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INTEGRATED APPLICATION OF ACTIVE CONTROLS (IAAC) TECHNOLOGY TO AN ADVANCED SUBSONIC TRANSPORT PROJECT— CONVENTIONAL BASELINE CONFIGURATION STUDY

FINAL REPORT

BOEING COMMERCIAL AIRPLANE COMPANY
P.O. BOX 3707, SEATTLE, WASHINGTON 98124

CONTRACTS NAS1-14742 AND NAS1-15325
JUNE 1980

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NASA
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Space Administration

Langley Research Center
Hampton, Virginia 23665

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N80-74609#

FOREWORD

This document constitutes the final report for the Conventional Baseline Configuration Design which was begun under Contract NAS1-14742 and completed under Contract NAS1-15325.

NASA Technical Monitors for this task were D. B. Middleton and R. V. Hood of the Energy Efficient Transport Project office at Langley Research Center.

The work was accomplished within the Preliminary Design Department of the Vice President-Engineering organization of the Boeing Commercial Airplane Company. Key contractor personnel who contributed were:

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H. A. Shomber	Project Manager (IAAC)
R. Olason/G. E. Seidel	Task Manager (Conventional Baseline)
J. F. Bueno-Varela/M.T. McIntosh	Structures Technology
C. E. Roth	Flight Control Technology
A. Maeshiro	Flight Control Technology
J. D. Brown	Weight Technology
W. N. Holmquist/R. L. Sullivan	Aerodynamic Technology
R. Olason/G. E. Seidel	Configuration
E. Heineman	Structures Design
K. A. Farnum	Systems Design
M. J. Omoth	Systems

Principal measurements and calculations used during this study were in customary units and were converted to Standard International units for this document.

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1.0 SUMMARY

1.0 SUMMARY

As the first major activity on the NASA/Boeing Integrated Application of Active Controls (IAAC) project within the Energy Efficient Transport (EET) program, a comprehensive data base was established for a modern Mach 0.8 transport design. The result is identified as the Conventional Baseline Configuration.

Characteristics of the U.S. domestic fleet were evaluated to determine the mission characteristics that would have the most impact on U. S. transport fuel use in the future. This resulted in selection of a 197-passenger (plus cargo), about 3710-km (2000 nmi) mission. This selection allowed Boeing to apply a considerable amount of available analytical and test data that had been derived during earlier preliminary design efforts.

The existing data base was reviewed and additional analysis was conducted as necessary to complete the technical descriptions. The resulting Baseline Configuration including significant characteristics is shown in Figure 1. It utilizes a double-lobe, but nearly circular, body with seven-abreast seating. External characteristics feature an 8.71 aspect ratio, 31.5-degree sweep wing, a T-tail empennage, and a dual CF6-6D2, wing-mounted engine arrangement. It provides for 22 LD-2 or 11 LD-3 containers plus bulk cargo in the lower lobe. Passenger/cargo loading, servicing provisions, taxi/takeoff speeds, and field length characteristics are all compatible with accepted airline operations and regulatory provisions.

The Baseline Configuration construction uses conventional aluminum structure except for advanced aluminum alloys and a limited amount of graphite epoxy secondary structure. Modern systems are used, including advanced guidance, navigation, and controls which emphasize application of digital electronics and advanced displays.

This initial task of the IAAC project resulted in a well defined Baseline Configuration that provides a firm base for definition and evaluation of the benefits offered by configurations that utilize active controls technology. Initiation of the activities planned to meet that objective is recommended.

Configuration	
Passengers	197 mixed class, 207 all tourist
Containers	22 LD-2, or 11 LD-3
Engines	2 (CF6-6D2)
Design mission	
Cruise Mach	0.8
Range	3590 km (1938 nmi)
Takeoff field length	2210m (7250 ft)
Approach speed	69.9 m/s (136 kn)
Noise	FAR 36 Stage 3
Flying qualities	Current commercial transport practice
Airplane technology	Current commercial transport practice (aerodynamics, structural, propulsion, etc.)

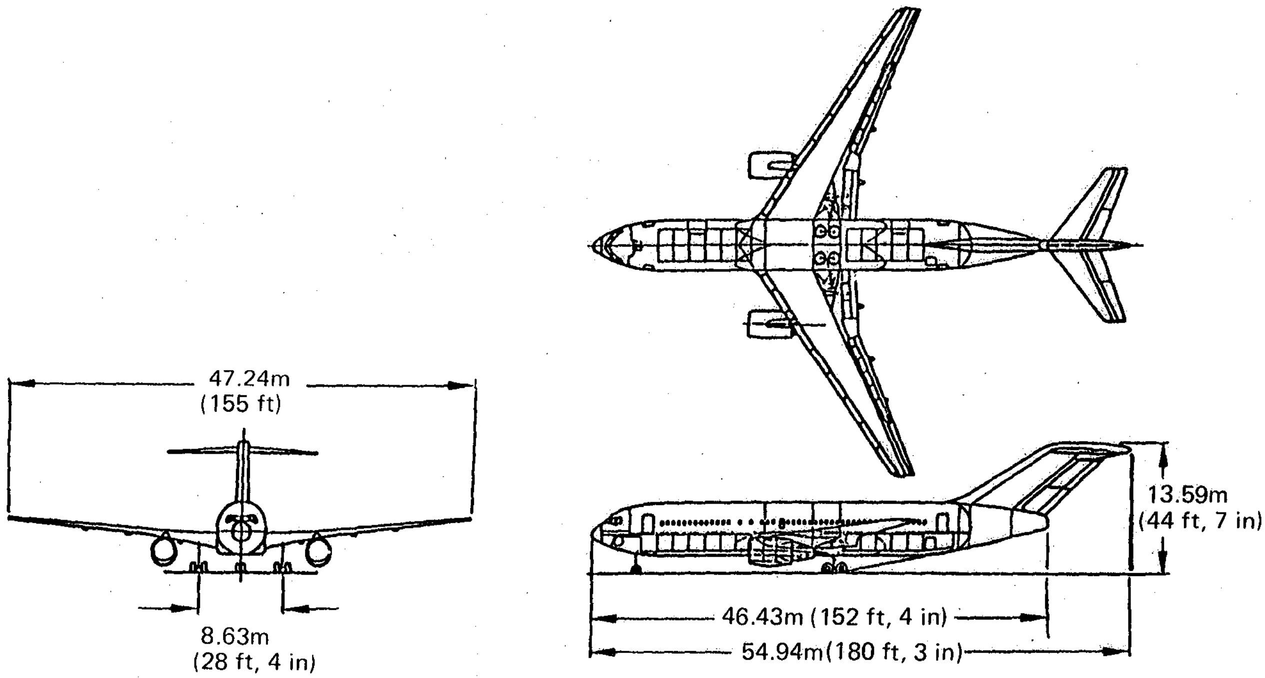

Figure 1. Baseline Configuration

2.0 INTRODUCTION

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2.0 INTRODUCTION

One of the projects in the NASA Energy Efficient Transport (EET) program is to assess the benefits associated with a major application of active controls technology (ACT) to the design of a modern subsonic commercial transport. This project was initially titled Maximum Benefit of ACT and is currently titled Integrated Application of Active Controls (IAAC) Technology to an Advanced Subsonic Transport. The IAAC project has three major constituent elements: the design of an airplane configuration and related current technology ACT system; an examination of advanced technology for use in the implementation of ACT functions; and the test and evaluation of selected components of the proposed ACT system. This document reports the work accomplished in the Conventional Baseline Configuration task of the Configuration/ACT System Design and Evaluation element.

Figure 2 shows the makeup of the Configuration/ACT System Design Task. One of the first tasks of this work was to select, define, analyze, and document a Conventional Baseline Configuration. This Baseline Configuration provided both a target mission and a modern transport airplane designed to accomplish that mission. The subsequent

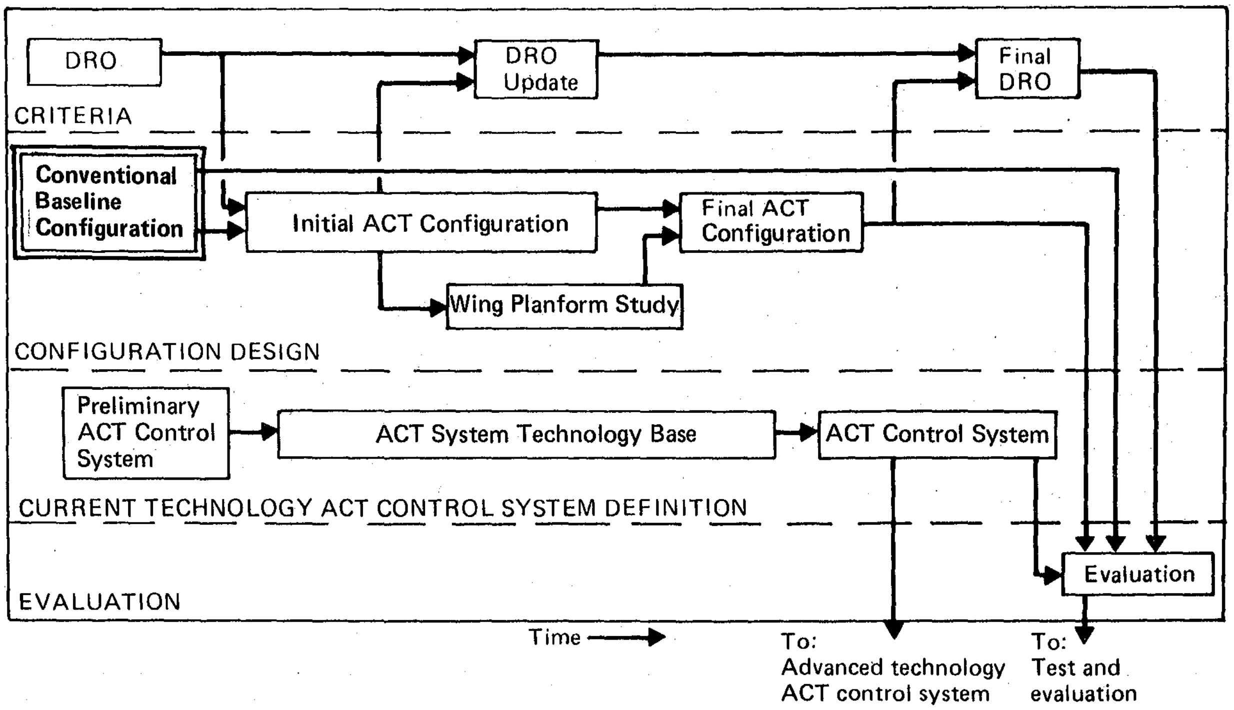

Figure 2. Configuration/ACT System Design and Evaluation Element

ACT configurations will be designed to meet this target mission and will be compared to the conventional solution as reflected in the Baseline design. The Baseline Configuration was developed under Contract NAS1-14742 and supplemented by flutter data established under Contract NAS1-15325.

2.1 OBJECTIVE

The overall objective of the EET program is to pursue technology developments that could lead to reduced fuel requirements of commercial transports.

The objectives of this first task were to:

- Establish the Baseline Configuration to sufficient depth that a high confidence technical data base existed to represent a conventional (nonactive controls) configuration. The configuration will be used as a point of departure, during later activities in the long-range IAAC project, for development of additional configurations that utilize ACT for reduced weight and improved aerodynamics.

- Provide the technical characteristics of a high confidence non-ACT configuration against which the benefits of configurations that utilize ACT can be measured during the later IAAC project activities.

- Prepare the IAAC team and the tools for the subsequent tasks.

The purpose of this document is principally to establish a data base of airplane and performance characteristics. The resulting design is identified as the Conventional Baseline Configuration, also referred to as the Baseline Configuration. The data will be used as a point of departure for configuration development and assessment of benefits of ACT within the IAAC project.

2.2 GENERAL APPROACH

A Conventional Baseline Configuration for the IAAC project was established from configuration development of a medium-range airplane in the Boeing New Airplane program. Figure 3 relates the two programs to each other and illustrates how the

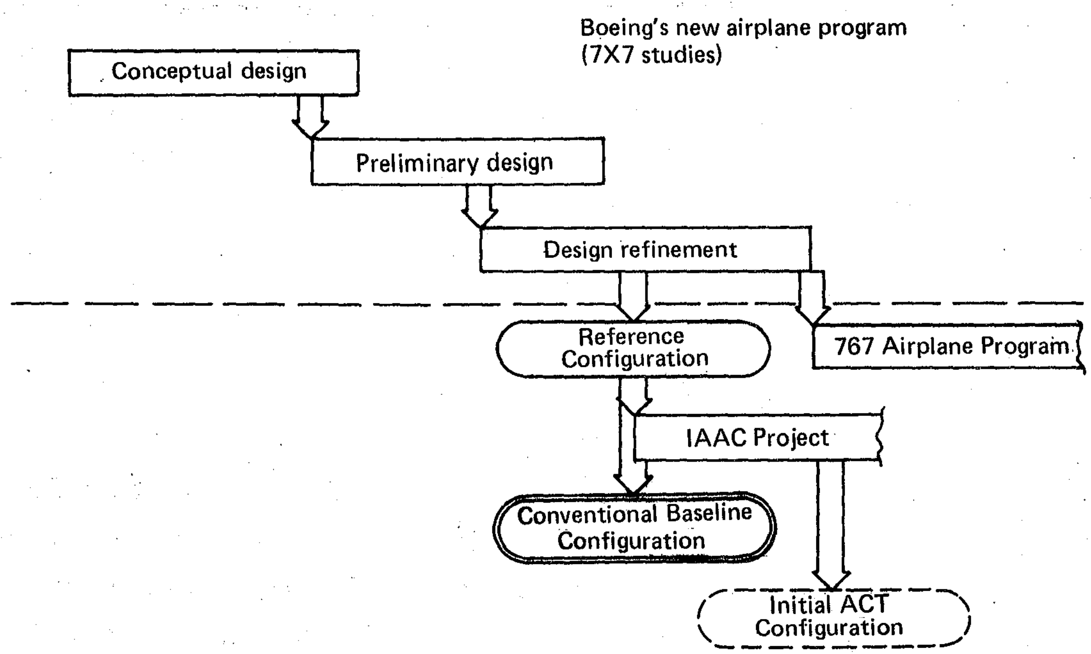

Figure 3. IAAC Conventional Baseline Selection

Baseline Configuration uses data from a developmental Reference Configuration. The Baseline Configuration has a design mission similar to those of commercial transport airplanes used in domestic fleet operations that account for the largest annual consumption of petroleum fuel. ACT airplanes developed from this Baseline Configuration potentially could save a significant amount of fuel.

Airplane systems are based on proven state-of-the-art components and Boeing experience in commercial transport development.

Aerodynamic lift and drag characteristics were established from wind tunnel tests of airplane configurations from which the Baseline Configuration was derived. These lift and drag data, although proprietary to the Boeing Commercial Airplane Company, will be used to define lift and drag differences between the Baseline and subsequent ACT configurations. Boeing experience in defining aerodynamic characteristics of full-scale airplanes from wind tunnel test results was applied in defining the performance characteristics of the Baseline Configuration.

The flying qualities were predicted from static wind tunnel data and damping estimates with classical incorporation of quasi-static-aeroelastic (QSAE) correction factors. The horizontal tail and elevator were sized for aft center of gravity (cg) deep stall recovery at landing, while the vertical tail and rudder were sized at the aft cg for engine-out control during takeoff ground acceleration. The lateral control size was determined by an engine-out trim requirement, employing the wheel only, at climbout just after takeoff.

2.3 SCOPE OF DOCUMENT

The document contains four major sections: 4.0 through 7.0. As described in Section 4.0 and illustrated in Figure 3, the design was derived from airplane configurations being considered by the Boeing New Airplane program. The size was based on those design missions of commercial passenger airplanes operating in domestic fleets comprising the largest annual fuel consumption; e.g., Model 727.

Drawings of the Baseline Configuration, which show the general arrangement, the major components, and the payload capabilities, are in Section 5.0. The illustrations comprise a general arrangement, inboard profile, body cross section, seating arrangement, cargo capability of lower and upper lobe, and principal characteristics. Also included is the primary flight control system. Shown and listed are the mission rules, speed schedules, performance and noise characteristics, as well as design weights and cg management.

Section 6.0 constitutes detailed data on the design of the airframe, propulsion, and flight control systems. A structures description of the major airplane components is presented. A description of the major airplane systems that will affect or be affected by an active control system is also given. Most of the design data were derived from a Reference Configuration in the Boeing New Airplane program. Where insufficient detail existed, additional effort was spent to make it a comprehensive data base.

Section 7.0, which describes characteristics data and analysis approaches, documents most of the work done under contract on this Baseline Configuration. Flying qualities data include trim, control, and stability characteristics about the longitudinal, lateral, and directional axes. Structural analysis data comprise definitions of critical design conditions and loads, material allowable stresses, flutter and fatigue analyses, and structural member sizing for wing, body, and horizontal tail. Aerodynamic charcteristics consist of high- and low-speed lift and drag predictions. The weight data include definitions of the weights of the major airplane components, weight distribution, and moments of inertia about the three reference axes of the airplane.

3.0 SYMBOLS AND ABBREVIATIONS

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3.0 SYMBOLS AND ABBREVIATIONS

This section is organized in four major parts: General Abbreviations in Subsection 3.1, Subscripts in 3.2, Symbols in 3.3, and Axes and Sign Nomenclature in 3.4. The parts are compiled in alphabetical order. Additionally, 3.2 is divided into three subsections: Coefficient (C) - related subscripts (3.2.1), Velocity (V) and Mach number (M) - related subscripts (3.2.2), and general subscripts (3.3.3).

3.1 GENERAL ABBREVIATIONS

ac	alternating current or aerodynamic center
ACT	active controls technology
ADF	automatic direction finder
AFC	automatic flight controls
AFCS	automatic flight control system
Ah	ampere-hour
AIDS	aircraft integrated data system
APS	auxiliary power system
APU	auxiliary power unit
AR	aspect ratio
ATA	Air Transport Association
ATC	air traffic control
ATDP	air-turbine driven pump
b	wing or tail span
BBL	body buttock line
BHD	bulkhead
BL	buttock line
BMS	Boeing Material Specification
BS	body station

BTB	bus tie breaker
BWL	body water line
c	chord
$\bar{c}$	mean aerodynamic chord (same as MAC)
C	coefficient
^{o}C	degree Celsius
CB	circuit breaker
C_c	compressibility correction
cfm	cubic feet per minute
cg	center of gravity
℄	centerline
CAS	calibrated airspeed
cm^3/s	cubic meters per second
CN	maximum continuous
CONT	continuous
CSEU	control system electronics unit
c_w	wing (local) chord
CWS	control wheel steering
D	drag, or dive
dc	direct current
deg	degrees
$\frac{d\alpha_T}{dL}$	tail angle of attack change due to tail load
DGS	drive generator system
DME	distance measuring equipment
DRO	design requirements and objectives
EAS	equivalent airspeed

EBU	engine buildup
ECS	environmental control system
EDP	engine-driven pump
E/E	electric/electronic
EET	Energy Efficient Transport (program)
EI	bending stiffness
EMP	electric-motor-driven pump
EP	electric panel
EPC	external power contractor
EPNdB	equivalent perceived noise in decibels
^{o}F	degrees Fahrenheit
FAR	Federal Aviation Regulations
FCC	flight control computer
FDAU	flight data acquisition unit
fig.	figure
FMC	flight management computer
f_s	shear stress
FS	front spar
F_{scr}	plate shear buckling stress
ft	feet
g	structural damping coefficient for neutral stability, or acceleration due to gravity
G	modulus of rigidity
gal	gallon
GCB	generator circuit breaker
GE	General Electric

gen	generator
GJ	torsional stiffness
h	hour (same as hr)
H	height
HAA	high angle of attack
hr	hour (same as h)
HSBL	horizontal stabilizer buttock line
HF	high frequency
hp	horsepower
Hz	hertz (cycles per second)
I	input, or moment of inertia (total airplane or section)
IAAC	Integrated Application of Active Controls
IAS	indicated airspeed
IDGS	integrated drive generator system
ILS	instrument landing system
in	inch
IRS	inertial reference system
k	kilo
KCAS	knots calibrated airspeed
KEAS	knots equivalent airspeed
kg	kilogram
km	kilometer
kn	knot
kVA	kilovoltampere
L	lift
LAS	lateral augmentation system

lb	pound
lbf	pounds force
LBL	left buttock line
LD- (2,3,4,8)	lower deck containers (various sizes)
LE	leading edge
l_H	horizontal tail arm
$\frac{L_E}{L_R}$	ratio of elastic to rigid lift
LRC	long-range cruise
LRRA	low-range radio altimeter
L_H	horizontal tail load
l_V	vertical tail arm
m	meter
M	Mach number
MAC	mean aerodynamic chord (same as $\bar{c}$)
MAX	maximum
MCU	modular concept unit
MID	middle position
min	minute
MLG	main landing gear
MLW	maximum design landing weight
mm	millimeter
m/s	meters per second
MTOW	maximum design takeoff weight (same as maximum TOGW)
MTW	maximum design taxi weight
MZFW	maximum design zero fuel weight
n	acceleration, or normal limit load factor

N	normal ultimate load factor
N_1	low-pressure compression speed
N_2	high-pressure compression speed
NAV	navigation
nmi	nautical mile
no.	number
OEW	operational empty weight
P	roll rate, or panel
$\hat{P}$	nondimensional roll rate
PFC	primary flight controls
PFCE	primary flight control electronics
q	dynamic pressure
Q	pitch rate
$\hat{Q}$	nondimensional pitch rate
QSAE	quasi-static aeroelastic
r	radius
R	yaw rate
$\hat{R}$	nondimensional yaw rate
rad	radian
RAT	ram air turbine
$\frac{R_E}{R_R}$	elastic-to-rigid ratio of rolling moment
REF	reference (also r)
RS	rear spar
s	second (same as sec)
S	area

sec	second (same as s)
SELCAL	selective calling
SFE	seller furnished equipment
SLST	sea level static thrust
SOB	side of body
SPD	speed
STA	station
t	thickness, or surface gage
$\bar{t}$	equivalent total gage
T	time
T_{2x}	time to double amplitude
$T_{x/2}$	time to halve amplitude
t/c	thickness ratio
TE	trailing edge
TMC	thrust management computer
T/O	takeoff
TOFL	takeoff field length
TOGW	takeoff gross weight (maximum TOGW = MTOW)
TR	taper ratio
T-R	transformer-rectifier
TRU	transformer rectifier unit
U	forward speed
$\hat{U}$	nondimensional forward speed
U_σ	true vertical gust velocity
V	velocity, speed

