1. These items are related to the pitch radius of gyration for the horizontal tail and the yaw radius of gyration for the vertical tail. Figures V.1.11 and V.1.12 show the definition of these terms in relation to the radii of gyration as shown in the VASCOMP program, Reference 1. This relationship is based on the regression analysis presented in Reference 2. Since this analysis mainly contained a large airplane data base some smaller airplanes were analyzed using these weight trend equations. Figure V.1.13 shows the resulting SK_y and SK_z terms for these smaller airplanes.

V.1.2.2.7 <u>Wing Weight</u>

The wing structural weight trend equation is based on a semi-empirical relationship developed from the approach outlined in Reference 3 and a regression analysis of the 18 aircraft shown in Figure V.1.14.

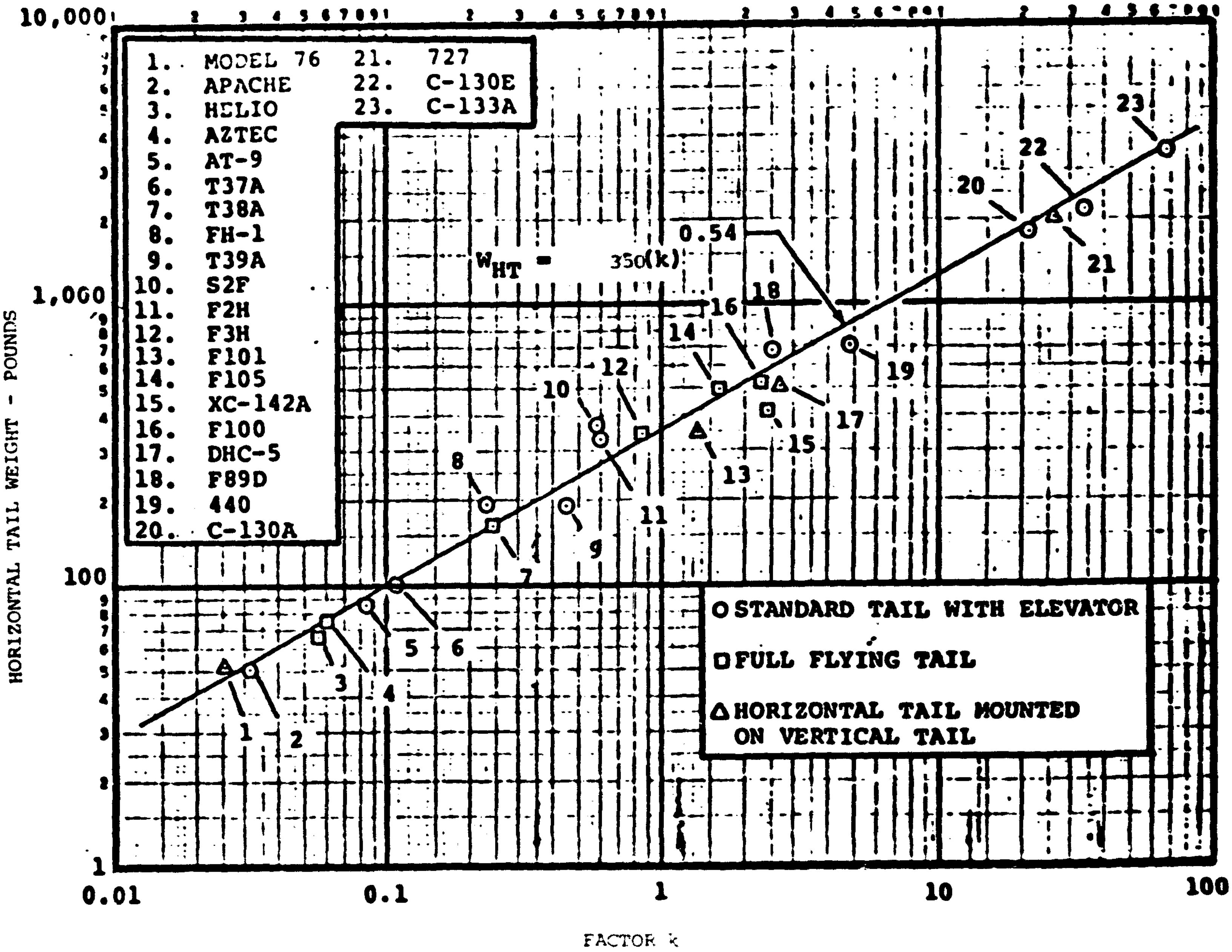


FIGURE V.1.9 - HORIZONTAL TAIL WEIGHT TRENDS

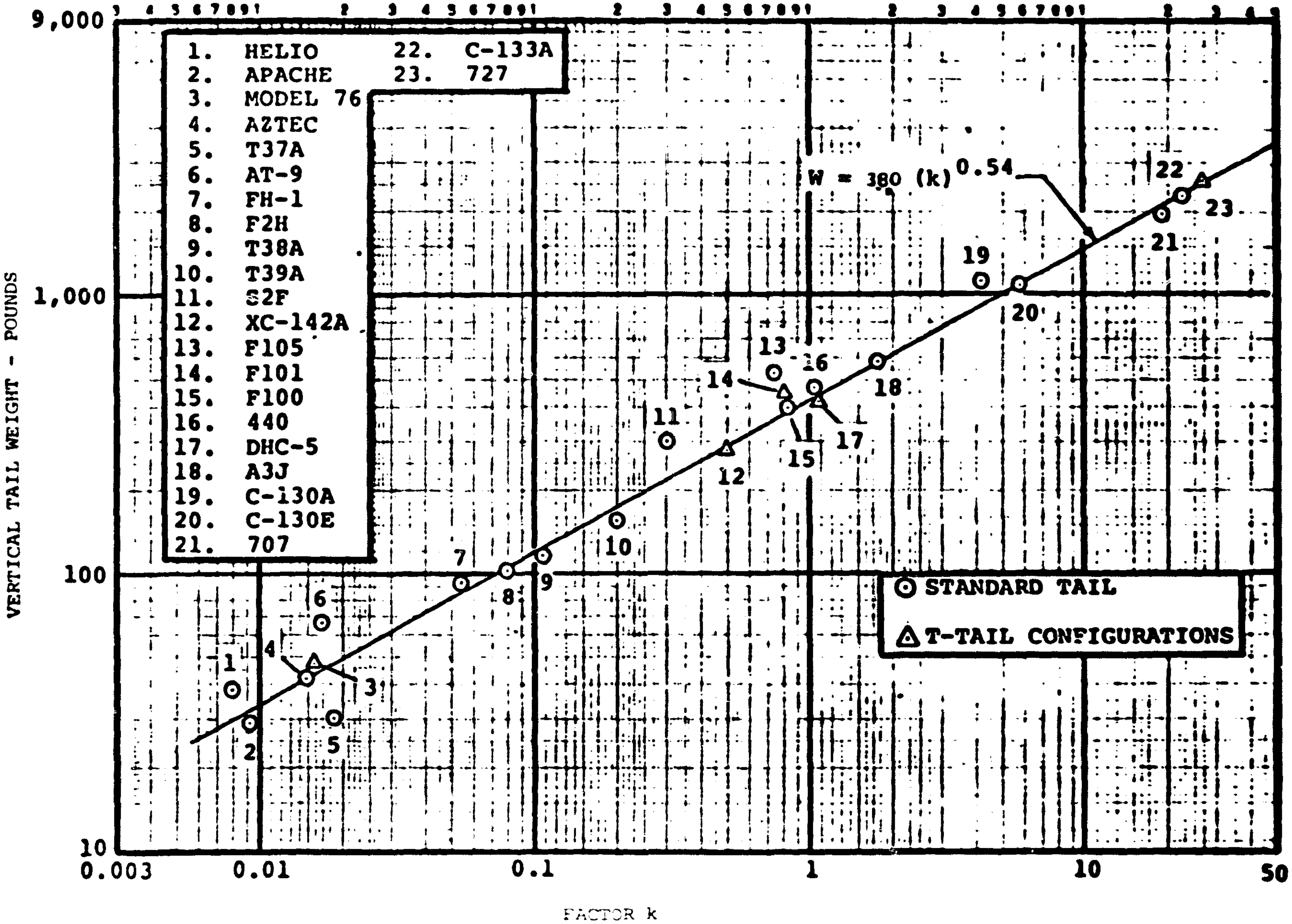


FIGURE V.1.10 - VERTICAL TAIL WEIGHT TRENDS

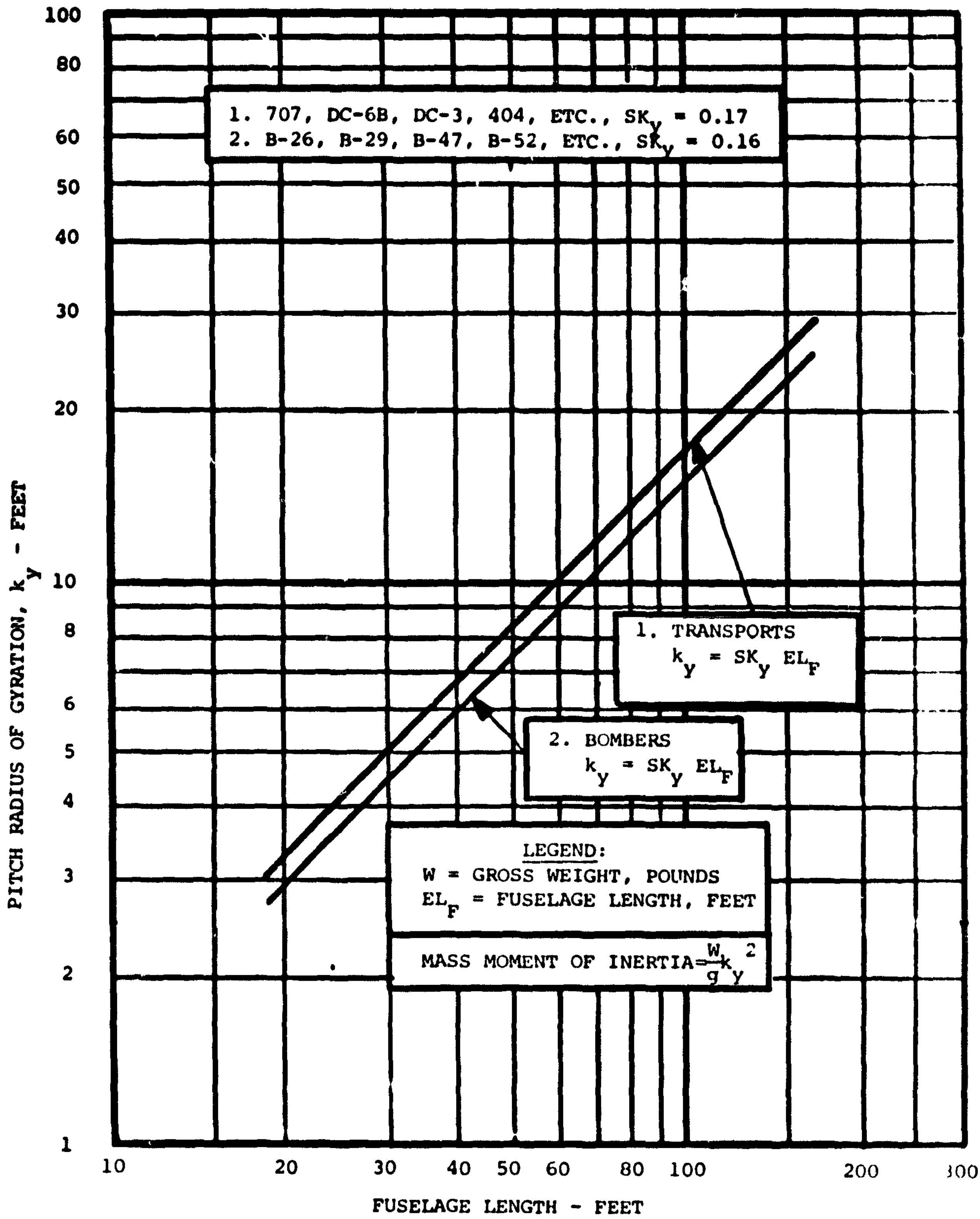


FIGURE V.1.11 - RADIUS OF GYRATION, PITCH.

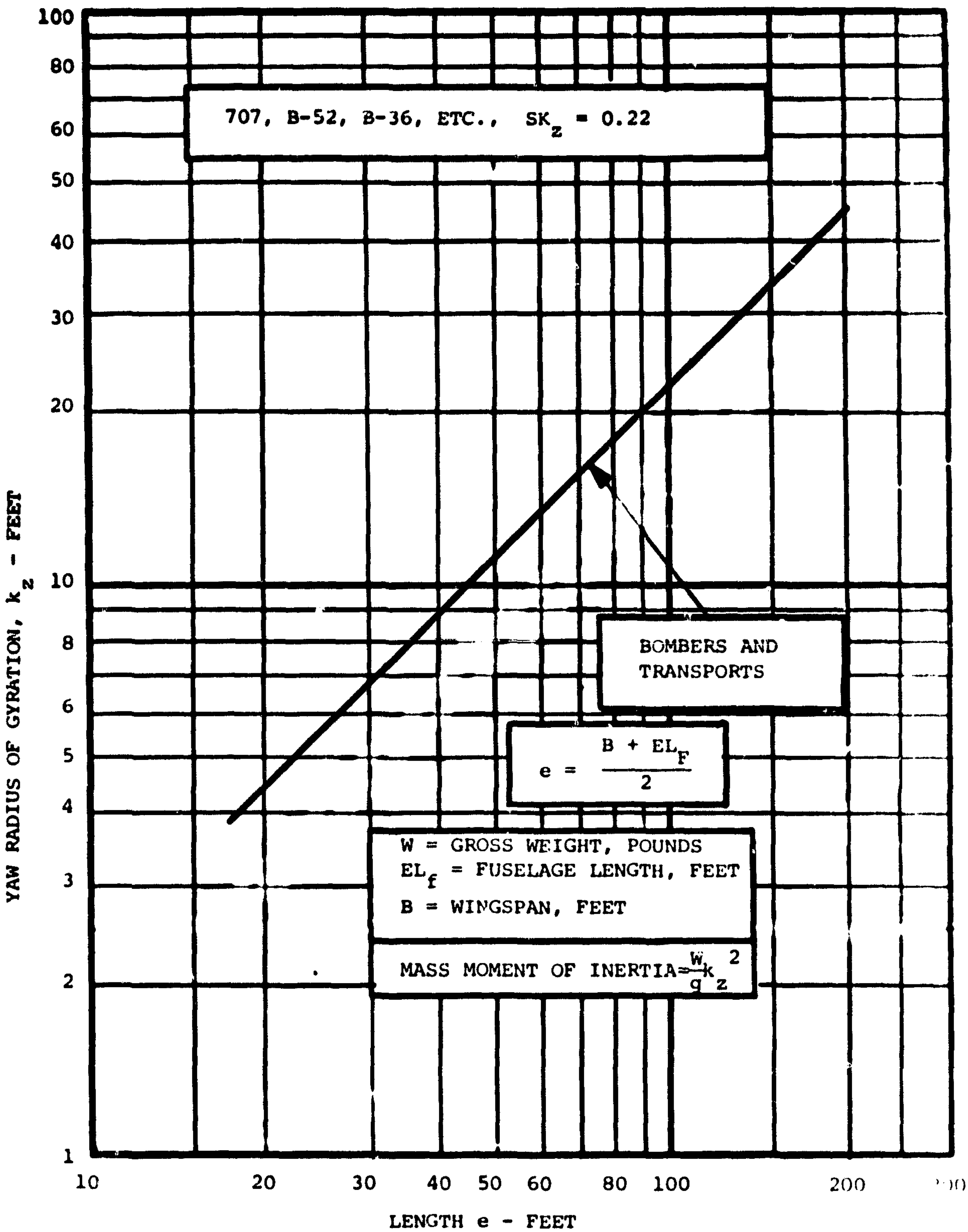


FIGURE V.1.12 - RADIUS OF GYRATION, YAW

Aircraft	SK_y	SK_z
Cessna 150	0.204	0.227
Bede 5J	0.14	0.190
Arrow	0.0747	0.220
Cessna 210	0.115	0.164
Cessna 340	0.063	0.10
Citation	0.102	0.200
Learjet	0.180	0.220
DHC-6	0.110	0.220
Gulfstream I	0.166	0.48

FIGURE V.1.13 FACTORS COMPUTED FOR SMALLER AIRCRAFT

	Aircraft	Gross Weight lbs.	Wing Area ft.	Actual Total Wing Weight lbs.	Predicted Total Wing Weight lbs.	Actual Weight of High Lift Devices** lbs.
1.	Bede 5J	1000	38	90	80.42	4.2
2.	Cessna 150*	1600	157	206	199.88	16.8
3.	Cessna 172*	2300	174	235	246.34	17.4
4.	Piper Arrow	2600	160	273	275.70	15.0
5.	Cessna 182*	2800	174	257	281.98	16.0
6.	Cessna 210	3800	175	375	395.87	22.3
7.	Cessna 340	5975	185	544	530.59	17.2
8.	CITATION	11650	268	1019	1064.61	85.6
9.	DHC-6*	12500	420	1212	1480.17	134.3
10.	Merlin IV	12500	277	1316	1249.80	100
11.	Learjet 24	13500	232	1175	1151.22	170
12.	Fregate	23810	600	2749	1981.87	320
13.	Gulfstream I	35100	610	3643	4162.57	472
14.	F-27	35700	754	4265	4121.93	516
15.	Jetstar I	40921	543	2954	3914.50	467
16.	F-28	65000	822	7135	7027.85	928
17.	737-200	100000	1005	10775	10332.96	2426
18.	DC-9-32	108000	1001	11391	11409.55	2241

*STRUT-BRACED WINGS **includes trailing edge and loading edge devices

FIGURE V.1.14 WING WEIGHT DATA

The method outlined in Reference 3 is based on a generalized expression for the material required to resist the root bending moment due to wing lift in a specified flight condition. The method does not include estimating the weight of the high-lift devices (see subroutine FLAPS) but does account for type of wing, wing relief factors, and landing gear location. The resulting trend equation is of the following form.

$$W_W = \frac{10^{-5} SK_{WW}\ SK_{NO}\ SK_{EPOS}\ SK_{GEAR}\ F_{OO}\ B^{1.049}(1 + SL_M)^{.4}}{TC_R^{\ .4}\ CS_{WC2}^{\ 1.535}} + W_{HLDEV} \qquad \text{(V.1.67)}$$

where

SK_{WW} = wing weight trend factor in namelist INGASP

(default = 133.4) (V.1.68)

SK_{NO} = correction factor for the non-optimum material

$= 1 - 2.5 / \sqrt{B / CS_{WC2}}$ (V.1.69)

SK_{EPOS} = engine position factor from Reference 3

= 1.0 propeller aircraft with no wing-mounted engines

= 0.98 propeller aircraft with 2 wing-mounted engines

= 0.95 propeller aircraft with 4 wing-mounted engines or high-subsonic jet aircraft with 2 wing-mounted engines

= 0.90 high-subsonic jet aircraft with 4 wing-mounted engines

= 1.05 high-subsonic jet aircraft with no wing-mounted engines

SK_{GEAR} = landing gear location factor from Reference 3

= 1.0 for wing-mounted landing gear (V.1.70)

= 0.95 for landing gear not mounted on wing

F_{OO} = wing loading parameter

$$= \left[SK_{STR}\ U_{LF} \left(W_G - .8\ W_{W_1} \right) \right]^{.757} \quad (V.1.71)$$

SK_{STR} = reduction in bending moment factor for street braced wing from Reference 3

$$= 1 - (STRUT)^2 \quad (V.1.72)$$

STRUT = ratio of spanwise location of wing strut to wing semi-span (equals 0 for cantilever wing)

U_{LF} = ultimate design load factor, g's

W_G = design gross weight, lbs.

W_{W_1} = wing weight, lbs

B = wing span, ft.

SL_M = wing taper ratio

TC_R = wing root thickness ratio

CS_{WC2} = cosine of wing half-chord sweeps angle

W_{HLDEV} = weight of wing high-lift devices, lbs. (computed in subroutine FLAPS)

Figure V.1.14 compares the predicted and actual wing weights for the 18 aircraft used in the regression analysis to determine the default value of SK_{WW} and the exponents on span, taper ratio, thickness, and sweep angle terms.

V.1.2.2.8 Fuselage Weight

The weight of the fuselage structure is determined from the weight trend equation developed in Reference 1 and presented below.

$$W_B = SK_B\ (k)^{0.508} + W_{BOOM} \quad (V.1.73)$$

where

$$k = \left(10^{-4} W_X\right)^{.7} \left(10^{-3} S_F\right) S_{WF} \left(EL_{FFC} + EL_{RW}\right)^{.5} \log\left(VD_{MIN}\right) \left(DEL_P + 1\right)^{.2} U_{LF}^{.3} \quad (V.1.74)$$

and

$$W_{BOOM} = 10^{-3} SK_{BM} EL_{BM} \sqrt{XAR_{BM}} \; W_G \quad (V.1.75)$$

LEGEND:

SK_B = fuselage weight trend factor in namelist INGASP (default = 136)

W_X = weight of fuselage and contents (includes empennage), lbs.

S_F = wetted area of fuselage, ft.^{2}

S_{WF} = fuselage width, ft.

EL_{FFC} = length of fuselage, ft.

EL_{RW} = length of pylon for fuselage mounted engines, ft.

VD_{MIN} = dive speed, kts

DEL_P = limit differential cabin pressure, psi

U_{LF} = ultimate load factor

W_{BOOM} = weight of fuselage tail boom, lbs. (KCONFG = 1)

SK_{BM} = tail boom weight trend factor in namelist INGASP

EL_{BM} = length of tail boom, ft.

XAR_{BM} = cross-sectional area of tail boom, ft.

W_G = aircraft design gross weight, lbs.

The above trend equation was developed using a number of commercial, military, and cargo aircraft. Figure V.1.15 indicates the relative body weight variation between these different families of aircraft. A mean line with an SK_B value of 124 is shown to represent the average of all the aircraft shown in Figure V.1.15. Also shown is a body adjustment

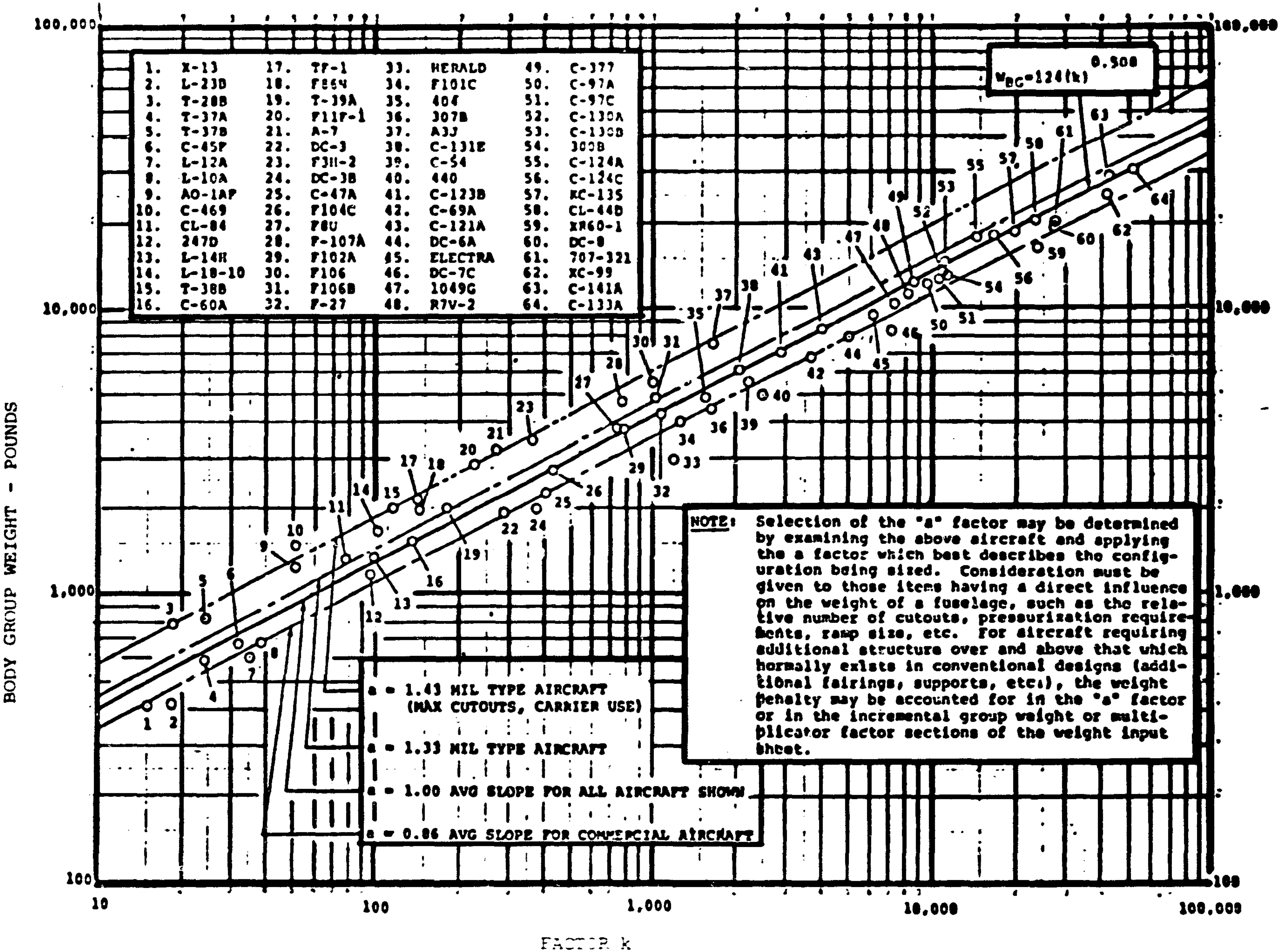


FIGURE V.1.15 - BODY GROUP WEIGHT TRENDS factor "a" which corrects the 124 constant in accordance with the specific family being considered. The revised constant, 124 "a", is the SK_B term to be input in namelist INGASP. The limit differential cabin pressure (DEL_p) is input also in INGASP or the program calculates the appropriate value to provide an 8000 ft. cabin altitude at the design cruise altitude. Since the analysis of Reference 1 mainly contained a large airplane data base some smaller airplanes were analyzed using the weight trend equation above. Figure V.1.16 shows the resulting SK_B for these smaller airplanes.

V.1.2.2.9 Flight Controls Group Weights

The weight of the total flight controls group is estimated first and then the weight of the cockpit controls component. The difference between these two weights is considered the fixed wing surface control portion of the flight controls.

The weight of the total flight controls group is estimated from the following trend equation which is based on a number of military and commercial aircraft.

$$W_{SCX} = SK_{FW} \, S_W^{\,.317} \, (10^{-3} \, W_G)^{.602} \, (U_{LF})^{.525} \, (Q_{DIV})^{.345} \qquad \text{(V.1.76)}$$

where

W_{SCX} = weight of total flight controls group, lbs.

SK_{FW} = flight controls weight trend factor in namelist INGASP (default = 0.404)

S_W = wing planform area, ft.^{2}

W_G = aircraft design gross weight, lbs.

U_{LF} = design ultimate load factor, g's

Q_{DIV} = dynamic pressure at design dive speed, lbs./ft.^{2}

Aircraft	Pressurized	SK_B
Cessna 150	No	130.0
Bede 5J	No	160.0
Arrow	No	128.0
Cessna 210	No	161.5
Cessna 340	Yes	101.0
Citation	Yes	87.0
Learjet	Yes	136.0
DHC-6	No	128.5
Gulfstream I	Yes	128.5

FIGURE V.1.16 COMPUTED FUSELAGE WEIGHT FACTORS

The cockpit controls weight is estimated as the following function of gross weight

$$W_{CC} = SK_{CC}\ (10^{-3}W_G)^{.41} \qquad (V.1.77)$$

where SK_{cc} = cockpit controls weight trend factor in namelist INGASP (default = 11.0)

The fixed wing surface controls portion is then the difference

$$WC_{FW} = W_{SCX} - W_{CC} \qquad (V.1.78)$$

Figure V.1.17 indicates some typical values of SK_{FW} and SK_{CC} for a selected number of small airplanes.

The sum of the aircraft flight control component weights is

$$W_{FC} = CK_{15}\ W_{CC} + CK_{18}\ WC_{FW} + CK_{19}\ W_{SAS} + DELW_{FC} \qquad (V.1.79)$$

where

$W_{SAS} = SK_{SAS}$ = weight of stability augmentation system input in namelist INGASP (default = 0.), lbs.

$DELW_{FC}$ = incremental flight controls weight, lbs., (this may be input in namelist INGASP)

and where the constants CK_I are currently the value of unity. If the flight controls weight trends are to be by-passed the user can input SK_{FW}, SK_{CC}, SK_{SAS} as zero and input the total flight controls weight as $DELW_{FC}$.

V.1.2.2.10 Optional Structural Weight Computation Bypass

If the structural weight trends are to be by-passed the user can input SK_{WW}, SK_y, SK_z, SK_B, SK_{LG}, SK_{PES} as zero and input the total structural weight as $DELW_{ST}$.

Aircraft	SK_{FW}	SK_{CC}
Cessna 150	0.485	11.5
Bede 5J	0.133	3.0
Arrow	0.383	13.5
Cessna 210	0.322	8.7
Cessna 340	0.350	9.8
Citation	0.430	20.6
Learjet	0.404	11.0
DHC-6	0.430	20.0
Gulfstream I	0.430	20.0

FIGURE V.1.17 COMPUTED FLIGHT CONTROLS WEIGHT FACTORS where

$DELW_{ST}$ = incremental structural weight, lbs. (this may be input in namelist INGASP)

V.1.2.2.11 Fuel System and Fuel Weight

The fuel system weight depends on the amount of fuel available (WFADES) for the design payload condition of the airplane. The design payload is input as WPLX in namelist INGASP and is identified as WPLDES in subroutine WGHT. If no value is input than WPLDES is a function of the number of passengers (PAX) and their unit weight (UMPAX).

The weight of the fuel system is basically a percentage of the fuel volume at the design payload condition as expressed by

$$W_{FSS} = CK_{21}\ (6.687/FUEL_D)\ SK_{FS}\ W_{FADES} \qquad (V.1.80)$$

Since the design fuel load is determined by subtracting the component weights and payload from the gross weight, the design fuel load is expressed as

$$W_{FADES} = (W_G - W_{PSTAR} - W_{ST} - W_{FC} - W_{FE} - W_{FUL} - W_{PLDES})/D \qquad (V.1.81)$$

where

$$D = 1 + CK_{21}\ SK_{FS}\ 6.687/FUEL_D \qquad (V.1.82)$$

W_{FUL} = fixed useful load weight

$FUEL_D$ = fuel density in lb. per gal. (program sets it to 6.687 for jet fuel and 6.0 for gasoline)

CK_{21} = weight multiplication factor for fuel system, currently set to 1.0 in program

SK_{FS} = .0195, default value of weight coefficient for fuel system

The value of SK_{FS} will vary depending on the capacity, type, and complexity of the system required. For commercial aircraft having simple fuel systems in the wing, the value for SK_{FS} would range between 0.02 and 0.07; for aircraft requiring self-sealing tanks with more complex systems, the value would range between 0.10 and 0.15. Figure V.1.18 contains the value of SK_{FS} for some general aviation aircraft.

The total propulsion weight is then computed as

$$W_P = W_{PSTAR} + W_{FSS} \tag{V.1.83}$$

where the first term has [illegible]n defined in Eq. (V.1.55) and the second is the weight of the fue[illegible] [illegible]ly system from equation V.1.80.

In addition to the design fuel load, two other fuel loading conditions are calculated for determining the payload-range calculations performed in XRANGE. One is the minimum fuel condition (WFAMIN) and the other the maximum fuel case (WFAMAX).

$$W_{FAMIN} = W_G - W_{PSTAR} - W_{ST} - W_{FC} - W_{FE} - W_{FUL} - W_{FSS} - W_{PLMAX} \tag{V.1.84}$$

The W_{FAMAX} is equal to either W_{FADES} or the fuel available for filling all the available volume without exceeding the gross weight.

The next group of statements deal with determining the available wing fuel volume and weight. The wing tank fuel volume is found in cu. ft. as

$$F_{VOLW} = \frac{.8889\ SK_{WF}\ TC\ S_W^{1.5}\ (2.*SL_M + 1)}{\sqrt{AR}\ \ (SL_M + 1)^2} \tag{V.1.85}$$

where TC is the average wing thickness to chord ratio, calculated on the previous line. SK_{WF} is a factor that relates wing volume available for

Aircraft	SK_{FS}
Bede 5J	.050
CE 150	.119
Piper Arrow	.138
CE 210	.0394
CE 340	.0415
CE Citation	.052
Twin Otter	.060
Learjet	.020
Gulfstream I	.0143

FIGURE V.1.18 CALCULATED FUEL SYSTEM WEIGHT COEFFICIENTS fuel to total theoretical wing volume. If SK_{WF} is input as zero than no fuel will be carried in the wing. For completely wet wings SK_{WF} is on the order of .43 to .45. Figure V.1.19 shows the value of SK_{WF} for some general aviation aircraft. The default value of SK_{WF} is .43.