$\bar{V}$	volume coefficient: $\bar{V}_H = \frac{S_H}{S_W} \frac{\ell_H}{\bar{c}}$ (horizontal tail) $\bar{V}_V = \frac{S_V}{S_W} \frac{\ell_V}{b}$ (vertical tail)
VA	voltampere
V ac	volt alternating current
V dc	volt direct current
VHF	very high frequency
VOR	very high frequency omnidirectional radio range
W	wing, wheel, weight, or width
WBL	wing buttock line
WL	water line
WS	wing station
wt	weight
x_T	horizontal tail quarter MAC to wing quarter MAC (dimension)

3.2 SUBSCRIPTS

3.2.1 Subscripts Related to Coefficient C

D	drag
D_H	horizontal tail drag
HM_α	hinge moment
HM_δ	change in hinge moment due to angle of attack
HM	change in hinge moment due to control deflection
HM_o	hinge moment at zero deflection
ℓ	stability axis rolling moment
ℓ_δ	change in rolling moment with control deflection
L	stability axis lift
L_H	horizontal tail lift

L_o	lift at zero angle of attack
L_{V_2}	lift at V_2 velocity
L_α	change in lift with angle of attack
m	stability axis pitching moment
M_o	pitching moment at zero lift
M_R	reference moment
M_T	pitching moment due to thrust
M	change in pitching moment with angle of attack
M	airplane pitching moment due to surface rotation
n	stability axis yawing moment
n_δ	change in yawing moment with control deflection
N_δ	change of body axis normal force due to control surface rotation
T	thrust
Y	side force

3.2.2 Subscripts Related to Velocity V or Mach Number M

A	design maneuvering
B	gust penetration
C	cruise
D	dive
e	equivalent air speed
g	gust
LO	lift-off
MC_A	air minimum control
MC_G	ground minimum control

MO	maximum operating
S	stall
T	true
X	cross wind

3.2.3 General Subscripts

A	aileron
B	body
C	cruise, or climb
E	elevator
F	flap
H	horizontal tail or stabilizer
inc	incompressible
LE	leading edge
MO	maximum operating
0	intercept, zero alpha, zero deflection (angle of attack and controls)
P	phugoid
R	rudder
r	roll mode
REF (or r)	reference
S	spiral mode, or stall
S.L.	sea level
SP	spoiler, or short period
ss	steady state
TE	trailing edge
V	vertical tail

(V)	function of velocity
W	wing
X,Y,Z	airplane reference axes

3.3 SYMBOLS

α	angle of attack
α_{MU}	angle of attack minimum unstick
$\hat{\alpha}$	nondimensional time rate of change of angle of attack
β	sideslip angle
$\hat{\beta}$	nondimensional time rate of change of sideslip angle
Δ	change in quantity
Δ ac	aerodynamic center shift due to elastics
ΔC_{L_O}	elastic increment of lift at zero angle of attack
ΔC_{M_O}	elastic increment of pitching moment at zero angle of attack
$\Delta\eta$	incremental normal load factor
ζ	damping ratio
δ	control deflection angle
ε	horizontal tail downwash angle
γ	flight path angle
η	fraction of semispan (2 y/b)
η_H	dynamic pressure loss at horizontal tail
θ	pitch orientation
$\ddot{\theta}$	pitch acceleration
ϕ	roll orientation

$\ddot{\phi}$	roll acceleration
τ	time constant
λ	sweep
σ	vertical tail sidewash angle
σ_w	RMS vertical gust velocity
$\ddot{\psi}$	yaw orientation
ψ	yaw acceleration
∞	infinity
ω_n	natural frequency

3.4 AXES AND SIGN NOMENCLATURE

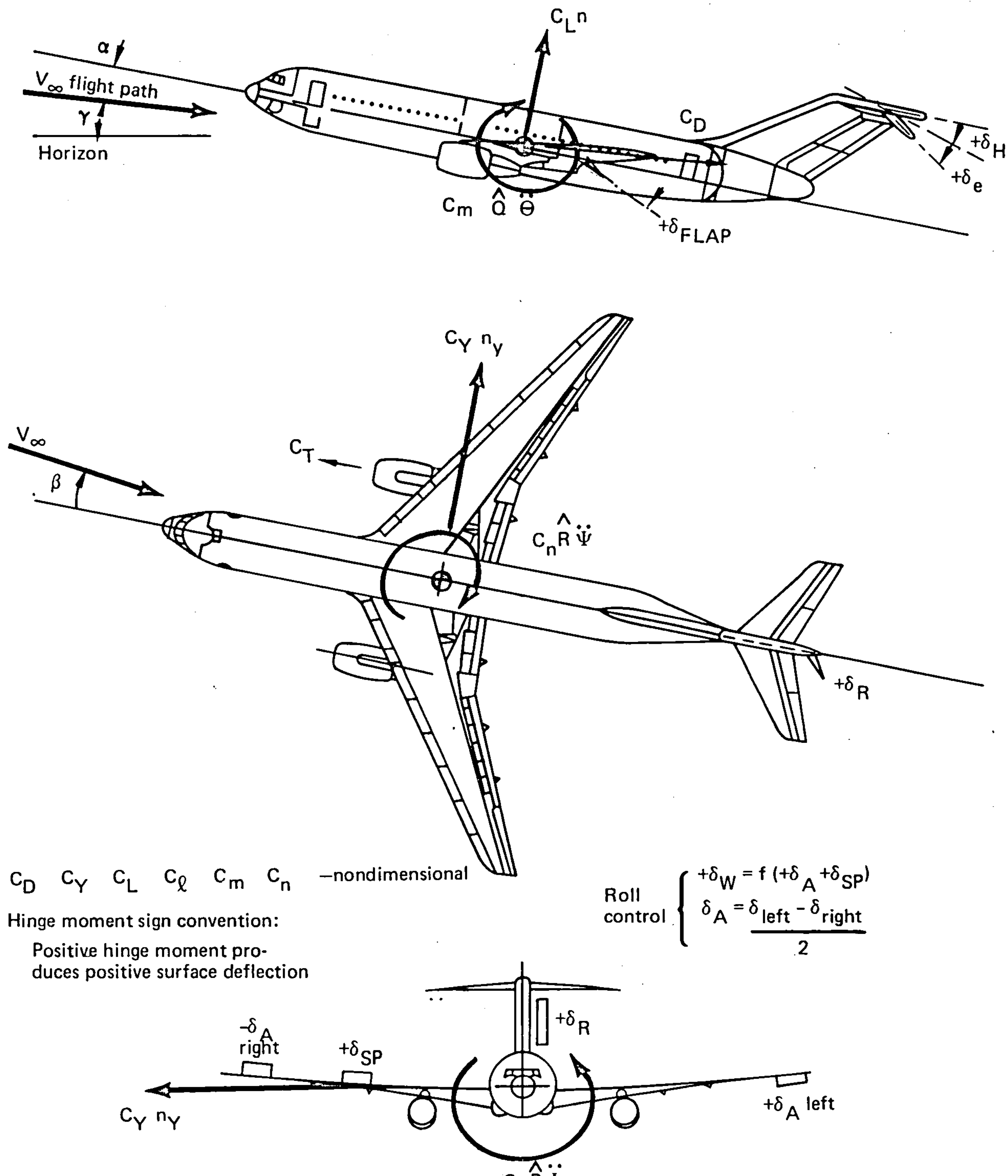


Figure 4. Axes and Sign Nomenclature

4.0 CONFIGURATION SELECTION

Page

4.0 CONFIGURATION SELECTION

The initial step was the selection and understanding of the Conventional Baseline Configuration. This selection was important for several reasons. The benefit of active controls technology (ACT) is extremely configuration-sensitive; consequently, it is theoretically possible to select a configuration for which there is little or no benefit. The elements in the design of the Baseline Configuration that are critical to the inclusion of ACT must be understood. For example, the selected configuration includes the use of powered controls, thus preventing a spurious benefit if they are introduced with active controls. Baseline Configuration technology will form the basis for the active control airplanes in all respects, except ACT.

To determine the design mission, one important consideration in the selection of the Baseline Configuration is the amount of fuel used in the market for which it is designed. Another consideration is the availability of a good data base.

4.1 MISSION REQUIREMENTS

The overall objective of the NASA Energy Efficient Transport (EET) program is to pursue technology developments that could lead to reduced fuel requirements of commercial transports. This objective dictated the need to establish the domestic fleet mission that would contribute most to reduction of fuel consumption in the time period beyond 1985.

4.1.1 FLEET FUEL REQUIREMENTS

United States air carriers consume about 37.9×10^9 liters (10 billion gallons) of jet fuel annually, which is about 44% of that consumed by the free-world air carriers. The U.S. domestic carriers use about 83% of the U.S. total; the U.S. domestic trunk air carriers use about 75% of the U.S. total.

Figure 5, based on the data of Reference 2, shows that the 727 domestic fleet uses approximately 9.5×10^9 (2.5 billion gallons) annually. This one airplane type (727), operating over an average stage length of about 930 km (500 nmi), utilizes one-half as

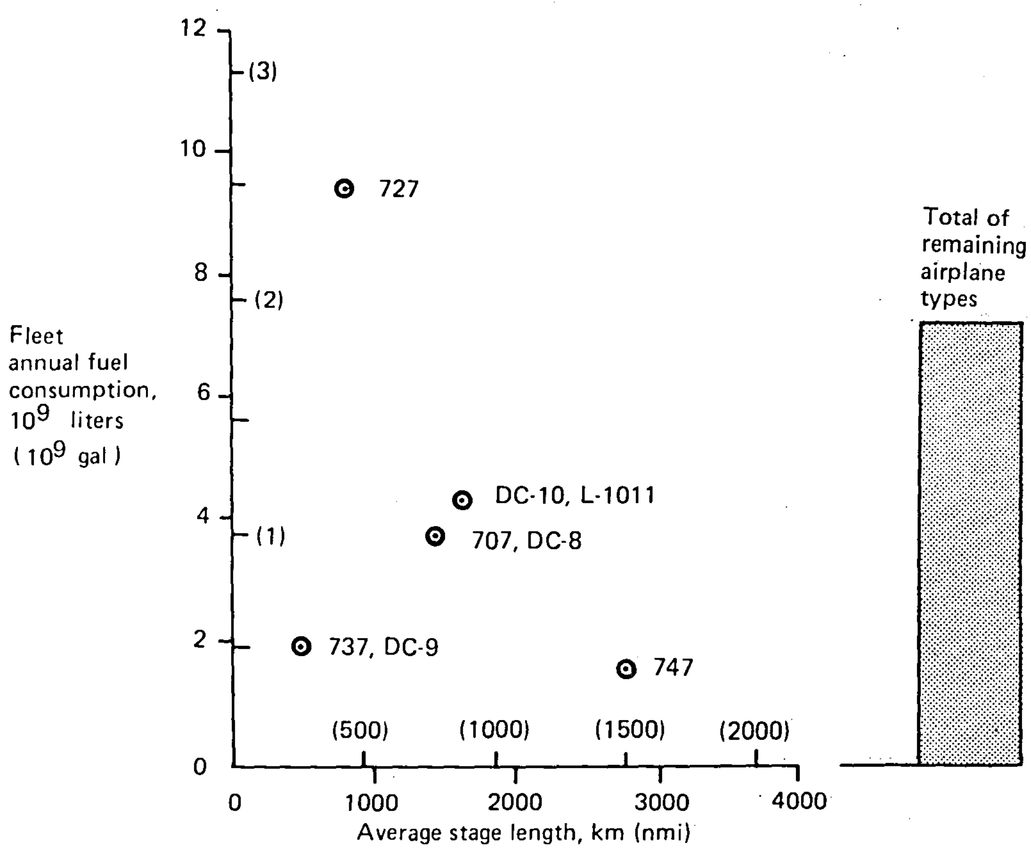


Figure 5. Fuel Usage—Domestic Airline Fleet

much fuel as all other domestic types combined. If it were possible to make a fleet substitution for one airplane type, substituting an active-control airplane for the 727 would provide the greatest leverage on fuel savings.

4.1.2 TARGET MISSION

Figure 6 illustrates the distribution of stage length for the 727 domestic airline fleet. Stage length, as used in this discussion, means the distance between takeoff and landing, so a multistop transcontinental trip would have several stages. Note that the 727, although operated at an average stage length of 930 km (500 nmi), does have flights out to almost 3710 km (2000 nmi).

The Baseline Configuration mission was defined to be a 3340 to 3710 km (1800 to 2000 nmi) design range with 150 to 200 passengers. The configuration designed to

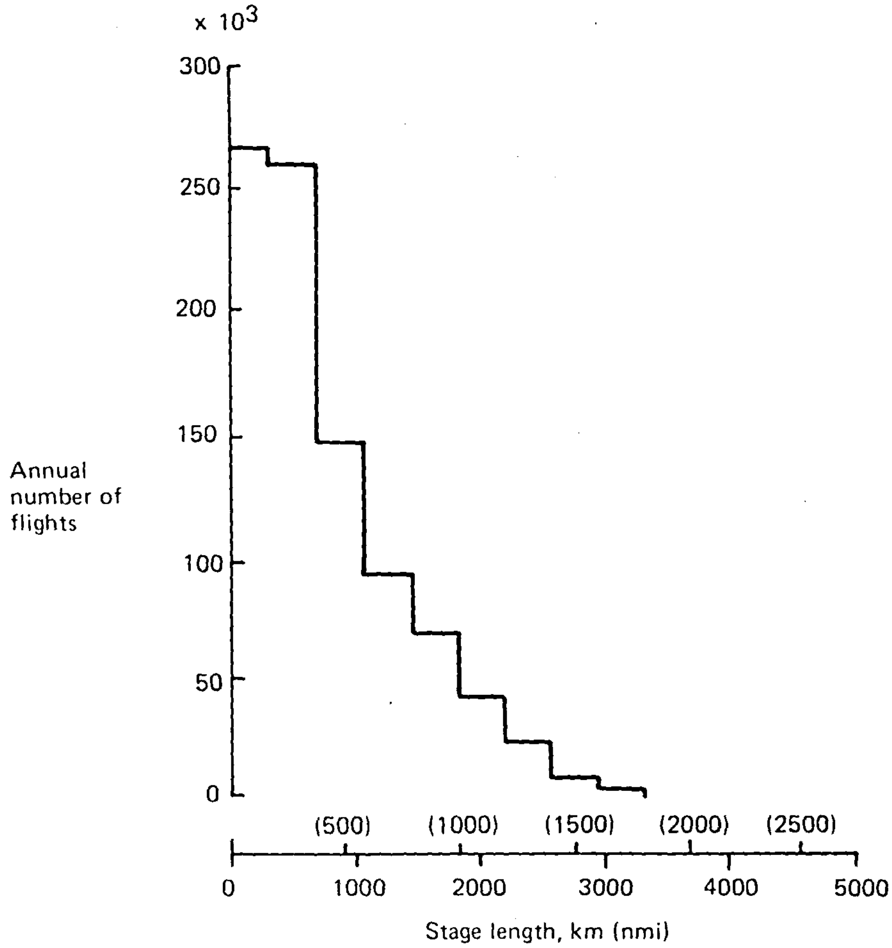


Figure 6. Flight Stage Length Distribution—727 Domestic Airline Fleet

meet this mission has performance characteristics (payload, range, cruise speed) similar to those airplanes of the current transport fleet that consume the most jet fuel. Therefore, such a Baseline Configuration conceivably could have the greatest impact on fuel used in domestic airline operation if it would replace the existing airplanes. Table 1 summarizes the target configuration characteristics that were used for the Integrated Application of Active Controls (IAAC) project.

4.2 DATA BASE AVAILABILITY

A key item in the selection of a Reference Configuration was the existence of wind tunnel data, as well as aerodynamic, flight control, structural, and weight analyses on the configuration developed under the Boeing New Airplane program, as illustrated in

Table 1. Airplane Characteristics Data

Item	Target	Actual Baseline Configuration
Configuration		
Passengers	150 to 200	197 mixed class, 207 all tourist
Containers	Compatible with wide-body transports	22 LD-2, or 11 LD-3
Engines	2 or 3	2 (CF6-6D2)
Design mission		
Cruise Mach	0.8	0.8
Range	3340 to 3710 km (1800 to 2000 nmi)	3590 km (1938 nmi)
Takeoff field length	2440m (8000 ft) maximum	2210m (7250 ft)
Approach speed	69.5 m/s (135 kn)	69.9 m/s (136 kn)
Noise	Current commercial transport practice	FAR 36 Stage 3
Flying qualities	Current commercial transport practice	
Airplane technology	Current commercial transport practice (aerodynamics, structural, propulsion, etc.)	

Figure 2 in Section 2.0. Under this present task, the analyses were refined and continued into more depth to prepare a data base for application of active controls at the future tasks. The refined product is referred to as Baseline Configuration throughout this document. The data will be used as a base for the development of ACT configurations and their comparison with the Baseline Configuration.

4.3 RESULTING CONFIGURATION

The resulting Baseline Configuration, including significant characteristics, is shown in Figure 1 in Section 1.0. The actual airplane characteristics are listed in the second part of Table 1. The Baseline Configuration carries 197 mixed-class passengers over a range of 3590 km (1938 nmi) at a cruise speed of Mach 0.8. The takeoff field length is 2210m (7250 ft), and the approach speed 69.9 m/s (136 kn).

The configuration utilizes a double-lobe, but nearly circular, body with seven-abreast seating. External characteristics feature an 8.71 aspect ratio, 31.5 deg sweep wing, a T-tail empennage, and a dual CF6-6D2, wing-mounted engine arrangement. It provides for 22-LD-2 as well as 11-LD-3 containers plus bulk cargo in the lower lobe.

Passenger/cargo loading, servicing provisions, taxi/takeoff speeds, and field length characteristics are all compatible with accepted airline operation and regulatory provisions.

The Baseline Configuration construction uses conventional aluminum structure except for advanced aluminum alloys and a limited amount of graphite epoxy secondary structure. Modern systems are used, including advanced guidance, navigation, and controls which emphasize application of digital electronics and advanced displays.

5.0 AIRPLANE DESCRIPTION

Page

5.0 AIRPLANE DESCRIPTION

The general arrangement, major components, and payload capabilities are illustrated in this section, supplemented by a description of the primary flight control system and principal configuration characteristics. The mission rules, speed schedules, performance and noise characteristics, design weights, and center of gravity (cg) management are presented.

5.1 CONFIGURATION

Physical data on the Baseline Configuration are described in this section, which is divided into geometric and characteristic data. The external shape of the airplane and the major internal views (systems, passengers, and cargo) are shown, and the primary flight control system is explained. Finally, the geometric data are supplemented by pertinent characteristics of engines, fuel capacity, and flight crew.

5.1.1 GENERAL ARRANGEMENT

The Baseline Configuration is a twin-engine, land-based, low-wing, T-tail airplane designed for commercial passenger and cargo transportation. The wide body provides for seven-abreast passenger seating with two longitudinal aisles and is sized for 197 mixed-class passengers. The lower cargo deck is designed for a 2.44-m (96-in) container width and provides for a full complement of 22 containers (type LD-2). The principal dimensions and general arrangement drawing of the airplane are shown in Figure 7.

5.1.2 EQUIPMENT

An inboard profile drawing of the airplane (fig. 8) shows the locations of the major airplane body components including passenger seats, cargo containers, electric and electronic bays, environmental control packs and mixing bays, and landing gear. Also shown are the door locations for passenger entry, galley, emergency escape, and cargo.

Geometry:

Body cross section, m (in):

Shape:	Vertical double lobe
Maximum width:	5.292 (198.00)
Maximum height:	5.410 (213.00)

Landing gear:	Nose	Main
Type:	Dual	Truck
Location, m (in):	6.896 (271.50)	56% MAC
Spacing, m (in):	0.609 (24)	1.143 x 1.422 (45 x 56)
Tire size, m,	0.939 x 0.330 -0.406	1.092 x 0.393 -0.508
(in):	(37 x 13-16)	(43 x 15.5-20)
Oleo stroke, m (in):	0.381 (15)	0.457 (18)

Aerodynamic surfaces	Wing	Vertical tail	Horizontal tail
Area m^2 (ft^2)	256.3 (2759)[a]	57.4 (618)	57.6 (620)
Aspect ratio	8.71[a]	0.67	4.00
Taper ratio	0.267[a]	0.700	0.400
Sweep at c/4 deg	31.5[a]	55.0	35.0
Incidence SOB	3.8[a]	–	–
Dihedral, deg	6.0[a]	–	-3
Root t/c %	15.1	12.0	11.0
Tip t/c %	10.3	12.0	9.0
Root chord, m (in)	8.567 (337.30)[a]	10.888 (428.69)	5.421 (213.45)
Tip chord, m (in)	2.286 (90.00)[a]	7.622 (300.08)	2.168 (85.37)
MAC, m (in)	6.031 (237.47)[a]	9.351 (368.17)	4.027 (158.55)
Span, m (in)	47.244 (1860.00)[a]	6.201 (244.14)	15.179 (597.61)
Tail arm, m (in)	–	19.972 (786.30)	27.134 (1068.30)
Tail arm, coefficient[b]	–	0.088	0.942

Engine toe-in angle–1° to a BBL

Nacelle incidence–2.625° to a BWL

Wing upper surface at side of body rib at WL 4.940 m (194.5 in)

[a]Trapezoid geometry quoted aero reference area –275.1m^2 (2961 ft^2)

[b]Based on aero reference area

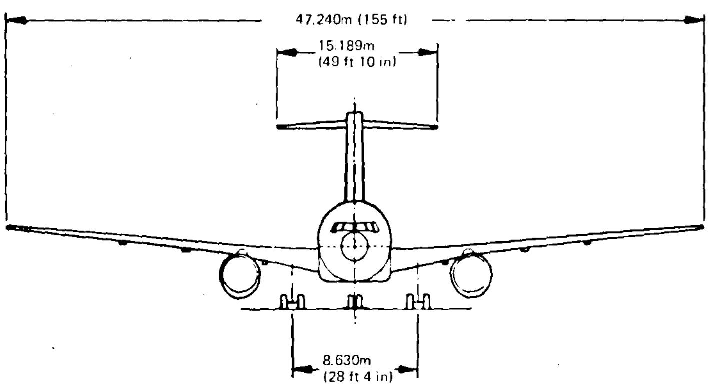
Passenger accommodations:	Passengers	Abreast	Pitch
First class:	18	6	0.965m (38 in)
Tourist:	179	7	0.864m (34 in)

Cargo and baggage, m^3 (ft^3):	22 LD-2	or	11 LD-3
Forward:	40.78 (1440)		26.85 (948)
Aft:	33.98 (1200)		22.37 (790)
Bulk cargo (aft only):	11.33 (400)		11.33 (400)
Total	86.09 (3040)		60.55 (2138)

Weights, kg (lb):

TOGW:	122 470 kg (270 000 lb)
OEW:	78 300 kg (172 610 lb)
MLW:	112 570 kg (248 160 lb)

Propulsion: Two CF6-6D2



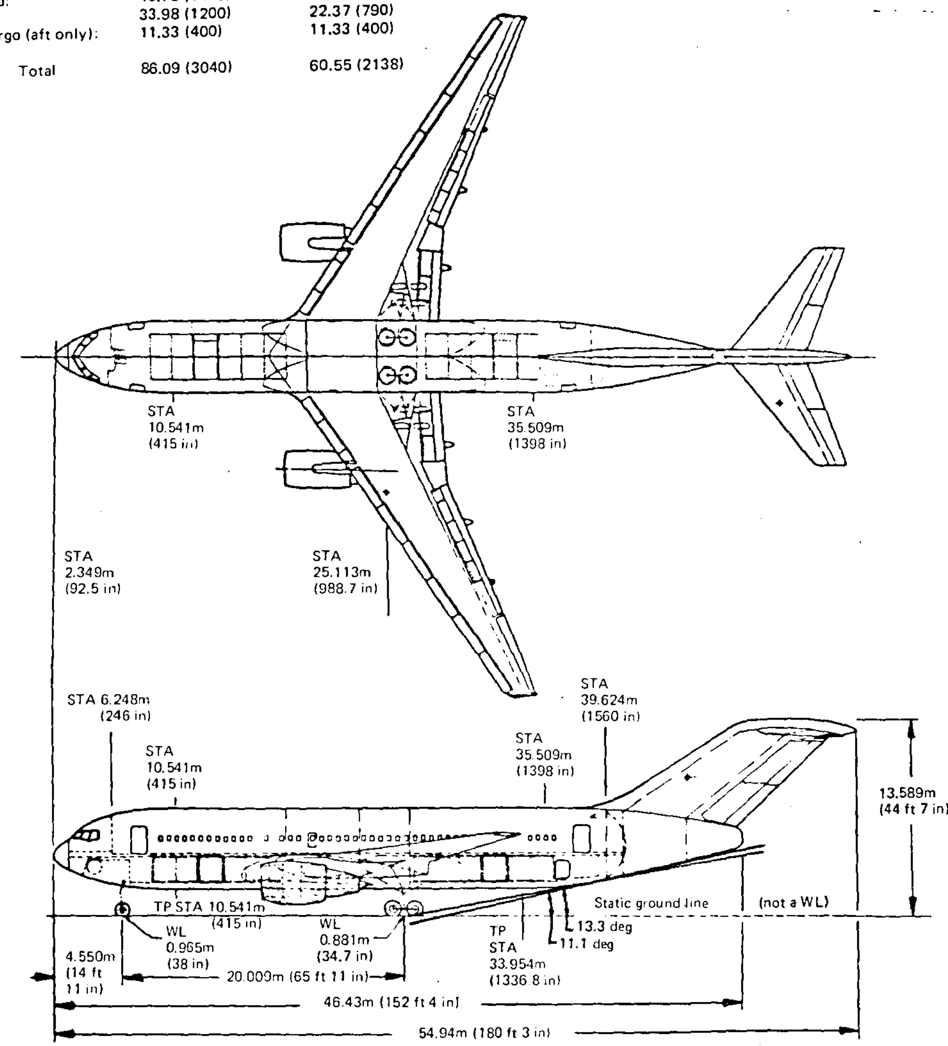


Figure 7. Baseline Configuration General Arrangement

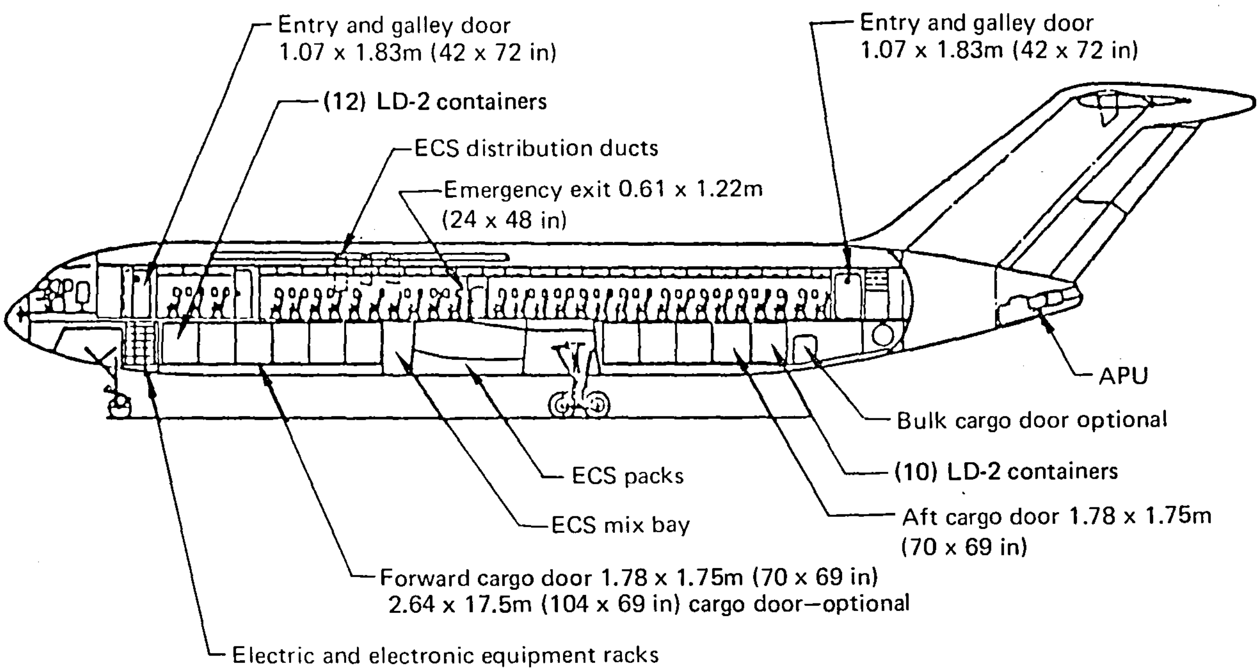


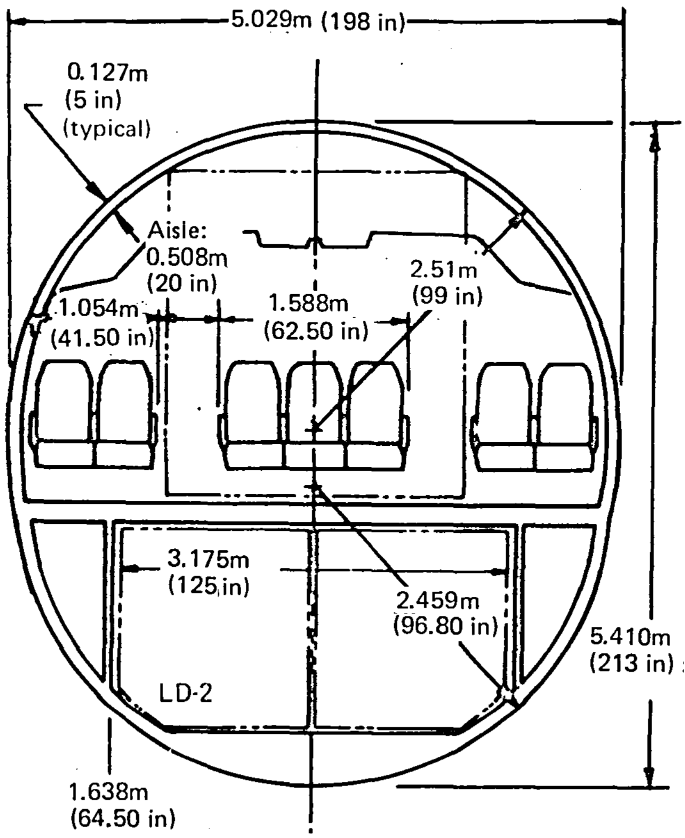
Figure 8. Inboard Profile

5.1.3 BODY CROSS SECTION

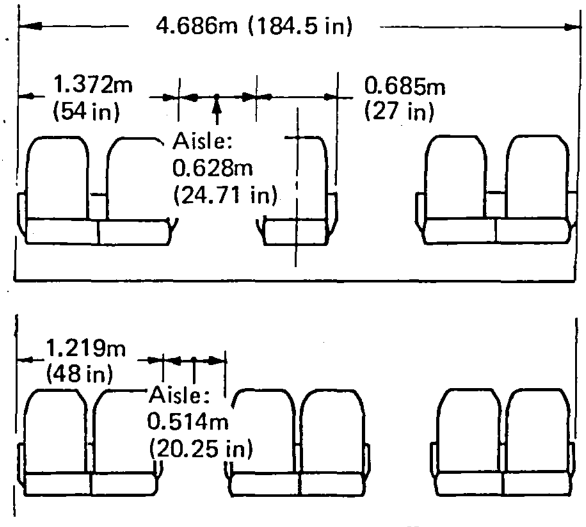
The body cross section upper lobe is 5.03m (198 in) in diameter and provides 4.67-m (184-in) seating width (fig. 9). Low density first-class and high density inclusive-tour seating arrangements are also shown. The lower lobe, sized for a 2.44-m (96-in) base width container, has a diameter of 4.92m (193.6 in), and the total section height is 5.41m (213 in).

5.1.4 SEATING ARRANGEMENT

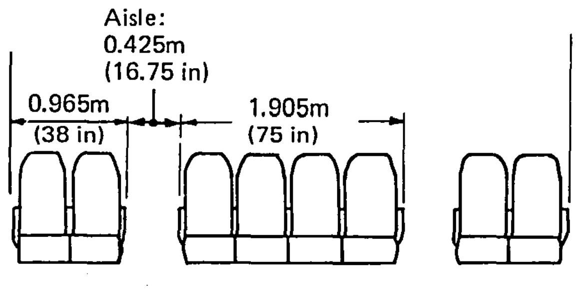
Passenger cabin seating arrangement for the basic two-class, 197-passenger version is shown in the upper part of Figure 10. The locations of galleys, lavatories, cabin attendants' seats, and cabin doors are also shown. In the all-tourist version, seating for 207 passengers is provided in a seven-abreast arrangement with two aisles and seats spaced at 0.86m (34 in) seat pitch (see the lower part of fig. 10). Figure 9 shows additional seating options for first class and inclusive tour.



Seating Options



First Class

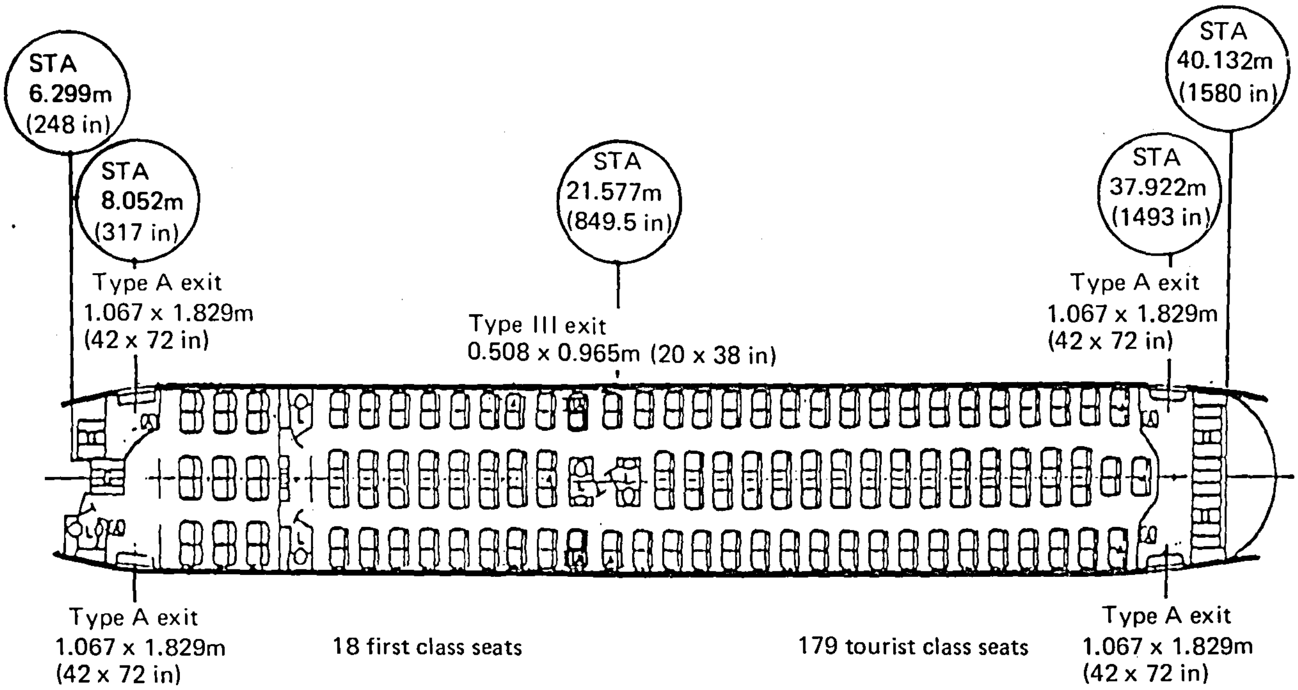


Eight-Abreast Inclusive Tour

- Perimeter = 16.502m (649.70 in)
- Area = 21.6 m^2 (232.556 ft^2)

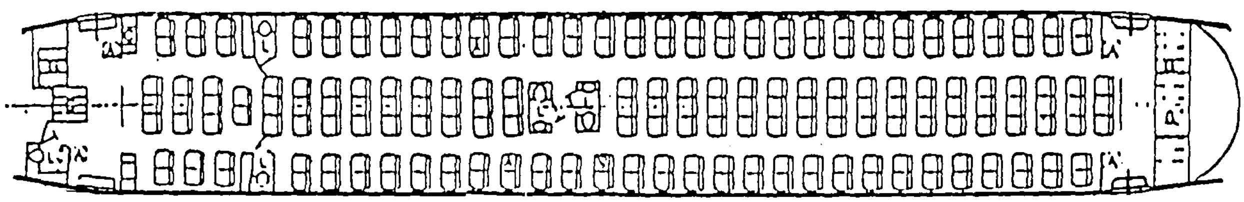
Figure 9. Body Cross Section

- 197-passenger mixed-class arrangement
- 0.965m (38 in) first class, 0.864m (34 in) tourist-class seat pitch
- Seven-abreast tourist seating



Two Classes

- 207-passenger all-tourist-class arrangement
- 0.864m (34 in) seat pitch
- Seven-abreast seating

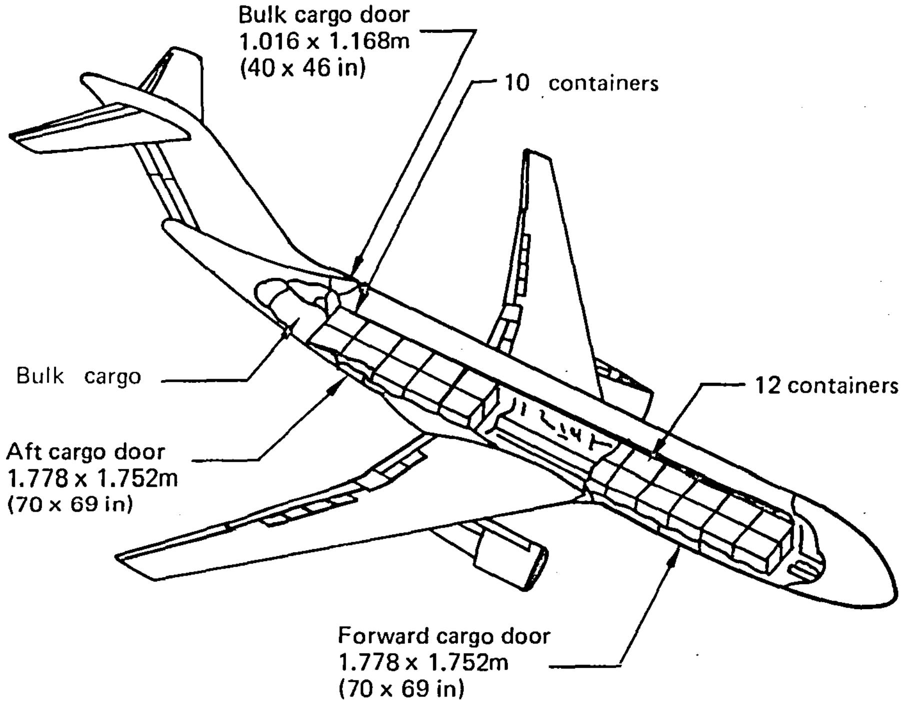


All Tourist

Figure 10. Interior Arrangement

5.1.5 CARGO CAPABILITY (LOWER LOBE)

Two compartments for containerized and bulk cargo are provided in the lower lobe. These will accommodate a dual row of LD-2 type containers or a single row of LD-3 containers. The forward cargo compartment will also accommodate three pallets, each with a base 2.44m (96 in) wide and 3.18m (125 in) long. The aft compartment accommodates containerized and bulk cargo. The lower lobe cargo system and cargo volumes are shown in Figure 11. The dimensions and volumes of cargo containers that can be carried in the lower lobe cargo compartment are illustrated in Figure 12.



Volume, m^3 (ft^3)										
Forward compartment			Aft compartment				Total containerized			Total bulk
LD-2	LD-8	LD-3	LD-2	LD-8	LD-3	Bulk	LD-2	LD-8	LD-3	Bulk
40.78	41.46	26.85	33.98	34.55	22.37	11.33	74.76	76.01	49.22	11.33
(1440)	(1464)	(948)	(1200)	(1220)	(790)	(400)	(2640)	(2684)	(1738)	(400)

Figure 11. Lower Deck Cargo

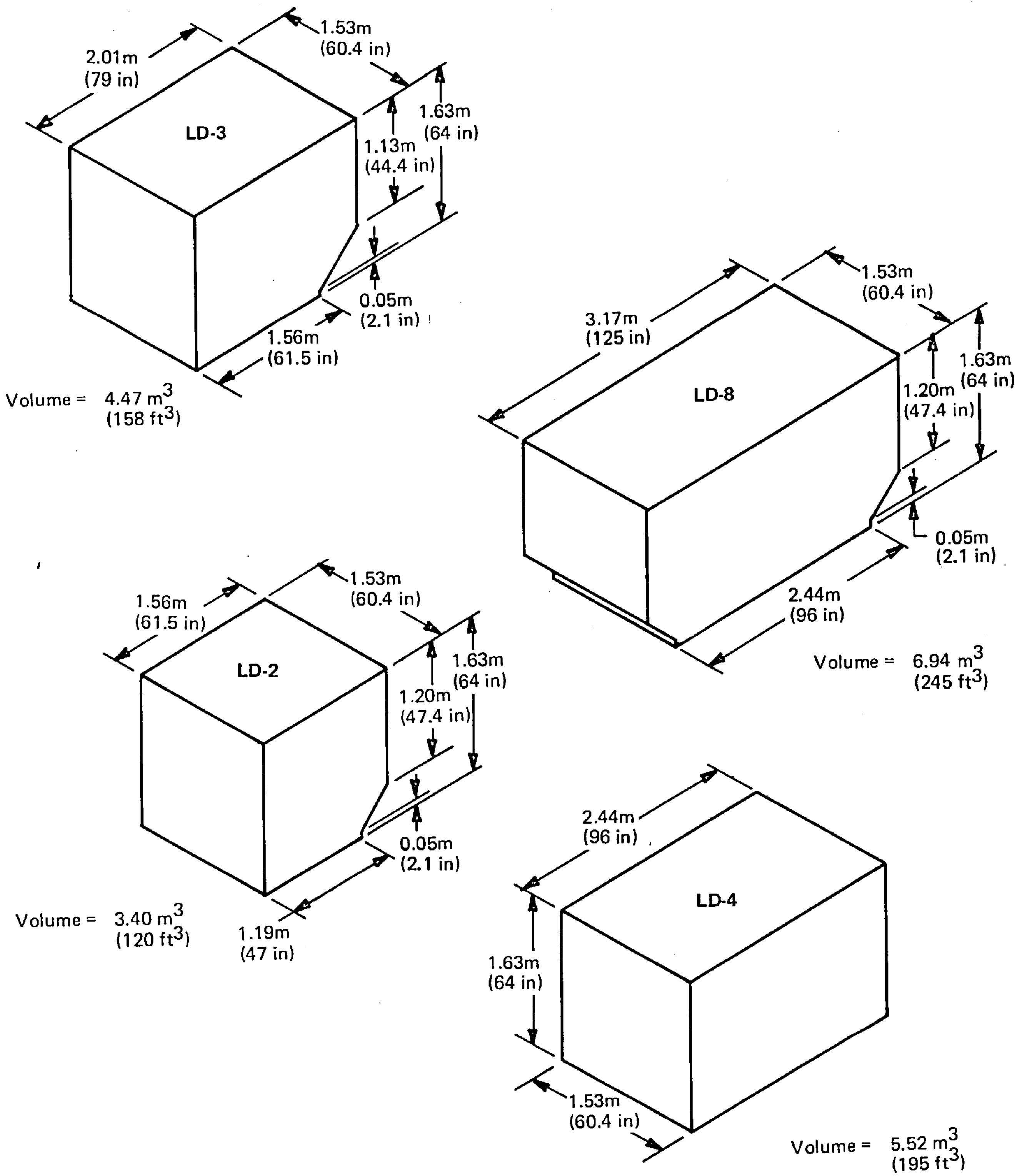


Figure 12. Cargo Containers

5.1.6 CARGO CAPABILITY (UPPER LOBE)

In all-cargo or in passenger/cargo combination versions, the upper lobe of the body will accommodate 2.44-m (8-ft) wide by 2.44-m (8-ft) high by 3.05-m (10-ft) long cargo containers. Cargo pallets on a 2.44-m (96-in) wide by 3.18-m (125-in) long base also can be carried. The capability to carry these cargo containers and/or pallets is provided using a large forward cargo door, 2.57m (101 in) high by 3.40m (134 in) long, which enables these cargo containers to be loaded. This cargo capability and large cargo door are optional, and space for installing this door has been provided in the Baseline Configuration.