The logic between statements 38 and 60 sets up an iteration to converge on a wing weight since fuel weight can be considered as a relief load factor.

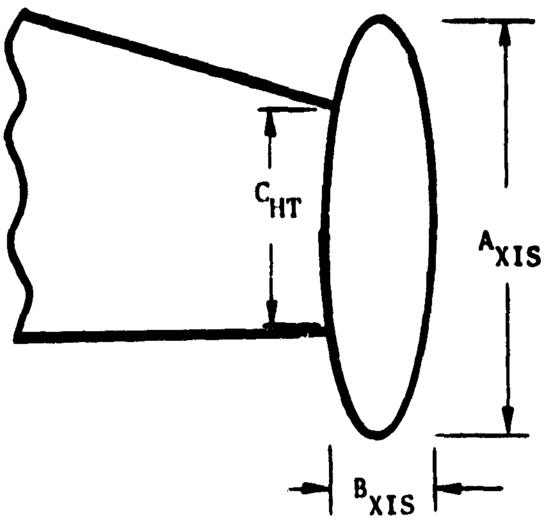
V.1.2.3 Tip Tank Sizing and Weight

After the wing weight iteration has converged, and if the maximum available fuel is more than the wing fuel capacity ($W_{FADES} > W_{FWMX}$), the tip-tank sizing calculations will start if that option has been selected (KTIP = 1). The tip-tank calculations determine the size of two tip tanks containing the excess fuel.

The tip tanks are simulated as prolate spheriods having a major axis, A_{XIS}, and minor axis, B_{XIS}. The dimensions of these axes must satisfy the constraints shown in V.1.20. The tip-tank sizing starts by sizing a tank based on the minimum tank length and growing the length by increments of .25 ft. until all the extra fuel is accomodated or the maximum allowable tank size is reached. Constraints are put on tank length, diameter, and fineness ratio (L/D). The six constraints are illustrated in Figure V.1.21, in which the length, diameter and volume vary as indicated. For $R_1 > L_1/D_2$ and $R_2 < L_2/D_1$, (as shown), two of the extreme corners of the rectangle cannot be used. Notice that the locus of fixed tank volume may not contact <u>any</u> of the four limiting values of length or diameter (locus V*).

Aircraft	SK_{WF}	Remarks
Bede 5J	0.43	wet wing
CE 150	0.0762	only uses portion of available volume
Arrow	0.0783	only uses portion of available volume
CE 340	0.114	majority of fuel carried in tip tanks
Twin Otter	0.	Fuselage fuel tanks
Learjet	0.43	wet wing plus tip tanks
Gulfstream I	0.3822	wet wing

FIGURE V.1.19 CALCULATED WING FUEL VOLUME COEFFICIENTS



$$A_{XMIN} = C_{HT}$$
$$A_{XMAX} = C_{HT}\ AT_{MXQC}$$
$$B_{XMIN} = TC_T\ C_{HT}$$
$$B_{XMAX} = A_{XMAX}/LD_{CKMX}$$

Figure V.1.20 - Tip Tank Geometry

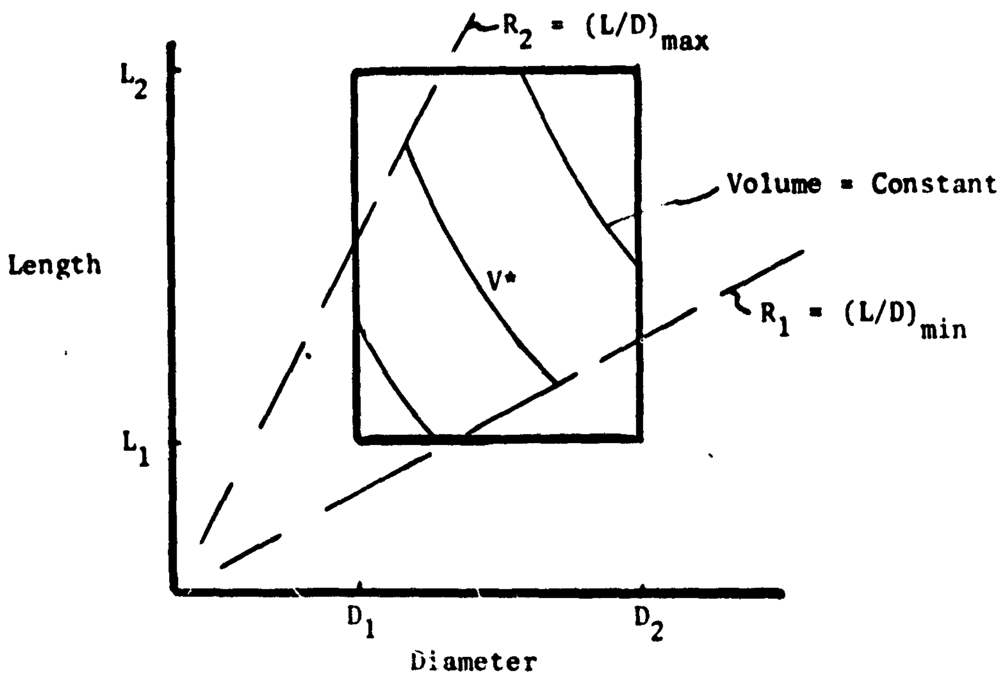


Figure V.1.21 - Geometric Constraints of Tank

The tip tank volume in cu.ft. is determined from the tip tank fuel requirement W_{FTP} and the fuel density, $FUEL_D$.

$$V_{FTP} = W_{FTP} / (7.4805 * FUEL_D) \qquad (V.1.87)$$

The wing root chord is derived from the mean chord ($CBAR_W$) and the taper ratio (SL_M) as

$$CH_R = \frac{3}{2} CBAR_W / \left[1 + SL_M - SL_M / (1 + SL_M)\right] \qquad (V.1.88)$$

and the tip chord is

$$CH_T = SL_M \; CH_R \qquad (V.1.89)$$

Limiting values for tip tank dimensions are defined, in terms of which the minimum and maximum values of tank volume are given by

$$V_{FTPMN} = SK_{FT} \; (1.046) \; AX_{MIN} \; BX^2_{MIN} \qquad (V.1.90)$$

and

$$V_{FTPMX} = SK_{FT} \; (1.046) \; AX_{MAX} \; BX^2_{MAX} \qquad (V.1.91)$$

In these expressions for volume of a prolate spheroid, the numerical factor is 2. (π/6.) to account for <u>both</u> tanks, and the factor SK_{FT} is input at .979 to account for volume of the tank that is not structure. The tank diameter needed for minimum length is found, if the volume exceeds the minimum volume, as by inverting the volume equation; i.e.,

$$B_{XIS} = \left[.956 \; V_{FTP} / (AX_{IS} * SK_{FT})\right]^{1/2} \qquad (V.1.92)$$

If this is within bounds, control is transferred to statement 30, where succeeding calculations include an equation for total surface area of the tip tanks. For two prolate spheroid tanks, the total area is found as

$$S_{TIP} = \Pi \left[B_{XIS}^2 + A_{XIS} B_{XIS} \sin^{-1} (E_{XIS})/E_{XIS} \right] - 2.\ \Pi * AX_{MIN} BX_{MIN}/4. \tag{V.1.93}$$

where

$$E_{XIS} = \left[1 - (B_{XIS}/A_{XIS})^2 \right]^{1/2} \tag{V.1.94}$$

E_{XIS} is the eccentricity of the ellipse of revolution. The last term in equation V.1.93 approximates the area of the wing tip as that of an ellipse with axes AX_{MIN} and BX_{MIN}, and it is subtracted because it is not part of the wetted area of the tip tank.

The empty weight of the tanks is proportional to the surface area

$$W_{TIP} = SK_{WTP} S_{TIP} \tag{V.1.95}$$

where $SK_{WTP} = 1.89$ is input as the weight in lb. per sq. ft. of the tank. The total weight of tip tank structure and fuel is

$$W_C = W_{TIP} + W_{FTP} \tag{V.1.96}$$

and the remaining equations in this portion of the program are concerned with the iterative tip tank parameters contained between statements numbered 5 and 80.

V.1.3 Aircraft Balance

During the task of concept formulation and aircraft definition, aircraft balance can be a major concern. This problem does not stand by itself. It reaches into other aspects of aircraft design such as stability, control, performance, structure weight, engine location, payload requirements, and cost. Two balance options are currently provided for in GASP. The first (LCWING = 1) is a simplified method and the second (LCWING = 2) goes into more detail stability, control, and loading requirements.

V.1.3.1 Simplified Balance Option

The simplified method is based on the following conditions:

(a) Horizontal tail volume coefficient is specified and held constant; i.e., the horizontal tail area varies inversely to the tail length.

(b) First-order values for wing-plus-body aerodynamic center, horizontal tail contribution, and static margin must be known.

(c) The airplane shall be balanced with its center of gravity on a certain point on the wing mean aerodynamic chord.

Following the tip tank calculations, the program deals with the relocation of the wing if the balance option flag is $LC_{WING} = 1$ or 2. This portion begins with the computation of remaining weight, W_{REMN}, the sum of fuselage, payload, fixed equipment and controls weights. This is found by subtracting many component weights from the gross weight, or

$$W_{REMN} = W_G - W_{HT} - W_{VT} - W_W - W_{PSTAR} - W_{FW} - W_{FTP} - W_{TIP} - W_{PES} - W_{BOOM} - CK_{21}\, W_{FSS} - XW_{LG} \qquad (V.1.97)$$

This is followed by a sequence of geometrically derivable quantities, related to wing, horizontal tail and vertical tail. For the wing, these include the wing centerline chord,

$$CR_{CLW} = 2.\ S_W / \left[B\ (1 + SL_M) \right] \qquad (V.1.98)$$

the tangent of the leading edge sweep angle,

$$T_{SWPLE} = TAN\ (RLM_{C4}) + (1 - SL_M) / \left[AR\ (1 + SL_M) \right] \qquad (V.1.99)$$

the mean aerodynamic chord,

$$X_{MAC} = B * T_{SWPLE} (1 + 2SL_M)/ \left[6. (1 + SL_M)\right] \qquad (V.1.100)$$

and the distance from the aircraft nose to the wing centerline quarter chord,

$$EL_{C4W} = EL_F - CR_{CLVT} + XV_{MAC} - EL_{TV} - X_{MAC} + .25 \left[CBAR_{VT} - CBAR_W + CR_{CLW}\right] \qquad (V.1.101)$$

where SL_M is the wing taper ratio, and where the various distances in V.1.101 are shown in Figure V.1.22. The root chord and mean aerodynamic chord are also developed for the vertical and horizontal tail surfaces, using analogous equations and symbols. Following these are expressions for the tip chord, which duplicates the equation found in Eq. (V.1.89)

$$CH_T = \frac{3}{2} SL_M \; CBAR_W/ \left[1 + SL_M - SL_M/ (1 + SL_M)\right] \qquad (V.1.102)$$

and for the distance from the aircraft nose to the half chord of the wingtip. This latter distance is given the name

$$EL_{TIP} = EL_{C4W} - .25CR_{CLW} + X_{MAC} + .25CBAR_W \qquad (V.1.103)$$

If the wing tip tanks are centered here, their trailing edge occurs at the distance

$$EL_{TIPE} = EL_{TIP} + .5\, A_{XIS}, \qquad (V.1.104)$$

aft of the nose. The wing tip trailing edge is located at

$$EL_{TPMX} = EL_{C4W} + .5 * B * TAN\ (RLM_{C4}) + .75\ C_{HT}, \qquad (V.1.105)$$

and if $EL_{TIPE} < EL_{TPMX}$, as occurs for a swept-back wing, the centers of the tip tanks are located 1/2 tank length ahead of the tip trailing edge.

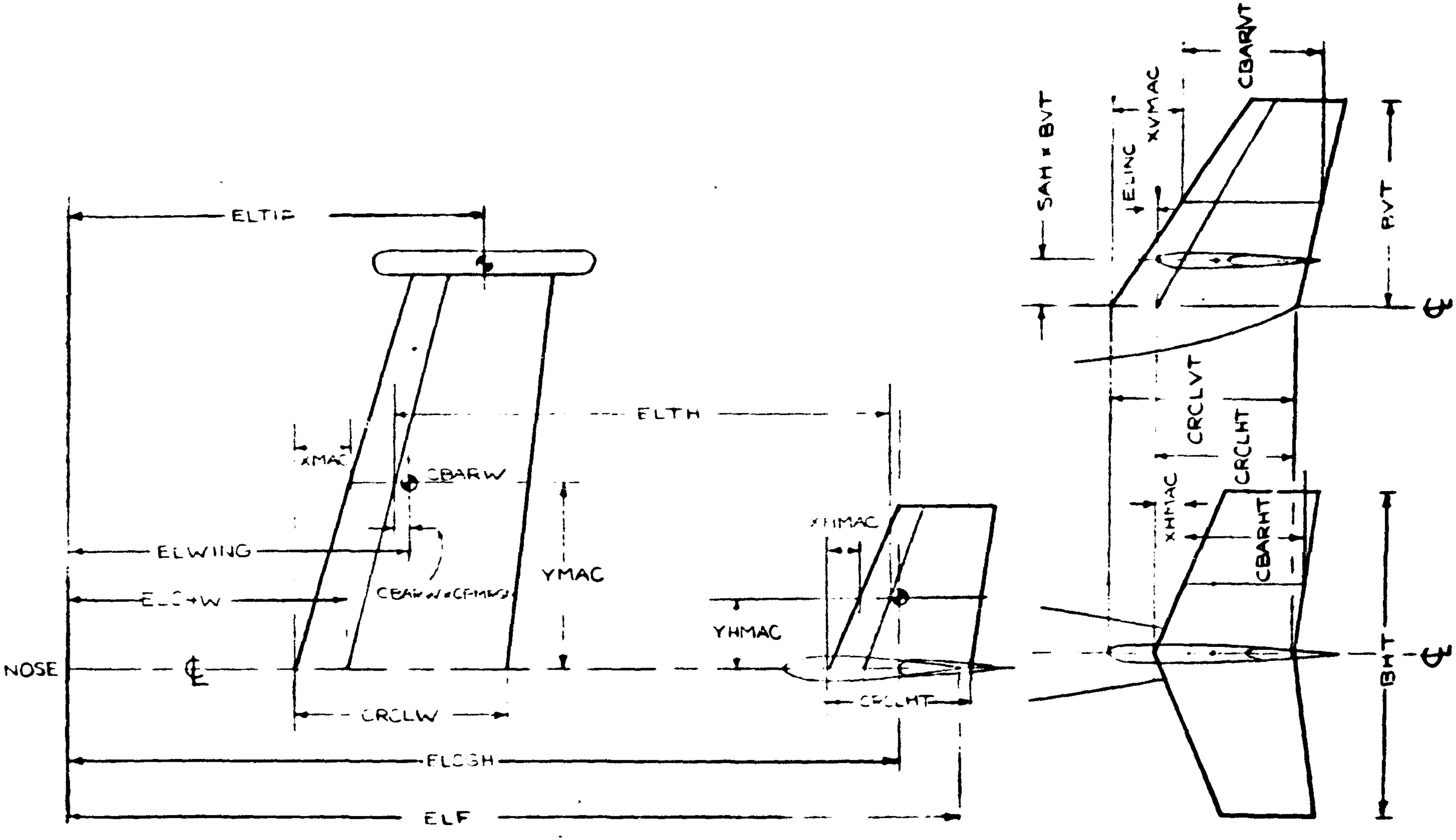


FIGURE V.1.22 - WING LOCATION GEOMETRY

The following computations deal with the moment arms from aircraft nose to the estimated center of gravity of vertical and horizontal tails,

$$EL_{CGV} = EL_{C4W} - .25CR_{CLW} + X_{MAC} + .25CBAR_{W} + EL_{TV} + .08CBAR_{VT} \quad (V.1.106)$$

$$EL_{CGH} = EL_{CGV} - X_{VMAC} - .25\ (CBAR_{VT} - CRCL_{VT}) + X_{HMAC}$$

$$+ .25\ (CBAR_{HT} - CRCL_{HT}) + EL_{INC} + SA_{H}\ B_{VT}\ \tan\ (SWPQC_{V}) \quad (V.1.107)$$

where the last term in Eq. V.1.106 assumes the c.g. of the vertical tail is at 8% aft of the mac 1/4 chord point for vertical tail. The other undefined quantities in this equation are

SA_H = vertical location of horizontal tail on vertical tail, $0 \leq SA_H \leq 1$.

B_{VT} = span of vertical tail

EL_{INC} = aft displacement of horizontal tail root chord leading edge relative to vertical tail root chord leading edge

The program follows with equations for the spanwise locations of the mean aerodynamic chord of the wing,

$$Y_{MAC} = B\ (1 + 2SL_M)/\left[6\ (1 + SL_M)\right] \quad (V.1.108)$$

and a similar expression for the horizontal tail mac location, where SL_M is the taper ratio of the wing.

For the simplified option ($LC_{WING} = 1$) the program now passes to statement 100 where the aircraft center of gravity location aft of the nose is then expressed as,

$$EL_{CG} = \Big[EL_{PROP} \left(W_{PSTAR} + W_{PES} \right) + CBAR_W \left(CP_{MRGN} - ST_{MRGN} \right)$$

$$\left(W_W + W_{FW} + XW_{LG} + CK_{21} W_{FSS} \right) + EL_{TIP} \left(W_{TIP} + W_{FTP} \right) \qquad (V.1.109)$$

$$+ EL_{CGH} W_{HT} + EL_{CGV} W_{VT} + EL_{REMN} W_{REMN} + W_{BOOM}$$

$$\left(EL_F - .5EL_{EM} \right) \Big] / \left(W_G - W_W - W_{FW} - XW_{LG} - CK_{21} W_{FSS} \right)$$

In terms of this distance, the wing center of gravity is the following distance aft of the nose,

$$EL_{WING} = EL_{CG} + CBAR_W \left(CP_{MRGN} - ST_{MRGN} \right) \qquad (V.1.110)$$

where

CP_{MRGN} = .10, location of wing CG in relation to the quarter cord of the wing mac

ST_{MRGN} = 0., location of aircraft CG in relation to the quarter cord of the wing mac

The distance from nose to the quarter chord of the wing mac is

$$EL_{CP} = EL_{CG} - ST_{MRGN} \; CBAR_W \qquad (V.1.111)$$

This equation (Statement 102) is also the point in the program where control returns after the more detail balance option. The next dozen equations are related to parameters which are descriptive of the horizontal and vertical tail. These include the moment arms for both tail surfaces, and these equations can be derived from study of Figure V.1.23.

$$EL_{TV} = EL_F - EL_{WING} - CR_{CLVT} + XV_{MAC} + .25 \; CBAR_{VT} \qquad (V.1.112)$$

$$+ CP_{MRGN} \; CBAR_W$$

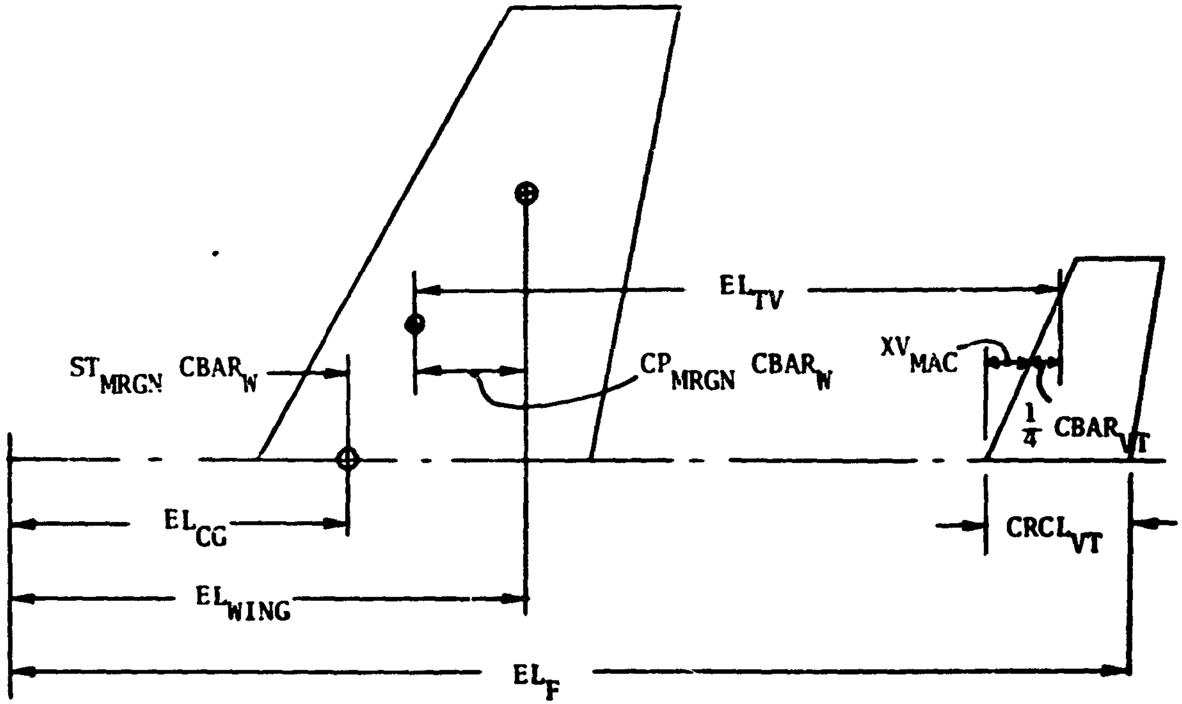


FIGURE V.1.23 - VERTICAL/HORIZONTAL TAIL PARAMETERS

$$EL_{TH} = EL_{TV} - XV_{MAC} - .25\ (CBAR_{VT} + CRCL_{VT} - CBAR_{HT}) \qquad (V.1.113)$$

$$+ XH_{MAC} + EL_{INC} + SA_H\ B_{VT} \tan\ (SWP_{QCV})$$

The areas of the horizontal and vertical tail surfaces are then recalculated

$$S_{HT} = VBAR_H\ S_W\ CBAR_W/EL_{TH} \qquad (V.1.114)$$

$$S_{VT} = VBAR_V\ S_W\ B/EL_{TV} \qquad (V.1.115)$$

where the parameters $VBAR_H$ and $VBAR_V$ are dimensionless tail volume coefficients. Other geometric calculations are concerned with span, root chord and mean aerodynamic chord of both tail surfaces; using straightforward geometric equations which have already been discussed.

The criterion for convergence in the tail sizing is that the difference in successive horizontal tail areas be less than .4 sq.ft. Otherwise, control returns to statement 4, where the tail sizing loop begins.

At statement 108, the cruise altitude, Mach number and weight are redefined, and subroutine CTAER is called to get the drag of the resized configuration. The aircraft center of gravity locaion on the wing mac is

$$G_{CGMAC} = \left[EL_{CG} - \dot{E}L_{CP} + .25\ CBAR_W\right] / \ CBAR_W \qquad (V.1.116)$$

Following a number of written statements regarding geometrical parameters, the subroutine then computes:

(a) leading edge sweep angles of wing, horizontal tail and vertical tail, in degrees, SWP_{LED}, SWH_{LED}, SWV_{LED},

(b) tip chords of horizontal and vertical tails CH_{TH}, CH_{TV}, after which 28 more geometrical parameters are printed.

The operating weight empty is determined as the sum of the weights of the propulsion group, flight controls, structures, fixed equipment and fixed useful load is then given by

$$O_{WE} = W_P + W_{FC} + W_{ST} + W_{FE} + W_{FUL} \tag{V.1.117}$$

and if the sum of this and the maximum available fuel weight exceeds the gross weight W_G, the maximum available fuel weight is redefined as

$$W_{FAMAX} = W_G - O_{WE} \tag{V.1.118}$$

Now, if $I_{DG} > 0$, or $K_{TIP} = 0$ or $NL_{WGHT} = 2$, the program returns. These flags correspond to no tip tanks being added or engine resized, in which case the drag and engine size do not require recomputation. Otherwise, the subroutines CTAER, ENGSZ and ENGWT are called prior to the second iteration of the loop beginning with the minimum wing fuel weight.

V.1.3.2 Detail Balance Option

The detail balance option tries to position the wing so that the c.g. travel from loading considerations falling within the allowable c.g. range from stability and control considerations. The logic for this option is as follows:

(1) The wing is initially positioned based on input or default tail volume coefficients and forward and aft stability limits are assumed.

(2) Based on c.g. limits the most forward and aft c.g. positions from a loading standpoint as well as the c.g. travel are determined. The loading rules are summarized later.

(3) Re-position wing so that c.g. at most forward loading condition (ELCGF) falls at forward c.g. stability and control limit (FCGLIM). Compute tail moment arms.

(4) Call subroutine TAIL with c.g. travel required to compute new tail areas and volumes and new stability and control limits.

(5) Compare tail size as in simplified balance option, if not within allowable tolerance repeat process starting at (1) above.

The group of statements in WGHT starting at the comment "COMPUTE MOST FWD AND AFT LOAD CONDITIONS" through statement 100 deal with the computation of the extreme center of gravity locations due to most forward and aft load conditions. These conditions are based on the loading rules summarized in the following sections V.1.3.3 to V.1.3.3.7. FCGLIM refers to the forward c.g. limit and ACGLIM is the aft c.g. limit from stability and control considerations.

The forward and aft loading conditions are specified in terms of the extreme values of center of gravity, expressed as a percent of mac. The forward c.g. location, in ft., is

$$F_{CGLIM} = EL_{WING} - (CP_{MRGN} + .25 - X_{CGWD})\ CBAR_{W} \qquad \text{(V.1.119)}$$

and the aft location A_{CGLIM} is of identical form, where the parameters X_{CGFWD} and X_{CGAFT} are given as percent mac. The center of gravity for the operating weight empty condition is

$$CG_{OWE} = EMO_{WE}/O_{WE} \qquad \text{(V.1.120)}$$

where the two factors are defined in terms of specified weights and moment arms for major aircraft components. The number of passengers, N_{PAX}, and the number of rows of seats, N_{ROW}, are then found under the assumption that each passenger weighs the input unit weight, with baggage, in terms of the number of seats abreast, S_{AB}.

The loop 81 is concerned with the moment produced by rows of passengers, measured relative to the nose of the aircraft. Thus, the moment arm is, for row number I,

$$EL_{ROW}\ (I) = (P_S/12.)\ I + EL_{ODN}\ H_N + EL_{PC} - P_S/36. \qquad (V.1.121)$$

where the last three terms measure from the aircraft nose to the front of the passenger compartment; i.e.,

EL_{ODN} = fineness ratio of fuselage nose

H_N = height of fuselage nose, ft.

EL_{PC} = length of pilot's compartment, ft.

P_S = lengthwise distance between seats, in.

This permits the computation of passenger-induced moment as the sum,

$$EM_{PAXT} = EM_{PAXT} + S_{AB}\ (170)\ EL_{ROW}\ (I) \qquad (V.1.122)$$

The more complex loop 88 is intended to develop the moment caused by several component weights when the c.g. is at the most aft (I_{LIMIT} = 1), the most forward (I_{LIMIT} = 2) and the design location (I_{LIMIT} = 3). The loop is called three times, corresponding to these c.g. positions, and the computations are complex chiefly because of the number of component weights which can vary on the aircraft.

The first third of the loop is devoted to computation of total weight and moment of the passengers, the moment being measured relative to the nose of the aircraft. These are denoted by W_{PAXD} (I_{LIMIT}) and EM_{PAXD} (I_{LIMIT}), respectively, and are found by straightforward means, being proportional to number of seats abreast, S_{AB} and to unit weight of passengers, UW_{PAX} .

Following this segment, the baggage weight and moment arm are found as $W_{BAG}\ (I_{LIMIT})$ and EL_{BAGS} in terms of payload, passenger weight and proportion of fuselage length, EL_F. The fuel weight and center of gravity are then found in terms of wing geometry and known limits of center of gravity location. Tip tanks are included in this computation, which ends with an expression for

$$X_{FUELD}\ (I_{LIMIT}) = W_{FUELM} - W_{FTPD}\ (I_{LIMIT}) - W_{FWD}\ (I_{LIMIT}), \qquad (V.1.123)$$

where $I_{LIMIT} = 1$ or 2, and

$$X_{FUELD}\ (3) = W_G - O_{WE} - W_{PAXD}\ (3) - W_{BAG}\ (3) - W_{FWD}\ (3) - W_{FTPD}\ (3) \qquad (V.1.124)$$

These weights must be positive, and they contribute to fuel weight,

$$W_{FUEL} = W_{FWD}\ (I_{LIMIT}) + W_{FTPD}\ (I_{LIMIT}) + X_{FUELD}\ (I_{LIMIT}) \qquad (V.1.125)$$

and hence to the limit weight,

$$W_T\ (I_{LIMIT}) = O_{WE} + W_{FUEL} + W_{PLD} \qquad (V.1.126)$$

where

O_{WE} = operating empty aircraft weight, lb.

$W_{FWD}\ (I_{LIMIT})$ = wing fuel weight, lb.

$WFTPD\ (I_{LIMIT})$ = tip tank fuel weight, lb.

W_{PLD} = weight of payload, lb.

If the limit weight exceeds W_G, baggage weight is redefined as

$$W_{BAG}(I_{LIMIT}) = W_{BAG}(I_{LIMIT}) - W_T(I_{LIMIT}) + W_G \qquad (V.1.127)$$

and

$$W_T(I_{LIMIT}) = W_G$$

Finally, the center of gravity is located the following distance aft,

$$EL_{CGD}(I_{LIMIT}) = EM_L(I_{LIMIT})/W_T(I_{LIMIT}) \qquad (V.1.128)$$

where the divisor is in Eq. V.1.126, and where the moment is the sum of five terms,

$$EM_L(I_{LIMIT}) = EM_{OWE} + EL_{SING}\left[W_{FWD}(I_{LIMIT}) + X_{FUELD}(I_{LIMIT})\right] + EM_{PAXD}(I_{LIMIT}) + EL_{BAGS}W_{BAG}(I_{LIMIT}) + EL_{TIP}W_{FTPD}(I_{LIMIT}) \qquad (V.1.129)$$

These terms correspond respectively to operating empty weight, wing fuel weight, passenger weight, baggage weight and tip tank fuel weight, for the three loading conditions, $I_{LIMIT} = 1, 2$ or 3.

A large number of parameters are printed, following which c.g. moment arms and wing 1/4 chord locations are calculated as EL_{WING}, EL_{CP}, EL_{CG} and G_{CGMAC}, in terms of CP_{MRGN}, X_{CGFWD}, $CBAR_W$ and related terms.

A call is next made to subroutine TAIL, which is described in section V.1.4, to determine tail size for the required c.g. travel, DELXCG.

V.1.3.3 Most Forward Load Condition

1. Pilot
2. All seats forward of FCGLIM are filled by passengers each weighing 170 lbs.

3. If baggage lies forward of FCGLIM, all baggage is loaded, otherwise none (baggage load defined below)

4. All fuel tanks forward of FCGLIM are full

5. If fuel forward of FCGLIM is less than minimum fuel (see below), add fuel to next most forward fuel tank until minimum fuel load is reached

V.1.3.4 Most Aft Load Condition

1. Pilot

2. All seats aft of ACGLIM are filled by passengers each weighing 170 lbs.

3. If baggage lies aft of ACGLIM, all baggage is loaded, otherwise none

4. All fuel tanks aft of ACGLIM are full

5. If fuel aft of ACGLIM is less than minimum fuel, add fuel to next most aft fuel tanks until minimum fuel load is reached

V.1.3.5 Design Load Condition

1. Pilot

2. Number of passengers is largest whole number contained in WPL/200. Passengers are loaded in forward seats

3. Baggage is WPL minus number of passengers x 170

4. Design fuel load contained in wing, tip tanks (if any), and fuselage tanks (if any)

V.1.3.6 Minimum Fuel for Weight and Balance* Condition

1. Piston engine aircraft: Minimum fuel (lbs) = $\frac{\text{maximum installed horsepower}}{2}$

---

*Min fuel for wt. and balance is not necessarily the same as minimum fuel for flight. See FAA Advisory Circular AC65-9, p. 50-51.

2. **Turbine Aircraft:** <u>Specified by Manufacturer</u> (no general rule)
The program currently assumes 20% of wing fuel tank capacity

V.1.3.7 <u>Maximum Baggage Condition</u>

1. Maximum baggage is assumed to be same as design load baggage (see above). Baggage is located by input variable RELB

V.1.4 <u>Stability and Control (TAIL)</u>

Subroutine TAIL is a computerized methodology used to determine the critical longitudinal and lateral stability and control requirements governing horizontal and vertical tail sizes of aircraft from two place trainers to business jets.

The procedures which were developed reflect the requirements of the Federal Aviation Regulations (FAR) and accepted engineering design practices. The tail sizing procedures account for the following:

<u>Horizontal</u>

(a) Meeting a specified static margin in relation to neutral static stability ($dCm/dC_L = 0$)

(b) The ability to rotate the aircraft to takeoff rotation speed

(c) The ability to obtain the maximum lift coefficient of the aircraft in the landing configuration in ground effect.

<u>Vertical</u>

(a) Static directional stability requirements (weathercock stability)

(b) Minimum control speed requirements for twin-engine aricraft.

To best understand the computations in this subroutine is to first review the output from the subroutine which summarizes the important tail sizing parameters. Figure V.1.24 is an example of this output. The first three lines of print are the parameters that depend on the particular flight condition for horizontal tail sizing. These are cruise, nose-wheel liftoff, and landing flare. The parameters are wing angle of attack (ALPHA), wing lift curve slope (WING CLA), tail lift curve slope (TAIL CLA), tail efficiency factor (TAIL EFF), downwash angle at tail (DOWNWASH), wing lift coefficient (WING CL), and moment derivatives (DCM) or moments (CM) for the fuselage, nacelle, flap and power effects. The next four lines of print deal with the elevator characteristics and effectiveness as identified on the output. The next five lines display the critical c.g. limits and the horizontal tail sizes required for the different criterion. The neutral point, aft and forward c.g. limits, and various tail sizes are determined in TAIL. The other items have been input or determined in WGHT. The last three lines of print are the vertical tail sizing requirements and the required vertical tail area.

Initially the subroutine contains about 60 COMMON and DATA statements that transfer data and store data for table interpolations. The last data statement is important in that it makes the assumption that the aerodynamic center of the wing with flaps up (XAC) and flaps extended (XACF) is at the quarter chord. The next statement sets up a function statement for calculating the lift curve slope of the wing or tail knowing the aspect ratio, sweep, and Mach number. This is the same method that is used in subroutine CLIFT.

---TAIL SIZING SUMMARY---

CONDITION	ALPHA	WING CLA	TAIL CLA	TAIL EFF.	DOWN WASH	WING CL	--FUSELAGE-- DCM	CM	---NACELLE-- DCM	CM	FLAP CM	------POWER----- DCM	CM	CT
CRUISE	2.6553	.0925	.0763	1.0000		.3567	.1212		.0029			0.0000		
LIFTOFF	1.0000	.0735	.0642	1.0000	.2595	.5169	.1524	0.0000	.0029	0.0000	-.1300		0.0000	
LANDING	13.3669	.0737	.0633	1.0000	3.0688	1.7108	.1522	.1386	.0029	.0026	-.1957		0.0000	

ELEVATOR PARAMETERS

CMALPHA(FLOATING TENDENCY) = -.00435 WING DE/DALPHA = .51895

CMDELTA(RESTORING TENDENCY) = -.01170

CMDELTA(CONTROL POWER) = -.01800

TAUH(EFFECTIVENESS) = .00800

	FRACTION MAC	STATION (DATUM NOSE)	HORIZONTAL TAIL SIZES	
NEUTRAL POINT	.3320	21.593	STATIC STABILITY AND TRIM	54.8898
STATIC MARGIN	.0300		STABILITY AND LIFTOFF	50.1587
AFT CG LIMIT(STABILITY)	.3020	21.381	LIFTOFF	49.2588
CG RANGE(LOADING)	.1338		REQUIRED TAIL SIZE	54.8898
FWD CG LIMIT(CONTROL)	.1686	20.439	TAIL ARM(ELTH)	18.5851

VERTICAL TAIL AREA = 29.5593 FOR DIRECTIONAL STABILITY OF -.00150

VERTICAL TAIL AREA= 43.3784 FOR MINIMUM CONTROL SPEED = 102.80 KTS

REQUIRED VERTICAL TAIL AREA = 43.3784 TAIL ARM(ELTV) = 15.1351

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FIGURE V.1.24 - TAIL SIZING OUTPUT

V.1.4.1 Horizontal Tail Sizing

After some initialization statements a number of parameters are determined for the cruise condition. These are the cruise lift coefficient (CLCR), cruise angle of attack (ALFC) and lift curve slope (CLALCR) from CLIFT, and for propeller aircraft the thrust coefficient,

$$C_{TCR} = \frac{\text{THRUST}}{1.4\ P_O\ EM^2\ D^2_{PROP}} \qquad (V.1.131)$$

where P_O is static pressure and D_{PROP} is propeller diameter. Next, is a cruise value for tail efficiency,

$$ETA_{CR} = \left[.9 + .1S_{AH} \right] \left[1. + 8.\ C_{TRC}/3.\ 14159 \right] \qquad (V.1.132)$$

where S_{AH} = position of horizontal tail on vertical tail.

The next portion of TAIL deals with conditions at nose-wheel liftoff for a speed 85 percent of the stall speed in takeoff configuration. For propeller aircraft this includes the thrust coefficient and tail efficiency calculations as in the preceding cruise portion.

The next section determines the "minimum control speed" and corresponding thrust used in the vertical tail sizing of multi-engined aircraft. The "minimum control speed" is an input fraction of the stall speed in the takeoff configuration;

$$V_{MC} = FV_{MCS}\ V_{SOGE} \qquad (V.1.133)$$

The corresponding thrust, as returned by ENGINE, is used in calculating the thrust with one engine failed

$$F_{NMC} = \text{THRUST}\ (EN_P - 1.) \qquad (V.1.134)$$

and for propeller aircraft, this is used in defining the thrust coefficient, C_{TMC}, and tail efficiency, ETA_{MC}, on the next lines.

If the configuration utilizes turbojet/turbofan engines, the next section computes the tail efficiency factors for the four flight conditions as a function of the horizontal tail position, SAH.

The position of the main gear from the aircraft center of gravity is determined

$$X_G = (CX_A - GCG_{MAC})\ CBAR_W \tag{V.1.135}$$

where X_G = distance of main gear from aircraft c.g., ft.

CX_A = location of gear on mac, fraction of mac

GCG_{MAC} = aircraft c.g. location on mac, fraction of mac

The following statements deal with determining stability factors for the fuselage, Q_{FUS}, and the nacelle, Q_{NAC}, or, Q_N, depending on whether the nacelle is on the wing or fuselage. The nacelle factors are only determined for multi-engine aircraft.

For the fuselage and wing mounted nacelles, Q_{FUS} and Q_{NAC} are determined from Figure V.1.25 where X_{4R} is the position of the wing root 1/4 chord on the fuselage and X_{4RN} is the position of the wing root 1/4 chord on the nacelle. These are determined by

$$X_{4R} = \left[EL_{C4W} + (S_{WF}/2.)\ TAN\ (DLM_{C4}) \right] / EL_P \tag{V.1.136}$$

$$X_{4RN} = \left[EL_N - .75\ CR_{NAC} - TSW_{TE}\ DBAR_N/2. \right] / EL_N \tag{V.1.137}$$

where $$CR_{NAC} = C_{RCLW} \left[1. - (1. - SL_M)\ YN_{ACR}/\ (B/2.) \right] \tag{V.1.138}$$

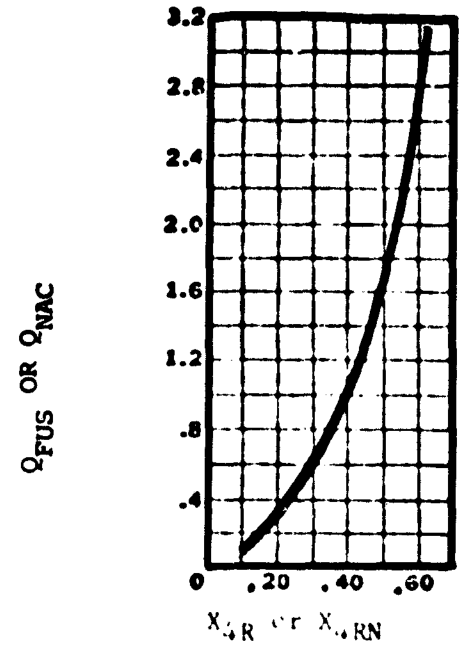


FIGURE V.1.25 - FUSELAGE/NACELLE STABILITY FACTOR (FROM NACA TR711)

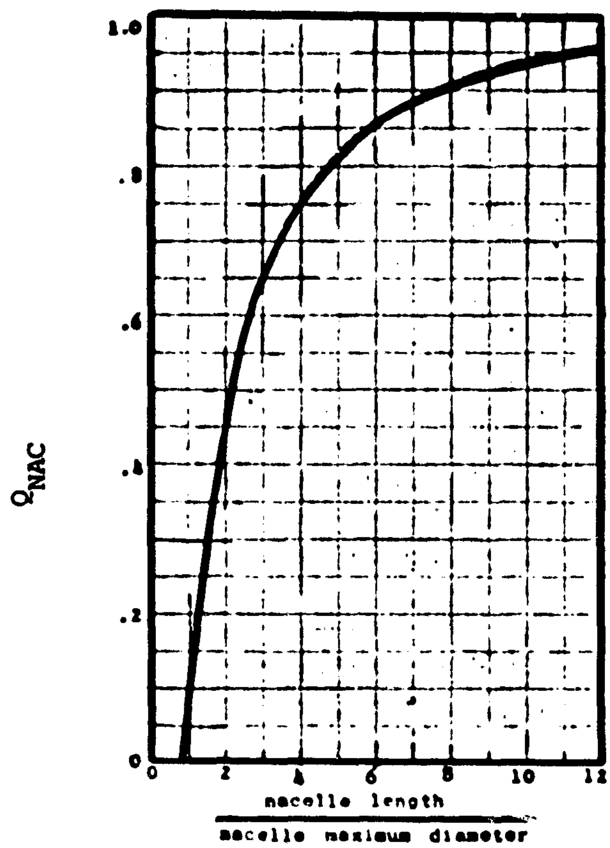


FIGURE V.1.26 - NACELLE FINENESS-RATIO CORRECTION FACTOR

$$YN_{ACR} = Y_P\ B/2 - DBAR_N/2. \qquad (V.1.39)$$

C_{RCLW} = wing chord at wing centerline, ft.

SL_M = wing taper ratio

S_{WF} = fuselage width, ft.

EL_N = nacelle length, ft.

$DBAR_N$ = nacelle diameter, ft.

TSW_{TE} = the tangent of the wing trailing edge sweep

If the nacelles are located on the fuselage the stability factor, Q_N, is determined from Figure V.1.26 as a function of nacelle fineness ratio.

The next parameter determined in TAIL is the downwash gradient at the horizontal tail. This is identified as DEDAW and is based on an empirical method of estimating the downwash gradient behind straight-tapered wings given in DATCOM.

$$DEDAW = \frac{CL_{AEMCRU}}{CL_{AO}}\ 4.44\left[F_{AR}\ F_{SLM}\ F_{HT}\ F_{LAM}\right]^{1.19} \qquad (V.1.140)$$

The factors F_{AR}, F_{SLM}, F_{HT}, F_{LAM} are wing aspect ratio, wing taper ratio, horizontal-tail-location, and wing sweep factors, respectively, determined from

$$F_{AR} = 1./AR - 1./\ (1. + AR^{1.7}) \qquad (V.1.141)$$

$$F_{SLM} = (10. - 3S_{LM})/7. \qquad (V.1.142)$$

$$F_{HT} = (1. - H_T/B)/\ (2\ EL_{TH}/B)^{1/3} \qquad (V.1.143)$$

$$F_{LAM} = COS\ (DLM_{C4})^{1/2} \qquad (V.1.144)$$

where AR, S_{LM}, DLM_{C4} and B are wing aspect ratio, taper ratio, 1/4 chord sweep, and span, respectively, H_T is the height of the tail relative to the wing chord plane and EL_{TH} is the distance between the 1/4 chord of the wing mac. and the 1/4 chord of the tail mac. The ratio of the wing lift-curve slopes of cruise Mach number and zero Mach number, CL_{AEMCRU}/ CL_{AO}) approximates the Mach number effect on the downwash.

Fuselage and nacelle stability contributions, dC_m/dC_L, are determined in the next few program statements. They are functions of geometry and the factors Q_{FUS}, Q_{NAC} and Q_N, which were previously determined. DCM_{FUS} and DCM_{NAC} are the fuselage and nacelle contributions at cruise. DCM_{FST} and DCM_{NCT} are the fuselage and nacelle contributions at nose-wheel liftoff. DCM_{FSL} and DCM_{NCL} are the respective contributions at landing conditions. The fuselage and nacelle pitching moments are then computed for the liftoff and landing condition. CM_{FUST} and CM_{FUSL} being the fuselage pitching moments and CM_{NACT} and CM_{NACL} the nacelle pitching moments.

The next items of importance are the hinge-moment coefficients and the elevator power and effectiveness. The two hinge-moment coefficients are, C_{HALF3}, the "floating tendency", and, C_{HDEL3}, the "restoring tendency". They are computed in terms of section parameters C_{HALF} and C_{HDEL} from Figure V.1.27 in the following equations

$$C_{HALF3} = C_{HALF}\ C_{LALFT}/C_{LALTO} \tag{V.1.145}$$

$$C_{HDEL3} = C_{HDEL} + TAU_H\ (C_{HALF3} - C_{HALF}) \tag{V.1.146}$$

where C_{LALFT} = lift curve slope of horizontal tail

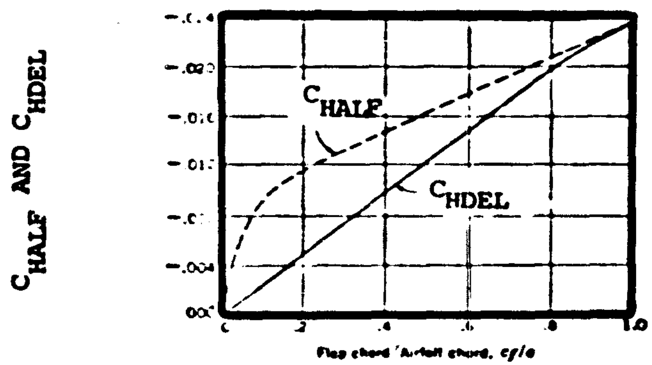


FIGURE V.1.27.

Section hinge moment parameters versus flap chord/airfoil chord ratio (NACA 0009 airfoil). From NACA WR L-663, "Wind-tunnel Data on the Aerodynamic Characteristics of Airplane Control Surfaces," by R. I. Sears.

C_{LALTO} = lift curve slope of horizontal tail with infinite aspect ratio

TAU_H = elevator effectiveness

The next few lines are concerned with downwash angles at touchdown, E_{GL}, and at nose wheel liftoff, E_{GT}, which are proportional to angle of attack at these times; AL_{WGL} and AL_{WGT}. The stall velocity V_{STALL} at liftoff is then found in ft. per sec., and the dynamic pressure at this airspeed is Q_{UE}. Subroutine C_{LIFT} returns the lift coefficient C_{LLO}, and the lift for nose-wheel liftoff is calculated (FL).

$$FL = C_{LLO}\, Q_{UE}\, S_W \tag{V.1.147}$$

The moment coefficient at nose-wheel liftoff due to power effects is computed in the following lines. This has the form,

$$C_{MPTO} = C_{TNWLO} \frac{2\, D^2_{PROP}\, T_P}{S_W\, C_{BARW}}\, E_{NP} \text{ (propeller aircraft)} \tag{V.1.148}$$

and

$$C_{MPTO} = F_{NNWLO}\, T_P\, C_{LLO} / (W_G\, C_{BARW}) \text{ (turbojet/turbofan)} \tag{V.1.149}$$

for propeller and turbojet aircraft, respectively. In these equations, the terms C_{TNWLO} and F_{NNWLO} are proportional to engine thrust at take-off, as were computed earlier in the subroutine.

Horizontal tail size calculations are initiated following the comment card "CALCULATION OF THE HORIZONTAL TAIL SIZE". The tail volume coefficients and tail areas are determined for the three horizontal sizing criterion. The volume coefficients are functions of previously determined stability derivatives and geometric parameters. V and S_{ST}

are the volume coefficient and tail area to satisfy static stability and trim criteria. X_{NO} is the neutral point and XCG_{AFT} and XCG_{FWD} the allowable c.g. travel limits. V_G and S_{TO} are the volume coefficient and tail area to meet nose-wheel liftoff criterion. V_{GX} and S_{TX} are for the stability and liftoff criteria. The most critical of the three criterion is determined next and the required horizontal tail area, span, and moment arm are computed as

$$S_{HT} = V * S_W \, CBAR_W / EL_H, \qquad \text{(V.1.150)}$$

$$B_{HT} = SQRT \, (AR_{HT} \, S_{HT}) \qquad \text{(V.1.151)}$$

$$EL_{TH} = EL_H + (X_{CGAFT} - .25) \, CBAR_W \qquad \text{(V.1.152)}$$

V.1.4.2 Vertical Tail Sizing

The vertical tail is sized to meet a directional stability requirement and in the case of multi-engine aircraft the "engine out" control requirement. Basically, the specifications for single-engine airplanes state that the directional stability criterion, $\frac{dC_n}{d\psi}$, should be negative (stable) for any anticipated speed greater than 1.2 times the stalling speed. In addition, the yawing moment control, usually the rudder, must be powerful enough to counteract the yawing moments encountered in various aspects of flight operations such as cross-wind takeoffs or landings. These requirements apply also to multi-engine airplanes, with the additional requirement that the rudder must be powerful enough to regain and maintain straight flight with one engine inoperative at a minimum speed not greater than 1.2 times the stalling speed. The minimum speed at which engine-out control can be maintained is known as the "minimum control speed", V_{mc}, and the requirement that it be less than 1.2 times the stall speed is the major design factor in determining the vertical tail and rudder size. Thus, in preliminary design calculations for the vertical tail, we are faced with a directional stability problem for light single-engine airplanes, and a directional control problem for multi-engine airplanes.

The vertical tail sizing computations begin at the comment "VERTICAL TAIL SIZING" in subroutine TAIL. The contribution of the various aircraft components, wing, fuselage, power, etc., to the directional stability of the aircraft are the first calculations.

The wing contribution to directional stability is in many cases negligible. The small contribution of the swept wing is estimated by

$$CNP_{WNG} = - 0.00006\ DLM_{C4}^{1/2} \tag{V.1.153}$$

The fuselage contribution to directional stability is CNP_{FUS}, which is a function of several parameters, as indicated in Figure V.1.28. The tail cone and nose cone approximations to the fuselage geometry are used to derive the equation for projected fuselage side area, S_S, and the fuselage contribution to the directional stability derivative is

$$CNP_{FUS} = (.96\ Q_B/57.3)\ (S_S/S_W)\ (EL_F/B)\ (D_{F1}/D_{F2})^{1/2} (D_{F2}/D_{F1})^{1/3} \tag{V.1.154}$$

where Q_B is determined from Figure V.1.29, and D_{FL} and D_{F2} are fuselage diameters at the quarter and three-quarter body length points, respectively.

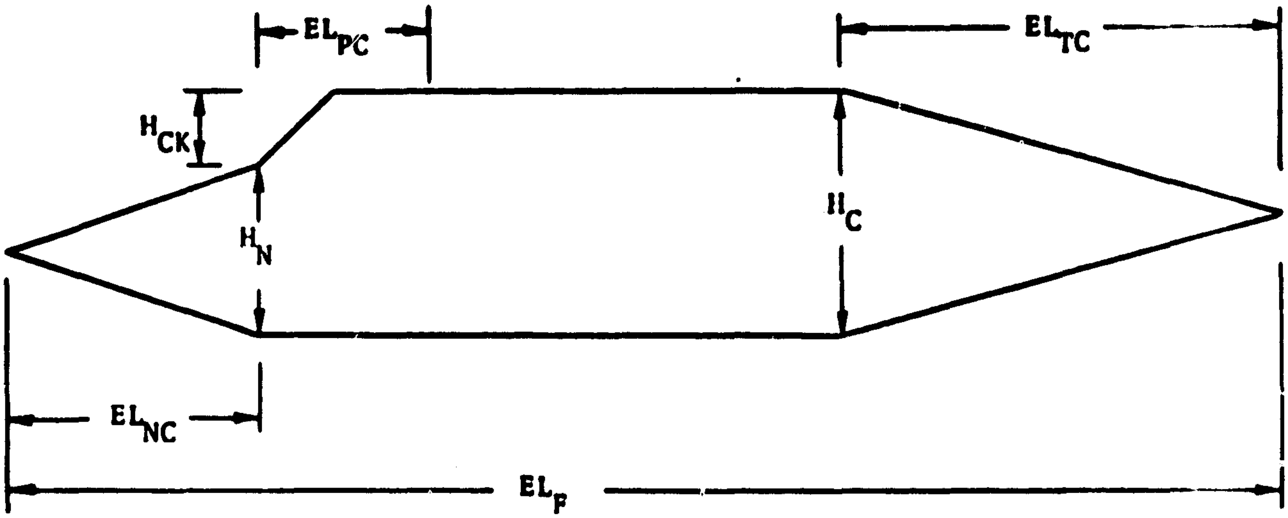
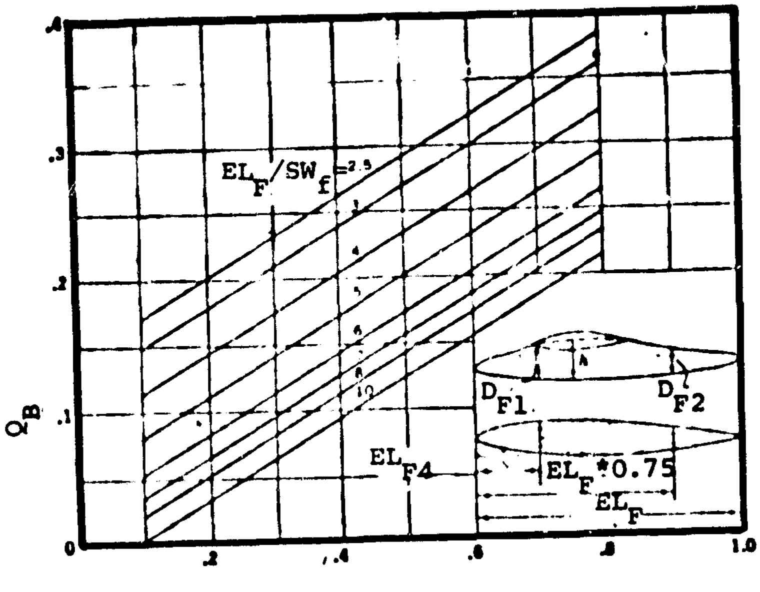


FIGURE V.1.28 - FUSELAGE GEOMETRY



EL_{CG}/EL_F

FIGURE V.1.29 - FUSELAGE DIRECTIONAL STABILITY COEFFICIENT

The wing-fuselage interference stability increment, which is computed next, depends on the wing location, H_{WING}.

$$DL1_{CNP} = 0.0002\ H_{WING} \qquad (V.1.155)$$

where H_{WING} = 1 for High wing

= 0 for Low wing

The next directional stability parameter computed is the one accounting for power effects. This is identified as CNP_P and is set to zero for turbojet/turbofan configurations (N_{TYE} = 7). For propeller aircraft the first item computed is the longitudinal distance from propeller disc to center of mass, which is denoted by EL_P. This term is equal to EL_{CG} for single engine aircraft, since this measures the center of gravity aft of the aircraft nose. For twin engine aircraft, it is more complex, and is given by

$$EL_P = EL_{CG} - (EL_{C4W} - .25\ C_{RCLW} + X_{MAC}) - EG_{MRGN} * CBAR_W + EL_N/2. \qquad (V.1.156)$$

The sideforce stability derivative caused by windmilling propellers is implied by the number of blades, B_L, and the derivative is named $DCYP_{DP}$. Similarly, the windmilling propeller's contribution to the directional stability derivative is CNP_{PO}, a function of propeller geometry and location, and of wing span and area; i.e.,

$$CNP_{PO} = \Pi * D^2_{PROP}\ EL_P\ DCYP_{DP}\ EN_P/\ (4.\ S_W B) \qquad (V.1.157)$$

The full-power value of this derivative is simply estimated by increasing the windmilling value by fifty percent,

$$C_{NPP} = 1.5\ C_{NPPO} \qquad (V.1.158)$$

The vertical tail moment arm from the aircraft center of gravity is computed as

$$EL_V = EL_H - DEL_{TVH} \qquad (V.1.159)$$

where EL_H is the horizontal tail moment arm from the aircraft center of gravity and DEL_{TVH} is the distance between the vertical and horizontal tails aerodynamic centers.

Another interference correction for directional stability calculations is the one that arises from the sidewash or interference from the wing-fuselage combination on the vertical tail. This is identified as $DL2_{CNP}$ and is determined from Figure V.1.30.

The lift-curve slope of the vertical tail, CLA_{LFV}, is determined from Figure V.1.31 and the "effective" aspect ratio, AR_{VTE}, of the vertical tail.

The lateral distance from centerline to engine thrust line is then calculated, for use in determining yawing moment due to engine failure. This is denoted Y, and it is given in terms of a percent semi-span Y_P or nacelle diameter $DBAR_N$ and maximum fuselage width SWF. The yawing moment coefficient caused by an asymmetric engine arrangement has either of two forms, depending on the propulsion type; i.e., for propeller aircraft,

$$C_{NP} = Y\ D^2_{PROP}\ C_{TMC} / \ (.5\ S_W\ B) \qquad (V.1.160)$$

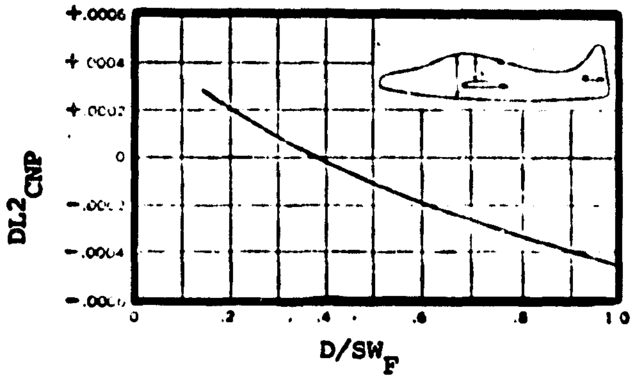


FIGURE V.1.30 - INTERFERENCE CORRECTION TO C_n

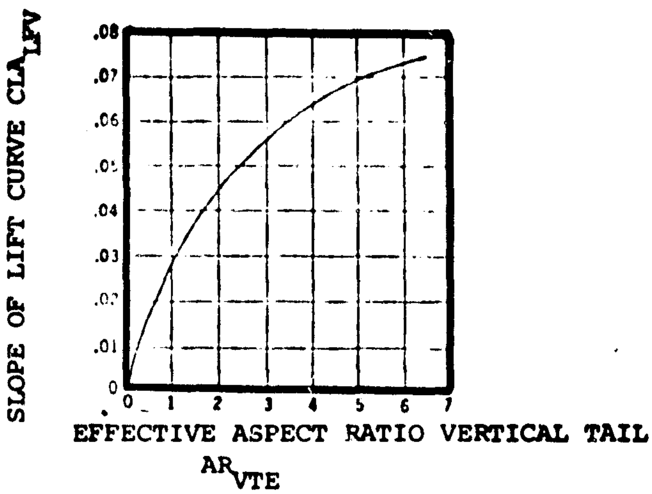


FIGURE V.1.31 - SLOPE OF LIFT CURVE, VERTICAL TAIL. FROM NACA TN 755 "ANALYSIS OF WIND TUNNEL DATA ON DIRECTIONAL STABILITY AND CONTROL" H. R. PASS while for turbojet or turbofan aircraft,

$$C_{NP} = Y\ F_{NHC}\ (.5\ RHO_{TO}\ S_W\ B\ V_{MC}^2) \tag{V.1.161}$$

The next group of calculations determine the vertical tail areas required for the two vertical tail sizing criterion. S_{DS} is the tail area required to satisfy the directional stability, CNP_{AC}, required for the aircraft. S_{VT2} is the tail area required to meet the engine-out requirement for multi-engine aircraft.

$$S_{DS} = \frac{(CNP_{WNG} + CNP_{FUS} + CNP_P + DL1_{CNP} + DL2_{CNP} - CNP_{AC})\ S_W B}{CLA_{LFV}\ EL_V\ ETA_{VC}} \tag{V.1.162}$$

$$S_{VT2} = \frac{C_{NP}\ S_W\ B}{CLA_{LFV}\ DR_{MAX}\ TAU_V\ EL_V\ ETA_{MC}} \tag{V.1.163}$$

where TAU_V = the vertical tail effectiveness

DR_{MAX} = maximum rudder deflection, deg

and all other variables have been previously defined. The most critical of the two criterion is determined if it is a multi-engine aircraft and then the span and moment arms are computed for the vertical tail.

$$B_{VT} = SQRT\ (AR_{VT}\ S_{VT}) \tag{V.1.164}$$

$$EL_{TV} = EL_V + (X_{CGAFT} - .25)\ CBAR_W \tag{V.1.165}$$

V.1.5 References

1. A. H. Schoen, "User's Manual for VASCOMP II - The V/STOL Aircraft Sizing and Performance Computer Program". The Boeing Vertol Company, Boeing Document D8-0375, Vol. VI, March 1968, revised Oct., 1971.

2. D. Garcia, "Empirical Formulae for Radii of Gyration of Aircraft, Revision A." Society of Aeronautical Weight Engineers Paper No. 78A, April, 1962.

3. E. Torenbeek, "Prediction of Wing Group Weight for Preliminary Design", Aircraft Engineering, July 1971, pgs. 16-21.

GASP - GENERAL AVIATION SYNTHESIS PROGRAM

VOLUME V - WEIGHT

PART 2 - USER'S MANUAL

JANUARY 1978

Prepared for

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Ames Research Center
Moffett Field, California

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AEROPHYSICS RESEARCH CORPORATION

V.2 WEIGHT MODEL USER'S MANUAL

After the aircraft geometric characteristics have been established, the component weights and balance characteristics are determined by straightforward methods. These methods require the use of many descriptive input parameters which are geometrical, initial or operational in nature, and which serve to describe the aircraft mass and its distribution in certain critical operational configurations.

The subroutine WGHT, ENGWGT, DLOAD and TAIL, subroutines have input-output quantities as defined on the followi g pages. It will be noted that WGHT and TAIL have many more inpu' utput parameters than most other subroutines, because of the relatively high level of detail required in the weight model analysis.

Input-output quantities for subroutine WAIT are in Volume IV--Propulsion.

FIGURE V.2.1 SUBROUTINE DLOAD, INPUT

VARIABLE	DESCRIPTION
CATD	0 to 3, aircraft category indicator
CBARW	wing mean aerodynamic chord, ft.
CLALPH	lift curve slope, per rad
VMLSFL	maximum sea level flight velocity, mi. per hr.
WOS	wing loading, lb. per sq. ft.

FIGURE V.2.2 SUBROUTINE DLOAD, OUTPUT

VARIABLE	DESCRIPTION
EMLF	maneuver load factor, g's
EMMO	maximum allowable operating Mach number
GLF	gust load factor, g's
ULF	ultimate load factor, g's
VDMIN	minimum design dive speed, kts. equivalent airspeed
VMO	maximum operating velocity, kts. equivalent airspeed

FIGURE V.2.3 SUBROUTINE ENGWGT, INPUT

VARIABLE	DESCRIPTION
AE	engine exhaust area, sq. ft.
AF	propeller activity factor, per blade
ANAC	wetted area of nacelle, sq. ft.
BL	number of propeller blades
DPROP	propeller diameter, ft.
EMCRU	cruise Mach number
ENP	number of engines
FPYL	pylon weight factor
GRATIO	gear ratio
HPMSLS	sea level static horsepower
KSPCHG	0 or 1 to indicate engine turbocharging
NCYL	number of cylinders
NTYEX	1 to 14 engine flag
NTYPX	1 to 16 propeller flag
RPM	engine rpm
SFNSLS	engine specific thrust at sea level
SKDIM	engine sizing factor
SKWGT	engine weight factor
UWNAC	weight per unit area of nacelle, lb. per sq. ft.
WASLS	sea level static airflow
XLQDE	length to diameter ratio of nacelle

FIGURE V.2.4 SUBROUTINE ENGWGT, OUTPUT

VARIABLE	DESCRIPTION
SWSLS	engine specific weight, lb./lb. thrust or lb./SHP
WENG	engine weight, lb.
WNAC	nacelle weight, lb.
WPROPI	propeller weight, lb.
WPYLON	pylon weight, lb.
WTGB	weight of gear box, lb.