5.1.7 PRIMARY FLIGHT CONTROL SYSTEM

The level of technology is consistent with current commercial transport practice and includes fully powered controls with some electrically commanded primary controls. Stability characteristics are such that a yaw damper is needed. The airplane will exhibit safe or satisfactory flying qualities following the failure or functional loss of its automatic flight control system (AFCS), which is not considered a part of the primary flight control system.

The longitudinal control surfaces consist of:

- an elevator control system, two surfaces on each side, each moved by two hydraulic actuators

- a stabilizer trim system, the whole horizontal tail surface hinged at the rear spar and hydraulically actuated by a fail-safe jack screw

The lateral control surfaces include (on each wing):

- an inboard aileron and an outboard aileron, hydraulically actuated

- six individually actuated spoiler panels, five of which are also used as speed brakes. The sixth is used only as a ground spoiler and is located over the main landing gear. All six spoilers are electrically commanded and hydraulically actuated.

The directional control system uses two double-hinged rudder surfaces. Each rudder is moved by two hydraulic actuators. Separate electrically controlled yaw damper servos are connected to the upper and lower rudder actuator control linkages, powered by separate hydraulic systems.

5.1.8 PRINCIPAL CHARACTERISTICS

Principal characteristics of the Baseline Configuration are shown in Table 2, page 38.

Table 2. Baseline Configuration Principal Characteristics

Airplane size	
Maximum takeoff weight, kg (lb)	122 470 (270 000)
Wing area, m^2 (ft^2)	275.1 (2961)
Wing span/sweep, m/deg (ft, in/deg)	47.24/31.47 (155, 0/31.47)
Location on body, percent body length	49.0
Location—engine pod on wing, percent b/2	33.6
Trailing-edge flaps	Single slot JP5
Leading-edge devices	Slats 6.29 lb/gall
Horizontal tail area/$\bar{V}_H$, m^2 (ft^2)	57.6/0.942 (620/0.942)
Sweep, deg	35
AR/taper	4.0/0.40
Vertical tail area/$\bar{V}_V$, m^2 (ft^2)	57.4/0.088 (618/0.088)
Sweep, deg	55
AR/taper	0.67/0.70
Body cross section, m (in)	5.03W/5.410H (198.0W/213.0H)
Body length/overall length, m (ft, in)	46.43/54.94 (152,4/180,3)
Cabin length, m (in)	33.38 (1314)
Doors, number, type, size, m (in)	4, type A, 1.07 x 1.83 (42 x 72) 2, type III, 0.51 x 0.97 (20 x 38)
Systems	
Engine: number/type	2/CF6-6D2
Engine thrust (SLST), N (lb)	182 377 (41 000)
Nacelle and acoustic treatment	FAR 36 Stage 3
Fuel capacity	
Wing tanks, m^3 (gal)	42.775 (11 300) 71,292 lb
Center tanks, m^3 (gal)	Dry
Total, m^3 (gal)	42.775 (11 300)
Main gear wheelbase/track, m (in)	1.42/1.14 (56.0/45.0)
Location, percent MAC	56
Stroke/extended length, m (in)	0.46/3.18 (18/125)
Tire size: wheel size, m (in)	1.09 x 0.39 - 0.51 (43 x 15.5 -20)
Nose gear type/tire spacing, m (in)	Dual/0.61 (Dual/24)
Stroke/extended length, m (in)	0.38/2.18 (15/86.0)
Tire size: wheel size, m (in)	0.94 x 0.33 - 0.41 (37 x 13 - 16)
Payload	
Flight crew/attendants	3/6
Mixed class passengers/split	197/9% first class/91% tourist
All tourist passengers	207
Containers: number/type	Twenty-two LD-2 or eleven LD-3
Cargo	
Containerized, m^3 (ft^3)	74.76 49.22 (2640) (1738)
Bulk, m^3 (ft^3)	11.33 11.33 (400) (400)
Total, m^3 (ft^3)	86.09 60.55 (3040) (2138)
cg location	
Forward, percent MAC	10.0
Average cruise, percent MAC	20.5

5.2 PERFORMANCE

5.2 PERFORMANCE

Estimated performance data for the Baseline Configuration described in this section are in compliance with the design requirements and objectives (DRO) in Table 3.

Table 3. Performance Characteristics

Item	Design requirements and objectives	Baseline Configuration
Still air range	3890 km (2100 nmi)	3590 km (1938 nmi)
FAR takeoff field length, sea level, 29°C (84°F)	2040m (6700 ft)	2210m (7250 ft)
V_{APP} KEAS, maximum landing weight	69.5 m/s (135 kn)	69.9 m/s (136 kn)
FAR dry landing field length, sea level, maximum landing weight	1525m (5000 ft)	1445m (4735 ft)

5.2.1 MISSION RULES

The mission is flown with a step cruise procedure beginning at 10.7-km (35 000-ft) altitude at a cruise Mach number of 0.8 with standard day cruise conditions. Reserve fuel includes allowances for 1 hour extended cruise and a 370m (200 nmi) alternate. Range capability is quoted for a typical U.S. domestic mission profile as presented in Figure 13 with full passenger payload and nominal performance. A summary for the design mission is included in Figure 14.

5.2.2 CLIMB AND DESCENT SPEED SCHEDULE

Climb performance was computed for a normal climb and speed schedule as follows:

- takeoff
- climb to 460m (1500 ft) altitude
- accelerate to 130 m/s (250 KCAS) at 460m (1500 ft) altitude
- climb to 3.05 km (10 000 ft) altitude
- accelerate to 150 m/s (300 KCAS) at 3.05 km (10 000 ft) altitude
- climb at 150 m/s (300 KCAS)/Mach 0.80 to cruise altitude

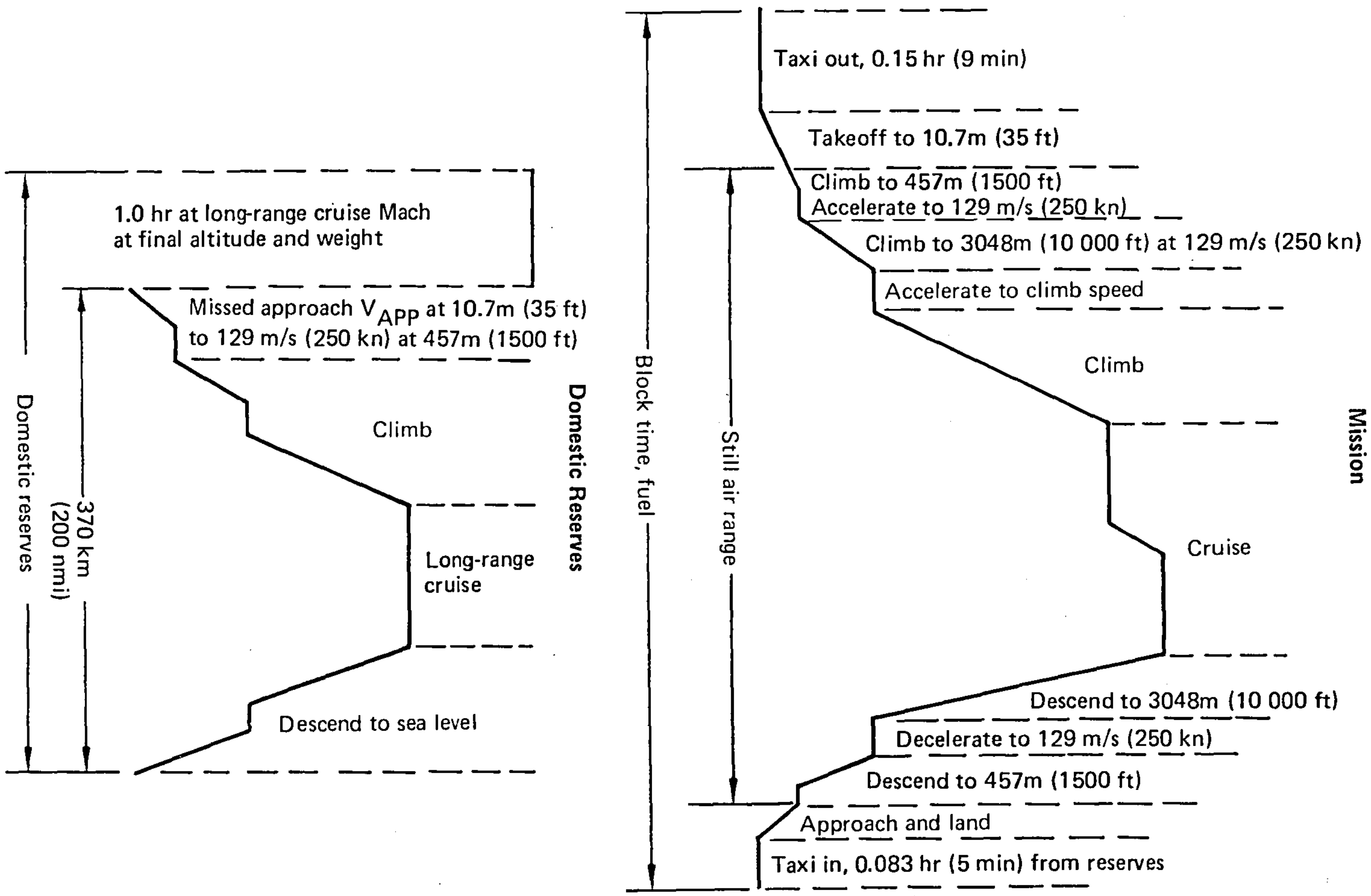
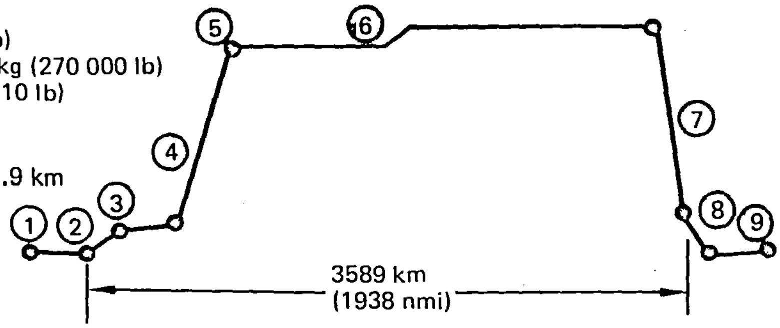


Figure 13. Typical Mission Profile

Taxi weight: 122 600 kg (270 350 lb)
Brake release gross weight: 122 450 kg (270 000 lb)
OEW and payload: 96 150 kg (212 010 lb)
Range: 3589 km (1938 nmi)
Fuel load: 26 450 kg (58 340 lb)
Type cruise: M = 0.80 at 10.7 to 11.9 km (35 000 to 39 000 ft)
Temperature: Standard day
Field elevation: Sea level
Block time: (4.682 hr)
Block fuel: 19 900 kg (43 940 lb)



Mission segments		Fuel burned, kg (lb)	Fuel remaining, kg (lb)	Weight at end of operation, kg (lb)	Time, hr	Distance, km (nmi)
1 Taxi out	0.15 hr (9 min)	160 (350)	26 300 (57 990)	122 450 (270 000)	0.150[a]	–
2 Takeoff, sea level to	10.6m (35 ft)	140 (300)	26 160 (57 690)	122 300 (269 700)	0.017	–
3 Retract gear and flap, accelerate 10.7 to 460m and 129 m/s (35 to 1500 ft and 250 KCAS)		360 (800)	25 800 (56 890)	121 950 (268 900)	0.033	11.1 (6)
4 Climb to	10.7 km (35 000 ft)	3435 (7575)	22 365 (49 315)	118 510 (261 325)	0.448	339 (183)
5 Accelerate to cruise speed (Mach = 0.80)						
6 Cruise	10.7 to 11.9 km (35 000 to 39 000 ft)	15 035 (33 155)	7330 (16 160)	103 480 (228 170)	3.554	3030 (1636)
7 Descend to	457m (1500 ft)	340 (755)	6985 (15 405)	103 130 (227 415)	0.297	209 (113)
8 ILS approach and land		360 (800)	6625 (14 605)	102 770 (226 615)	0.100	–
9 Taxi in	0.083 hr (5 min)	90 (200)[a]	–	–	0.083[a]	–
Total mission			19 835 (43 735)	Trip air time	4.449	3590 (1938)
10 Total reserves		6625 (14 605)				

[a]Not included in total mission fuel or trip air time; taxi time included in block time, taxi-in fuel (from reserves) included in block fuel

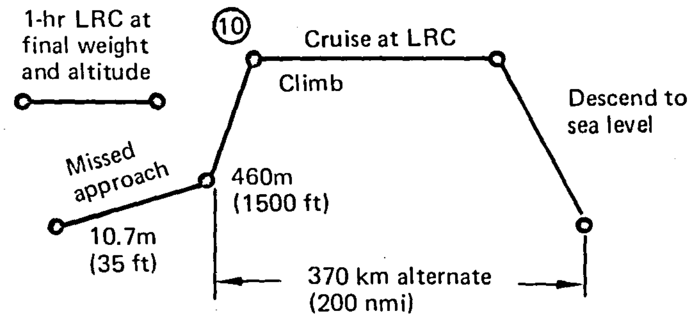


Figure 14. Mission Summary

The identical speed schedule, in reverse, is followed for descent to landing. The altitude/speed relationship of this schedule is illustrated in Figure 15.

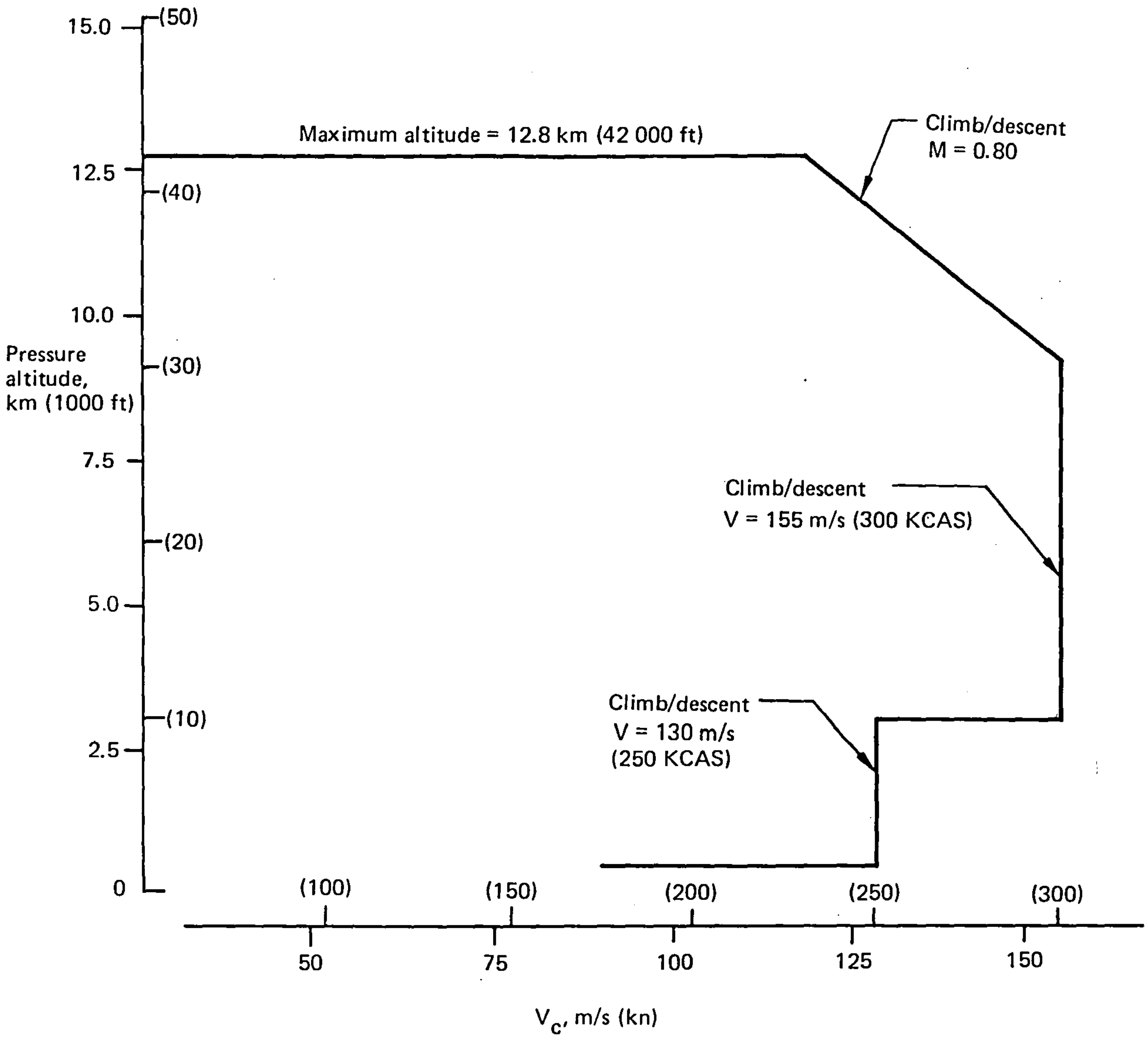


Figure 15. Climb and Descent Speed Schedule

5.2.3 PERFORMANCE CHARACTERISTICS

The performance characteristics computed for the Baseline Configuration are presented and compared to the DRO in Table 3.

5.2.4 NOISE

The Baseline Configuration's nominal noise levels are listed below:

- takeoff
  - without cutback--95 EPNdB
  - with cutback--90 EPNdB
- sideline--93 EPNdB
- landing approach--101 EPNdB

The engine nacelles of the Baseline Configuration are acoustically treated to achieve these noise levels with reasonable confidence. As a reference, airframe noise alone for landing approach is 94 EPNdB and represents a limitation on how far the total noise can be reduced.

The above nominal predictions, with proper design tolerances, are sufficient to meet Stage 3, FAR 36 Noise Certification Requirements.

5.3 WEIGHT, BALANCE, AND INERTIA

Page

5.3 WEIGHT, BALANCE, AND INERTIA

This section presents the weight and balance definition of the Baseline Configuration from the overall airplane standpoint. Detailed weight and inertia data are provided in Section 7.4.

An overview of the Baseline task in Subsection 5.4.1 presents objectives for the Weight Analysis task. Design weights are described in Subsection 5.4.2; airplane moments of inertia are described in Subsection 5.4.3; and cg management data are contained in Subsection 5.4.4.

5.3.1 TASK OVERVIEW

The objectives of the weight and balance analyses were to:

- develop a weight data base of the selected Baseline Configuration that is as detailed as possible

- improve the quality of the weight and balance definition of the Reference Configuration within pre-established constraints of resources and configuration definition

- calibrate the methods of weight analysis to the Reference Configuration definition. The relationship of the Reference Configuration to the Boeing New Airplane program and the Integrated Application of Active Controls (IAAC) project is described in Section 2 and illustrated in Figure 3.

The Baseline Configuration was derived from the Reference Configuration. The primary objective of developing the Baseline Configuration weight data base was achieved and is adequate to accomplish succeeding tasks in the IAAC project (active controls technology [ACT], Wing Planform Study, Final ACT).

Improvements to the design weight definition of the Reference Configuration were made by incorporating a new wing box analysis, a revised main landing gear design, and a new cg analysis.

The methods of weight analysis to be used on the IAAC project were calibrated to correlate with the Reference Configuration weights.

5.3.2 DESIGN WEIGHTS

Design weights used for structural loads analysis are listed in Table 4. The

Table 4. Design Weights

Item	Weight	
	kg	(lb)
Operational empty weight (OEW)	78 300	(172 610)
Maximum design zero fuel weight (MZFW)	104 400	(230 160)
Maximum design landing weight (MLW)	112 570	(248 160)
Maximum design takeoff weight (MTOW)	122 470	(270 000)
Maximum design taxi weight (MTW)	122 900	(271 000)

corresponding terminology in Table 4, which is in accordance with Air Transport Association (ATA) Specification 100, Section 2-8-3 (ref 3), is:

- Operational empty weight (OEW): Manufacturer's empty weight plus standard and operational items. Standard items are unusable fuel, engine oil, emergency equipment, toilet fluid and chemical, galley structure, etc. Operational items are crew baggage, manuals, removable service equipment for cabin and galley, food and beverages, baggage and cargo containers, etc.