FIGURE V.2.5 SUBROUTINE TAIL, INPUT

VARIABLE	DESCRIPTION
ALPHLO	zero lift angle of attack, deg
AR	aspect ratio of wing
ARHT	aspect ratio of horizontal tail
ARVT	aspect ratio of vertical tail
B	wing span, ft.
BL	number of propeller blades
BMLOD	fineness ratio of tail boom
CBARW	mean aerodynamic chord of wing, ft.
CLMXLD	maximum lift coefficient, landing configuration
CLMXTO	maximum lift coefficient, take-off configuration
CMFLPL	pitching moment due to flaps, landing configuration
CMFLPT	pitching moment due to flaps, take-off configuration
CMPLD	pitching moment due to power, landing configuration
CRCLW	wing chord at fuselage enterline, ft.
CROOTW	root chord of wing, ft.
CXA	horizontal displacement of main landing gear wheel, percent mac
DBARN	mean nacelle diameter, ft.
DELXCG	center of gravity range, percent mac
DEMAX	maximum trailing-edge-up elevator deflection, deg
DLMC4	sweep back of wing quarter chord, deg
DPROP	propeller diameter, ft.
DRMAX	maximum rudder deflection, deg

FIGURE V.2.5 SUBROUTINE TAIL INPUT (Continued)

VARIABLE	DESCRIPTION
DWPQCH	sweep of quarter chord of horizontal tail, deg
EGMRGN	engine c.g. relative to leading edge mac, ft.
ELBM	length of tail boom, ft.
ELCG	distance from aircraft nose to aircraft c.g., ft.
ELC4W	position on fuselage centerline of wing quarter chord, ft.
ELF	length of fuselage, ft.
ELODN	length to diameter ratio of fuselage nose cone
ELODT	length to diameter ratio of fuselage tail cone
ELPC	length of pilot compartment, ft.
ELTH	distance from wing ac to horizontal tail ac, ft.
ELTV	distance from wing ac to vertical tail ac, ft.
EMCRU	cruise Mach number
ENP	number of engines
EYET	incidence angle of tail, deg
EYEW	incidence angle of wing, deg
GCGMAC	position of aircraft c.g. in percent mac
HC	main cabin height, ft.
HCK	cockpit windshield height, ft.
HN	nose cone diameter, ft.
HNCRU	cruise altitude, ft.
HWING	wing location parameter
KCONFG	type of fuselage indicator
KNAC	nacelle drag accounting indicator

C-2

FIGURE V.2.5 SUBROUTINE TAIL INPUT (Continued)

VARIABLE	DESCRIPTION
KWRITE	output indicator
LC	length of fuselage cabin, ft.
NTYE	engine type indicator
RELP	fuselage-mounted engine location, fraction of fuselage length
RH	ratio of elevator chord to horizontal tail chord
RV	ratio of rudder chord to vertical tail chord
RVMCS	ratio of minimum control speed to takeoff stall speed
SAH	horizontal tail location on vertical tail
SLM	taper ratio of wing
STATIC	static margin, fraction mac
SW	wing area, sq. ft.
SWF	maximum fuselage width, ft.
TP	vertical distance c.g. to thrust line, positive downward, ft.
UM	coefficient of rolling friction of aircraft on dry runway
WG	aircraft gross weight, lb.
WGS	wing loading, lb. per sq. ft.
XLN	length of nacelle, ft.
XLQDE	ratio of nacelle length to diameter
XMAC	location of wing leading edge of mac, aft from wing centerline leading edge, ft.

FIGURE V.2.5 SUBROUTINE TAIL INPUT (Concluded)

VARIABLE	DESCRIPTION
YMAC	spanwise location from wing centerline chord to wing mac chord
YP	spanwise engine locations on wing/wingspan

FIGURE V.2.6 SUBROUTINE TAIL, OUTPUT

VARIABLE	DESCRIPTION
ALFC	cruise angle of attack, deg
ARVTE	effective aspect ratio of vertical tail
BHT	horizontal tail span, ft.
BVT	vertical tail span, ft.
CHALF	section hinge moment gradient with angle-of-attack
CHDEL	section hinge moment gradient with deflection
CLALPH	aircraft lift curve slope, per rad
CMFVST	pitching moment due to fuselage, take-off configuration
CMNACT	pitching moment due to each nacelle, take-off configuration
CNPAC	aircraft directional stability; yaw moment due to yaw angle
DCMCLP	stability contribution of propulsion unit
ELN	overall length of nacelle, ft.
EM	Mach number
H	altitude, ft.
SHT	horizontal tail area, sq. ft.
SVT	vertical tail area, sq. ft.
TAUV	vertical tail control surface effectiveness factor
VSTALL	aircraft stall speed, knots
XCGAFT	extreme aft c.g. position, percent mac
XCGFWD	extreme forward c.g. position, percent mac
ZCG	vertical displacement of c.g. above runway at nosewheel liftoff, ft.

FIGURE V.2.7 SUBROUTINE WGHT, INPUT

VARIABLE	DESCRIPTION
AR	wing aspect ratio
ARHT	horizontal tail aspect ratio
ARVT	vertical tail aspect ratio
ATMXQC	maximum tip tank length, divided by wing tip chord
B	wing span, ft.
BHT	horizontal tail span, ft.
BVT	vertical tail span, ft.
CATD	1 to 3, specifies structural design criteria
CBARW	mean aerodynamic chord of wing, ft.
CK1-21	wing scale factors, nominally unity
CPMRGN	distance from wing c.g. to aircraft c.g., divided by CBARW
DELP	fuselage cabin pressure differential, lb. per sq. in.
DELWFC	incremental weight input to flight control system
DELWST	incremental structural weight input
DLMC4	wing quarter chord sweep, deg
DWPQCH	horizontal tail quarter chord sweep, deg
DWPQCV	vertical tail quarter chord sweep, deg
EGMRGN	location nacelle c.g. aft of mac leading edge, divided by CBARW
ELBM	tail boom length, ft.
ELF	fuselage length, ft.
ELFFC	cabin length for tail boom fuselage, ft.
ELINC	distance from vertical tail LE to horizontal tail LE, ft.
ELODN	length to diameter ratio of nose cone

FIGURE V.2.7 SUBROUTINE WGHT INPUT (Continued)

VARIABLE	DESCRIPTION
ELPC	length of pilot's compartment, ft.
ELRW	length of pylon attachment for fuselage mounted engines
EMCRU	cruise Mach number
EMLF	maximum manuver load factor, g's
EMMO	maximum operating Mach number
ENP	number of engines
GLF	maximum gust load factor, g's
HN	altitude, ft.
HCRU	cruise altitude, ft.
HPMSLS	maximum sea level static horsepower
KNAC	= 0, nacelle drag not included in aircraft drag
	= 1 or 2, nacelle drag included in aircraft drag
LDCKMX	maximum allowable tip tank fineness ratio
NTYE	1 to 14; type of engine, see ENGWGT
NTYP	1 to 16; type of propeller, see ENGWGT
PAX	number of passengers
RELP	location of propulsion on fuselage, fraction of fuselage length
RELR	location of c.g. of fuselage and contents, fraction of fuselage length
SAB	seats abreast in fuselage
SAH	horizontal tail location on vertical tail
SF	surface area of fuselage

FIGURE V.2.7 SUBROUTINE WGHT INPUT (Continued)

VARIABLE	DESCRIPTION
SHT	horizontal tail area, sq. ft.
SKB-SKZ	weight coefficients of aircraft components
SLM	taper ratio of wing
SLMH	taper ratio of horizontal tail
SLMV	taper ratio of vertical tail
STMRGN	aircraft c.g. location relative to mac quarter chord
STRUT	wing strut attachment point, fraction of semispan
SVT	vertical tail area, sq. ft.
SW	wing area, sq. ft.
SWF	fuselage width, ft.
TCHT	thickness to chord ratio, horizontal tail
TCR	thickness to chord ratio, wing root
TCT	thickness to chord ratio, wing tip
TCVT	thickness to chord ratio, vertical tail
ULF	ultimate load factor, g's
UWPAX	weight per passenger, lb.
VBARH	volume coefficient, horizontal tail
VBARV	volume coefficient, vertical tail
VDMIN	minimum design dive speed, knots EAS
VMLFSL	maximum level flight velocity of sea level, mph
VMO	maximum allowable operating airspeed, knots EAS
WENG	engine weight, lb.
WFA	weight of fuel available

FIGURE V.2.7 SUBROUTINE WGHT INPUT (Concluded).

VARIABLE	DESCRIPTION
WFUL	fixed useful load weight, lb.
WG	gross weight, lb.
WGS	wing loading, lb. per sq. ft.
WHLDEV	weight of high lift devices, lb.
WNAC	nacelle weight, lb.
WPROP1	propeller weight, lb.
WPYLON	engine pylon weight, lb.
WTGB	weight of engine gearbox
XARBM	cross-sectional area of tail boom
YMG	main gear location on wing; fraction of semispan
YP	engine location on wing; fraction of semispan

FIGURE V.2.8 SUBROUTINE WGHT, OUTPUT

VARIABLE	DESCRIPTION
AXIS	length of tip tank, ft.
BHT	horizontal tail span, ft.
BVT	vertical tail span, ft.
BXIS	diameter of tip tank, ft.
CBARHT	mac of horizontal tail, ft.
CBARVT	mac of vertical tail, ft.
CLALPH	aircraft lift curve slope, per rad
ELTH, ELTV	moment arms of horizontal and vertical
EM	Mach number
H	altitude, ft.
OWE	operating weight empty, lb.
SHT	horizontal tail area, sq. ft.
STIP	wetted tip tank area, sq. ft.
SVT	vertical tail area, sq. ft.
UWPAX	weight of one passenger, lb
VFTP	tip tank fuel volume, cu. ft.
VFTPMN	minimum tip tank fuel volume, cu. ft.
VFTPMX	maximum tip tank fuel volume, cu. ft.
W	weight of aircraft, lb.
WB	fuselage weight, lb.
WCC	cockpit controls weight, lb.
WCFW	fixed wing controls weight, lb.
WEP	total engine weight, lb.

FIGURE V.2.8 SUBROUTINE WGHT OUPUT (Concluded)

VARIABLE	DESCRIPTION
WFADES	design value of fuel weight, lb.
WFAMAX	maximum value of fuel weight, lb.
WFAMIN	minimum value of fuel weight, lb.
WFC	flight control system weight, lb.
WFE	fixed equipment weight, lb.
WFSS	fuel system weight, lb.
WFTP	tip tank fuel weight, lb.
WFW	wing fuel weight, lb.
WHT	horizontal tail weight, lb.
WLG	landing gear weight, lb.
WP	propulsion group weight, lb.
WPEI	primary engine installation weight, lb.
WPES	primary engine section weight (pylon, nacelle, etc.) lb.
WPL	payload weight, lb.
WSAS	stability augmentation system weight, lb.
WST	structure weight, lb.
WTIP	tip tank weight, lb.
WVT	vertical tail weight, lb.
WW	wing weight, lb.

FIGURE V.2.9 INPUT-SUBROUTINE HOPWSZ

VARIABLE	DESCRIPTION
AHPSL (I)	array of sea level static horsepowers, nonsupercharged engines
AHPSLX (I)	array of seal level static horsepowers, supercharged engines
ASWN (I)	array, normalized specific engine weight, lb. per HP
ASWSC (I)	array of specific weight increments of engine due to supercharger
BMEP	brake mean effective pressure, lb. per sq. in.
HPMSLS	maximum horsepower, sea level standard conditions, HP
HPQAB	horsepower per unit base area, HP per sq. in.
KSPCHG	0, engine does not have supercharger 1, engine has supercharger
NCYL	number of cylinders in engine (4 to 8)
NTYE	engine type (11, 12 or 13)
RWH	ratio of width to height of piston engine
SKWDTH	multiplier for engine width
SKWGT	multiplier for engine weight
XNMAX	maximum engine speed, rpm

FIGURE V.2.10 OUTPUT-SUBROUTINE HOPWSZ

VARIABLE	DESCRIPTION
ANAC	nacelle surface area, sq. ft.
B	multiplier function of NCYL
SWSLS	specific weight of engine (sea level static) lb. per HP
XLN	length of engine nacelle, ft.

FIGURE V.2.11 INPUT-SUBROUTINE RCWS7

VARIABLE	DESCRIPTION
AHPSL (I)	array of sea level static horsepowers, nonsupercharged engines
AHPSLX (I)	array of sea level static horsepowers, supercharged engines
AROTN (I)	array of number of rotors
ASWSC (I)	array of specific weight increments of engine due to supercharger
ASW75 (I)	array of specific weights, 1975 technology
ASW80 (I)	array of specific weights, 1980 technology
GR	gear ratio
HPMSLS	maximum sea level static horsepower
IDATE	estimated technology date
KSPCHG	0, engine has no supercharger 1, engine has supercharger
NTYE	engine type (14)
ROTN	number of rotors
SKDIAM	multiplier for engine diameter
SKWGT	multiplier for engine weight
SWSLS	specific weight of engine (sea-level static), lb. per HP

FIGURE V.2.12 OUTPUT-SUBROUTINE RCWSZ.

VARIABLE	DESCRIPTION
ANAC	nacelle surface area, sq. ft.
GR	gear ratio
SWSLS	specific weight of engine (sea-level static), lb. per HP
XLN	length of engine nacelle, ft.
XNMAX	maximum engine rpm

No Input or Output Figures for HOPWSZ or RCWSZ in this volume or the propulsion volume.

GASP - GENERAL AVIATION SYNTHESIS PROGRAM

VOLUME V - WEIGHT

PART 3 - PROGRAMMER'S MANUAL

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Ames Research Center
Moffett Field, California

Under

CONTRACT NAS 2-9352

AEROPHYSICS RESEARCH CORPORATION

V.3 WEIGHT MODEL PROGRAMMER'S MANUAL

As shown in the flow charts on the following pages, subroutine WGHT involves several minor iterative loops and computational sequences associated with variation of the aircraft geometry. The fuel weight (wing and tip tanks) and the engine sizing loops are interior to the tail area loop, and all of these three major loops must converge before the component weights and wing location are fixed.

The flow charts are given for the aircraft structure weights routines WGHT, DLOAD, ENGWGT and TAIL. It may be noted that the input to WGHT includes over 70 lengths, angles, and weights which are descriptive of the aircraft geometry and size. In addition, aerodynamic data is required by TAIL, which requires some 85 input parameters, while the other two programs are relatively short. Propulsion system weights are computed by the routines ENGWGT, HOPWSZ, RCWSZ and WAIT.

V.3.1 Structural Weight

V.3.1.1 Subroutine WGHT, Structural Weight and Wing Positioning Routine

This routine controls the calculation of aircraft structural weight and the wing positioning calculation. Routines called are DLOAD for design load ratios, TAIL for tail sizing based on longitudinal and directional stability requirements, CTAER for cruise aerodynamics (described in Volume III), and finally a call tc ENGWGT passes program control to propulsion system sizing.

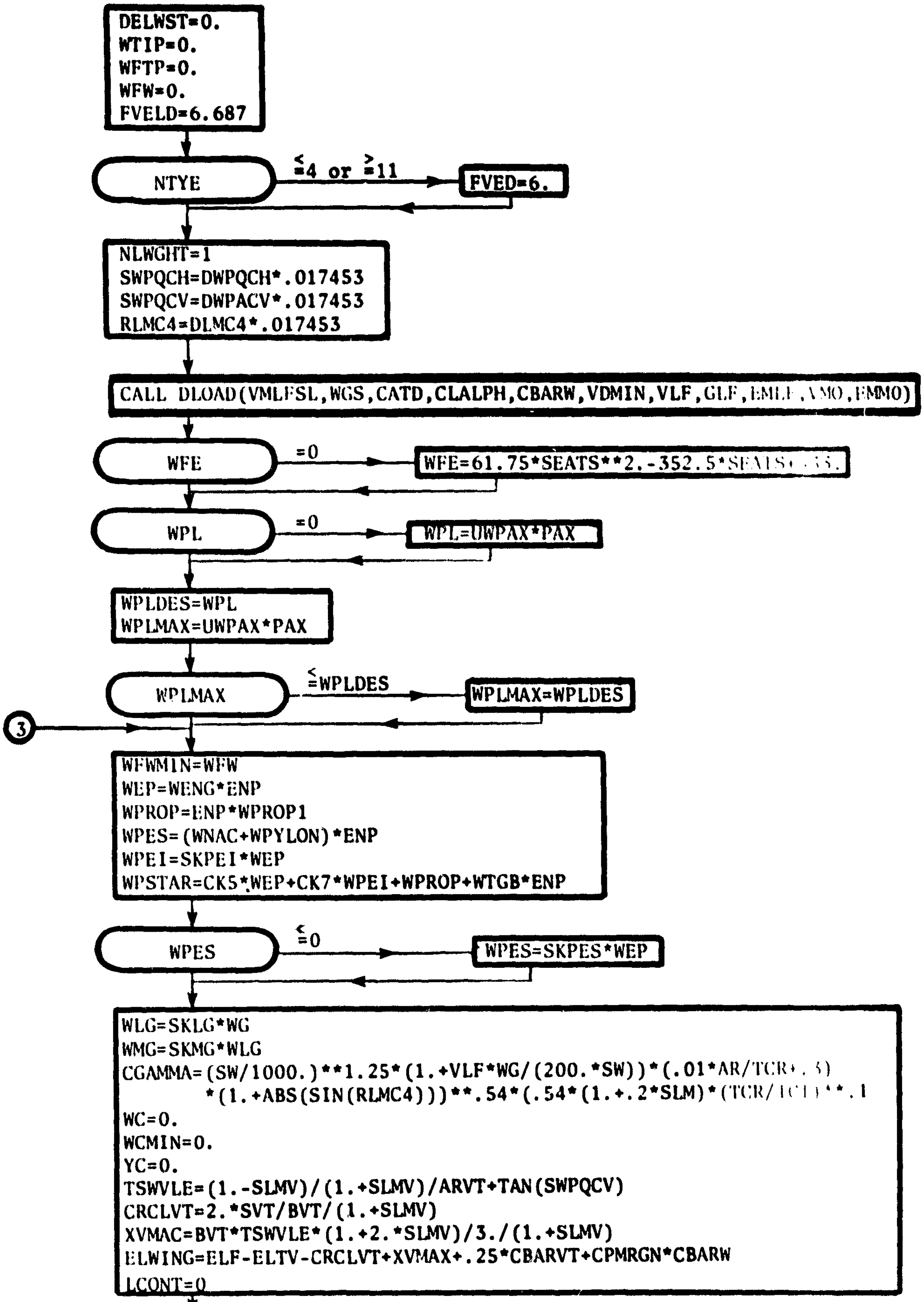
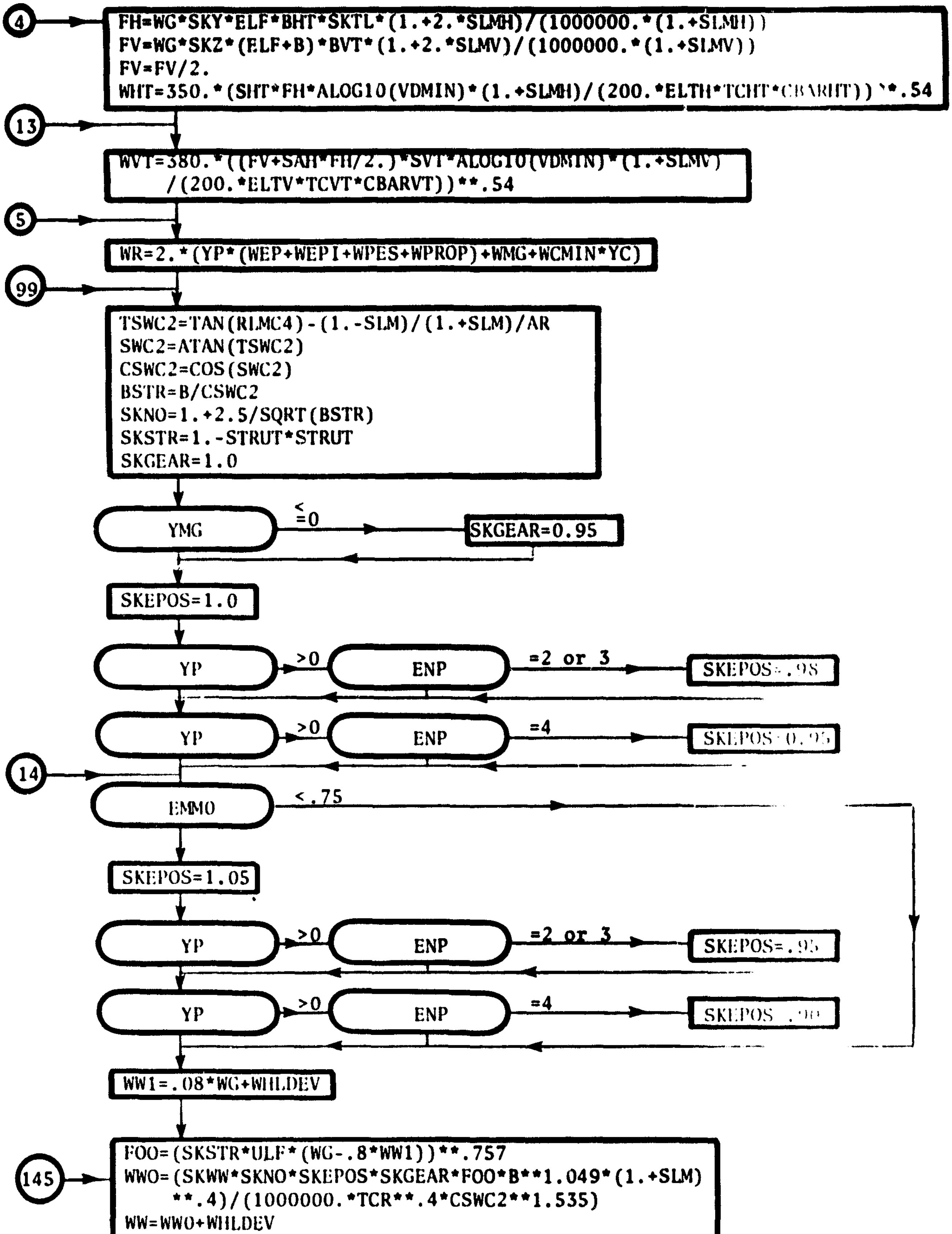
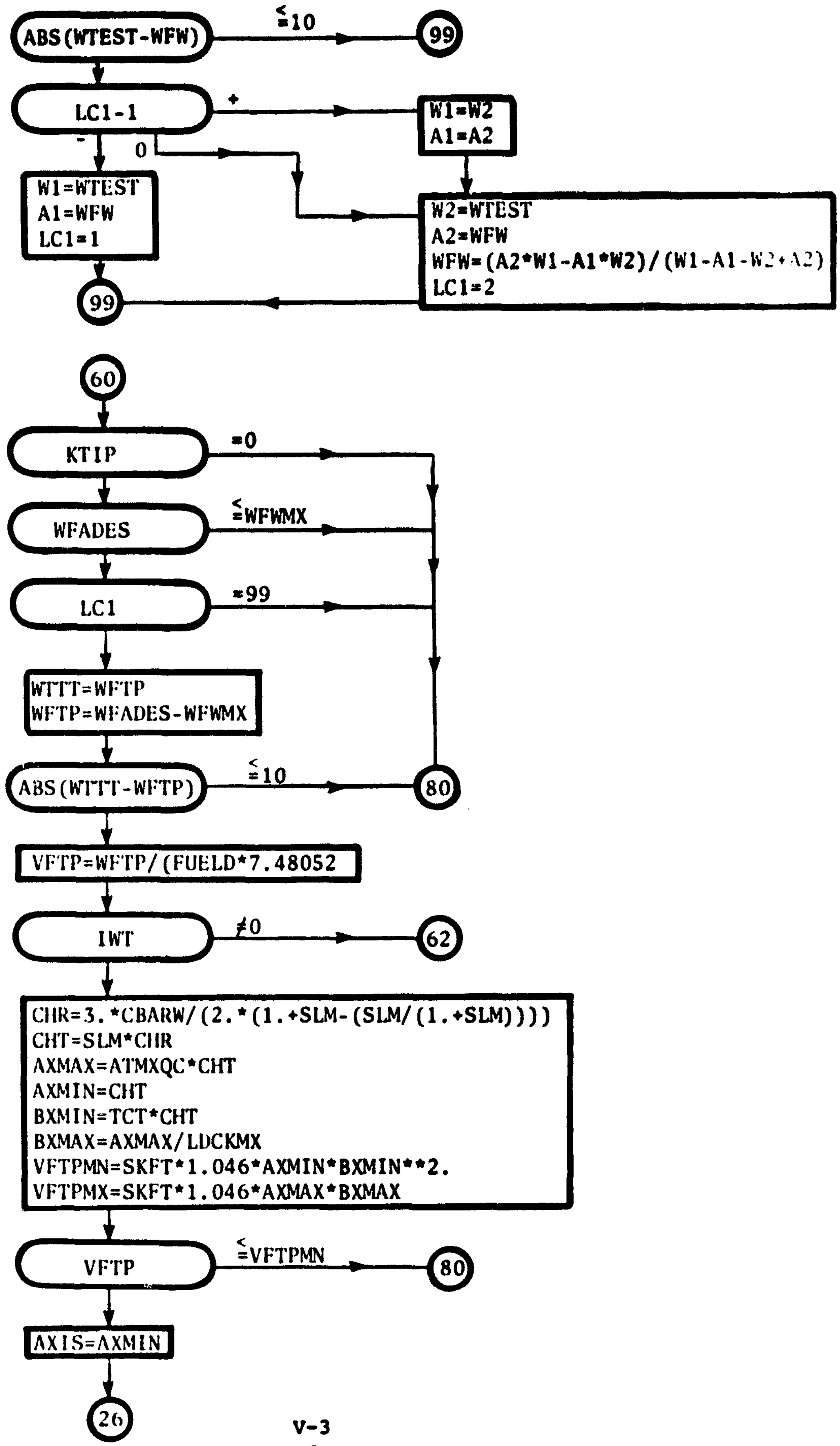
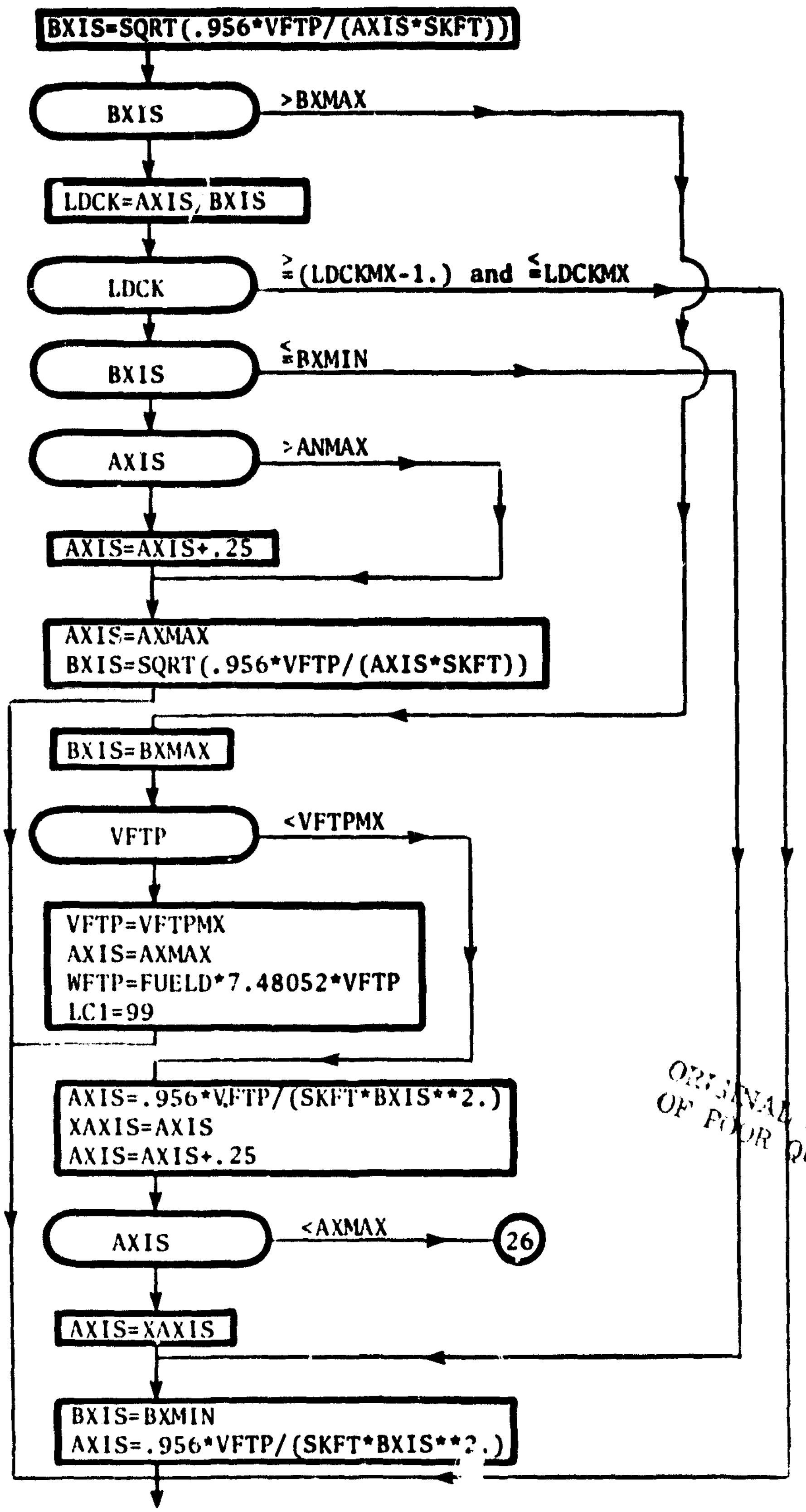


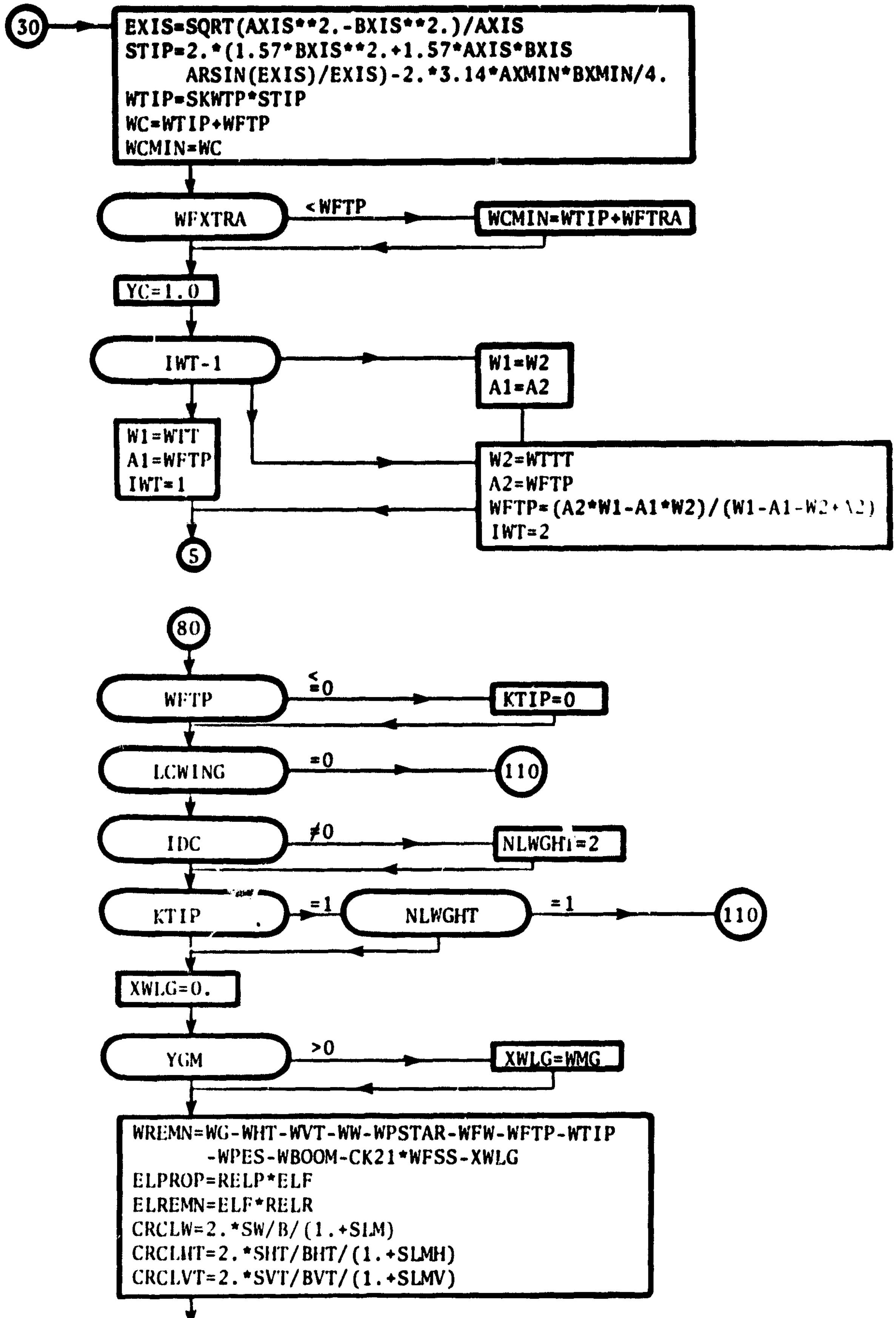
FIGURE V.3-1 DETAILED FLOW CHART SUBROUTINE WGHT


4
FH=WG*SKY*ELF*BHT*SKTL*(1.+2.*SLMH)/(1000000.*(1.+SLMH))
FV=WG*SKZ*(ELF+B)*BVT*(1.+2.*SLMV)/(1000000.*(1.+SLMV))
FV=FV/2.
WHT=350.*(SHT*FH*ALOG10(VDMIN)*(1.+SLMH)/(200.*ELTH*TCHT*CBARHT))**.54
13
WVT=380.*((FV+SAH*FH/2.)*SVT*ALOG10(VDMIN)*(1.+SLMV)
/(200.*ELTV*TCVT*CBARVT))**.54
5
WR=2.*(YP*(WEP+WEPI+WPES+WPROP)+WMG+WCMIN*YC)
99
TSWC2=TAN(RLMC4)-(1.-SLM)/(1.+SLM)/AR
SWC2=ATAN(TSWC2)
CSWC2=COS(SWC2)
BSTR=B/CSWC2
SKNO=1.+2.5/SQRT(BSTR)
SKSTR=1.-STRUT*STRUT
SKGEAR=1.0
YMG
≤0
SKGEAR=0.95
SKEPOS=1.0
YP
>0
ENP
=2 or 3
SKEPOS=.98
YP
>0
ENP
=4
SKEPOS=0.95
14
EMMO
<.75
SKEPOS=1.05
YP
>0
ENP
=2 or 3
SKEPOS=.95
YP
>0
ENP
=4
SKEPOS .90
WW1=.08*WG+WHLDEV
145
FOO=(SKSTR*ULF*(WG-.8*WW1))**.757
WWO=(SKWW*SKNO*SKEPOS*SKGEAR*FOO*B**1.049*(1.+SLM)
.4)/(1000000.*TCR.4*CSWC2**1.535)
WW=WWO+WHLDEV

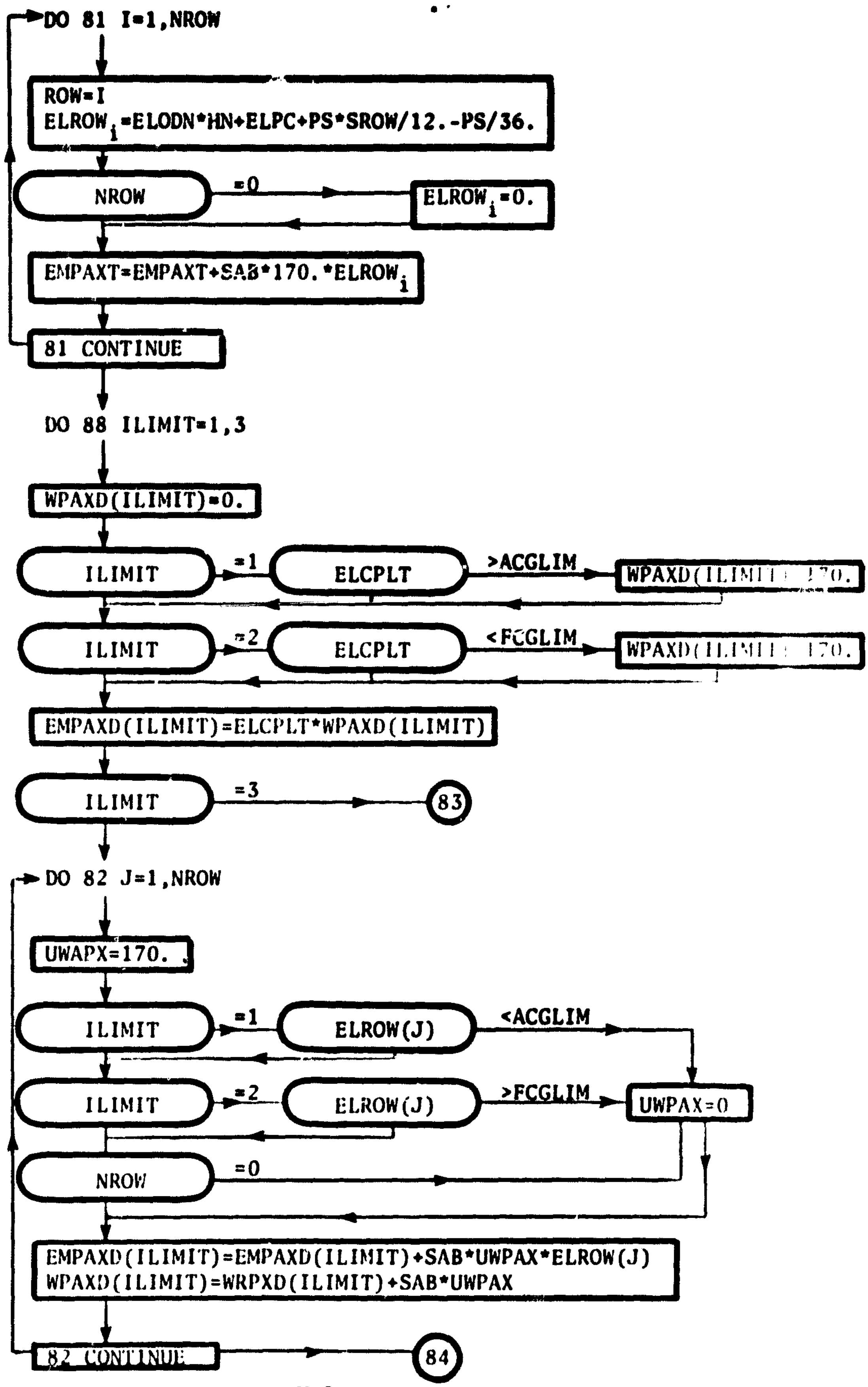
```
ABS(WW1-WW)/WW)   ≤.005 →
WW1=WW → 145

WM=YP/(YP+.001)*(WEP+WPEI+WPES+WPROP)+WMG*YMG/
   (YMG+.001)+WCMIN*YC?(YC+.001)
WX=WG-WW-WFWMIN-WM
WBOOM=SKBM*ELBM*SQRT(XARBM)*WG/1000.
WB=SKB*((WX/10000.)**.7*(SF/1000.)*SWF*(ELFFC+ELRW)**.5*ALOG10
  (VDMIN)*(DELP+1.)**2.*ULF**.3)**.508+WBOOM
WST=CK8*WW+CK9*WHT+CK10*WVT+CK11*WB+CK12*WLG+CK14*WPES+DELWST+WTIP
QDIV=(1.15*VDMIN)**2/391.
WSCX=SKFW*SW**.317*(WG/1000.)**.602*ULF**.525*QDIV**.345
WCC=SKCC*(WG/1000.)**.41
WCFW=WSCX-WCC
WSAS=SKSAS
WFC=CK15*WCC+CK18*WCFW+CK19*WSAS+DELWFC
WFADES=(WG-WPSTAR-WST-WFC-WFE-WFUL-WPLDES)/(1.+CK21*SKFS*6.687/FUED)
       /(1.+CK21*SKFS*6.687/FUELD)
WFSS=CK21*(6.687/FUELD)*SKFS*WFADES
WP=WPSTAR+WFSS
WFA=WG-WPSTAR-WST-WFC-WFE-WFUL-WPL-WFSS
WFAMIN=WG-WPSTAR-WST-WFC-WFE-WFUL-WPLMAX-WFSS
WFADES=WFA
WFAMAX=WFADES

IWT   ≠0 → 60

WTEST=WFW
TC=((TCR-SWF/B*(TCR-TCT))*(1.-SWF/B*(1.-SLM))
   +SLM*TCT)/(1.+SLM-SWF/B(1.-SLM))
FVOLW=(SKWF*.888889*TC*(SW**1.5)*(2.*SLM+1.))/((AR**.5)*((SLM+1.)**2.))

WFWMX=WFW
WFXTRA=0.

WFA       <WFW     → WFW=WFA
WFAMIN    <WFWMIN  → WFWMIN=WFAMIN
WFAMIN    ≤WFWMIN  → WFAMIN=WFWMIN

WFWMIN=WFW
WFXTRA=WFAMIN-WFW
```

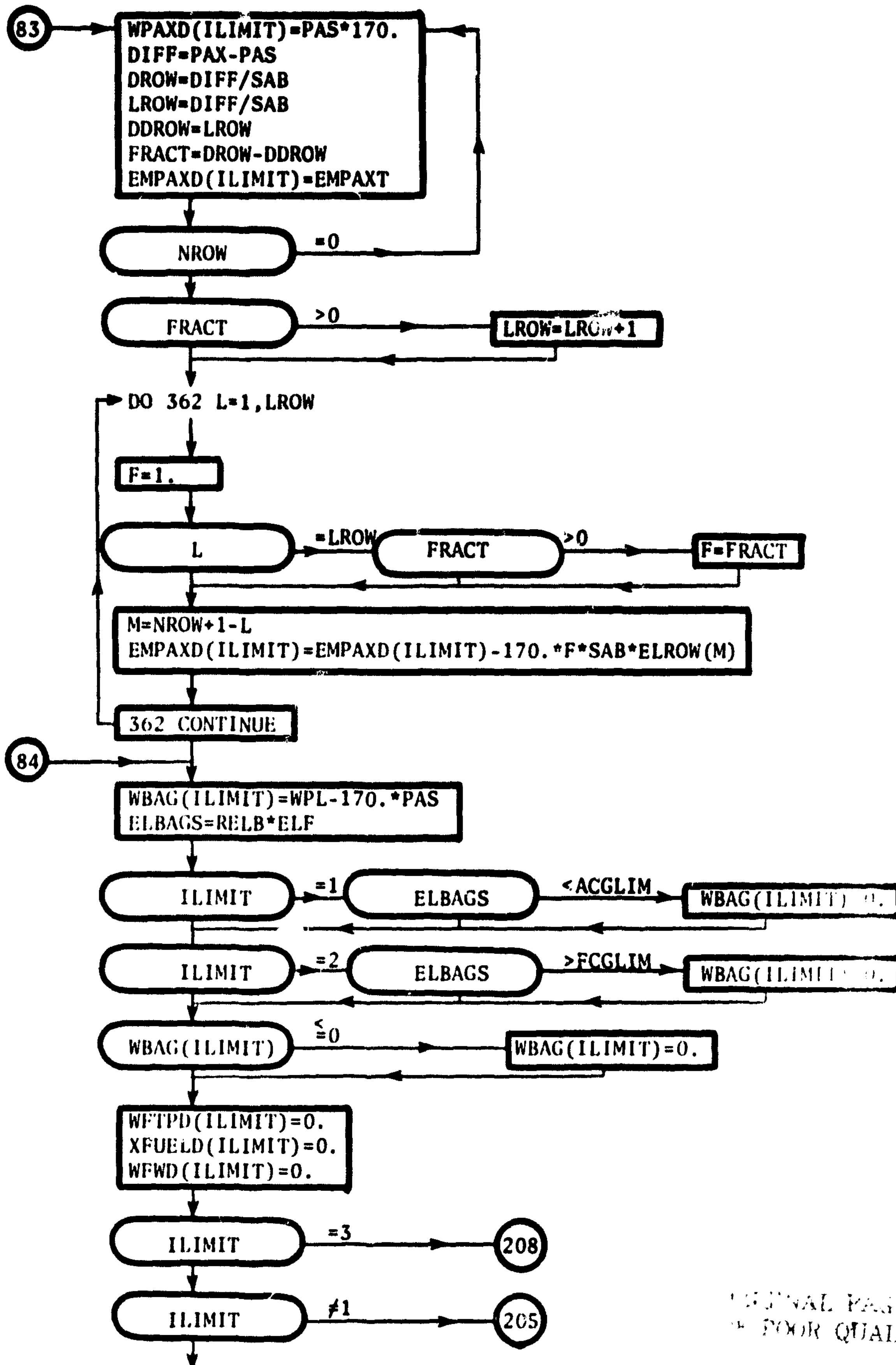

ABS(WTEST-WFW)
≤10
99
LC1-1
+
W1=W2
A1=A2
-
0
W1=WTEST
A1=WFW
LC1=1
W2=WTEST
A2=WFW
WFW=(A2*W1-A1*W2)/(W1-A1-W2+A2)
LC1=2
99
60
KTIP
=0
WFADES
≤WFWMX
LC1
=99
WTTT=WFTP
WFTP=WFADES-WFWMX
ABS(WTTT-WFTP)
≤10
80
VFTP=WFTP/(FUELD*7.48052
IWT
≠0
62
CHR=3.*CBARW/(2.*(1.+SLM-(SLM/(1.+SLM))))
CHT=SLM*CHR
AXMAX=ATMXQC*CHT
AXMIN=CHT
BXMIN=TCT*CHT
BXMAX=AXMAX/LDCKMX
VFTPMN=SKFT*1.046*AXMIN*BXMIN**2.
VFTPMX=SKFT*1.046*AXMAX*BXMAX
VFTP
≤VFTPMN
80
AXIS=AXMIN
26


BXIS=SQRT(.956*VFTP/(AXIS*SKFT))
BXIS
>BXMAX
LDCK=AXIS/BXIS
LDCK
≥(LDCKMX-1.) and ≤LDCKMX
BXIS
≤BXMIN
AXIS
>ANMAX
AXIS=AXIS+.25
AXIS=AXMAX
BXIS=SQRT(.956*VFTP/(AXIS*SKFT))
BXIS=BXMAX
VFTP
<VFTPMX
VFTP=VFTPMX
AXIS=AXMAX
WFTP=FUELD*7.48052*VFTP
LC1=99
AXIS=.956*VFTP/(SKFT*BXIS**2.)
XAXIS=AXIS
AXIS=AXIS+.25
AXIS
<AXMAX
26
AXIS=XAXIS
BXIS=BXMIN
AXIS=.956*VFTP/(SKFT*BXIS**2.)


30
EXIS=SQRT(AXIS**2.-BXIS**2.)/AXIS
STIP=2.*(1.57*BXIS**2.+1.57*AXIS*BXIS
ARSIN(EXIS)/EXIS)-2.*3.14*AXMIN*BXMIN/4.
WTIP=SKWTP*STIP
WC=WTIP+WFTP
WCMIN=WC
WFXTRA
<WFTP
WCMIN=WTIP+WFTRA
YC=1.0
IWT-1
W1=W2
A1=A2
W1=WTT
A1=WFTP
IWT=1
W2=WTTT
A2=WFTP
WFTP=(A2*W1-A1*W2)/(W1-A1-W2+A2)
IWT=2
5
80
WFTP
≤0
KTIP=0
LCWING
=0
110
IDC
≠0
NLWGHT=2
KTIP
=1
NLWGHT
=1
110
XWLG=0.
YGM
>0
XWLG=WMG
WREMN=WG-WHT-WVT-WW-WPSTAR-WFW-WFTP-WTIP
-WPES-WBOOM-CK21*WFSS-XWLG
ELPROP=RELP*ELF
ELREMN=ELF*RELR
CRCLW=2.*SW/B/(1.+SLM)
CRCLHT=2.*SHT/BHT/(1.+SLMH)
CRCLVT=2.*SVT/BVT/(1.+SLMV)

```
TSWPLE=(1.-SLM)/(1.+SLM)/AR+TAN(RLMC4)
TSWHLE=(1.-SLMH)/(1.+SLMH)/ARHT+TAN(SWPQCH)
TSWVLE=(1.-SLMV)/(1.+SLMV)/ARVT+TAN(SWPQCV)
XMAC=B*TSWPLE*(1.+2.*SLM)/6./(1.+SLM)
XHMAC=BHT*TSWHLE*(1.+2.*SLMH)/6./(1.+SLMH)
XVMAC=BVT*TSWVLE*(1.+2.*SLMV)/6./(1.+SLMV)

ELC4W=ELF-CRCLVT+XVMAC+.25*CBARVT-ELTV-.25*CBARW-XMAC+.25*CRCLW

ELPROP  =0  -> ELPROP=ELC4W-.25*CRCLW+XMAC+
                 EGMRGN*CBARW

CHT=SLM*3.*CBARW/(2.*(1.+SLM-(SLM/(1.+SLM))))
ELTIP=ELC4W-.25*CRCLW+XMAC+.25*CBARW
ELTIPE=ELTIP+.5*AXIS
ELTPMX=ELC4W+B*TAN(RLMC4)/2.+.75*CHT

ELTIPE  <ELTPMX  -> ELTIP=ELTPMX-.5*AXIS

WFTP  =0  -> ELTIP=0.

ELCGV=ELC4W-.25*CRCLW+XMAC+.25*CBARW+ELTV+.08*CBARVT
ELCGH=ELCGV-XVMAC-.25*CBARVT+.25*CRCLVT+SAH*BVT*TAN(SWPQCV)
      -.25*CRCLHT+XHMAC+.25*CBARHT+ELINC
YMAC=B*(1.+2.*SLMH)/6./(1.+SLM)
YHMAC=BHT*(1.+2.*SLMH)/6./(1.+SLMH)

LCWING  ≠2  -> (100)

FCGLIM=ELWING-(CPMRGN+.25-XCGFWD)*CBARW
ACGLIM=ELWING-(CPMRGN+.52-XCAFT)*CBARW
WFUELM=HPMSLS/2.

NTYE  =7  -> WFUELM=.2*WFW

OWE=WP+WFC+WST+WFE+WFUL
WREMN1=WFUL+WFC+WB+WFE+WLG-XWLG
EMOWE=ELPROP*(WPSTAR+WPES)+ELCGH*WHT+ELCGV*WVT+ELWING*(WW+XWLG+CK2
      *WFSS)+WBOOM*(ELF-.5*ELBM)+ELTIP*WTIP+ELREMN*WREMN1
CGOWE=EMOWE/OWE
NPAX=WPL/200.
PAS=NPAX
NROW=(PAX-1.)/SAB
ELCPLT=ELODN*HN+ELPC/2.
EMPAXT=ELCPLT*170.
WPAX=PAX*170.
```


DO 81 I=1,NROW
ROW=I
ELROW$_i$=ELODN*HN+ELPC+PS*SROW/12.-PS/36.
NROW
=0
ELROW$_i$=0.
EMPAXT=EMPAXT+SAB*170.*ELROW$_i$
81 CONTINUE
DO 88 ILIMIT=1,3
WPAXD(ILIMIT)=0.
ILIMIT
=1
ELCPLT
>ACGLIM
ILIMIT
=2
ELCPLT
<FCGLIM
EMPAXD(ILIMIT)=ELCPLT*WPAXD(ILIMIT)
ILIMIT
=3
83
DO 82 J=1,NROW
UWAPX=170.
ILIMIT
=1
ELROW(J)
<ACGLIM
ILIMIT
=2
ELROW(J)
>FCGLIM
UWPAX=0
NROW
=0
EMPAXD(ILIMIT)=EMPAXD(ILIMIT)+SAB*UWPAX*ELROW(J)
WPAXD(ILIMIT)=WRPXD(ILIMIT)+SAB*UWPAX
82 CONTINUE
84



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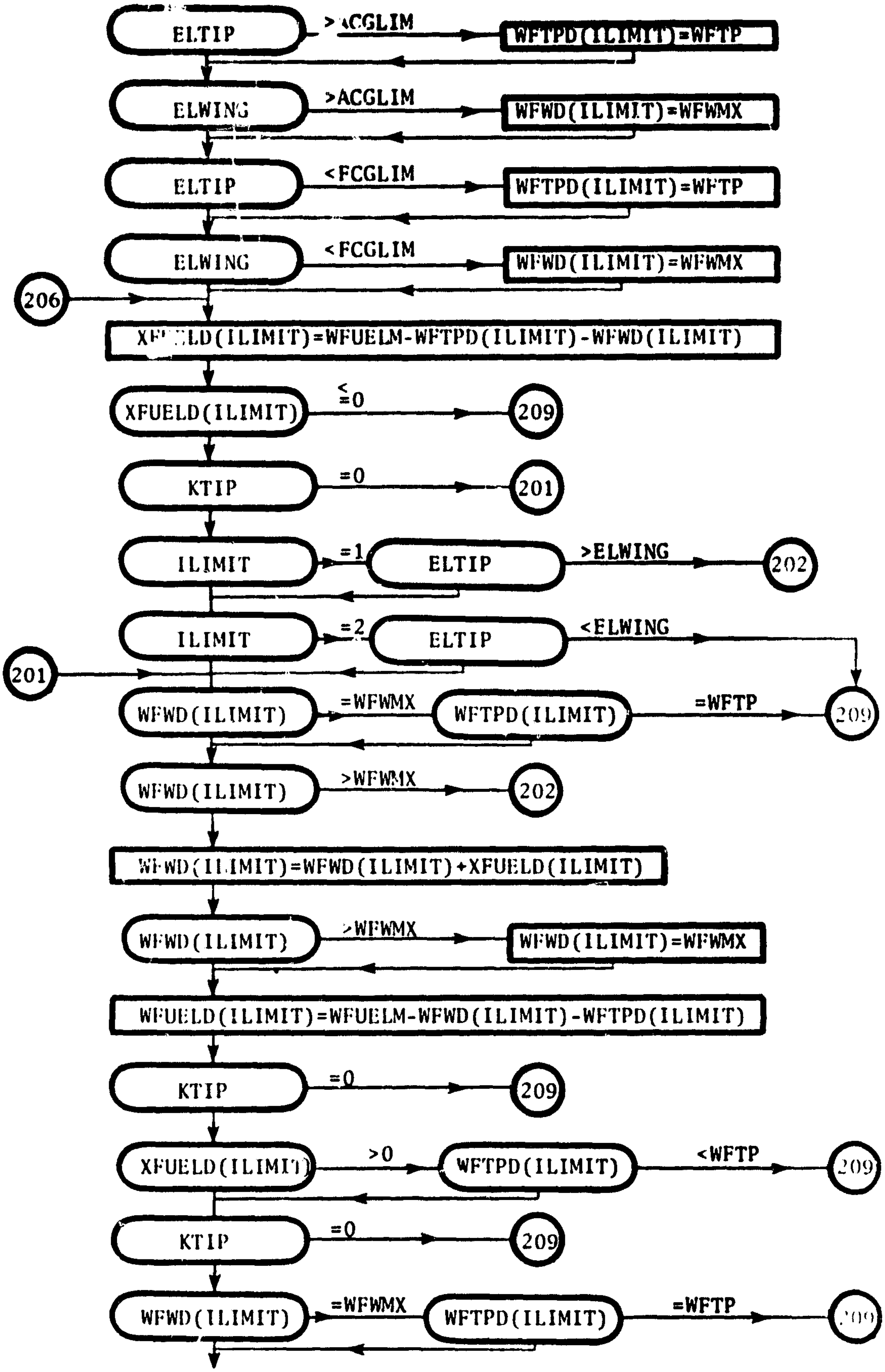
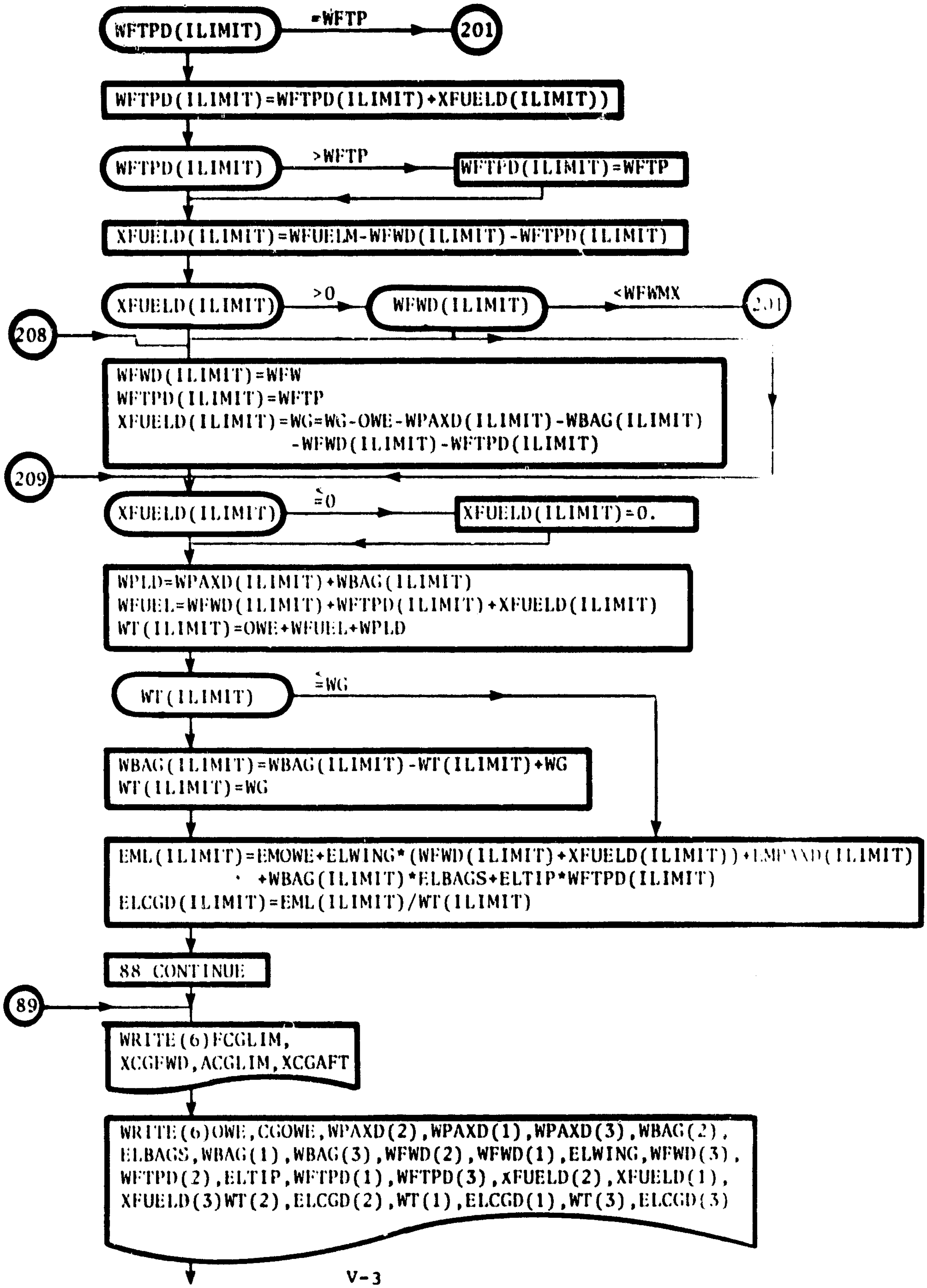
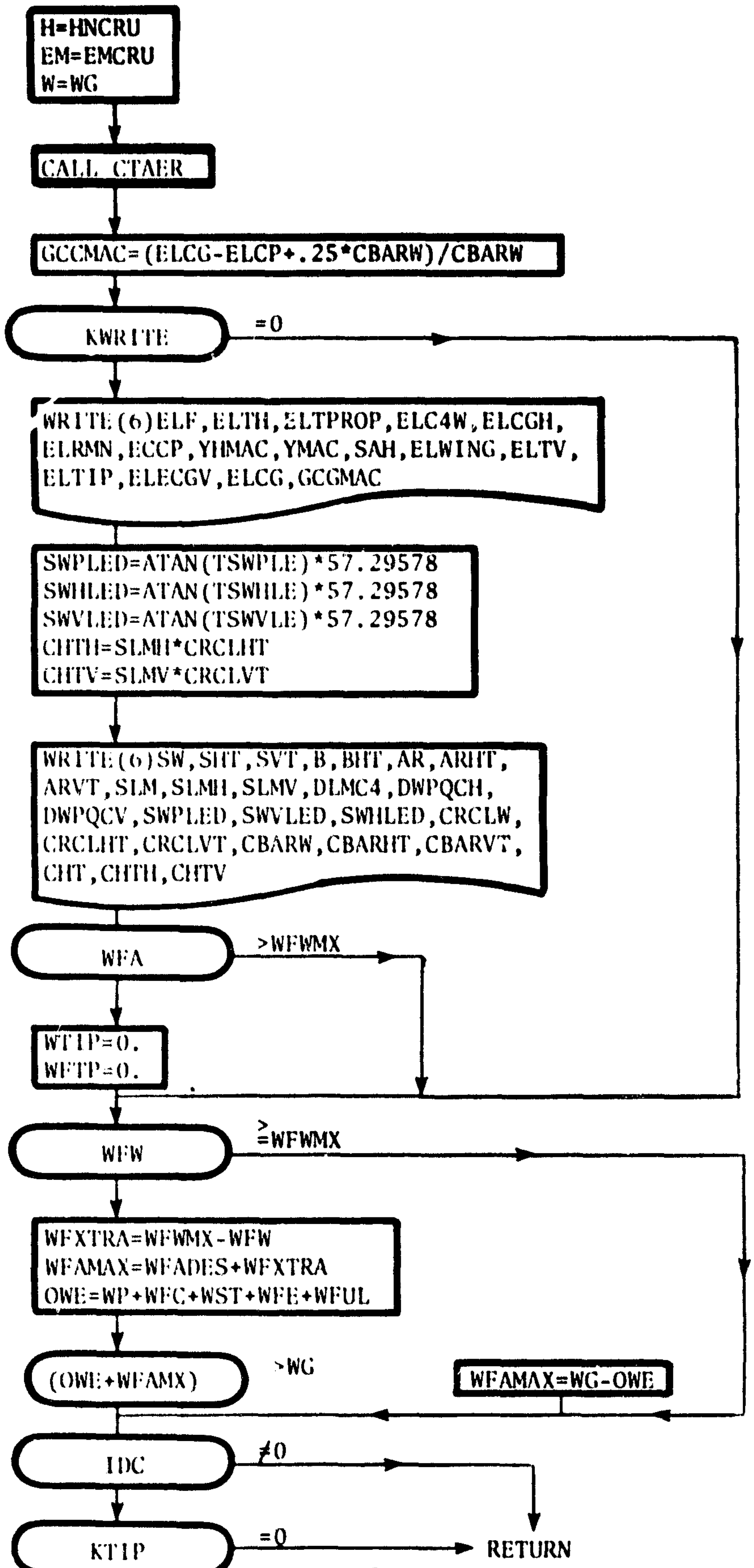
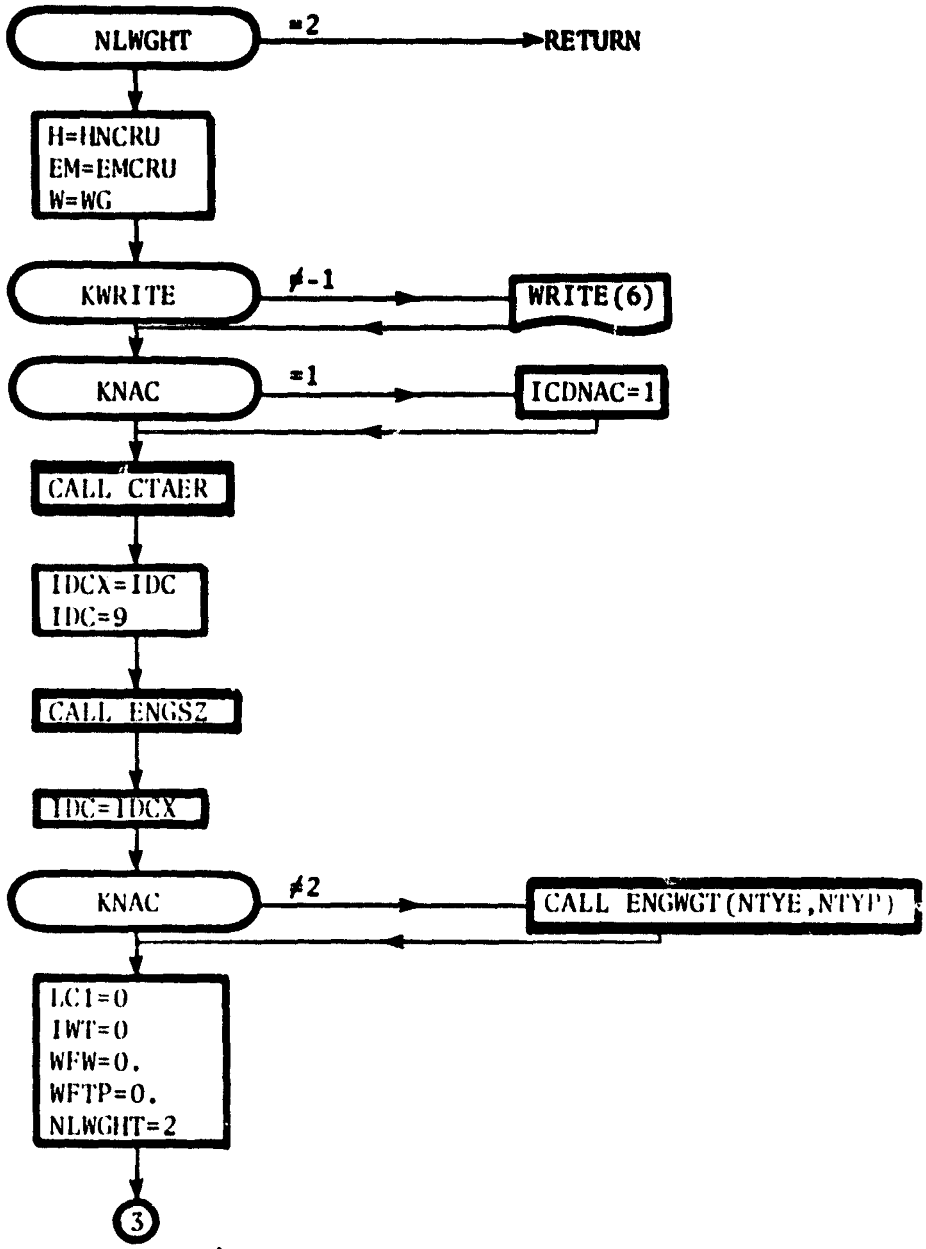