- Maximum design zero fuel weight (MZFW): Maximum weight allowed before usable fuel and other specified usable agents must be loaded in defined sections of the aircraft as limited by strength and airworthiness requirements. It is the sum of OEW and maximum allowable payload.

- Maximum design landing weight (MLW): Maximum weight for landing as limited by aircraft strength and airworthiness requirements.

- Maximum design takeoff weight (MTOW): Maximum weight for takeoff as limited by aircraft strength and airworthiness requirements. This is the maximum weight at the start of the takeoff run. It is occasionally defined as maximum brake release weight.

- Maximum design taxi weight (MTW): Maximum weight for ground maneuver as limited by aircraft strength and airworthiness requirements. It includes weight of taxi and runup fuel.

5.3.3 AIRPLANE MOMENTS OF INERTIA

Airplane moments of inertia for the Baseline Configuration about the three airplane reference axes X, Y, Z (global system) are shown in Figures 16 through 18. The X and Y axes are parallel to a waterline plane.

5.3.4 CENTER OF GRAVITY MANAGEMENT

A cg management (loadability) diagram is presented in Figure 19. In determining the cg loading range requirements, a tolerance (+3% MAC and -4% MAC) was applied to the nominal OEW cg to account for manufacturing variations and airline options, such as increased passenger accommodations and engine substitution. The aft payload envelope was critical for 197 mixed-class passengers (18/179), establishing the aft operating cg limit. The forward envelope was critical for 207 tourist-class passengers and established the forward operating cg limit required. The "typical payload (mixed)" line is a prediction of passenger loading obtained from many airline surveys conducted throughout the airline industry.

The moment vectors of the cargo stored in the forward and aft cargo compartments were based on 22 LD-2s at 105 kg/m^3 (6.58 lb/ft^3). Adding the bulk cargo (located aft of the containerized cargo) completed the payload envelope for the zero fuel weight airplane.

Center of gravity locations required for fuel usage were established with the application of in-flight tolerances of +81 120 Nm (718 000 in-lbf) aft, and -37 280 Nm

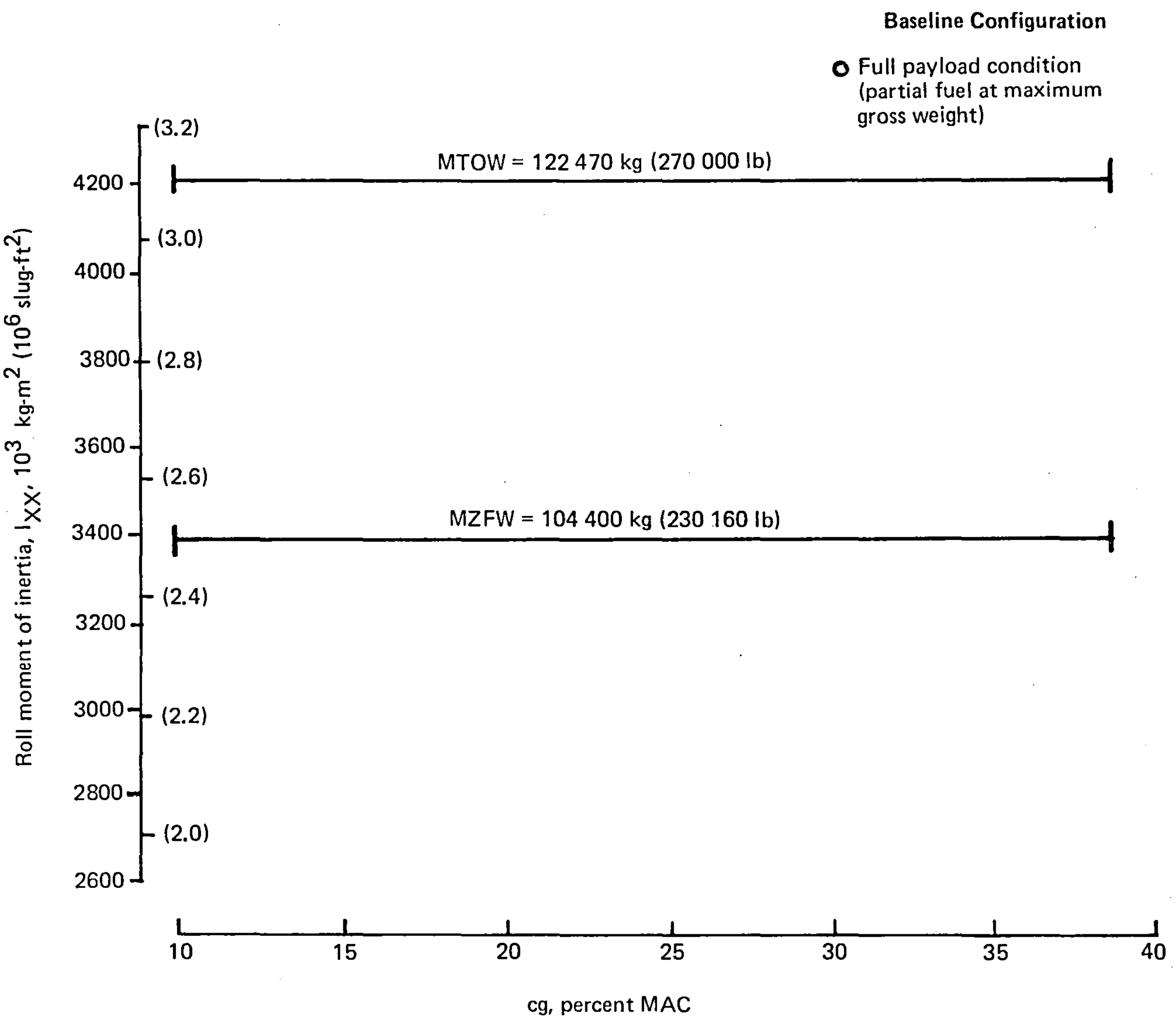


Figure 16. Roll Moment of Inertia

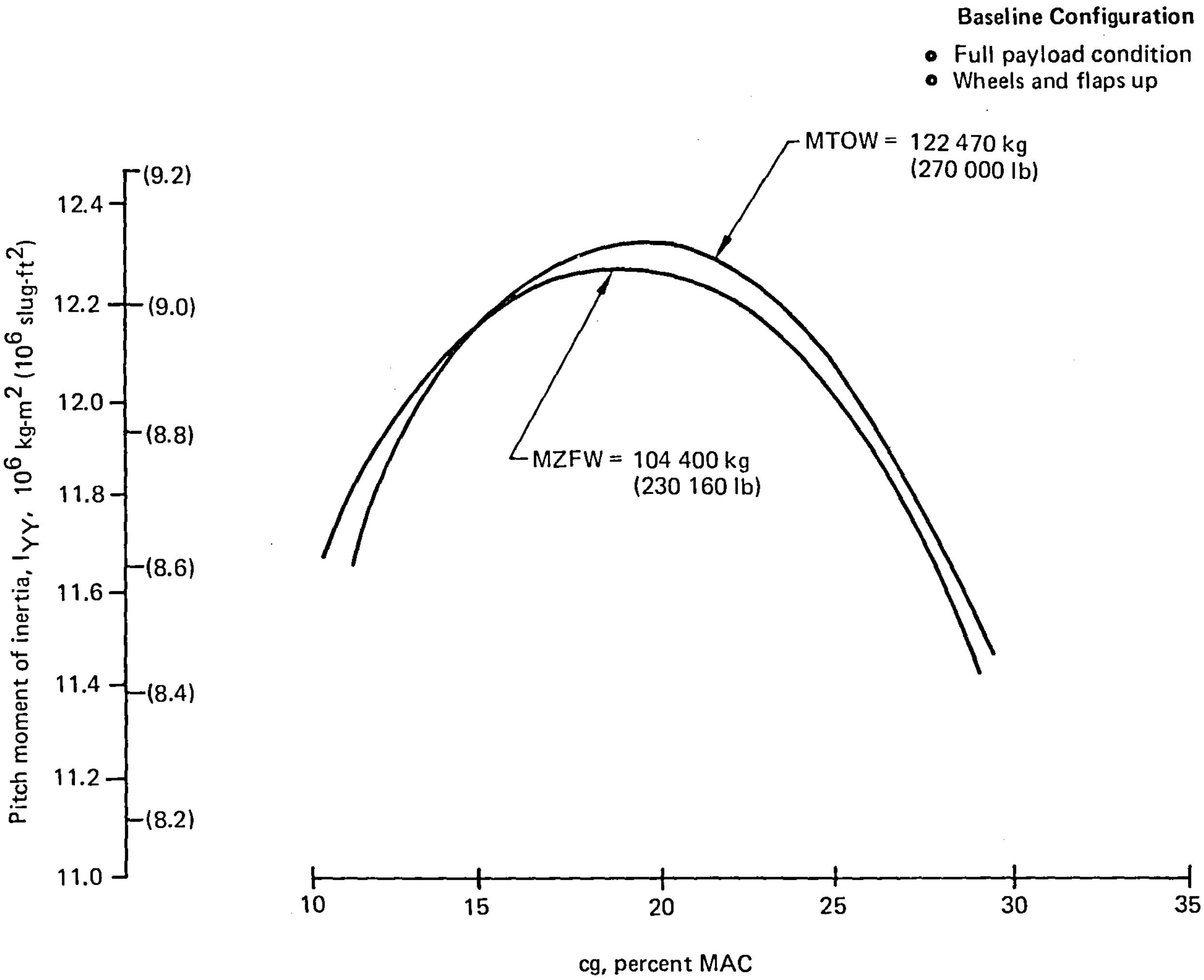


Figure 17. Pitch Moment of Inertia

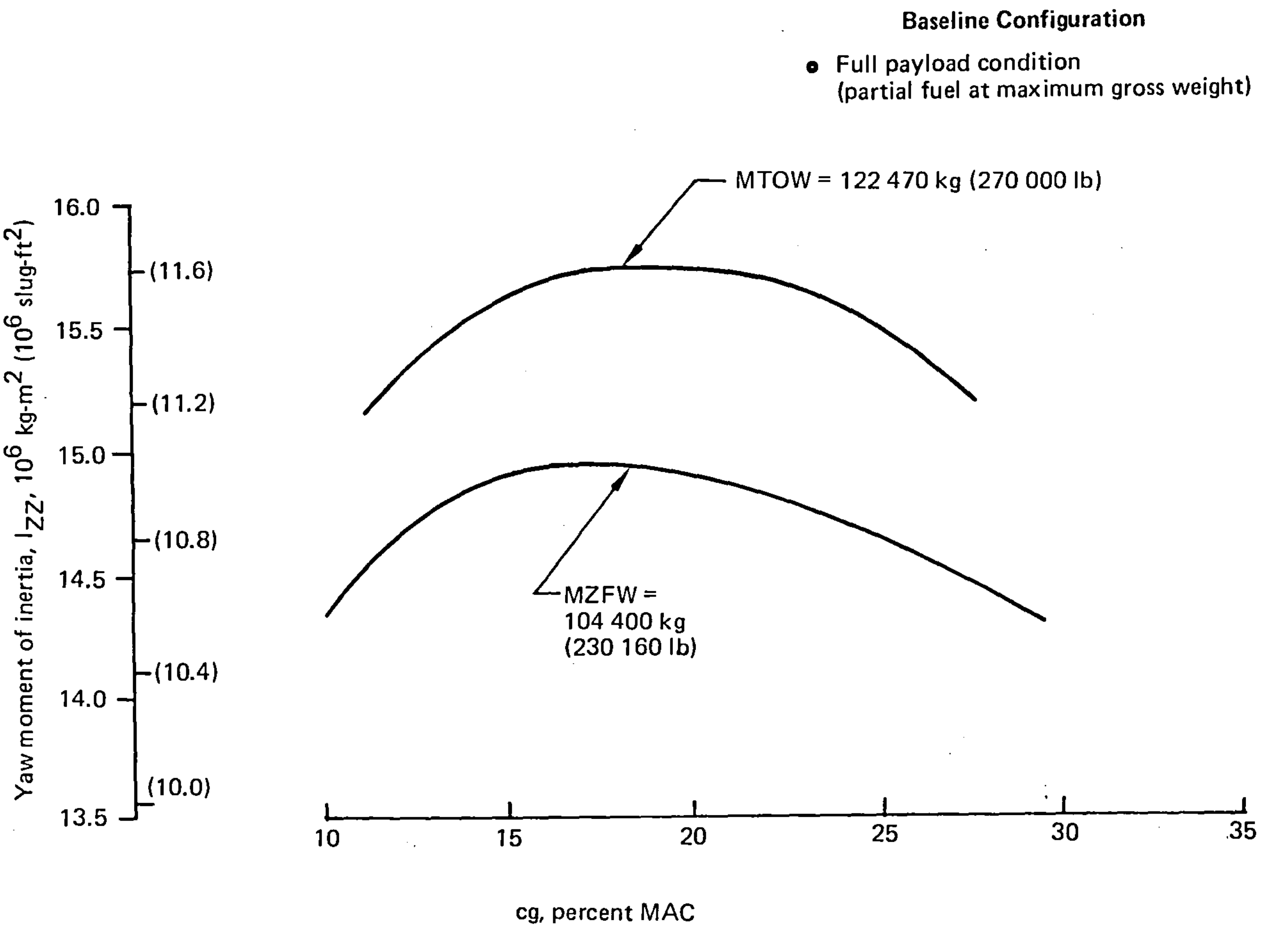


Figure 18. Yaw Moment of Inertia

Baseline Configuration

- $\frac{\bar{c}}{4}$ balance arm = 25.11m (988.7 in)
- MAC = 6.03m (237.5 in)
- Main landing gear location = 56.1% MAC

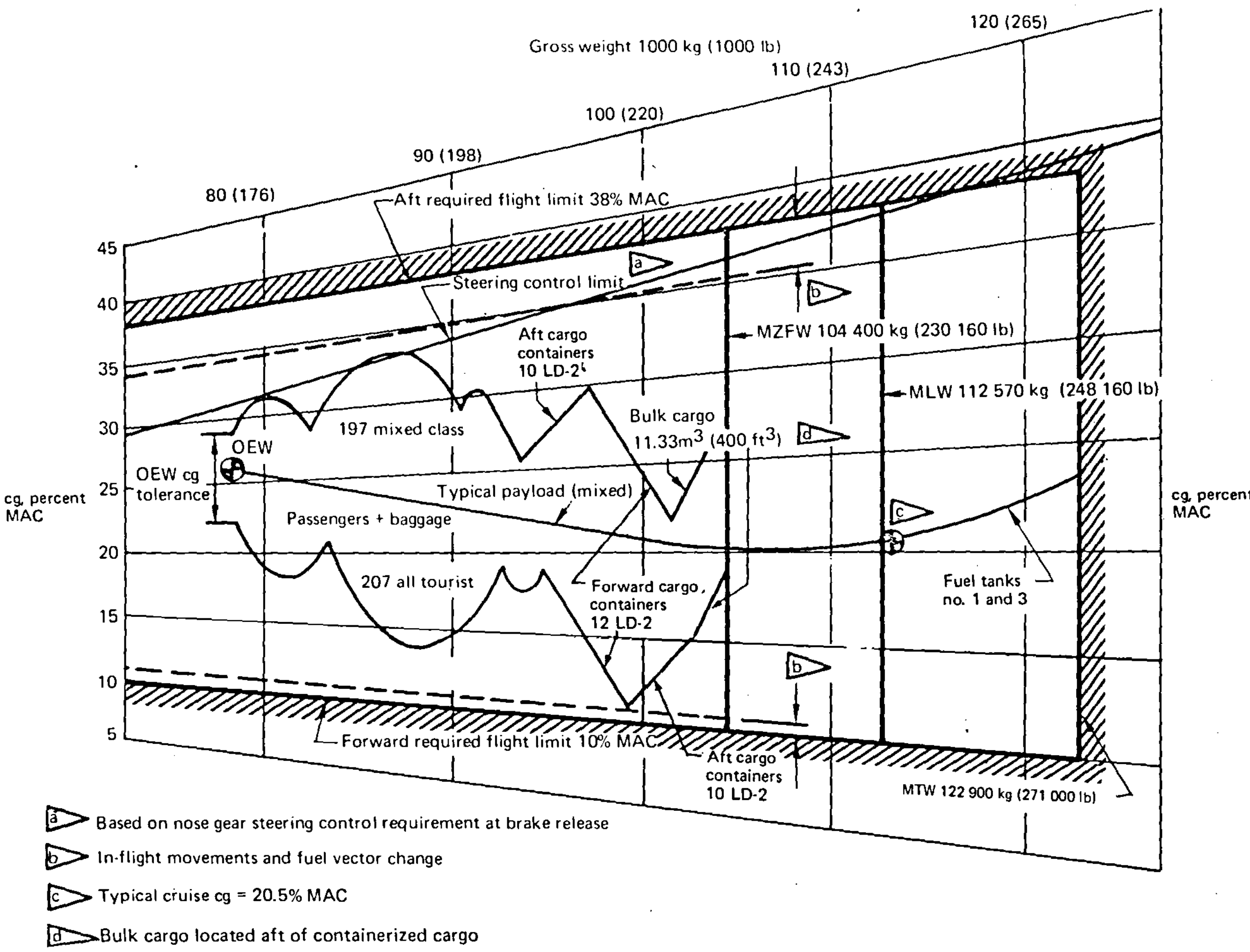


Figure 19. Center of Gravity Management

(330 000 in-lbf) forward. These are representative of passenger and crew movement, control surface deflections, landing gear movement, etc. Refer to note "(b)" in Figure 19.

The fuel system consists of one main tank per side. The aft required flight cg (38% MAC) limit was established by the full fuel condition. The forward required flight cg limit was established by a partial fuel condition. "Off loading" (i.e., the capability to trade passengers for fuel) requires the fuel volume to be larger than needed for the design mission.

The typical cruise cg was based on a definition consistent with the performance analysis ground rules used for a typical airline customer.

6.0 DESIGN DATA

6.0 DESIGN DATA

The design data, established for the Baseline Configuration, are described in this section. A structures description of the major airplane components is presented in Subsection 6.1, followed by a description of the major airplane systems that will affect or be affected by active control systems (subsec 6.2).

6.1 AIRPLANE STRUCTURE

The airplane structure is presented in six major elements and is described in the following subsections:

- wing (subsec 6.1.1)
- body general (subsec 6.1.2)
- flight deck (subsec 6.1.3)
- horizontal tail (subsec 6.1.4)
- vertical tail (subsec 6.1.5)
- main and nose landing gear (subsec 6.1.6)

Conventional materials and construction are used in the design and fabrication of the airframe, except for a limited amount of graphite epoxy composite secondary structure. The airframe consists primarily of aluminum alloys, including advanced alloys selected to offer a high degree of structural reliability for the operational requirements and service life of the airplane. Highly stressed landing gear components are fabricated from high-strength vacuum melt steel.

6.1.1 WING

The wing consists of left and right main outboard sections, joined to a wing center section at the side-of-body (see fig. 20 for a structure diagram of the wing). The outboard sections include the wing box, the fixed leading- and trailing-edge structure, the leading-edge slats and trailing-edge flaps, the ailerons, the spoilers, and the wing tip. The wing box structure is of conventional two-spar construction. Use of flush fasteners and butt skin splices ensures that all external surfaces of the wing are smooth. Nonstructural access panels are provided throughout the wing for inspection, maintenance, and repair.

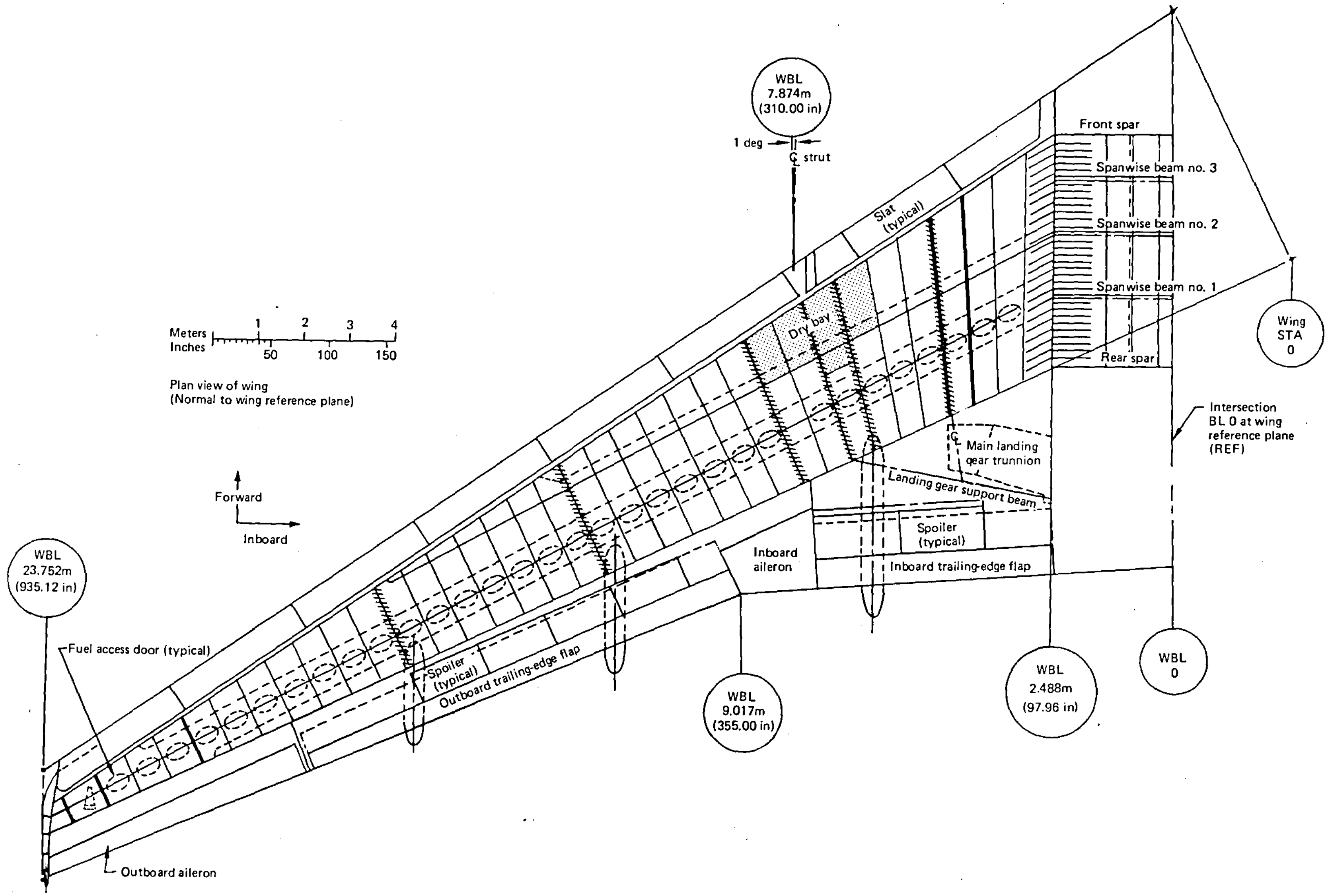


Figure 20. Wing Structure Diagram

The outboard wing box structure consists of stringer-stiffened upper and lower panels and built-up spars and ribs. It is joined to the center section at the side-of-body rib. The lower panel side-of-body splice is a double shear design to reduce eccentricity and improve durability. The spars consist of upper and lower machined chords, machined webs with pads around cutouts, and machined, extruded web stiffeners. The intermediate ribs are built up with extruded chords, stiffeners, and sheet webs. Tank end ribs incorporate fuel pans.