ELTIP
>ACGLIM
WFTPD(ILIMIT)=WFTP
ELWING
>ACGLIM
WFWD(ILIMIT)=WFWMX
ELTIP
<FCGLIM
WFTPD(ILIMIT)=WFTP
ELWING
<FCGLIM
WFWD(ILIMIT)=WFWMX
206
XFUELD(ILIMIT)=WFUELM-WFTPD(ILIMIT)-WFWD(ILIMIT)
XFUELD(ILIMIT)
≤0
209
KTIP
=0
201
ILIMIT
=1
ELTIP
>ELWING
202
ILIMIT
=2
ELTIP
<ELWING
201
WFWD(ILIMIT)
=WFWMX
WFTPD(ILIMIT)
=WFTP
209
WFWD(ILIMIT)
>WFWMX
202
WFWD(ILIMIT)=WFWD(ILIMIT)+XFUELD(ILIMIT)
WFWD(ILIMIT)
>WFWMX
WFWD(ILIMIT)=WFWMX
WFUELD(ILIMIT)=WFUELM-WFWD(ILIMIT)-WFTPD(ILIMIT)
KTIP
=0
209
XFUELD(ILIMIT)
>0
WFTPD(ILIMIT)
<WFTP
209
KTIP
=0
209
WFWD(ILIMIT)
=WFWMX
WFTPD(ILIMIT)
=WFTP
209


WFTPD(ILIMIT)
=WFTP
201
WFTPD(ILIMIT)=WFTPD(ILIMIT)+XFUELD(ILIMIT))
WFTPD(ILIMIT)
>WFTP
WFTPD(ILIMIT)=WFTP
XFUELD(ILIMIT)=WFUELM-WFWD(ILIMIT)-WFTPD(ILIMIT)
XFUELD(ILIMIT)
>0
WFWD(ILIMIT)
<WFWMX
201
208
WFWD(ILIMIT)=WFW
WFTPD(ILIMIT)=WFTP
XFUELD(ILIMIT)=WG=WG-OWE-WPAXD(ILIMIT)-WBAG(ILIMIT)
-WFWD(ILIMIT)-WFTPD(ILIMIT)
209
XFUELD(ILIMIT)
≤0
XFUELD(ILIMIT)=0.
WPLD=WPAXD(ILIMIT)+WBAG(ILIMIT)
WFUEL=WFWD(ILIMIT)+WFTPD(ILIMIT)+XFUELD(ILIMIT)
WT(ILIMIT)=OWE+WFUEL+WPLD
WT(ILIMIT)
≤WG
WBAG(ILIMIT)=WBAG(ILIMIT)-WT(ILIMIT)+WG
WT(ILIMIT)=WG
EML(ILIMIT)=EMOWE+ELWING*(WFWD(ILIMIT)+XFUELD(ILIMIT))+EMPAXD(ILIMIT)
+WBAG(ILIMIT)*ELBAGS+ELTIP*WFTPD(ILIMIT)
ELCGD(ILIMIT)=EML(ILIMIT)/WT(ILIMIT)
88 CONTINUE
89
WRITE(6)FCGLIM,
XCGFWD,ACGLIM,XCGAFT
WRITE(6)OWE,CGOWE,WPAXD(2),WPAXD(1),WPAXD(3),WBAG(2),
ELBAGS,WBAG(1),WBAG(3),WFWD(2),WFWD(1),ELWING,WFWD(3),
WFTPD(2),ELTIP,WFTPD(1),WFTPD(3),XFUELD(2),XFUELD(1),
XFUELD(3)WT(2),ELCGD(2),WT(1),ELCGD(1),WT(3),ELCGD(3)


DELXCG=(ELCGD(1)-ELCGD(2))/CBARW
ELWING=ELCGD(2)+(CPMRGN+.25-XCGFWD)*CBARW
ELCP=ELWING-CBARW*CPMRGN
ELC4W=ELWING-CBARW*CPMRGN-YMAC*TAN(DLMC4*.017453)
ELCG=ELWING-CBARW*CPMRGN+STMRGN*CBARW
GCGAMC=(ELCG-ELCP+.25*CBARW)/CBARW
100
ELCG=(ELPROP*(WPSTAR+WPES)+(CBARW*(CPMRGN-STMRGN))
*(WW+WFW+XWLG+CK21*WFSS)+ELTIP+WFTP)+ELCGH*WHT
+ELCGV*WVT+ELREMN*WREMN+WBOOM*(ELF-.5*ELBM))/
(WG-WW-WFW-XWLG-CK21*WFSS)
ELWING=ELCG+(CBARW*(CPMRGN-STMRGN))
103
ELCP=ELCG-STMRGN*CBARW
ELTV=ELF-ELWING-CRCLVT+XVAMC+.25*CBARVT+CPMRGN*CBARW
ELTH=ELTV-XVMAC-.25*CBARVT+.25*CRCLVT+SAH*
BVT*TAN(SWPQCV)-.25*CRCLHT+XHMAC+.25*CBARHT+ELINC
SHTX=SHT
LCWING
≠2
CALL TAIL(ELC4W,XMAC,ELCG,XAMC,GCGMAC,CLALFN,DELXCG,XCGFWD,XCGAFT)
LCONT=LCONT+1
SHT=VBARH*SW*CBARW/ELTH
SVT=VBAV*SW*B/ELTV
BHT=SQRT(SHT*ARHT)
BVT=SQRT(SVT*ARVT)
CRCLHT=2.*SHT/BHT/(1.+SLMH)
CRCLVT=2.*SVT/BVT/(1.+SLMV)
CBARHT=(2.*CRCLHT/3.)*((1.+SLMH)-(SLMH/(1.+SLMH)))
CBARVT=(2.*CRCLVT/3.)*((1.+SLMV)-(SLMV/(1.+SLMV)))
LC1=0
IWT=0
ERR=ABS(SHT-SHTX)
123
KWRITE
≠1
WRITE(6)ERR,SHT,SHTX,
ELTH,WREMN,WHT,WVT,
WW,WPSTAR,WFW,WFTP
WTIP,WFA,WPES
ERR
<.4
LCONT
>15
LCONT
=16
WRITE (6)


H=HNCRU
EM=EMCRU
W=WG
CALL CTAER
GCCMAC=(ELCG-ELCP+.25*CBARW)/CBARW
KWRITE
=0
WRITE(6)ELF,ELTH,ELTPROP,ELC4W,ELCGH,
ELRMN,ECCP,YHMAC,YMAC,SAH,ELWING,ELTV,
ELTIP,ELECGV,ELCG,GCGMAC
SWPLED=ATAN(TSWPLE)*57.29578
SWHLED=ATAN(TSWHLE)*57.29578
SWVLED=ATAN(TSWVLE)*57.29578
CHTH=SLMH*CRCLHT
CHTV=SLMV*CRCLVT
WRITE(6)SW,SHT,SVT,B,BHT,AR,ARHT,
ARVT,SLM,SLMH,SLMV,DLMC4,DWPQCH,
DWPQCV,SWPLED,SWVLED,SWHLED,CRCLW,
CRCLHT,CRCLVT,CBARW,CBARHT,CBARVT,
CHT,CHTH,CHTV
WFA
>WFWMX
WTIP=0.
WFTP=0.
WFW
≥WFWMX
WFXTRA=WFWMX-WFW
WFAMAX=WFADES+WFXTRA
OWE=WP+WFC+WST+WFE+WFUL
(OWE+WFAMX)
>WG
WFAMAX=WG-OWE
IDC
≠0
KTIP
=0
RETURN


NLWGHT
=2
RETURN
H=HNCRU
EM=EMCRU
W=WG
KWRITE
≠-1
WRITE(6)
KNAC
=1
ICDNAC=1
CALL CTAER
IDCX=IDC
IDC=9
CALL ENGSZ
IDC=IDCX
KNAC
≠2
CALL ENGWGT(NTYE,NTYP)
LC1=0
IWT=0
WFW=0.
WFTP=0.
NLWGHT=2
3

V.3.1.2 Subroutine DLOAD, Design Load Factors

This subroutine is called by WGHT to obtain design speeds and load factors in accordance with Federal Aviation Regulations. Design speeds and load factors are then employed by WGHT in the structural weight analysis as described in Volume V, Part I of this report.

DLOAD

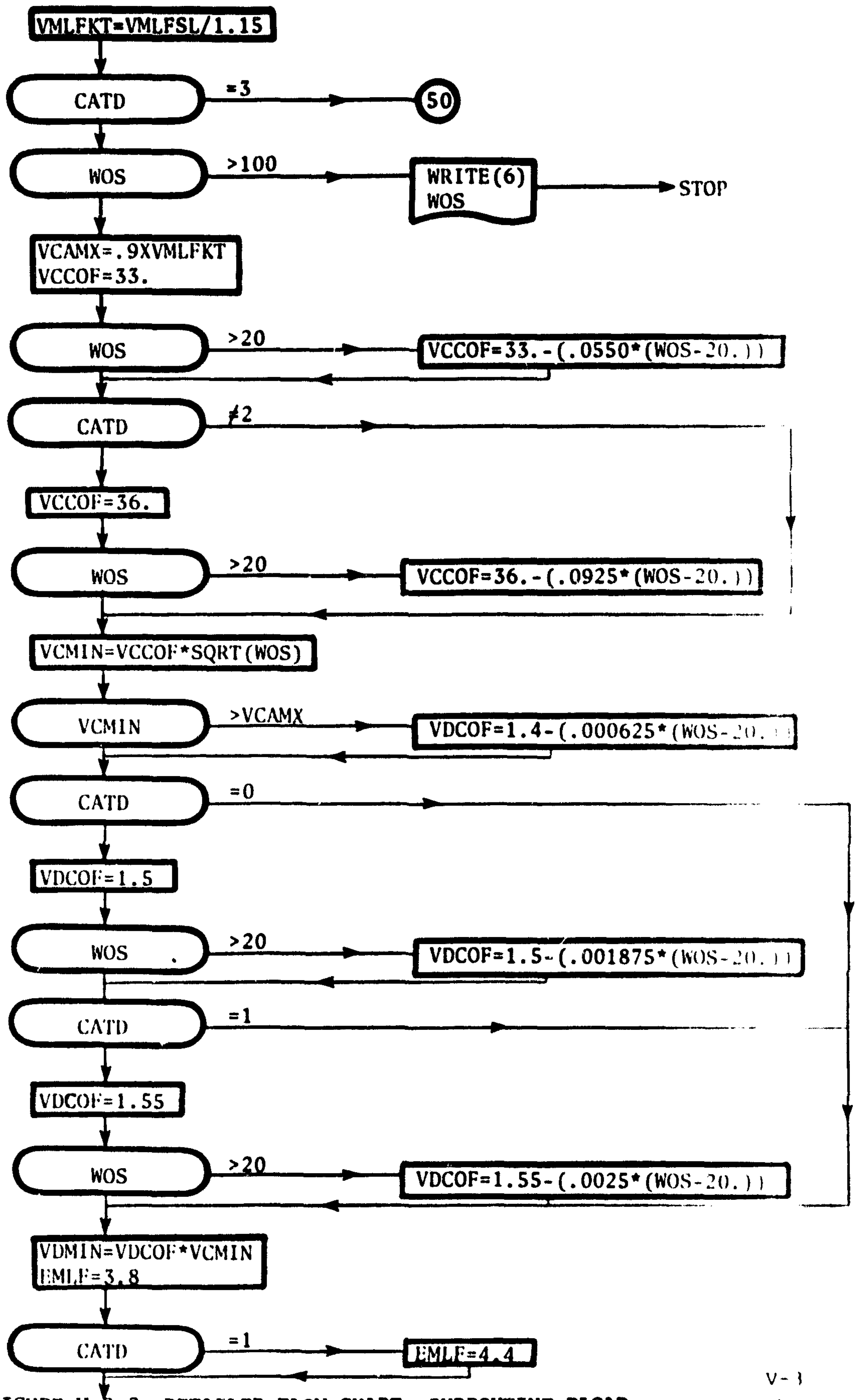
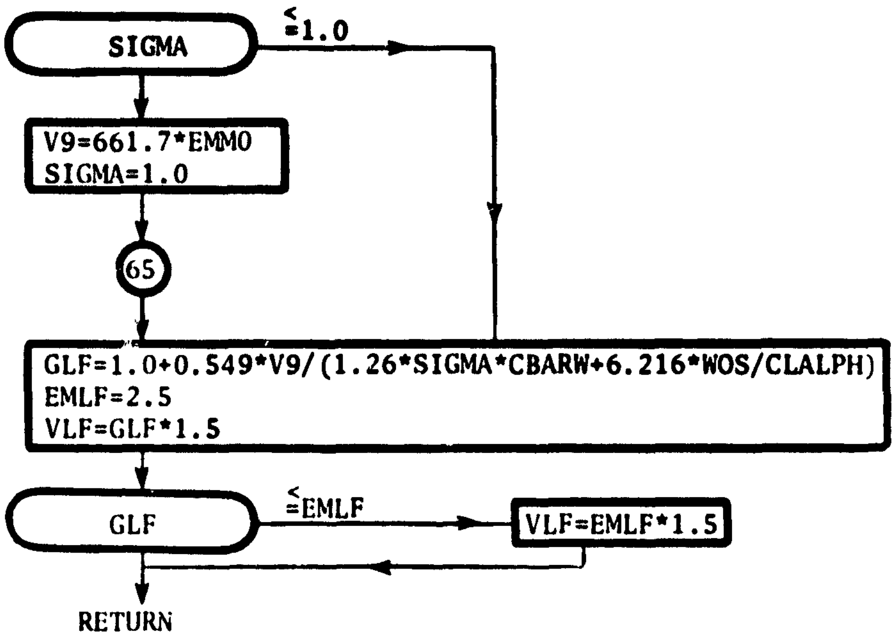


FIGURE V.3.2 DETAILED FLOW CHART, SUBROUTINE DLOAD


SIGMA
≤1.0
V9=661.7*EMMO
SIGMA=1.0
65
GLF=1.0+0.549*V9/(1.26*SIGMA*CBARW+6.216*WOS/CLALPH)
EMLF=2.5
VLF=GLF*1.5
GLF
≤EMLF
VLF=EMLF*1.5
RETURN

V.3.1.3 Subroutine TAIL, Tail Sizing Routine

Subroutine TAIL determines vertical and horizontal tail sizes on the basis of directional and longitudinal stability requirements. These requirements are based on Federal Aviation Regulations and current design practices as discussed in Volume V, Part I of the present report.

TAIL

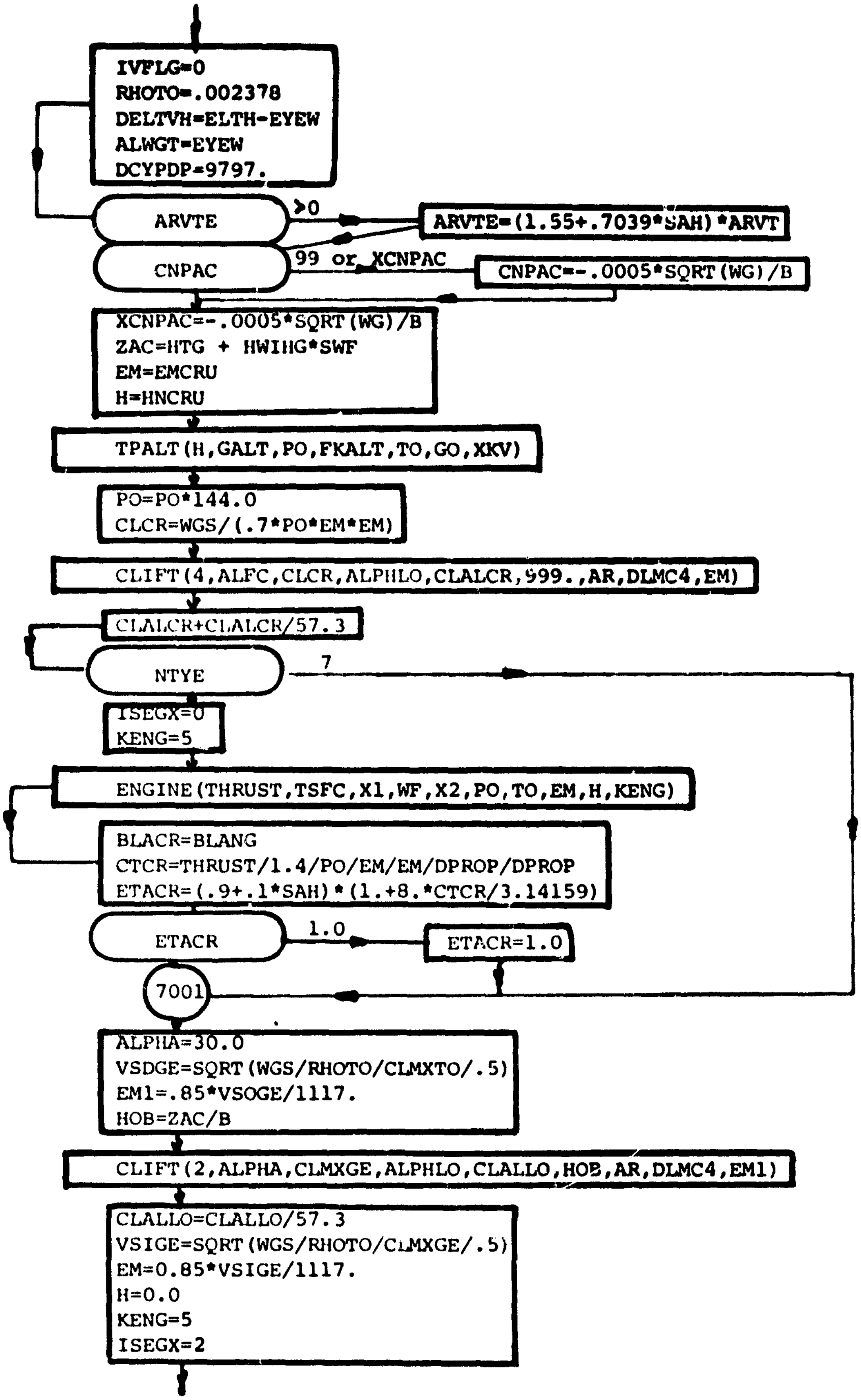
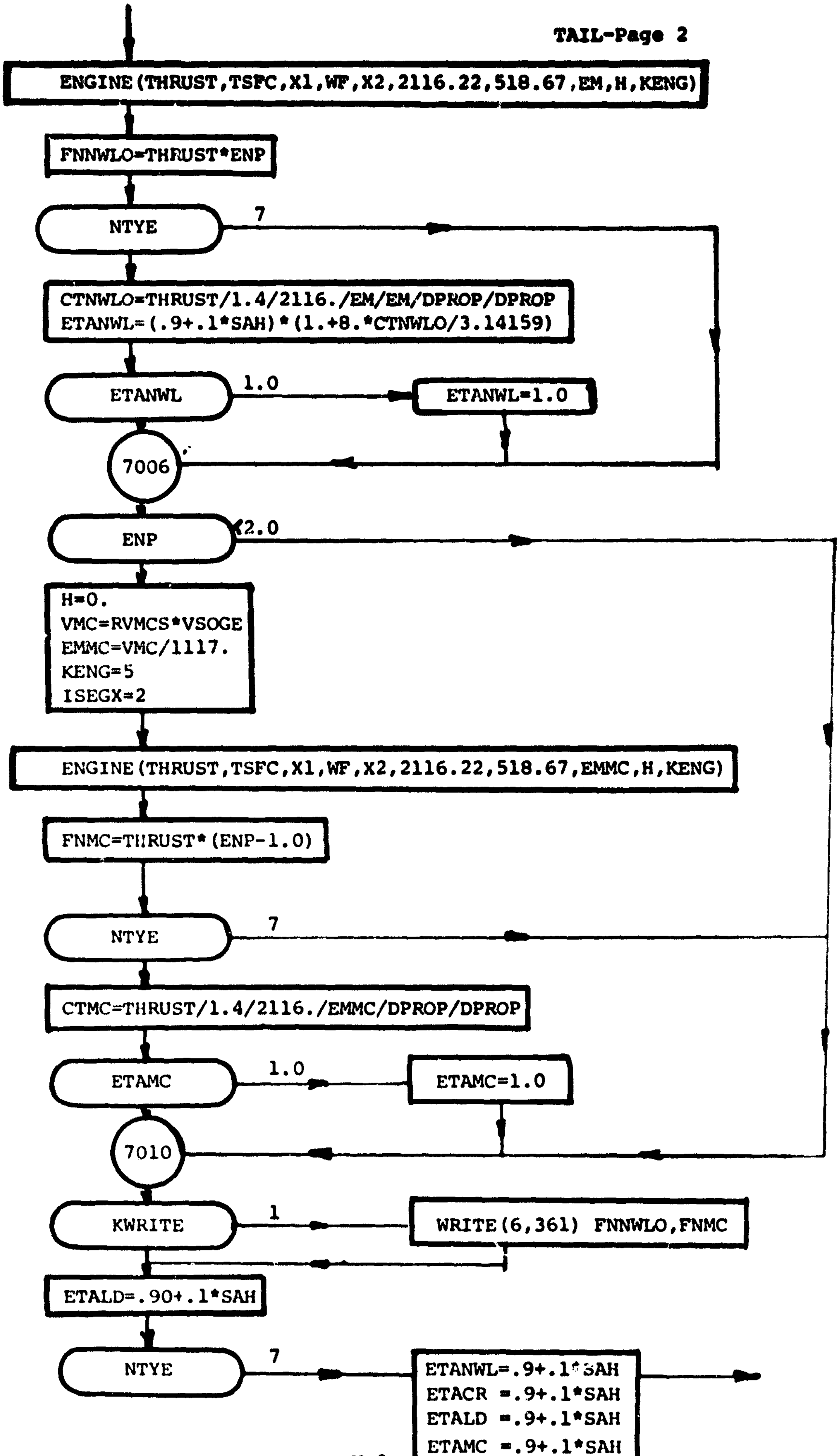
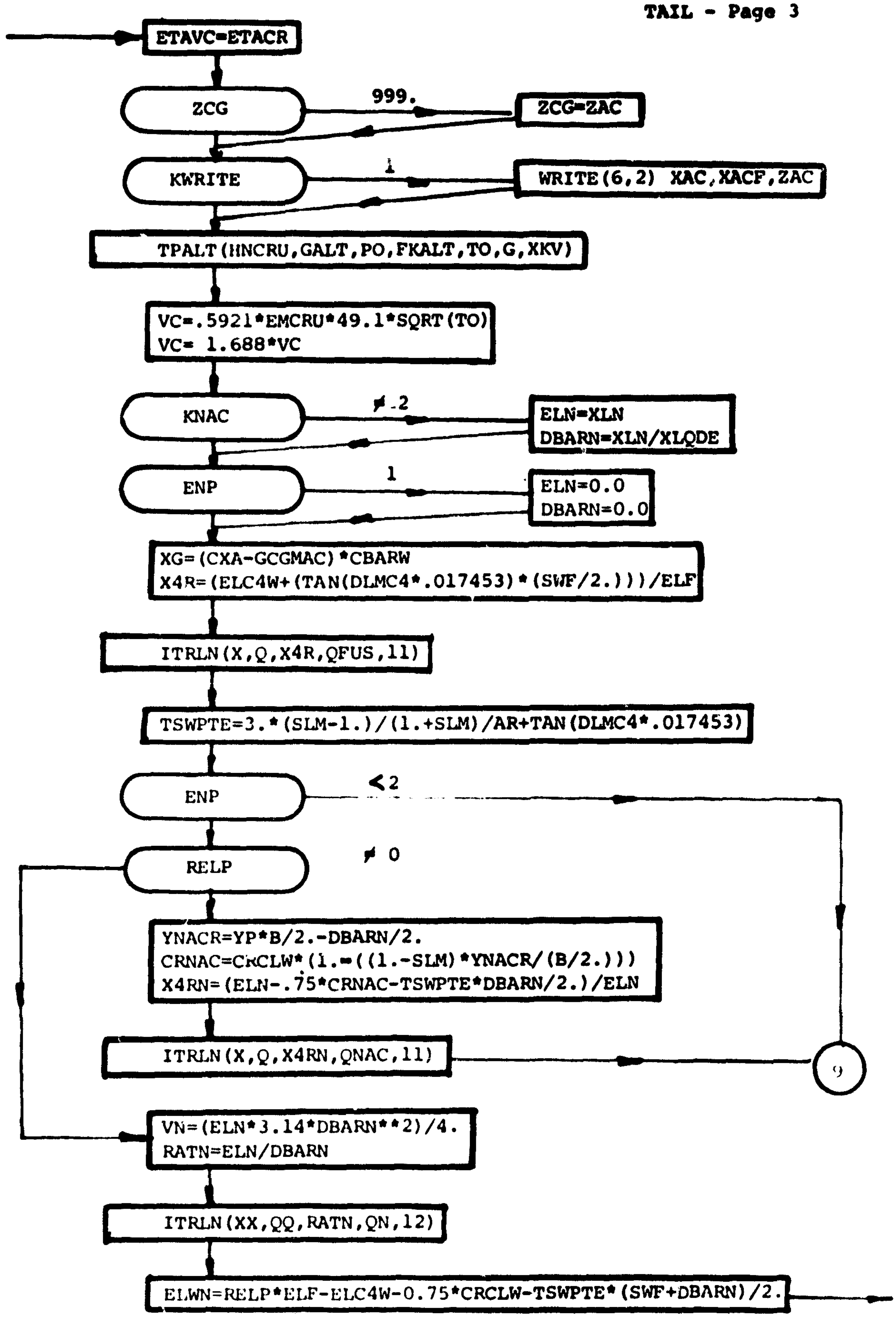
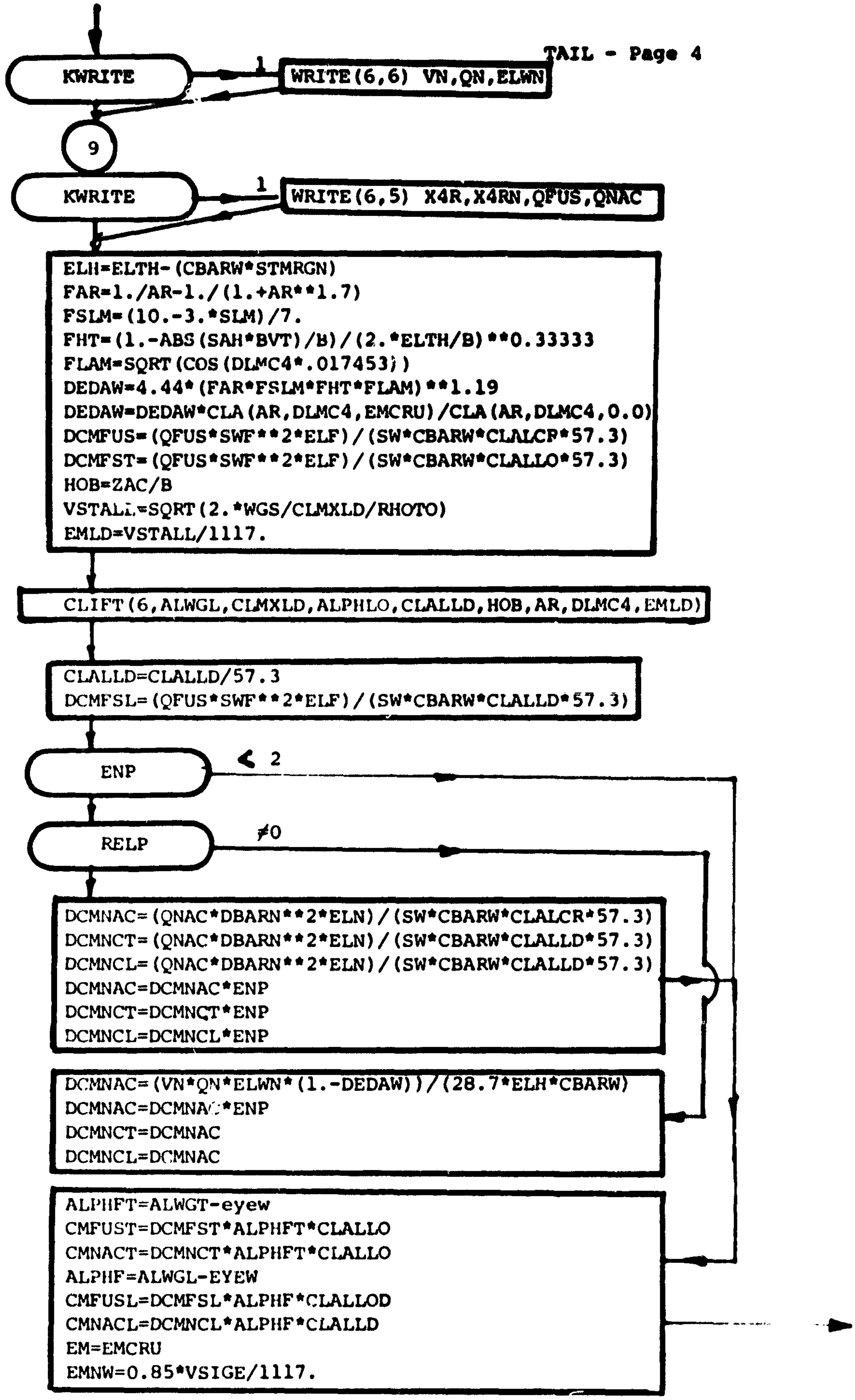
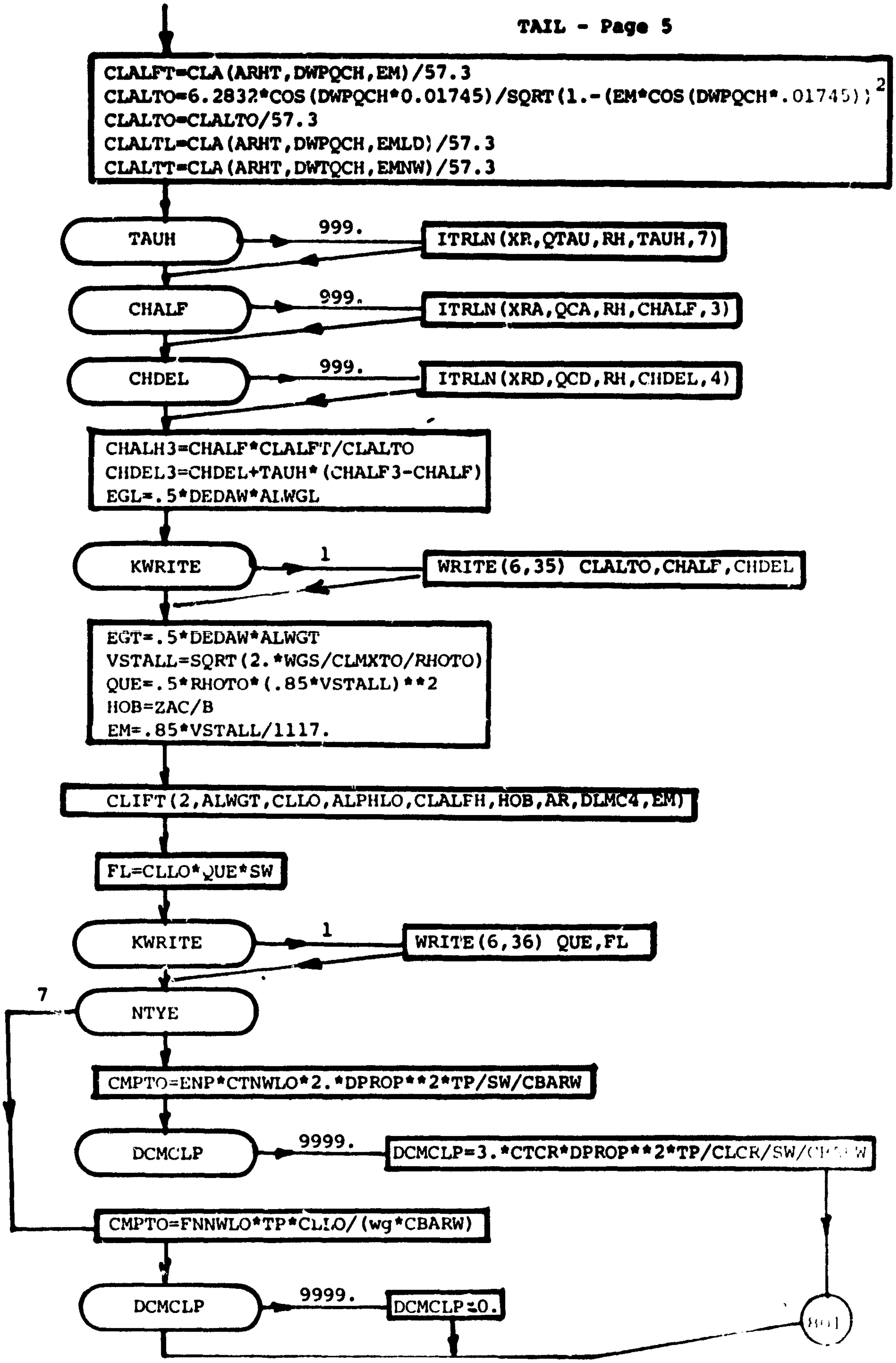


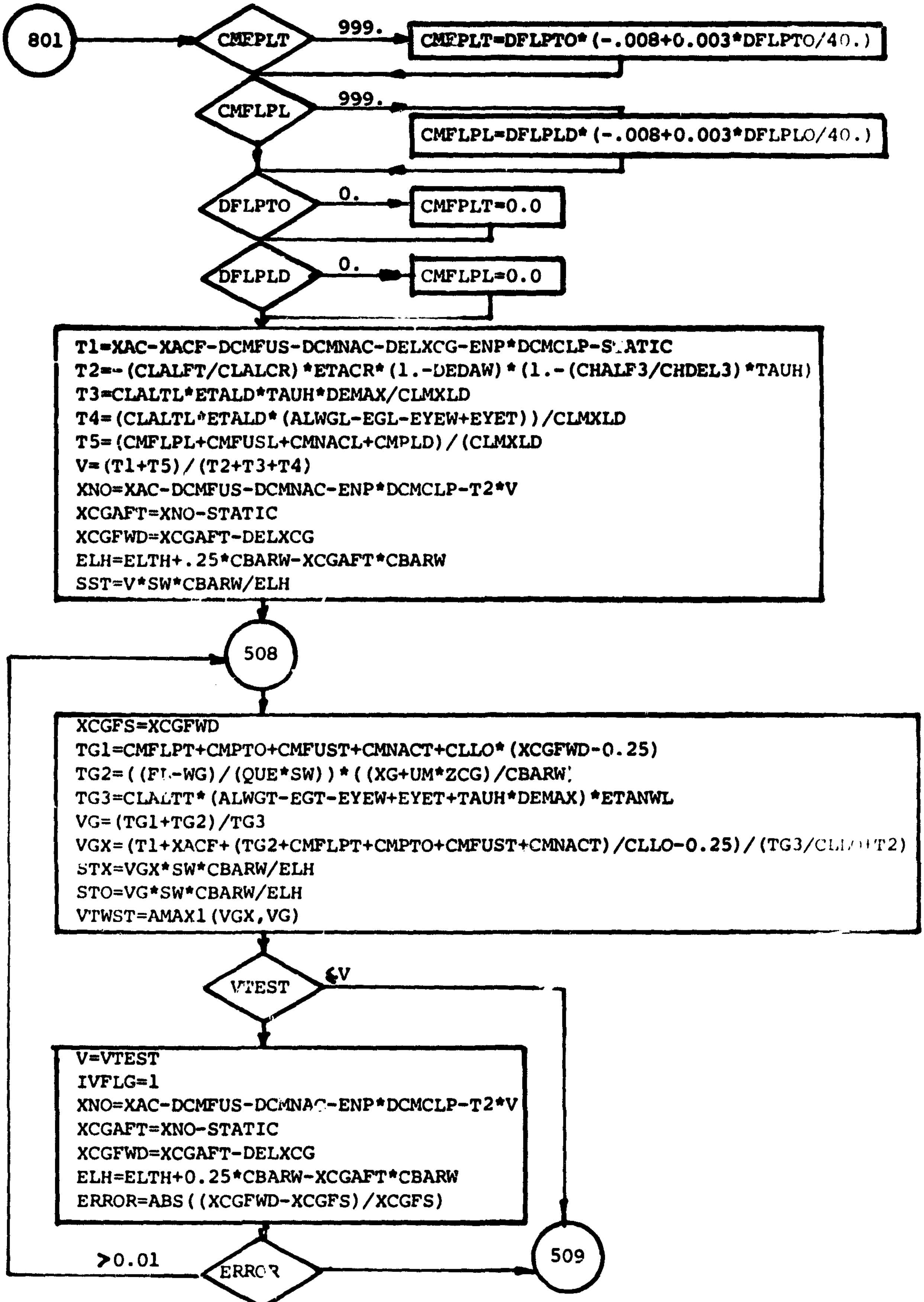
FIGURE V.3.3 DETAILED FLOW CHART, SUBROUTINE TAIL

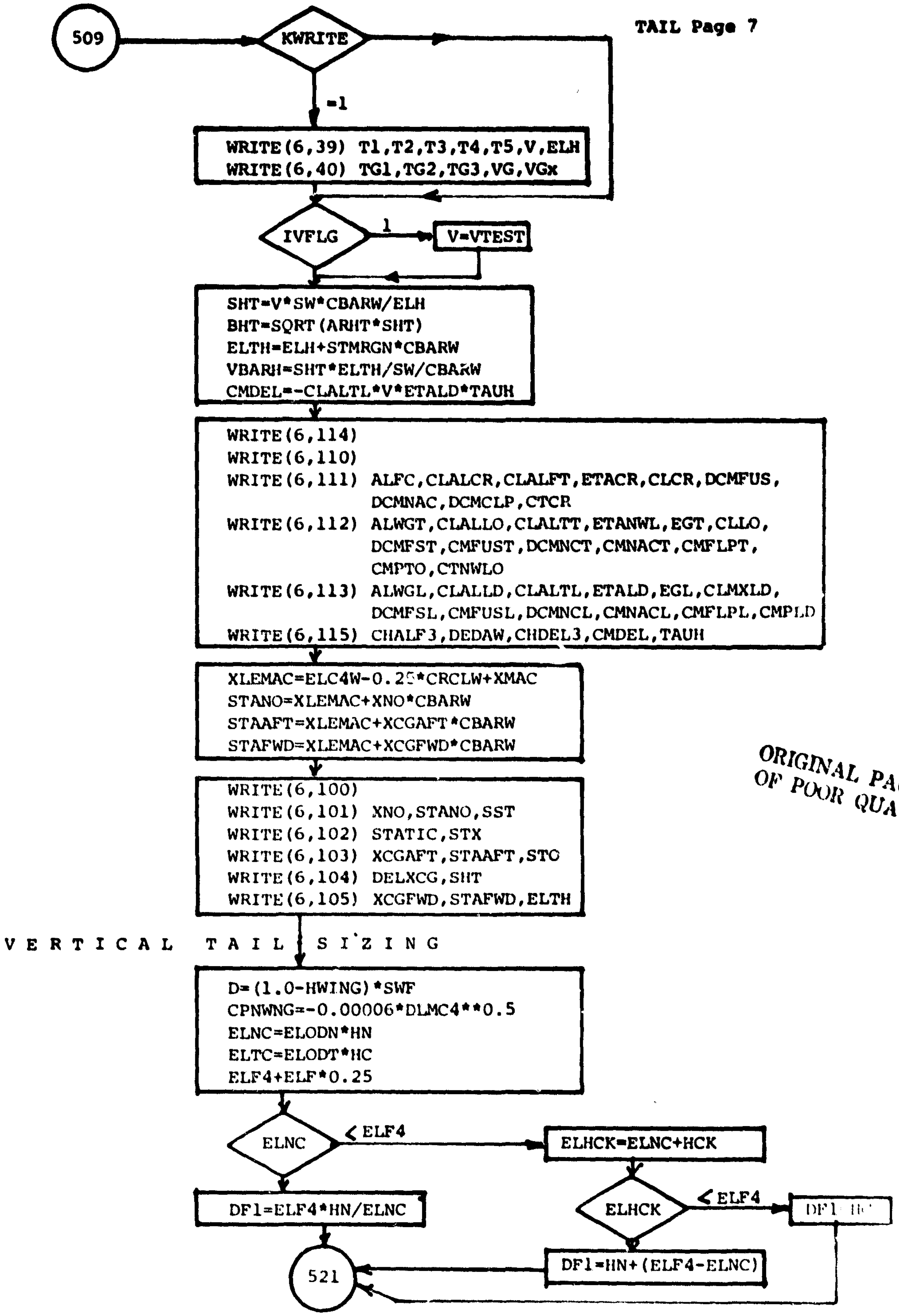

ENGINE(THRUST,TSFC,X1,WF,X2,2116.22,518.67,EM,H,KENG)
FNNWLO=THRUST*ENP
NTYE
7
CTNWLO=THRUST/1.4/2116./EM/EM/DPROP/DPROP
ETANWL=(.9+.1*SAH)*(1.+8.*CTNWLO/3.14159)
ETANWL
1.0
ETANWL=1.0
7006
ENP
2.0
H=0.
VMC=RVMCS*VSOGE
EMMC=VMC/1117.
KENG=5
ISEGX=2
ENGINE(THRUST,TSFC,X1,WF,X2,2116.22,518.67,EMMC,H,KENG)
FNMC=THRUST*(ENP-1.0)
NTYE
7
CTMC=THRUST/1.4/2116./EMMC/DPROP/DPROP
ETAMC
1.0
ETAMC=1.0
7010
KWRITE
1
WRITE(6,361) FNNWLO,FNMC
ETALD=.90+.1*SAH
NTYE
7
ETANWL=.9+.1*SAH
ETACR =.9+.1*SAH
ETALD =.9+.1*SAH
ETAMC =.9+.1*SAH

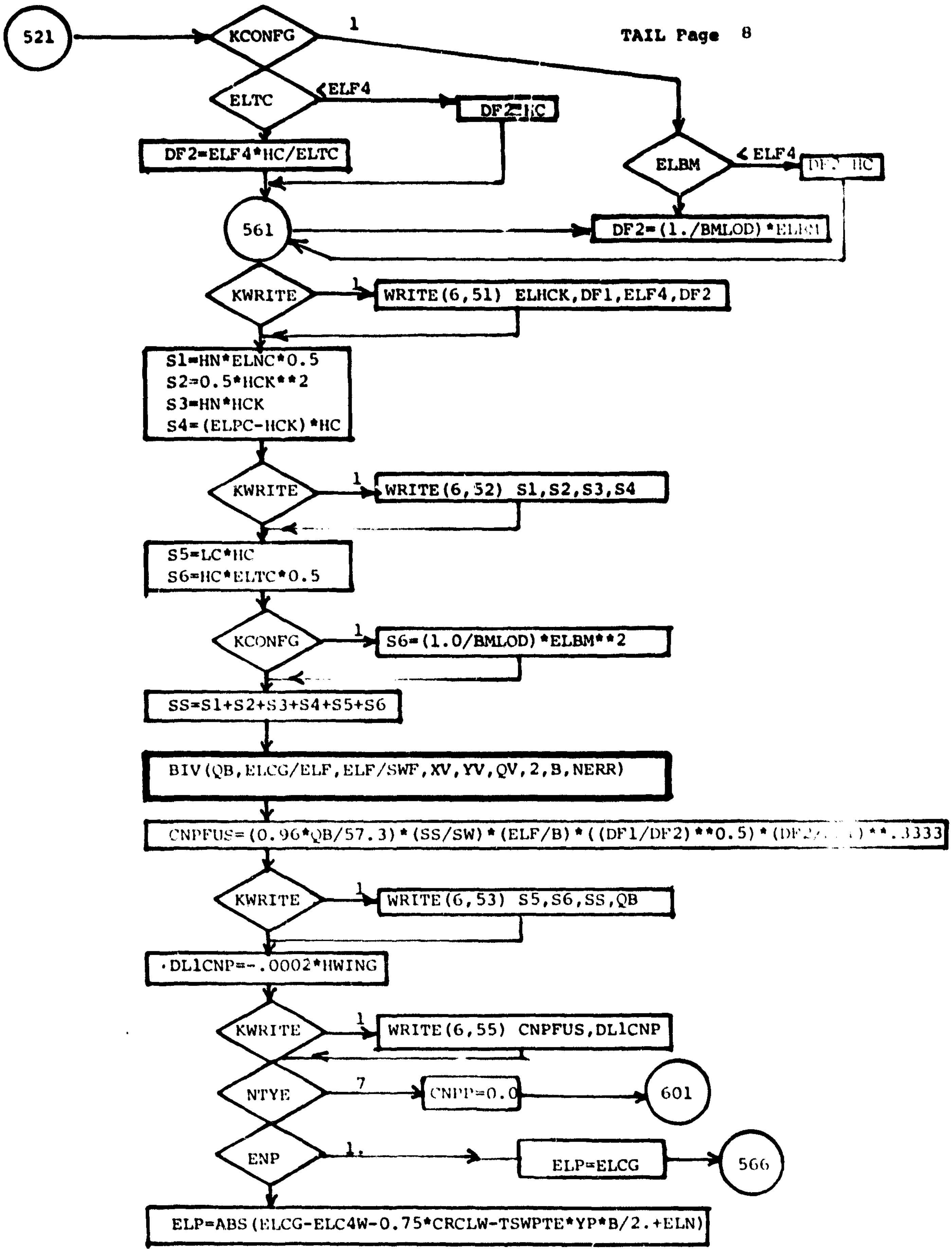

ETAVC=ETACR
ZCG
999.
ZCG=ZAC
KWRITE
1
WRITE(6,2) XAC,XACF,ZAC
TPALT(HNCRU,GALT,PO,FKALT,TO,G,XKV)
VC=.5921*EMCRU*49.1*SQRT(TO)
VC= 1.688*VC
KNAC
≠2
ELN=XLN
DBARN=XLN/XLQDE
ENP
1
ELN=0.0
DBARN=0.0
XG=(CXA-GCGMAC)*CBARW
X4R=(ELC4W+(TAN(DLMC4*.017453)*(SWF/2.)))/ELF
ITRLN(X,Q,X4R,QFUS,11)
TSWPTE=3.*(SLM-1.)/(1.+SLM)/AR+TAN(DLMC4*.017453)
ENP
<2
RELP
≠ 0
YNACR=YP*B/2.-DBARN/2.
CRNAC=CRCLW*(1.-((1.-SLM)*YNACR/(B/2.)))
X4RN=(ELN-.75*CRNAC-TSWPTE*DBARN/2.)/ELN
ITRLN(X,Q,X4RN,QNAC,11)
9
VN=(ELN*3.14*DBARN**2)/4.
RATN=ELN/DBARN
ITRLN(XX,QQ,RATN,QN,12)
ELWN=RELP*ELF-ELC4W-0.75*CRCLW-TSWPTE*(SWF+DBARN)/2.


KWRITE
1
WRITE(6,6) VN,QN,ELWN
9
KWRITE
1
WRITE(6,5) X4R,X4RN,QFUS,QNAC
ELH=ELTH-(CBARW*STMRGN)
FAR=1./AR-1./(1.+AR**1.7)
FSLM=(10.-3.*SLM)/7.
FHT=(1.-ABS(SAH*BVT)/B)/(2.*ELTH/B)**0.33333
FLAM=SQRT(COS(DLMC4*.017453))
DEDAW=4.44*(FAR*FSLM*FHT*FLAM)**1.19
DEDAW=DEDAW*CLA(AR,DLMC4,EMCRU)/CLA(AR,DLMC4,0.0)
DCMFUS=(QFUS*SWF**2*ELF)/(SW*CBARW*CLALCR*57.3)
DCMFST=(QFUS*SWF**2*ELF)/(SW*CBARW*CLALLO*57.3)
HOB=ZAC/B
VSTALL=SQRT(2.*WGS/CLMXLD/RHOTO)
EMLD=VSTALL/1117.
CLIFT(6,ALWGL,CLMXLD,ALPHLO,CLALLD,HOB,AR,DLMC4,EMLD)
CLALLD=CLALLD/57.3
DCMFSL=(QFUS*SWF**2*ELF)/(SW*CBARW*CLALLD*57.3)
ENP
< 2
RELP
≠0
DCMNAC=(QNAC*DBARN**2*ELN)/(SW*CBARW*CLALCR*57.3)
DCMNCT=(QNAC*DBARN**2*ELN)/(SW*CBARW*CLALLD*57.3)
DCMNCL=(QNAC*DBARN**2*ELN)/(SW*CBARW*CLALLD*57.3)
DCMNAC=DCMNAC*ENP
DCMNCT=DCMNCT*ENP
DCMNCL=DCMNCL*ENP
DCMNAC=(VN*QN*ELWN*(1.-DEDAW))/(28.7*ELH*CBARW)
DCMNAC=DCMNAC*ENP
DCMNCT=DCMNAC
DCMNCL=DCMNAC
ALPHFT=ALWGT-eyew
CMFUST=DCMFST*ALPHFT*CLALLO
CMNACT=DCMNCT*ALPHFT*CLALLO
ALPHF=ALWGL-EYEW
CMFUSL=DCMFSL*ALPHF*CLALLOD
CMNACL=DCMNCL*ALPHF*CLALLD
EM=EMCRU
EMNW=0.85*VSIGE/1117.


CLALFT=CLA(ARHT,DWPQCH,EM)/57.3
CLALTO=6.2832*COS(DWPQCH*0.01745)/SQRT(1.-(EM*COS(DWPQCH*.01745))2
CLALTO=CLALTO/57.3
CLALTL=CLA(ARHT,DWPQCH,EMLD)/57.3
CLALTT=CLA(ARHT,DWTQCH,EMNW)/57.3
TAUH
999.
ITRLN(XR,QTAU,RH,TAUH,7)
CHALF
999.
ITRLN(XRA,QCA,RH,CHALF,3)
CHDEL
999.
ITRLN(XRD,QCD,RH,CHDEL,4)
CHALH3=CHALF*CLALFT/CLALTO
CHDEL3=CHDEL+TAUH*(CHALF3-CHALF)
EGL=.5*DEDAW*ALWGL
KWRITE
1
WRITE(6,35) CLALTO,CHALF,CHDEL
EGT=.5*DEDAW*ALWGT
VSTALL=SQRT(2.*WGS/CLMXTO/RHOTO)
QUE=.5*RHOTO*(.85*VSTALL)**2
HOB=ZAC/B
EM=.85*VSTALL/1117.
CLIFT(2,ALWGT,CLLO,ALPHLO,CLALFH,HOB,AR,DLMC4,EM)
FL=CLLO*QUE*SW
KWRITE
1
WRITE(6,36) QUE,FL
7
NTYE
CMPTO=ENP*CTNWLO*2.*DPROP**2*TP/SW/CBARW
DCMCLP
9999.
DCMCLP=3.*CTCR*DPROP**2*TP/CLCR/SW/
CMPTO=FNNWLO*TP*CLLO/(wg*CBARW)
DCMCLP
9999.
DCMCLP=0.


801
CMFPLT
999.
CMFPLT=DFLPTO*(-.008+0.003*DFLPTO/40.)
CMFLPL
999.
CMFLPL=DFLPLD*(-.008+0.003*DFLPLO/40.)
DFLPTO
0.
CMFPLT=0.0
DFLPLD
0.
CMFLPL=0.0
T1=XAC-XACF-DCMFUS-DCMNAC-DELXCG-ENP*DCMCLP-STATIC
T2=-(CLALFT/CLALCR)*ETACR*(1.-DEDAW)*(1.-(CHALF3/CHDEL3)*TAUH)
T3=CLALTL*ETALD*TAUH*DEMAX/CLMXLD
T4=(CLALTL*ETALD*(ALWGL-EGL-EYEW+EYET))/CLMXLD
T5=(CMFLPL+CMFUSL+CMNACL+CMPLD)/(CLMXLD
V=(T1+T5)/(T2+T3+T4)
XNO=XAC-DCMFUS-DCMNAC-ENP*DCMCLP-T2*V
XCGAFT=XNO-STATIC
XCGFWD=XCGAFT-DELXCG
ELH=ELTH+.25*CBARW-XCGAFT*CBARW
SST=V*SW*CBARW/ELH
508
XCGFS=XCGFWD
TG1=CMFLPT+CMPTO+CMFUST+CMNACT+CLLO*(XCGFWD-0.25)
TG2=((FL-WG)/(QUE*SW))*((XG+UM*ZCG)/CBARW)
TG3=CLALTT*(ALWGT-EGT-EYEW+EYET+TAUH*DEMAX)*ETANWL
VG=(TG1+TG2)/TG3
VGX=(T1+XACF+(TG2+CMFLPT+CMPTO+CMFUST+CMNACT)/CLLO-0.25)/(TG3/CLLO+T2)
STX=VGX*SW*CBARW/ELH
STO=VG*SW*CBARW/ELH
VTWST=AMAX1(VGX,VG)
VTEST
≤V
V=VTEST
IVFLG=1
XNO=XAC-DCMFUS-DCMNAC-ENP*DCMCLP-T2*V
XCGAFT=XNO-STATIC
XCGFWD=XCGAFT-DELXCG
ELH=ELTH+0.25*CBARW-XCGAFT*CBARW
ERROR=ABS((XCGFWD-XCGFS)/XCGFS)
>0.01
ERROR
509


509
KWRITE
=1
WRITE(6,39) T1,T2,T3,T4,T5,V,ELH
WRITE(6,40) TG1,TG2,TG3,VG,VGx
IVFLG
1
V=VTEST
SHT=V*SW*CBARW/ELH
BHT=SQRT(ARHT*SHT)
ELTH=ELH+STMRGN*CBARW
VBARH=SHT*ELTH/SW/CBARW
CMDEL=-CLALTL*V*ETALD*TAUH
WRITE(6,114)
WRITE(6,110)
WRITE(6,111) ALFC,CLALCR,CLALFT,ETACR,CLCR,DCMFUS,
DCMNAC,DCMCLP,CTCR
WRITE(6,112) ALWGT,CLALLO,CLALTT,ETANWL,EGT,CLLO,
DCMFST,CMFUST,DCMNCT,CMNACT,CMFLPT,
CMPTO,CTNWLO
WRITE(6,113) ALWGL,CLALLD,CLALTL,ETALD,EGL,CLMXLD,
DCMFSL,CMFUSL,DCMNCL,CMNACL,CMFLPL,CMPLD
WRITE(6,115) CHALF3,DEDAW,CHDEL3,CMDEL,TAUH
XLEMAC=ELC4W-0.25*CRCLW+XMAC
STANO=XLEMAC+XNO*CBARW
STAAFT=XLEMAC+XCGAFT*CBARW
STAFWD=XLEMAC+XCGFWD*CBARW
WRITE(6,100)
WRITE(6,101) XNO,STANO,SST
WRITE(6,102) STATIC,STX
WRITE(6,103) XCGAFT,STAAFT,STO
WRITE(6,104) DELXCG,SHT
WRITE(6,105) XCGFWD,STAFWD,ELTH
VERTICAL TAIL SIZING
D=(1.0-HWING)*SWF
CPNWNG=-0.00006*DLMC4**0.5
ELNC=ELODN*HN
ELTC=ELODT*HC
ELF4+ELF*0.25
ELNC
< ELF4
ELHCK=ELNC+HCK
DF1=ELF4*HN/ELNC
ELHCK
< ELF4
DF1=HN+(ELF4-ELNC)
521


521
KCONFG
1
ELTC
< ELF4
DF2=HC
DF2=ELF4*HC/ELTC
ELBM
< ELF4
DF2=HC
561
DF2=(1./BMLOD)*ELBM
KWRITE
1
WRITE(6,51) ELHCK,DF1,ELF4,DF2
S1=HN*ELNC*0.5
S2=0.5*HCK**2
S3=HN*HCK
S4=(ELPC-HCK)*HC
KWRITE
1
WRITE(6,52) S1,S2,S3,S4
S5=LC*HC
S6=HC*ELTC*0.5
KCONFG
1
S6=(1.0/BMLOD)*ELBM**2
SS=S1+S2+S3+S4+S5+S6
BIV(QB,ELCG/ELF,ELF/SWF,XV,YV,QV,2,B,NERR)
CNPFUS=(0.96*QB/57.3)*(SS/SW)*(ELF/B)*((DF1/DF2)**0.5)*(DF2/...)**.3333
KWRITE
1
WRITE(6,53) S5,S6,SS,QB
DL1CNP=-.0002*HWING
KWRITE
1
WRITE(6,55) CNPFUS,DL1CNP
NTYE
7
CNPP=0.0
601
ENP
1.
ELP=ELCG
566
ELP=ABS(ELCG-ELC4W-0.75*CRCLW-TSWPTE*YP*B/2.+ELN)

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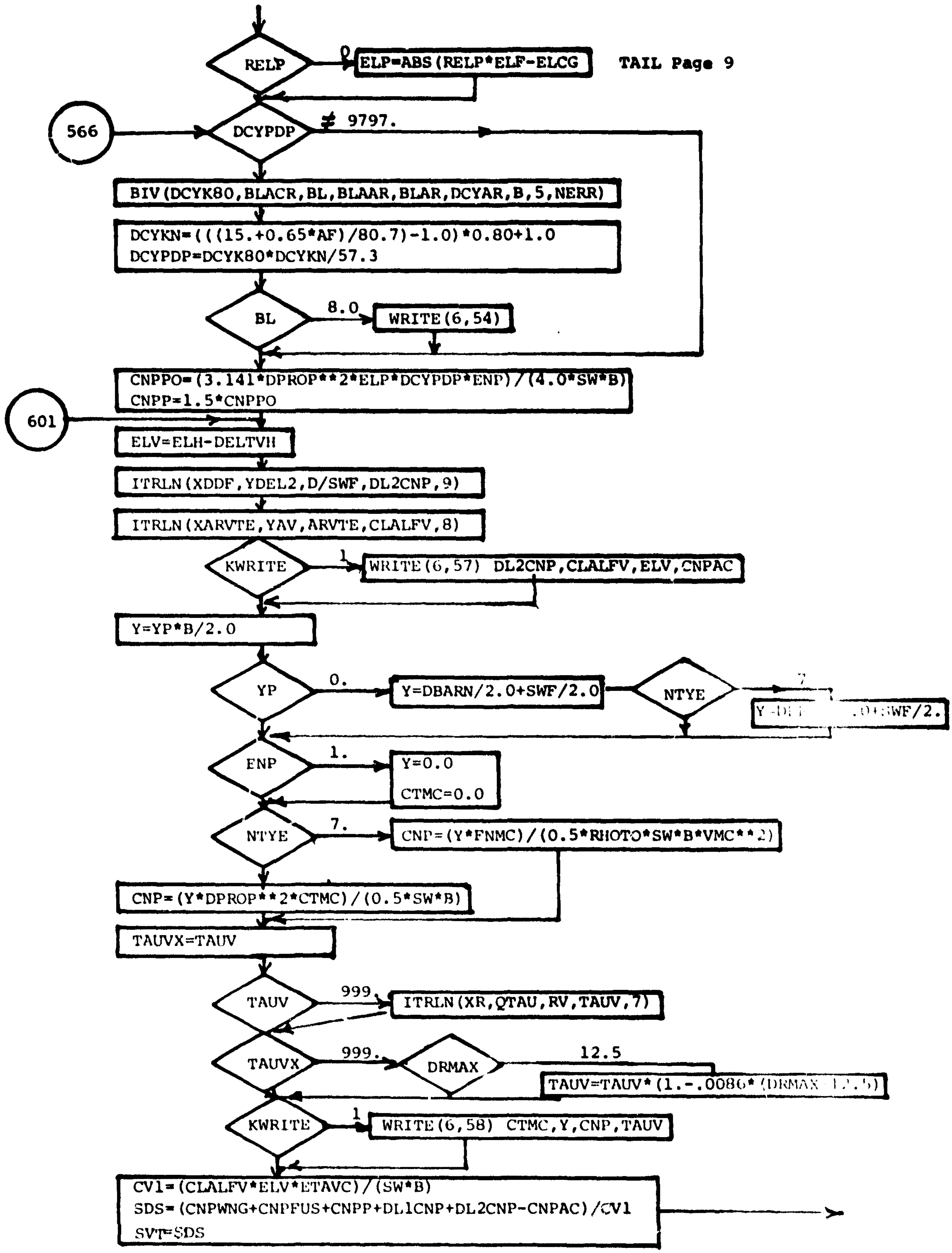
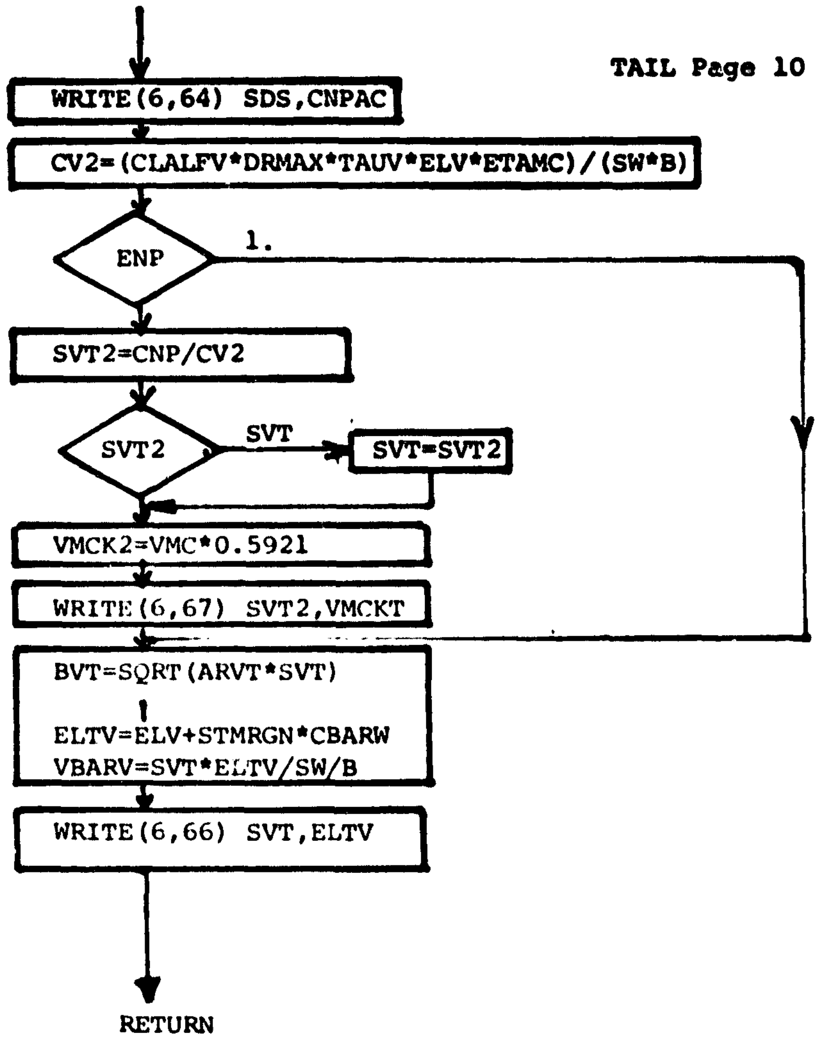

RELP
0
ELP=ABS(RELP*ELF-ELCG
566
DCYPDP
≠ 9797.
BIV(DCYK80,BLACR,BL,BLAAR,BLAR,DCYAR,B,5,NERR)
DCYKN=(((15.+0.65*AF)/80.7)-1.0)*0.80+1.0
DCYPDP=DCYK80*DCYKN/57.3
BL
8.0
WRITE(6,54)
CNPPO=(3.141*DPROP**2*ELP*DCYPDP*ENP)/(4.0*SW*B)
CNPP=1.5*CNPPO
601
ELV=ELH-DELTVH
ITRLN(XDDF,YDEL2,D/SWF,DL2CNP,9)
ITRLN(XARVTE,YAV,ARVTE,CLALFV,8)
KWRITE
1
WRITE(6,57) DL2CNP,CLALFV,ELV,CNPAC
Y=YP*B/2.0
YP
0.
Y=DBARN/2.0+SWF/2.0
NTYE
7
ENP
1.
Y=0.0
CTMC=0.0
NTYE
7.
CNP=(Y*FNMC)/(0.5*RHOTO*SW*B*VMC**2)
CNP=(Y*DPROP**2*CTMC)/(0.5*SW*B)
TAUVX=TAUV
TAUV
999
ITRLN(XR,QTAU,RV,TAUV,7)
TAUVX
999.
DRMAX
12.5
KWRITE
1
WRITE(6,58) CTMC,Y,CNP,TAUV
CV1=(CLALFV*ELV*ETAVC)/(SW*B)
SDS=(CNPWNG+CNPFUS+CNPP+DL1CNP+DL2CNP-CNPAC)/CV1
SVT=SDS


WRITE(6,64) SDS,CNPAC
CV2=(CLALFV*DRMAX*TAUV*ELV*ETAMC)/(SW*B)
ENP
1.
SVT2=CNP/CV2
SVT2
SVT
SVT=SVT2
VMCK2=VMC*0.5921
WRITE(6,67) SVT2,VMCKT
BVT=SQRT(ARVT*SVT)
ELTV=ELV+STMRGN*CBARW
VBARV=SVT*ELTV/SW/B
WRITE(6,66) SVT,ELTV
RETURN

V.3.2 Propulsion System Weight

V.3.2.1 Subroutine ENGWGT, Control Program for Propulsion System Weight Calculation

This routine controls propulsion system weight calculations. Routines called by ENGWGT are:

- HOPWSZ - Horizontal opposed piston engine weight and size
- RCWSZ - Rotary combustion engine weight and size
- WAIT - Propeller weight (Volume IV)
- ENGINE - Turbojet and turbofan weight and size (Volume IV)

Subroutines ENGINE and WAIT are described in Volume IV of this report, Propulsion. These descriptions are not repeated here.