Special ribs at engine and landing gear supports, trailing-edge flap supports, and the side-of-body joint incorporate backup and terminal fittings and skin panel shear ties as required. Chords and stiffeners are machined extrusions, and the webs are machined plate. Pin joints attach the landing gear support beam to the rear spar and fuselage. The space between the front and rear spars and upper and lower wing panels of the outboard wing sections is liquid-vapor sealed to provide fuel storage. The volume is divided for fuel system requirements by "tank end" ribs, and baffles are provided to control the fuel center of gravity (cg).

The wing center section structure consists of stringer-stiffened upper and lower panels, built-up front and rear spars, three spanwise, full-depth beams, and a centerline rib. Fore and aft internal intercostals are used on the lower surface to provide fixity for stiffeners. External fore and aft floor beams on the upper surface provide fixity for the upper surface stiffeners. The center section is a dry bay area but includes fuel seal planes and structural provisions for an integral fuel tank.

The outboard wing leading-edge slats consist of eight three-position slat assemblies per side. Each slat is supported by two machined tracks, programmed by two auxiliary tracks and actuated by a ball screw actuator. An additional two-position slat seals to the inboard side of the nacelle strut in the extended position. The strut seal slat is supported by two machined tracks and is actuated by a rotary gear box and linkage mechanism.

The trailing-edge flaps are a single-slotted design and consist of one inboard and one outboard section. The flaps are supported and extended by chordwise-oriented linkage mechanisms actuated by rotary gear boxes (see fig. 21). A portion of each flap support mechanism extends below the wing contour and is enclosed in streamwise fairings.

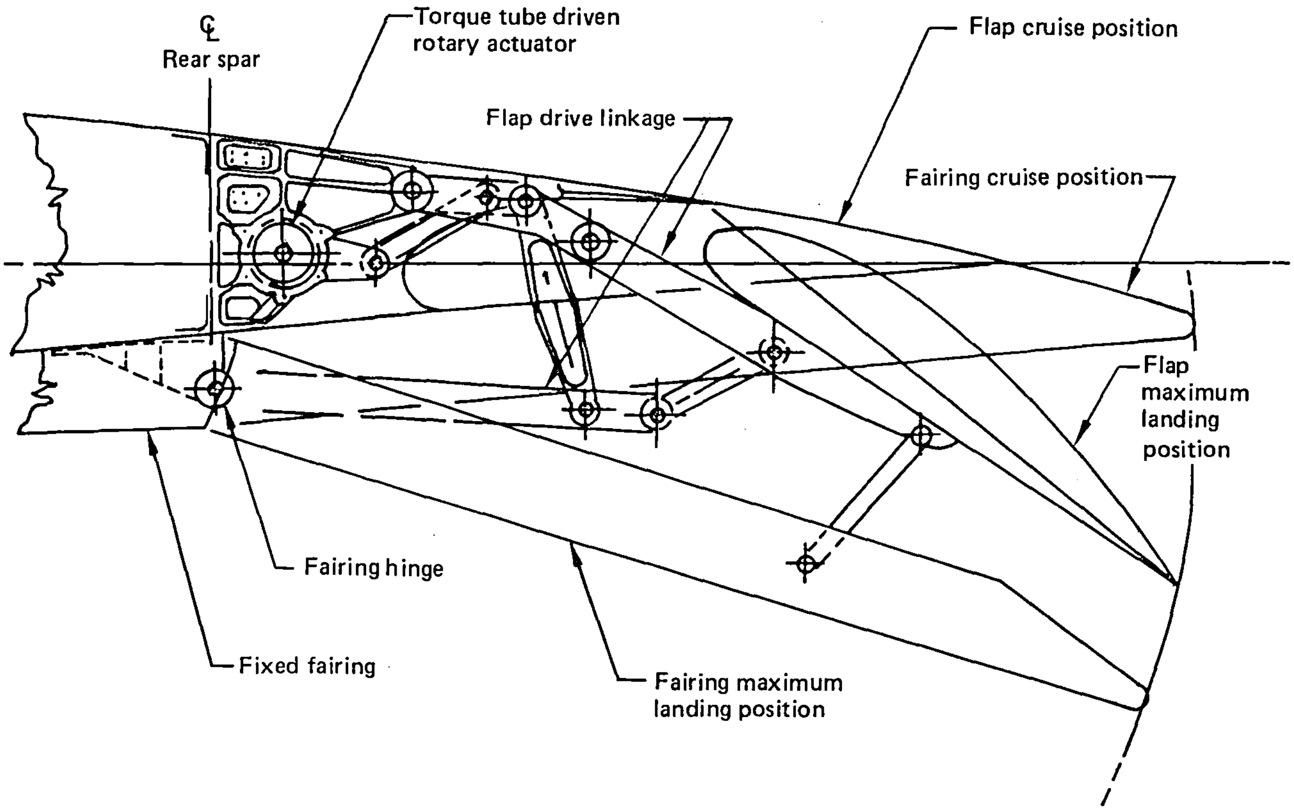


Figure 21. Trailing - Edge Flap—Single-Slotted, Linkage Supported

Inboard and outboard ailerons are provided on the Baseline Configuration (fig. 20). Each aileron is hydraulically actuated and attached to the rear spar on self-aligning bearings.

Six hydraulically operated, flush spoilers are provided in the upper surface of each wing aft of the rear spar. The four outboard spoilers on the left and right wing are identical. The two inboard spoilers on the left wing are opposite to the two on the right wing.

6.1.2 BODY

The body consists of permanently joined major subsection assemblies, with a double-lobe cross section formed by upper and lower radii, faired together with a second-degree curve. The basic body structure of aluminum alloy material is of semi-monocoque construction with formed hat section longitudinal stiffeners attached to the skin panels. Basic body frames are pitched at 0.559m (22 in). The longitudinal skin splices are lap joints, while the circumferential skin splices are butt joints. In noncritical aerodynamic areas, the upper fastener row of the longitudinal joints incorporates protruding head fasteners to meet durability objectives. A centerline diagram of the body structure is shown in Figure 22.

The nose section assembly contains the radome, flight deck compartment, electric panels and electronic racks, the wheel well for the nose landing gear, overhead emergency hatch, and access hatches in the lower lobe. A radome of fiberglass sandwich construction is installed on the nose of the aircraft to accommodate the localizer, glide slope, and weather radar antennas.

The upper lobe between the flight deck compartment and the aft pressure bulkhead contains the passenger compartment with four entry/service doors, and two Type III exits. The lower lobe contains the forward and aft cargo compartments, landing gear wheel wells, wing center section, two 1.778 x 1.753m (70 x 69 in) cargo doors, one bulk cargo door, electric/electronic equipment, and air-conditioning pack. The unpressurized portion of the body behind the aft pressure bulkhead contains the auxiliary power unit (APU) and firewall and has space provisions for a tail skid.

The pressure boundary includes the body shell, forward bulkhead, aft bulkhead, the walls of the two landing gear compartments, and the wing center section. In the region of the wing-to-body joint, the pressure boundary is the wing front spar and bulkhead below the front spar, the wing center section upper skin panel, a horizontal pressure web which extends aft over the main landing gear wheel well, and the aft wheel well bulkhead. The aft pressure boundary is a domed pressure bulkhead. A floor shear tie beam extending fore and aft at the top of the floor beams is constructed as a truss to permit intercompartmental venting.

The body construction behind the aft pressure bulkhead is similar to the forward body described in the preceding paragraphs, except for the additional machined bulkheads, firewalls, and vertical tail attachment fittings. Support structure for mounting the APU equipment is also located in this section.

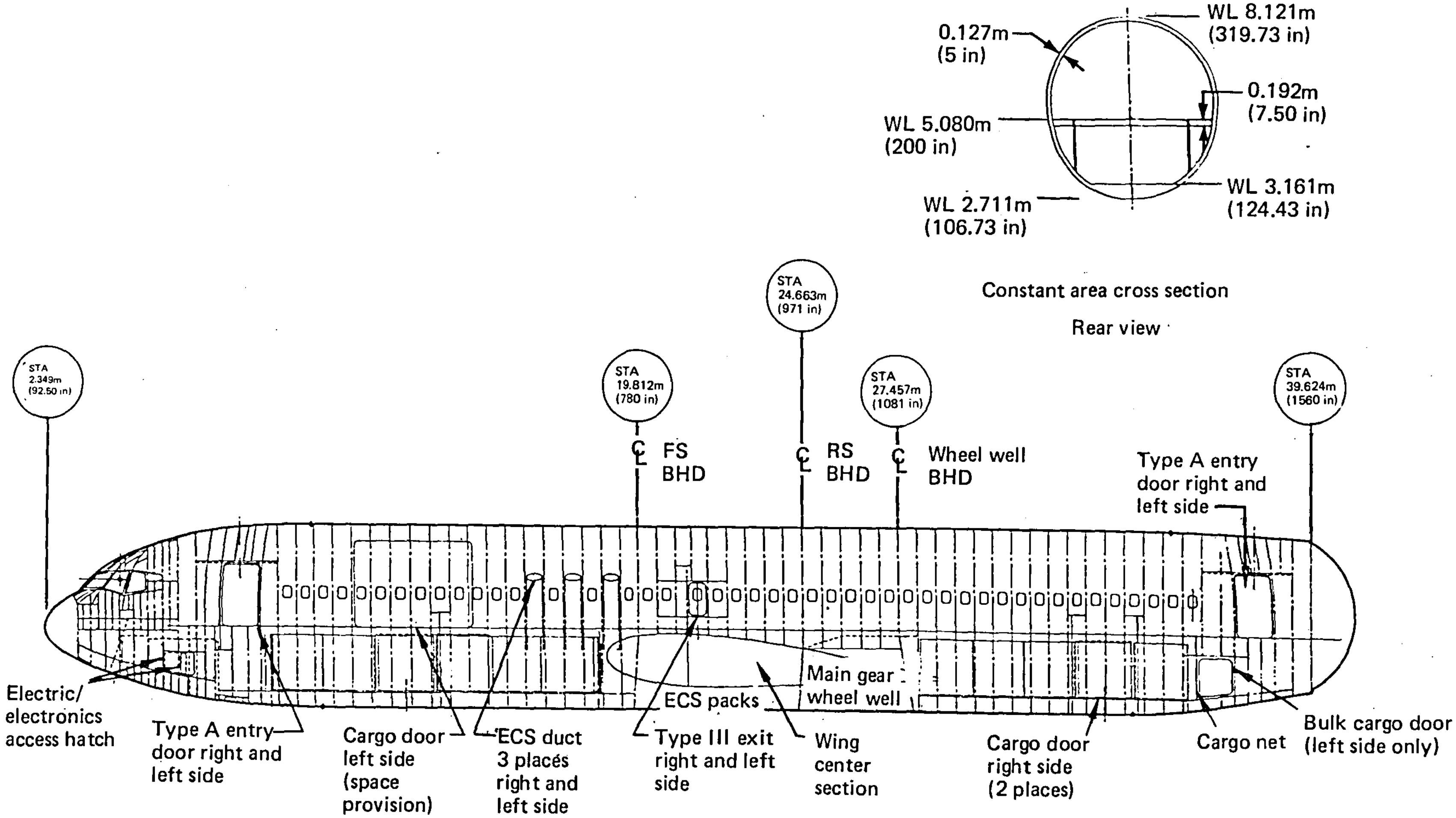


Figure 22. Body Centerline Diagram

The wing-to-body joint is designed so that the full body depth is effective in the vertical bending mode in the area of the wing center section. The wing-to-body joint consists of machined fittings, which attach the main body bulkheads to the front and rear spars, and the body skin which attaches to the wing upper surface "plus" chord. The intermediate frames attach to the wing at the side-of-body rib and to the outboard longitudinal floor beams. The wing-to-body fairing is readily removable for inspection and maintenance. The fairing panels are of fiberglass sandwich construction.

There are two wheel well cavities, one for the main landing gear and one for the nose landing gear. The main wheel well cavity is encompassed by an upper pressure deck, wing rear spar, aft wheel well bulkhead, keel beam, and main gear doors. The gear doors consist of single piece doors attached to the lower keel box and a small door attached to the main gear strut. The nose wheel well cavity is enclosed by vertical sidewalls, a horizontal pressure deck, forward and aft pressure bulkheads, and the nose-gear doors. Two pairs of clamshell doors are actuated mechanically by the gear; the forward pair is closed when the gear is down.

The passenger cabin windows are 0.254 x 0.356m (10 x 14 in) in size and consist of stretched acrylic outer and cast acrylic center panes. Four main entry/service doors are 1.067 x 1.829m (42 x 72 in) Type A overhead opening plug type. Doors are provided on left and right sides of the passenger compartment at the forward and aft ends.

Two cargo compartment doors on the right-hand side of the airplane provide clear openings that are 1.753m (69 in) high by 1.778m (70 in) wide. The doors are outward opening nonstructural plug-type doors that are similar to the 727-200 cargo doors except they are electrically actuated. An aft bulk cargo compartment door, which is provided on the left side of the airplane, is a 0.965 x 1.219m (38 x 48 in) inward-opening plug-type door. Two emergency exit doors are provided, over the wing root near the center and on each side of the passenger compartment. They are 0.508 x 0.965m (20 x 38 in) Type III inward-opening plug-type doors. An inward-opening, plug-type, access hatch, located at the bottom of the body, is provided for the electric/electronic equipment bay and is operable from either inside or outside the airplane. An inward-opening, plug-type, overhead hatch, operable from either inside or outside the airplane, is provided for crew emergency exit. An inward-opening, plug-type, access hatch located at the bottom of the body is provided for access to the lower lobe below the flight deck compartment. One outward-opening unpressurized door is provided for access to the interior of the aft body section. Also, unpressurized doors are provided for access to the APU.

The main cabin floor installation provides for track-mounted passenger seats. The seat tracks are installed throughout the cabin except where permanently installed facilities are provided. The tracks are flush with the floor, and 0.0254-m (1-in) incremental seat adjustment is provided. The main cabin floor panels are constructed of fiberglass face sheets with Nomex core that are service proven and lightweight. The floors under lavatories, galleys, and entryways are sealed to prevent liquid leakage through the floor. Floor panels are designed so that all exposed panels can be replaced without removing major components such as galleys and lavatories.

The flight deck floor aft of the pilot seats is similar to that in the main cabin. Structural aluminum sheet flooring is used in the forward portion of the flight deck.

The forward and aft cargo compartment support structure is designed to provide for LD-1, LD-2, LD-3, and LD-8 containers.

6.1.3 FLIGHT DECK

The flight deck is designed for a three-person crew consisting of captain, first officer, and flight engineer. Critical controls and displays are accessible to or duplicated for at least two crew members, allowing safe operation to be maintained by any two crew members. A first observer's station is provided, and a second observer's station is available as an option. The flight deck is designed to accommodate persons 1.575 to 1.905m (62 to 75 in) tall at all crew stations.

The flight deck from the forward bulkhead to the right-hand partition is 2.819m (111 in) and to the left-hand partition 2.616m (103 in). The flight deck door is offset left of the centerline to allow crew entry/exit without interfering with the flight engineer. The flight deck floor is located 0.203m (8 in) above the passenger floor with a step at the entry door. The flight deck compartment is separated from the passenger compartment by a nonstructural bulkhead which includes a lockable door.

The aft partition of the flight deck is designed to withstand a maximum of 7.584 kPa (1.1 psi) differential pressure. The flight deck door is designed to blow open to provide pressure relief in case of rapid depressurization.

A crew escape hatch is located in the ceiling just forward of the aft partition. Four inertia-type escape tapes are located next to the hatch and space for a fifth tape is provided for the optional second observer. Two fixed, forward windshields and four fixed, side windows are installed in the flight deck. The windshields, which are flat, are of glass and plastic laminate construction. The flight deck side windows are curved and of plastic laminate construction. The total window area is less than 3.226 m^2 (5000 in^2). Visibility meets the requirements of AS-SAE-580B (ref 4) 3-second rule with full flaps.

6.1.4 HORIZONTAL TAIL

The horizontal tail (see fig. 23) is adjustable for airplane pitch trim. It is hinged at the rear spar and actuated by a fail-safe jack screw actuator. The elevator consists of inboard and outboard segments controlled by hydraulic actuators and is removable at the actuators and hinges. The horizontal tail tapers in thickness and width. Space provisions for logo lights are included.

The horizontal tail primary structure consists of a torque box from the side-of-fairing rib to the tip rib. The torque box is constructed of stiffened panels supported by built-up ribs and spars. The leading edge consists of a forward removable assembly of skin supported by closely spaced sheet metal ribs. The center section consists of the front and rear spars.

6.1.5 VERTICAL TAIL

The vertical tail is a fixed surface attached to the aft body which supports the horizontal tail (see fig. 24). The rudder consists of an upper and lower double-hinged segment controlled by hydraulic actuators; the rudders are removable at the actuators