ENGWGT

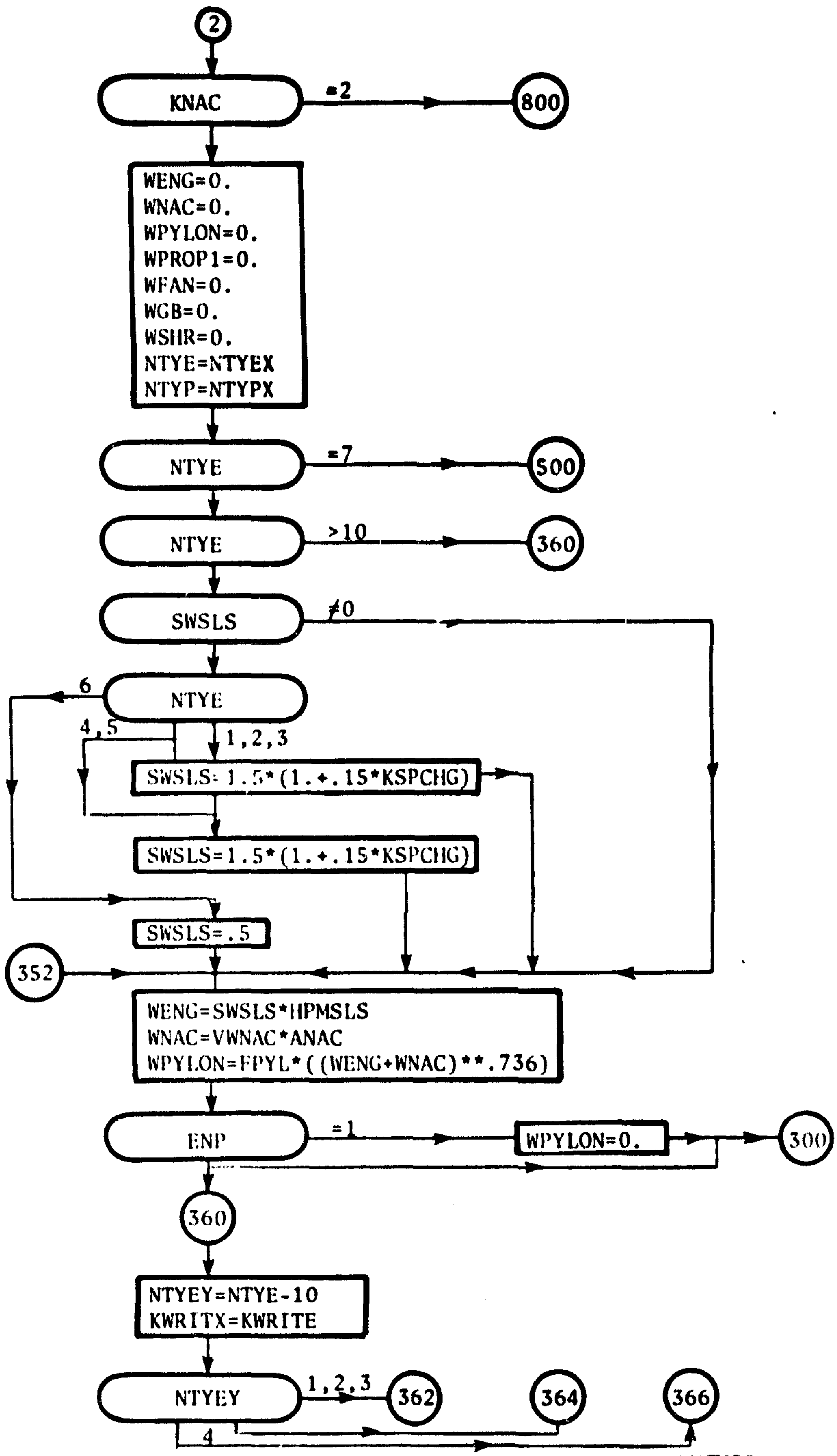
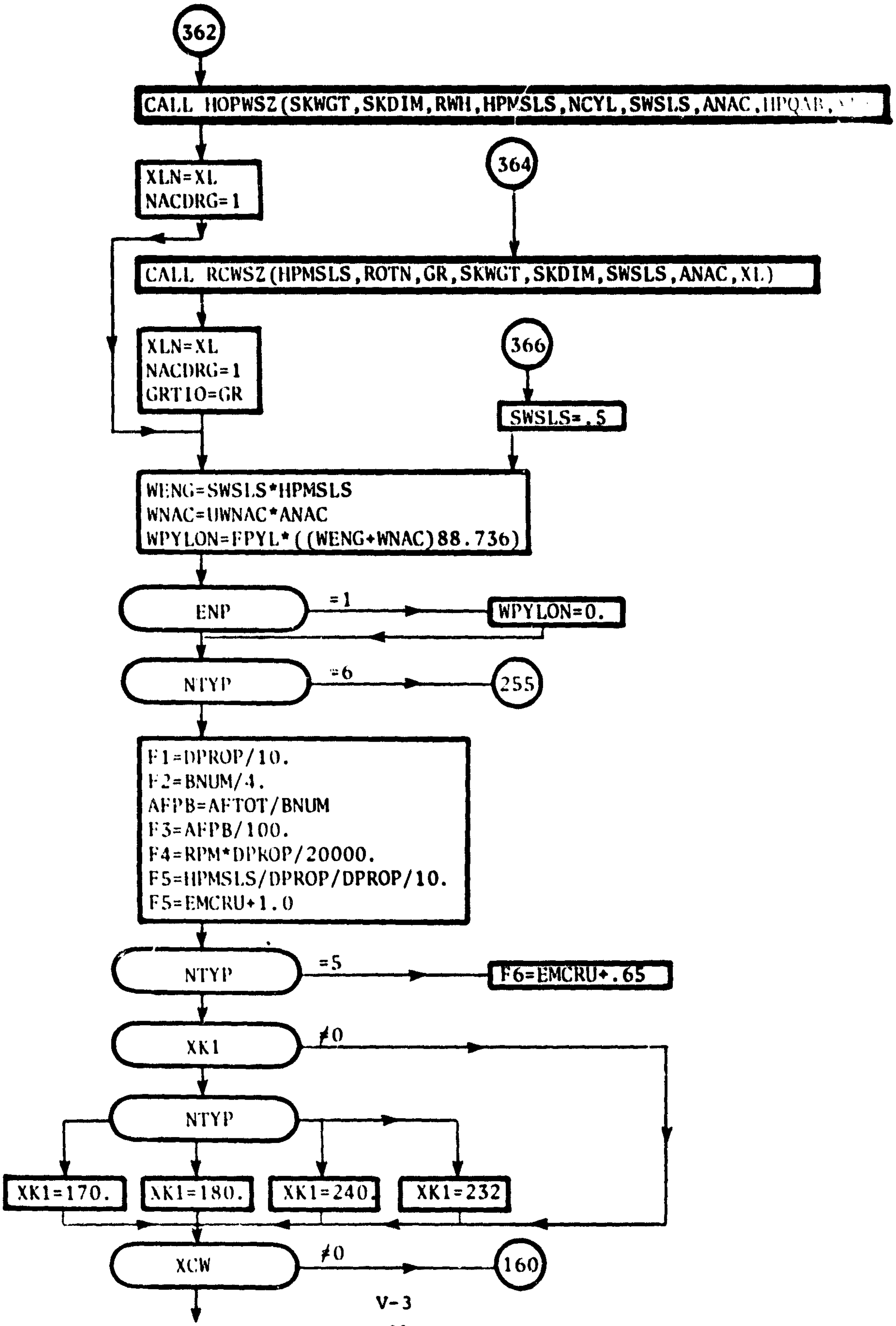
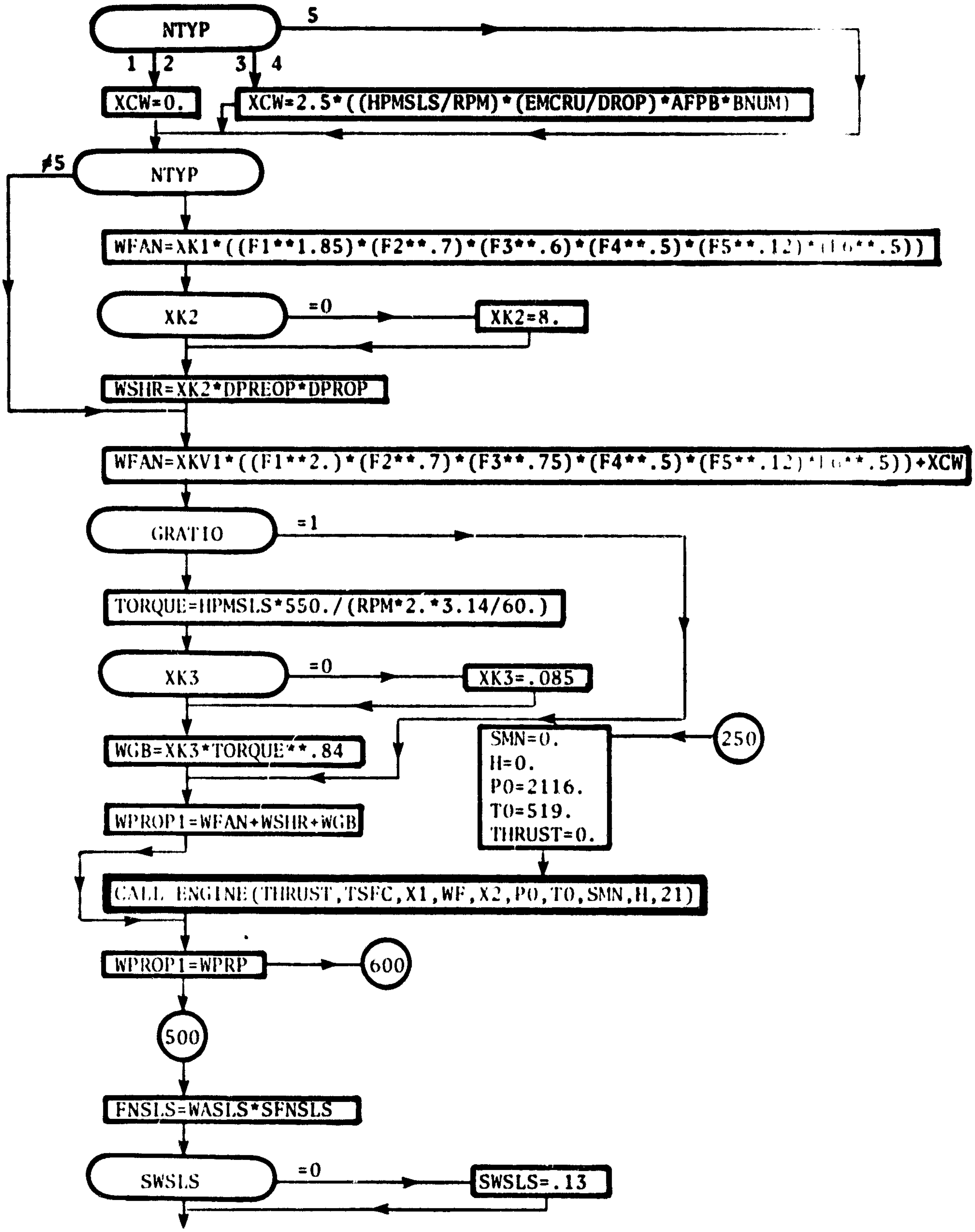
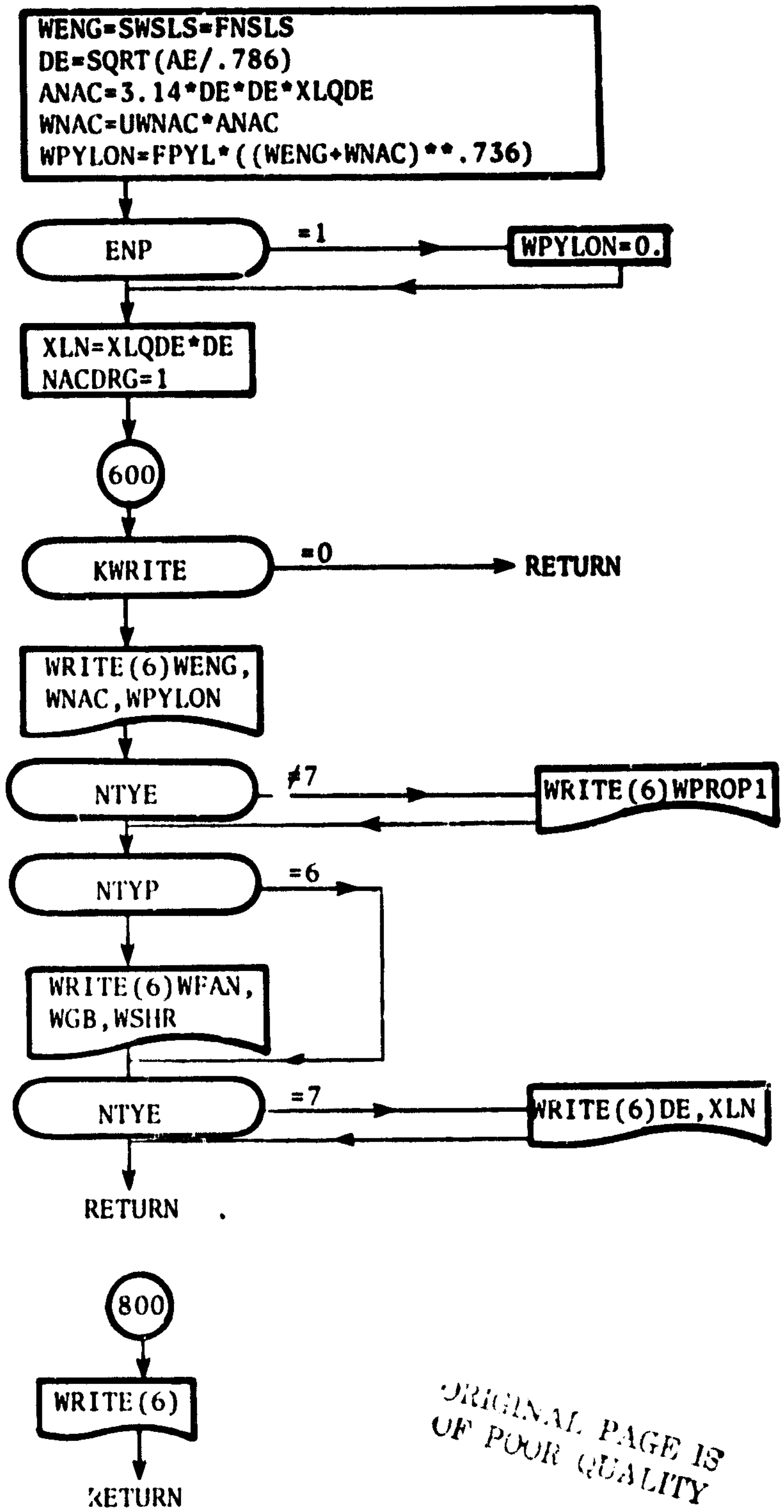


FIGURE V.3.4 DETAILED FLOW CHART, SUBROUTINE ENGWGT


362
CALL HOPWSZ(SKWGT,SKDIM,RWH,HPMSLS,NCYL,SWSLS,ANAC,HPQAB,
XLN=XL
NACDRG=1
364
CALL RCWSZ(HPMSLS,ROTN,GR,SKWGT,SKDIM,SWSLS,ANAC,XL)
XLN=XL
NACDRG=1
GRTIO=GR
366
SWSLS=.5
WENG=SWSLS*HPMSLS
WNAC=UWNAC*ANAC
WPYLON=FPYL*((WENG+WNAC)88.736)
ENP
=1
WPYLON=0.
NTYP
=6
255
F1=DPROP/10.
F2=BNUM/4.
AFPB=AFTOT/BNUM
F3=AFPB/100.
F4=RPM*DPROP/20000.
F5=HPMSLS/DPROP/DPROP/10.
F5=EMCRU+1.0
NTYP
=5
F6=EMCRU+.65
XK1
≠0
NTYP
XK1=170.
XK1=180.
XK1=240.
XK1=232
XCW
≠0
160


NTYP
5
1 2
3 4
XCW=0.
XCW=2.5*((HPMSLS/RPM)*(EMCRU/DROP)*AFPB*BNUM)
≠5
NTYP
WFAN=XK1*((F1**1.85)*(F2**.7)*(F3**.6)*(F4**.5)*(F5**.12)
XK2
=0
XK2=8.
WSHR=XK2*DPREOP*DPROP
WFAN=XKV1*((F1**2.)*(F2**.7)*(F3**.75)*(F4**.5)*(F5**.12)
GRATIO
=1
TORQUE=HPMSLS*550./(RPM*2.*3.14/60.)
XK3
=0
XK3=.085
WGB=XK3*TORQUE**.84
SMN=0.
H=0.
P0=2116.
T0=519.
THRUST=0.
250
WPROP1=WFAN+WSHR+WGB
CALL ENGINE(THRUST,TSFC,X1,WF,X2,P0,T0,SMN,H,21)
WPROP1=WPRP
600
500
FNSLS=WASLS*SFNSLS
SWSLS
=0
SWSLS=.13


WENG=SWSLS*FNSLS
DE=SQRT(AE/.786)
ANAC=3.14*DE*DE*XLQDE
WNAC=UWNAC*ANAC
WPYLON=FPYL*((WENG+WNAC)**.736)
ENP
=1
WPYLON=0.
XLN=XLQDE*DE
NACDRG=1
600
KWRITE
=0
RETURN
WRITE(6)WENG,
WNAC,WPYLON
NTYE
≠7
WRITE(6)WPROP1
NTYP
=6
WRITE(6)WFAN,
WGB,WSHR
NTYE
=7
WRITE(6)DE,XLN
RETURN
800
WRITE(6)
RETURN

V.3.2.2 Subroutine HOPWSZ, Horizontally Opposed Piston Engine Weight and Size

This routine computes weight and size of horizontally opposed piston engines. Methods employed are discussed in Section V.1.1.2 of the present volume.

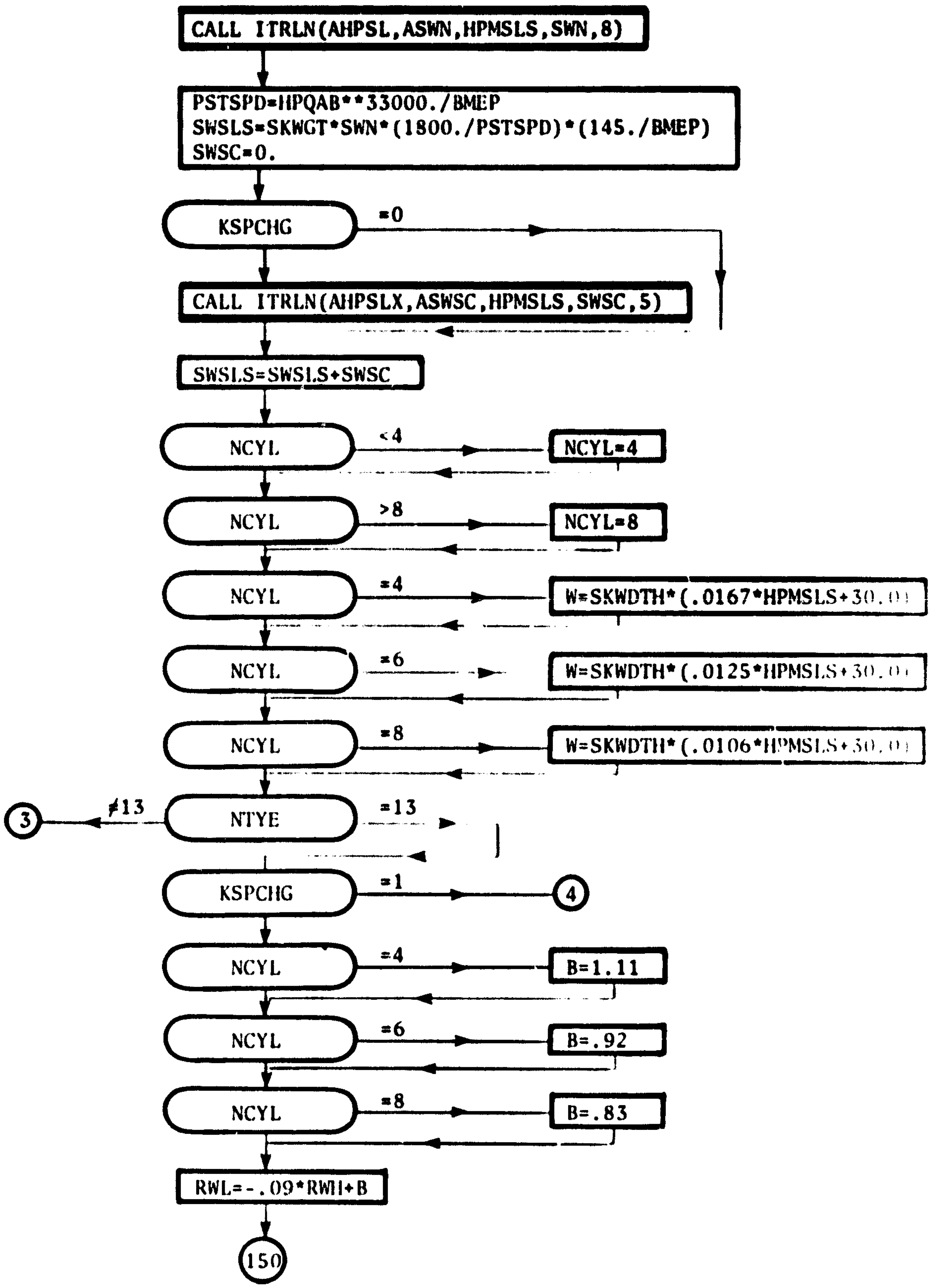
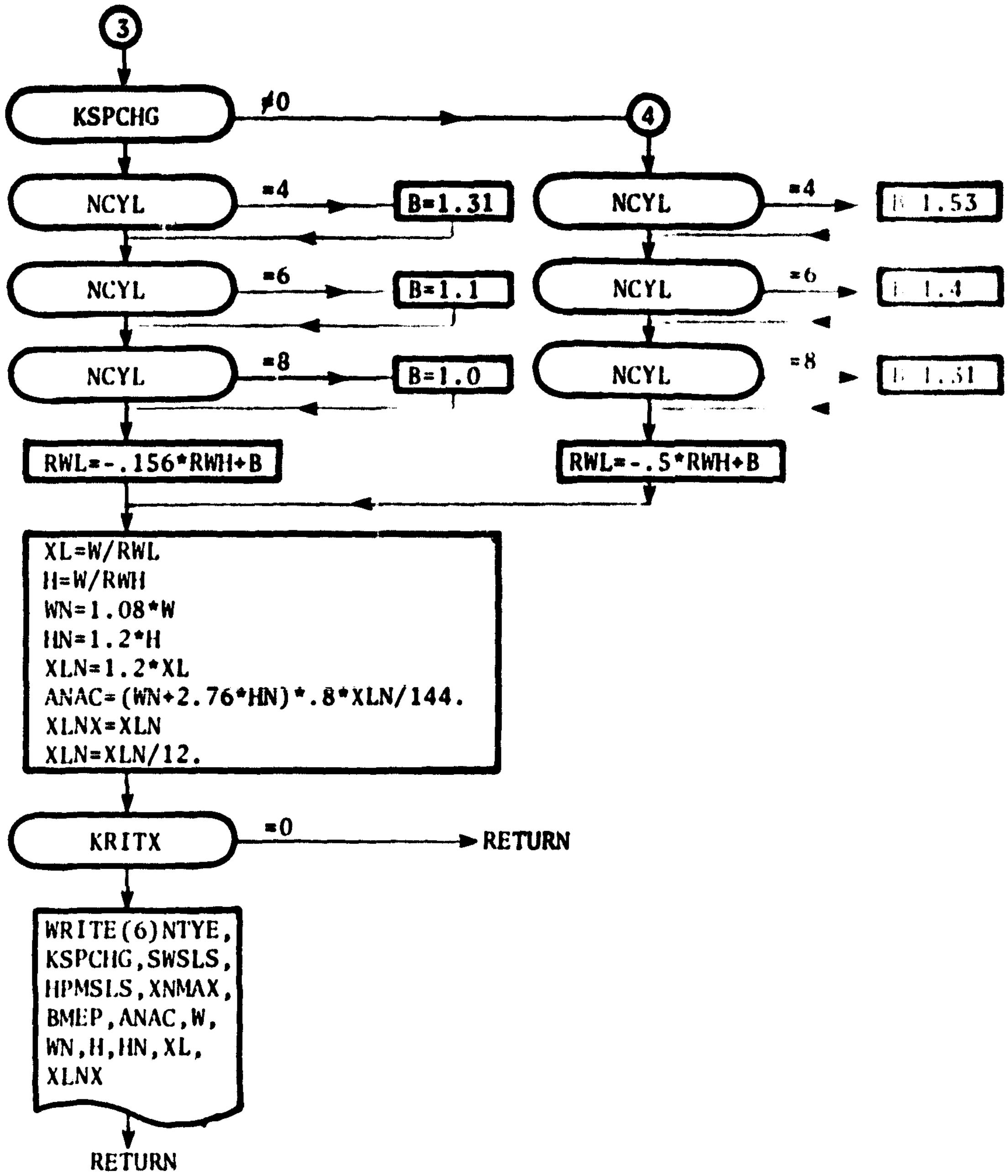


FIGURE V.3.5 DETAILED FLOW CHART, SUBROUTINE HOPWSZ



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V.3.2.3 Subroutine RCWSZ, Rotary Combustion Engine Weight and Size

This routine computes weight and size of rotary combustion engines. Methods employed are discussed in Section V.1.1.3 of the present report.

RCWSZ

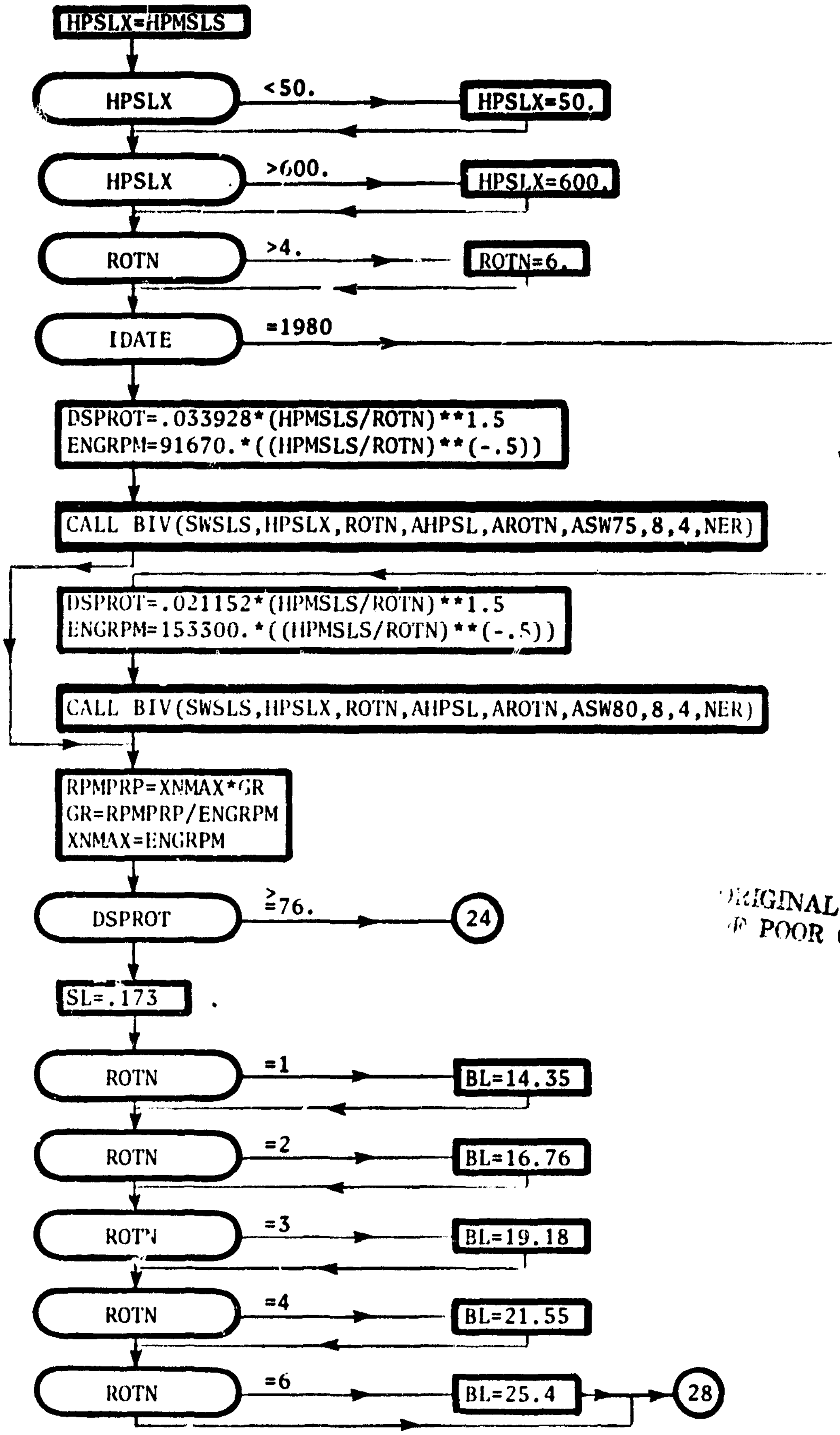

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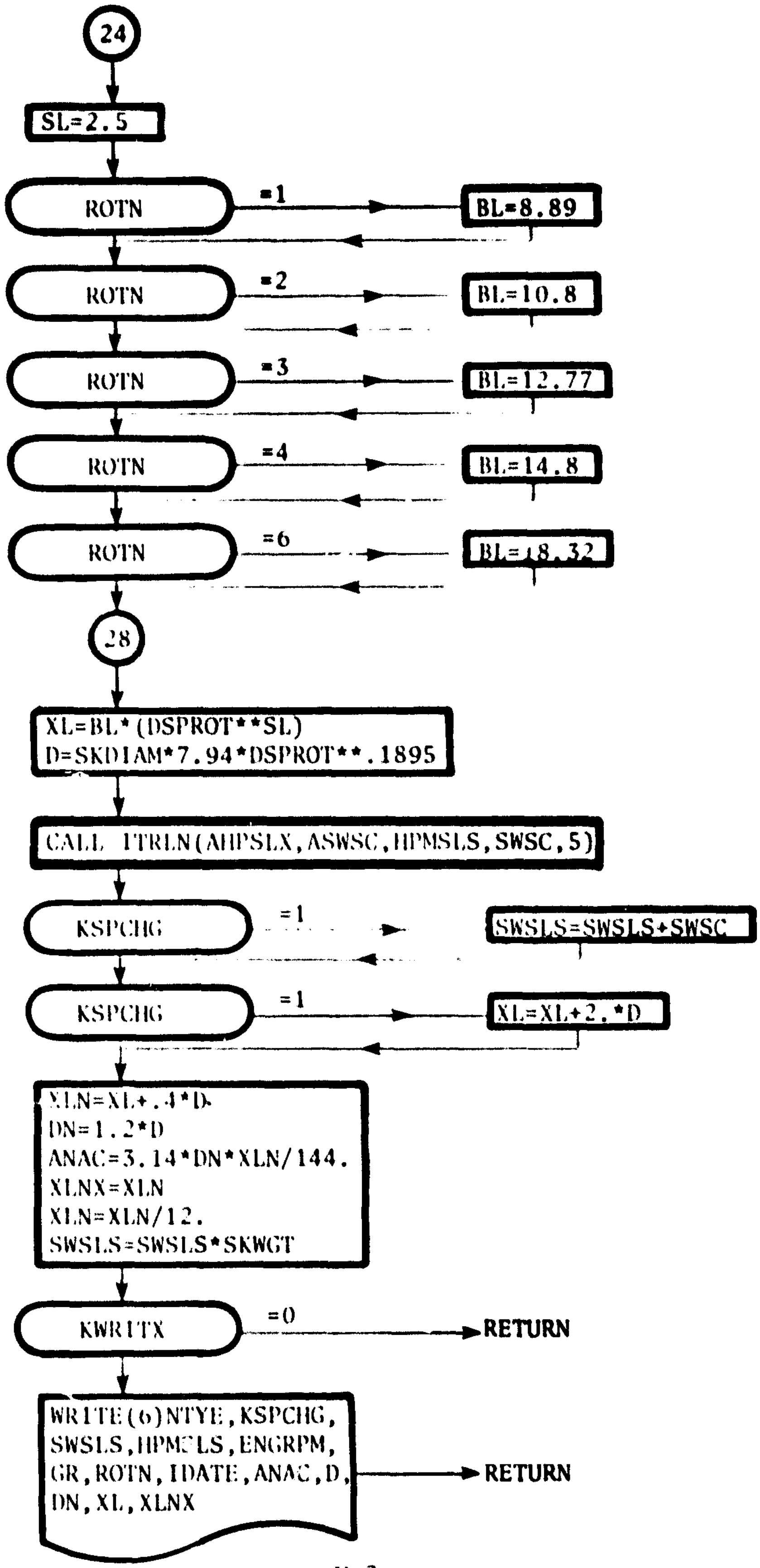
FIGURE V.3.6 DETAILED FLOW CHART, SUBROUTINE RCWSZ


24
SL=2.5
ROTN
=1
BL=8.89
ROTN
=2
BL=10.8
ROTN
=3
BL=12.77
ROTN
=4
BL=14.8
ROTN
=6
BL=18.32
28
XL=BL*(DSPROT**SL)
D=SKDIAM*7.94*DSPROT**.1895
CALL ITRLN(AHPSLX,ASWSC,HPMSLS,SWSC,5)
KSPCHG
=1
SWSLS=SWSLS+SWSC
KSPCHG
=1
XL=XL+2.*D
XLN=XL+.4*D
DN=1.2*D
ANAC=3.14*DN*XLN/144.
XLNX=XLN
XLN=XLN/12.
SWSLS=SWSLS*SKWGT
KWRITX
=0
RETURN
WRITE(6)NTYE,KSPCHG,
SWSLS,HPMSLS,ENGRPM,
GR,ROTN,IDATE,ANAC,D,
DN,XL,XLNX
RETURN