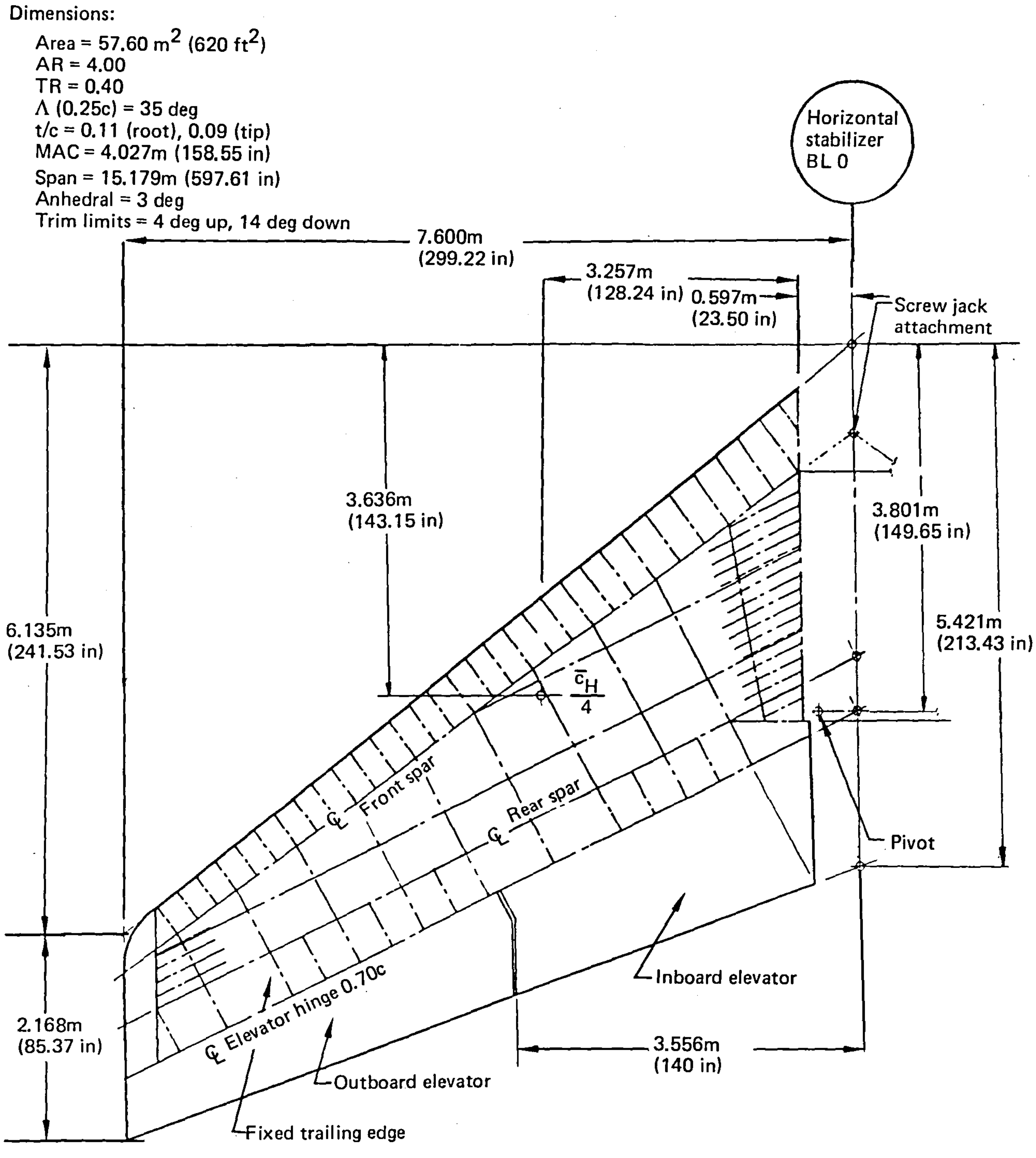
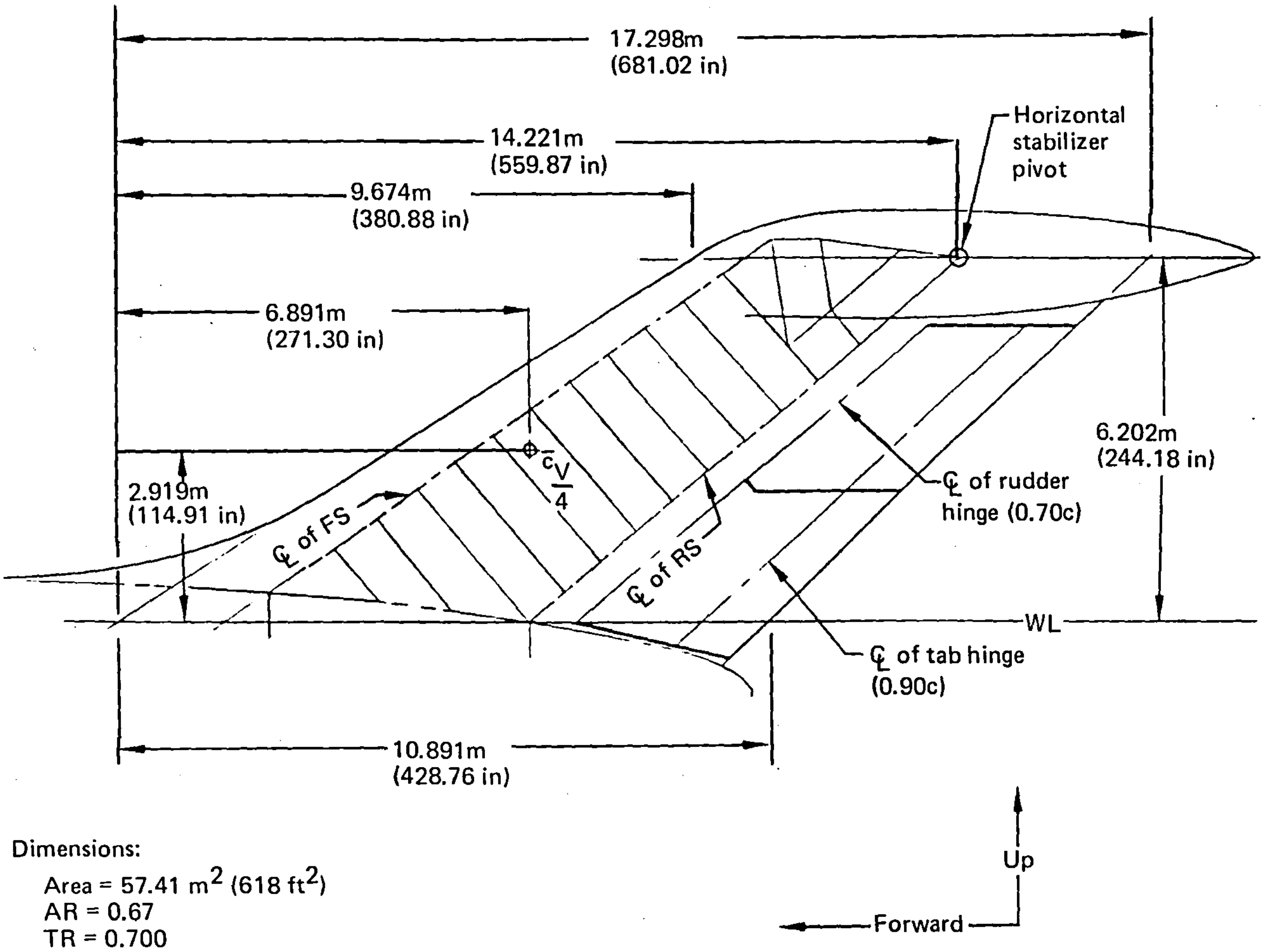


Figure 23. Horizontal Stabilizer Geometry, Plan View



Dimensions:

Area = 57.41 m^2 (618 ft^2)
AR = 0.67
TR = 0.700
Λ (0.25c) = 55 deg
t/c = 0.12
Root chord = 10.891m (428.76 in)
Tip chord = 7.623m (300.14 in)
MAC = 9.353m (368.23 in)
Span = 6.202m (244.18 in)

Figure 24. Vertical Stabilizer Geometry, Left Hand Side View

and hinges. The vertical tail tapers in thickness and width. Space provisions for a very high frequency omnidirectional radio range (VOR) antenna are included.

The vertical tail primary structure consists of a full span torque box. The torque box is constructed of stiffened panels supported by built-up ribs and spars. The leading edge consists of a forward removable assembly supported by closely spaced sheet metal ribs.

6.1.6 LANDING GEAR

One nose gear and two main gears are used in the landing gear system. All three shock absorbers are single-stage oleo-pneumatic struts. Main and nose gears have hydraulically operated uplock and downlock mechanisms. Structural and space provisions are incorporated for a weight and balance system and a brake temperature monitor system.

Landing gear primary structure is steel while the wheels are forged aluminum alloy. Brakes with steel heat sinks are provided and space is incorporated in the wheel design for structural carbon brakes. Sleeve bearings are aluminum-nickel-bronze. All structural joints, static or dynamic, are bushed and lubricated.

The main landing gear consists of two four-wheel trucks, each attached to a single shock strut. Each strut attaches to wing structure through a trunnion and two folding braces and folds sideways, stowing the wheels in the fuselage. Figure 25 is a drawing of the main landing gear.

The nose landing gear consists of dual wheels mounted on a common axle attached to the shock strut. The nose gear retracts by swinging forward and wheel rotation is stopped upon gear retraction. Hydraulic linear actuators are used to steer the nose wheels. A drawing of the nose landing gear is shown in Figure 26.

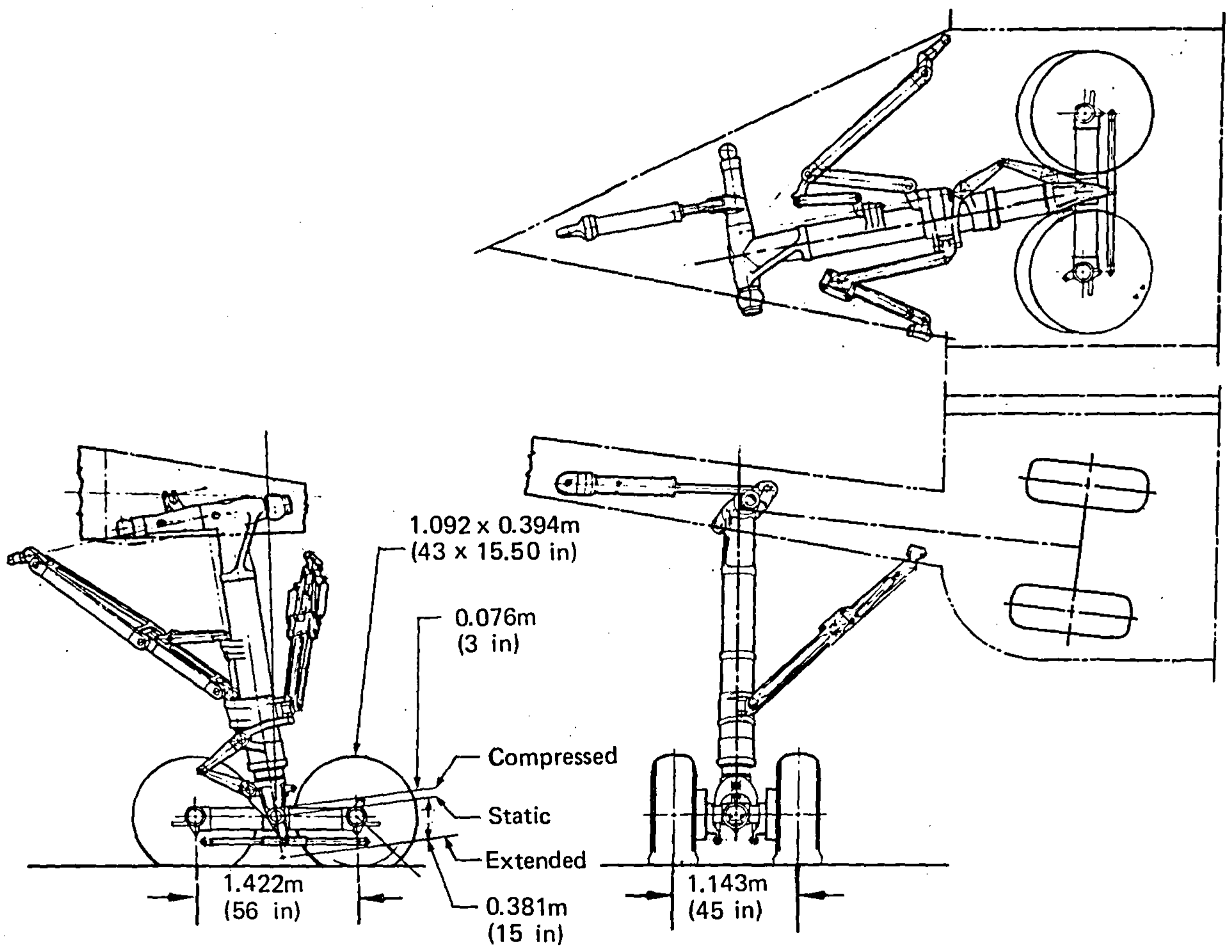


Figure 25. Main Landing Gear

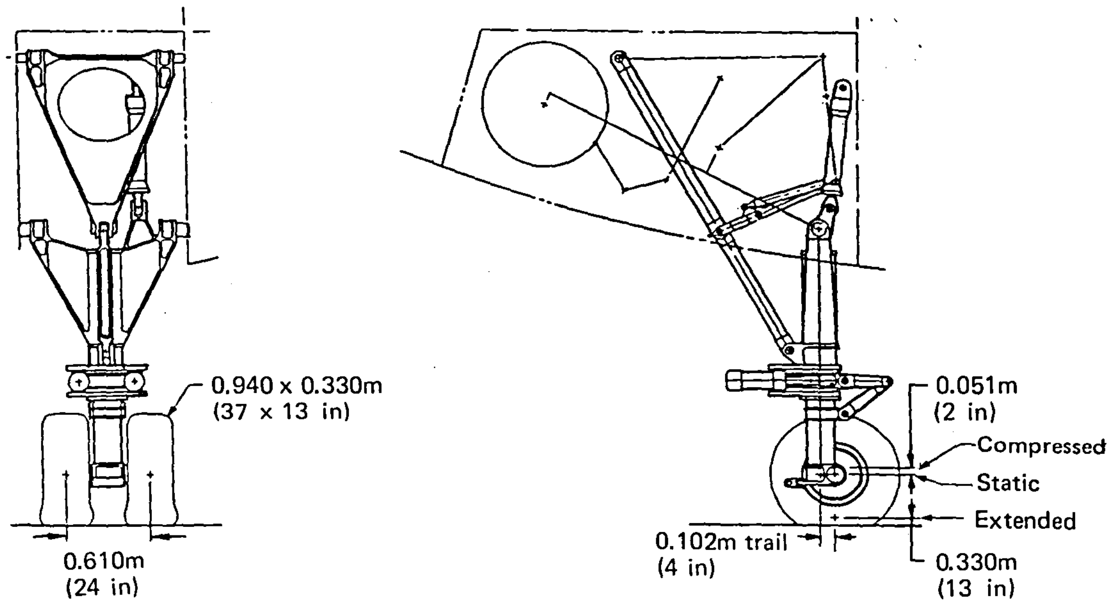


Figure 26. Nose Landing Gear

6.2 AIRPLANE SYSTEMS

6.2 AIRPLANE SYSTEMS

Baseline Configuration systems are described in this section including propulsion (subsec 6.2.1), flight controls (subsec 6.2.2), hydraulic power (subsec 6.2.3), electric power (subsec 6.2.4), flight management (subsec 6.2.5), and environmental control (subsec 6.2.6). Current proven state-of-the-art concepts were used in defining these airplane systems which will be used in defining and evaluating the systems for the ACT configurations.

6.2.1 PROPULSION SYSTEM

The propulsion system includes the propulsion control, the propulsion pod, the engine buildup and the fuel system, and the auxiliary power system (APS).

6.2.1.1 Propulsion Control System

The engine control is a hybrid comprising a basic hydromechanical and electronic supervisory control. It is capable of providing the following features:

- forward and reverse thrust operation under steady-state and transient conditions when operated with other airplane systems (i.e., reverser)
- linear thrust variation with flight deck thrust level travel throughout the airplane flight and temperature envelope (the thrust variation is free of noticeable flat spots, detents, or discontinuities)
- surveillance and limiting of maximum rotor speeds
- engine stability (start, acceleration, deceleration, steady-state, etc.)
- means to switch fuel on or off and enrich the fuel on cold-day engine start, if required

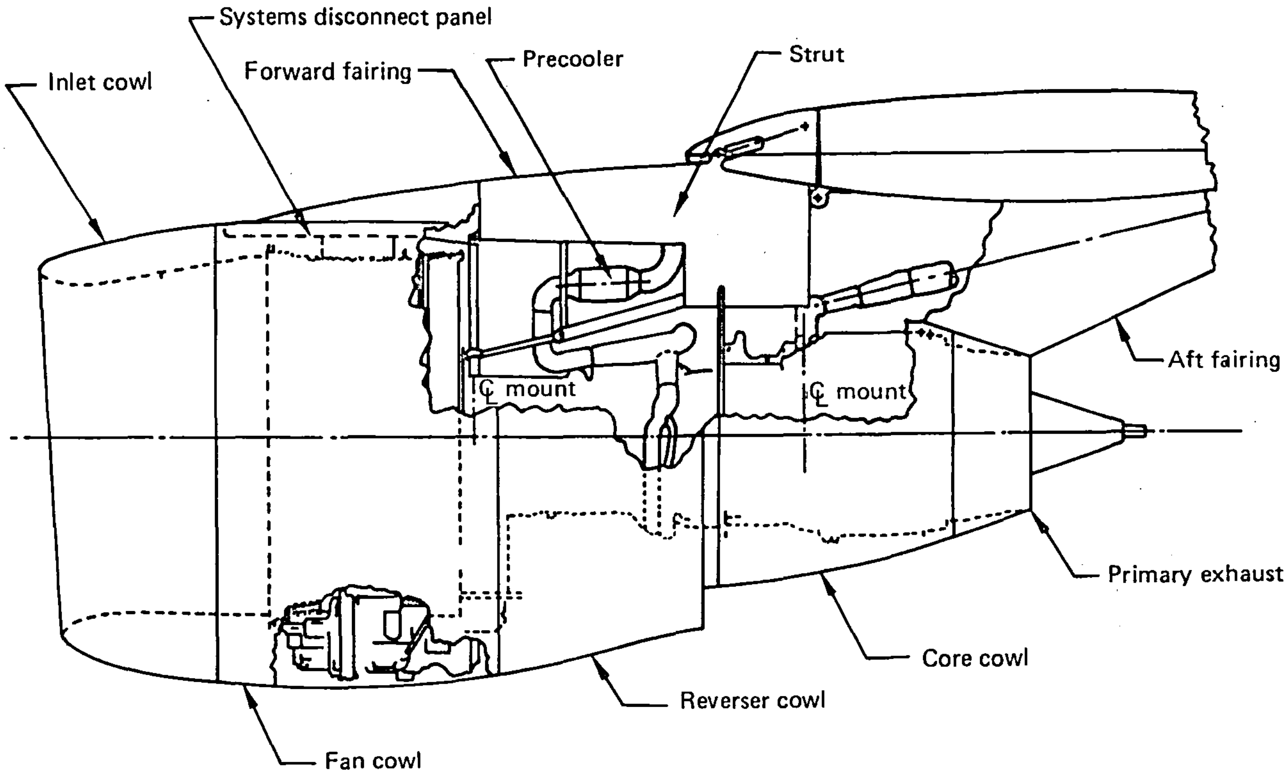


Figure 27. GE CF6-6D2 Baseline Installation

6.2.1.2 Propulsion Pod

The pod installation for the General Electric (GE) CF6-6D2 engine is shown in Figure 27. Equipment and systems comprising hydraulic and fire system tubing, electric/electronic wiring, strut-mounted fire detector, pneumatic ducting, and engine and reverser controls are located in the strut and are accessible when access panels are removed or cowls opened. Wiring, plumbing, and services from the wing leading edge terminate at a systems disconnect panel.

6.2.1.3 Engine Buildup

The engine buildup (EBU) assembly, identical for both engine positions on the airplane, consists of the engine, engine-mounted airplane accessories, equipment, instrumentation, ducting, plumbing, and wiring as described in the following subsections.

The engine includes a full-duty accessory gearbox mounted on the bottom of the engine fan case. The accessory gearbox is supplied with all engine accessories installed (fuel pump, fuel control, and engine lubrication pump) and provides pads for installation of the airplane accessories (drive generator systems [DGS], hydraulic pump, starter, and high-pressure compression speed [N_2] tach generator). The integrated drive generator system (IDGS), mounted on the engine gearbox, is rated to supply 90-kVA electric power. Electric power is transmitted by four copper conductors. Power feeder cables and the IDGS instrumentation harness terminate at the disconnect panel. A pneumatic starter is also mounted on the pad of the engine accessory gearbox.

Pneumatically supplied air from 16th-stage high-pressure engine bleed ports is manifolded and interconnected with an eighth-stage low-pressure engine bleed manifold. The starter duct is connected to the pneumatic supply from the strut at the strut-to-engine interface joint. Inlet cowl anti-ice air is ducted directly from the engine 13th-stage bleed.

Airframe instrumentation, installed on the EBU, and engine instrumentation monitor the functioning of the engine and engine-mounted accessories. The instrumentation consists of sensors, transducers, and switches to provide the signal to the flight deck instruments and indicators. Two fire detector loops are installed. Each loop consists of two detectors: one in the engine core area and one above the engine-airplane accessory gearbox.

6.2.1.4 Fuel System

A thrust control cable is installed on the engine fan case with a disconnect coupling at the strut disconnect panel. The sealed wing box structure forms an integral fuel tank for each engine fuel supply plus two surge (or collector) tanks. Access into fuel tanks is through access doors in the wing lower panel. The upper panel of the tanks contains an overwing fill port for fueling at facilities where pressure fueling equipment is not available, for defueling, and for ventilation for tank maintenance. Selected "baffle" ribs are fitted with directional flapper check valves to restrict fuel movement (slosh) and to maintain an adequate fuel supply at the boost pumps. A portion of the wing box above the engines and aft of the front spar is sealed off as a dry bay to minimize possibility of fuel spilling onto the engine in the event of an engine burst that ruptures a fuel tank.

6.2.1.5 Auxiliary Power System

The APS, which is basic equipment on the airplane, is ground and flight operable. It provides: (1) compressed air to start the propulsion engines, to operate the environmental control system, and to drive hydraulic pumps, and (2) shaft power to drive an electric generator. The APS consists of the APU, APU controller, structural support for the APU, air intake, exhaust, supporting systems (ventilation/cooling, fuel, pneumatic, fire protection, controls/instrumentation), and drains.

The APS installation involves four locations in the airplane:

- The APU is located in a dedicated compartment in the fuselage aft of the passenger cabin.

- The APU controller is mounted in the aft electric/electronics rack.

- The APS controls and instrumentation are contained in the flight deck.

- The APU ground control panel is located in the main wheel well.

Power for APU starting is obtained from a dedicated transformer rectifier unit (TRU) when the airplane has ac power, or from the airplane standby battery at other times.

The APU is a small gas turbine engine of 2.5 kg/s (5.5 lb/s) airflow and 370 to 450 kW (500 to 600 hp) with the following accessories:

- gearbox with engine accessory equipment

- gearbox-driven air compressor (load compressor) with variable inlet guide vanes for airflow regulation

- gearbox mount pad and lube/cooling oil service for a 900-kVA oil cooled electric generator

- gearbox-driven fan for cooling and ventilation

The APU starting is operable in flight and provides compressed air bleed and generator shaft power, but not simultaneously. It provides generator shaft power only to 10 670-m (35 000-ft) altitude.

6.2.2 FLIGHT CONTROL SYSTEM

The flight control system controls the position of the ailerons, spoilers, rudders, elevators, horizontal stabilizer, and the leading- and trailing-edge flaps to maneuver the airplane, control lift on the wing, and trim out the steady-state control load. Based on their functions, the system is divided into a primary, a secondary, and an automatic flight control system as described in the following subsections; these descriptions are followed by a subsection describing systems implementation.

The general arrangement of the flight control surfaces and wing flap surfaces is shown in Figure 28. The location of major mechanical control components is shown in Figure 29.

6.2.2.1 Primary Flight Controls

Primary flight controls provide the basic capability for maneuvering, stabilizing, and trimming the airplane in three axes. Control in the lateral (roll) axis is provided by positioning the inboard ailerons, the outboard ailerons, and spoilers. Control in the longitudinal (pitch) axis is provided by positioning the elevators. Trim in the longitudinal (pitch) axis is provided by positioning the horizontal stabilizer. The upper and lower rudders are positioned to provide directional (yaw) control.

Longitudinal Controls—The longitudinal controls consist of the elevator controls and the horizontal stabilizer trim. The elevator controls, which are illustrated in Figure 30, position four sections of the elevator control surface, two on each semispan of the horizontal stabilizer. Each elevator section is moved by two hydraulic

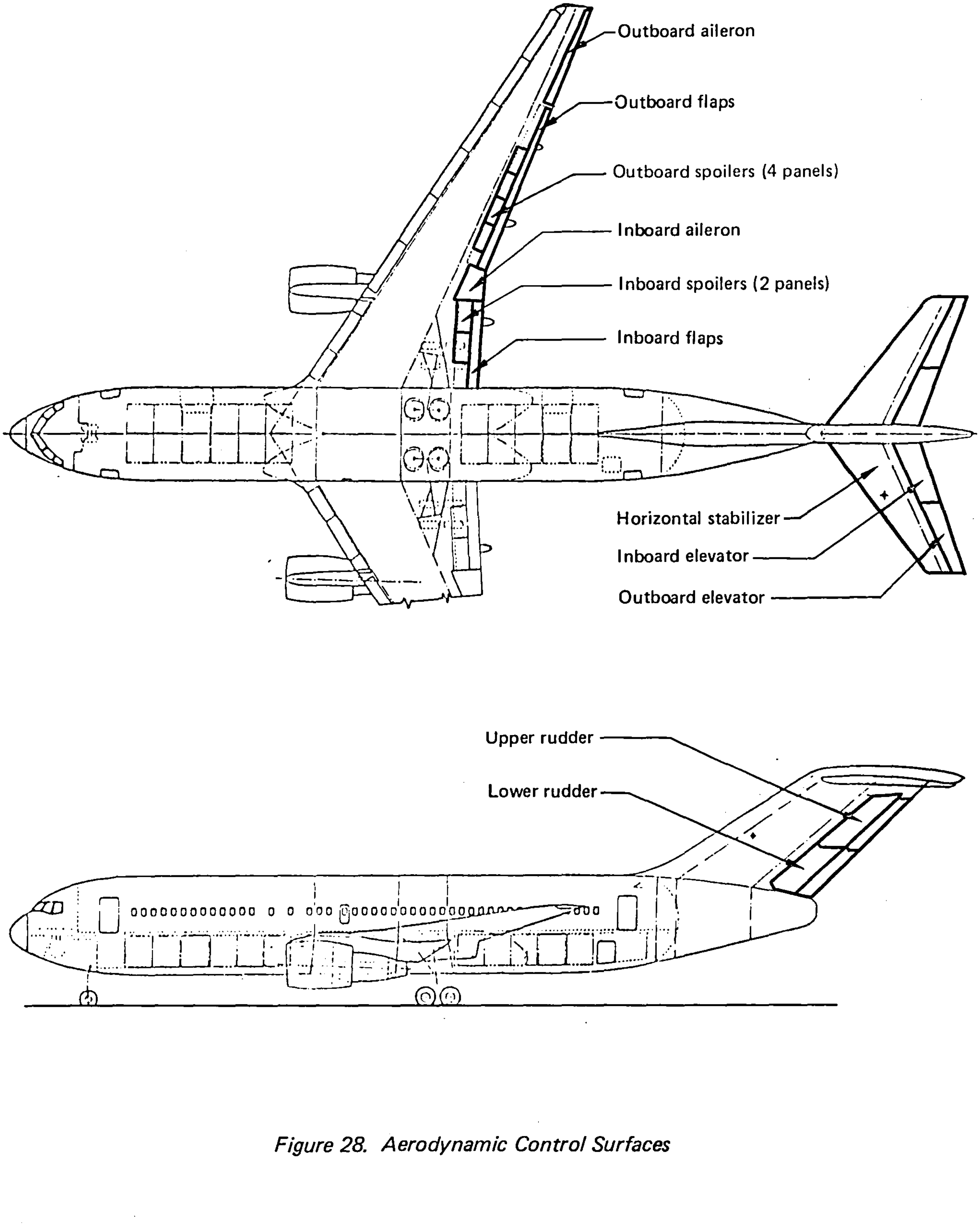


Figure 28. Aerodynamic Control Surfaces

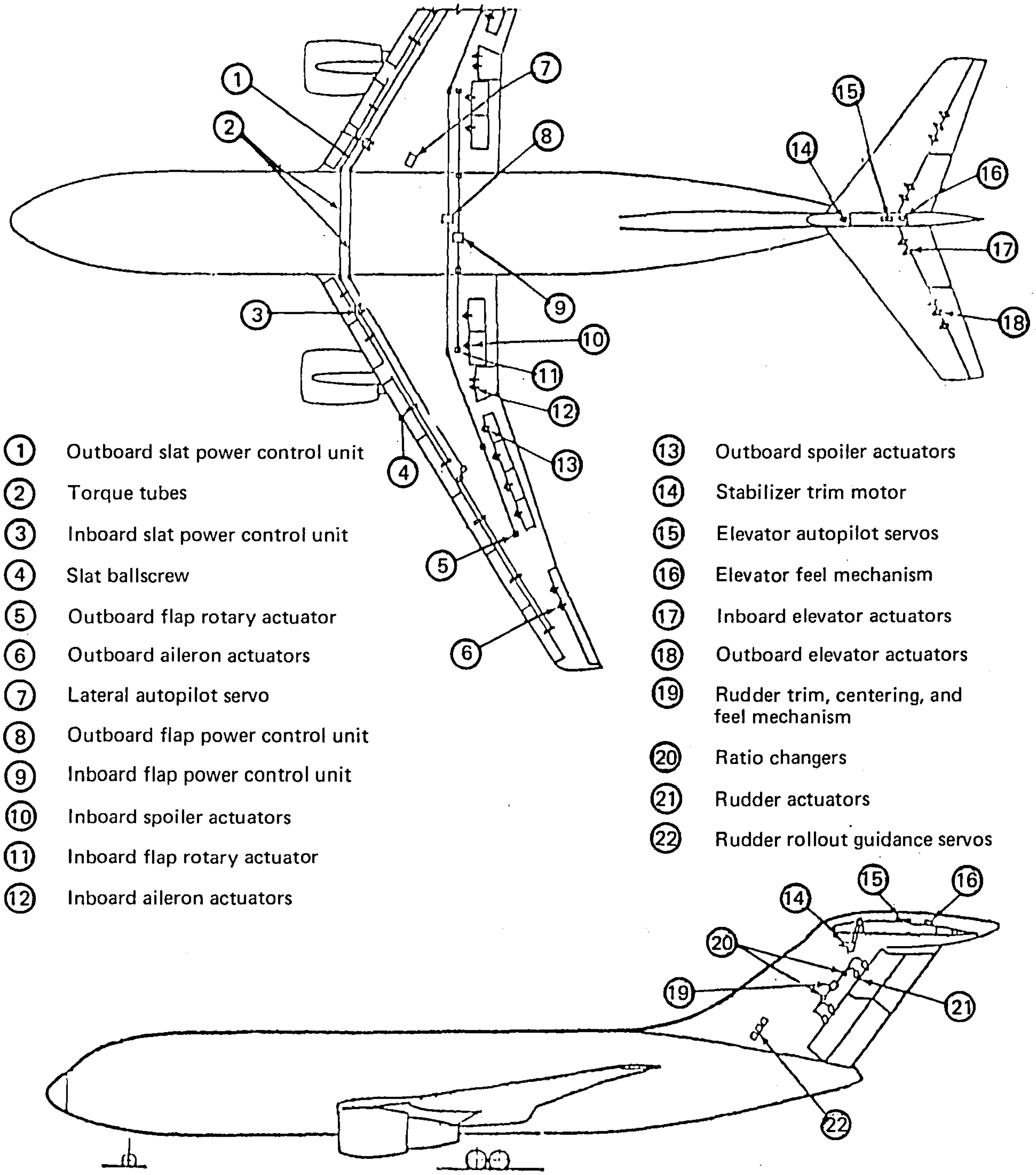


Figure 29. Flight Control System Components

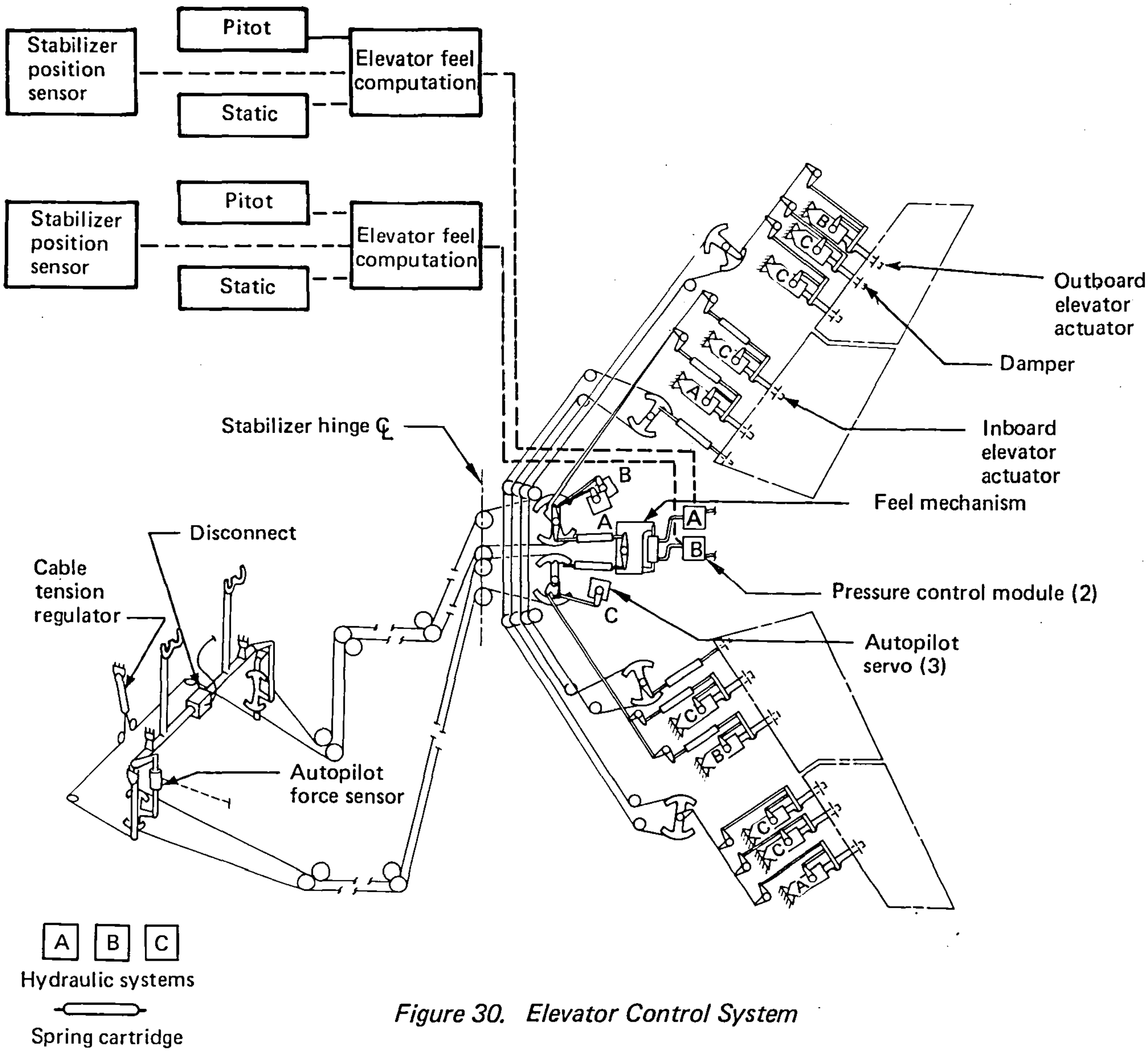


Figure 30. Elevator Control System

actuators. Actuators for the four elevator sections are powered by three different hydraulic systems. To provide flutter stability with dual hydraulic failure, a flutter damper is also connected to each of the two outboard sections. Elevator balance weights are not used. Each set of outboard elevator section actuators is controlled by cables driven by the opposite inboard elevator section movement. The two inboard elevator control cable systems are connected to a common artificial feel mechanism. The feel force is modulated by variable hydraulic pressure controlled by dual pressure modules. The two pilots' control columns are interconnected by a torque shaft incorporating a jam override spring and a manual disconnect mechanism.

The horizontal stabilizer trim control is illustrated in Figure 31. The stabilizer trim actuator consists of a ballscrew and nut, upper and lower gimbals, primary no-back brake, reduction gearing, and two hydraulic motors that drive the reduction gearing via a differential gear assembly. Each motor is also connected to a hydraulic pressure-released brake, controlled by a separate control module, and powered by a separate hydraulic system. Each control module contains two valves: one to arm the module by turning on hydraulic power, and one to control flow to one motor. The arm and control valves are commanded either electrically or mechanically. The electric commands are controlled by dual trim interface units. Manual electric trim commands originate at two dual switches, one on each pilot's control wheel. The trim interface units prohibit electric trim commands in opposition to elevator control inputs. Manual mechanical trim commands are transmitted to the control modules by dual control cables connected to dual levers on the pilots' aisle stand. All electric commands may be overridden by mechanical commands. Each control module contains a stabilizer rate-limiting valve which reduces hydraulic fluid flow as a function of airspeed. Dual electric stabilizer position indicators are mounted on the aisle stand and connected to dual stabilizer position sensors.

Lateral Controls—The lateral controls position the inboard aileron, the outboard aileron, and the six spoiler panels on each wing, as shown in Figure 32. The manual input for the lateral controls is provided by two control wheels. The ailerons' positions are controlled through dual cable systems which are bussed together fore and aft with force-limiting type disconnects at each bus. The control cables are connected to a dual feel and centering mechanism at each forward quadrant. The spoiler/speedbrake positions are electrically commanded and hydraulically actuated. The distribution of the three hydraulic systems to the lateral control and flight/ground speedbrake actuators is shown in Figure 32. The spoiler position authority for lateral control and flight/ground speedbrakes is also shown. Flight spoilers may be used as speedbrakes while continuing to provide lateral control. The spoilers are controlled by a four-channel set of electronics. The cable system provides input to the two hydraulic servo actuators at each inboard aileron and also back-drives the control wheels. Position sensors on the control wheels electrically command the spoiler control system. The outboard aileron hydraulic servo actuators receive cable-driven input commands from

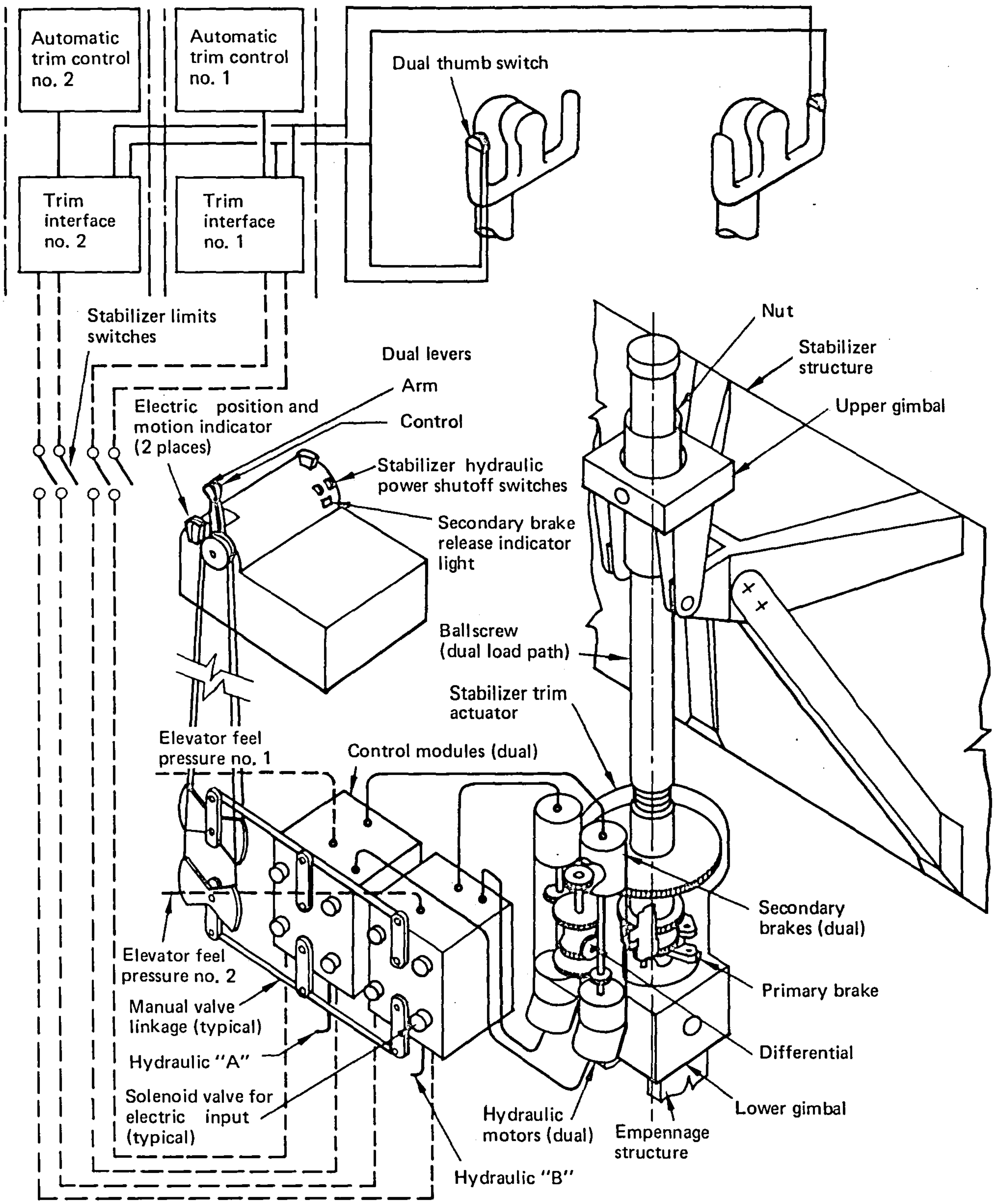


Figure 31. Stabilizer Trim System

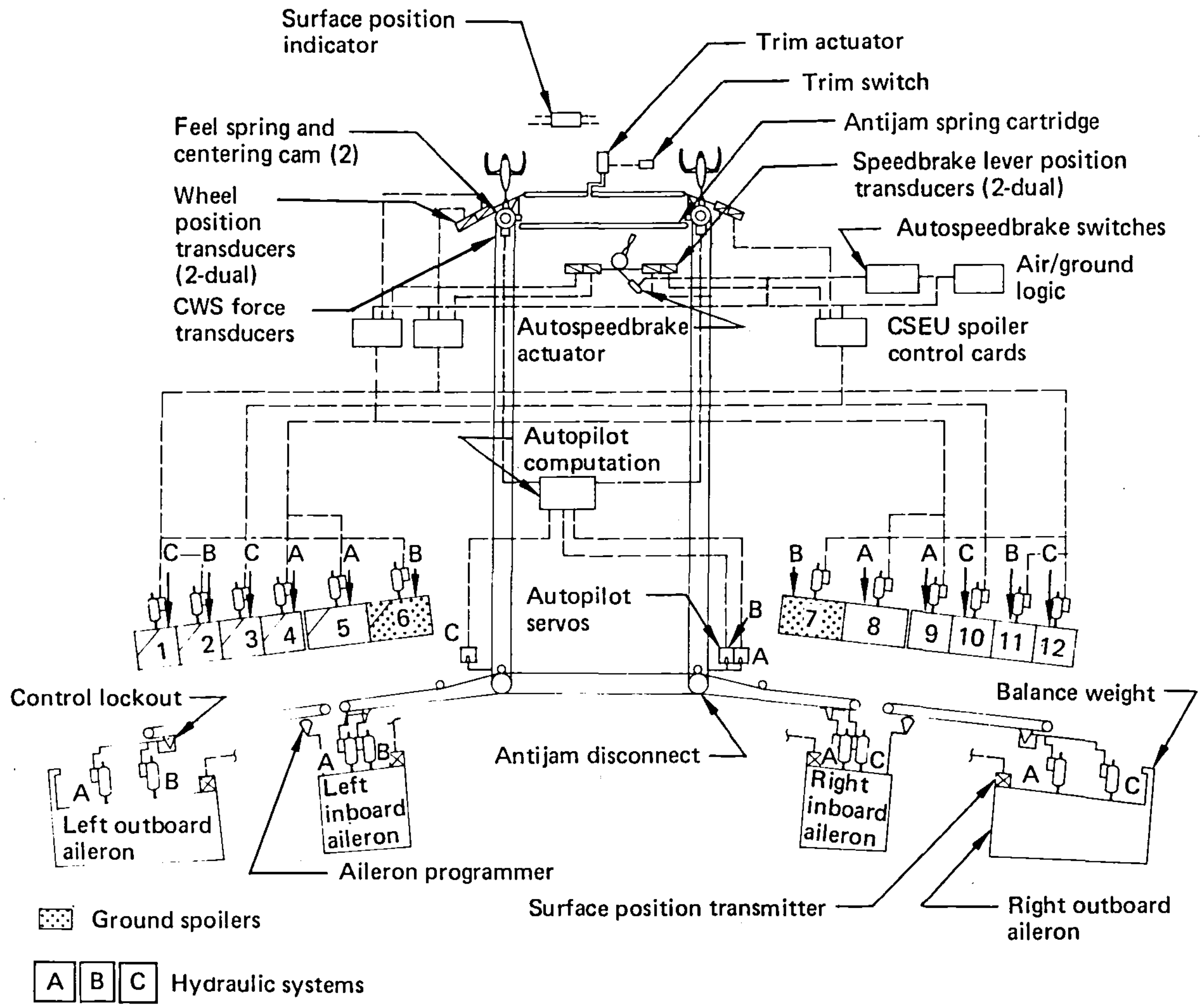


Figure 32. Lateral Control System

the inboard aileron motion. The outboard aileron control input is locked in neutral at high airspeed and unlocked at low airspeed; balance weights are utilized on the outboard ailerons only. Lateral trim is accomplished using electric switches on the aisle stand that control an electronically operated actuator, which connects to the trim, feel, and centering mechanisms at the forward quadrant bus and changes the neutral position of the lateral control system.

Directional Controls—The directional controls, shown in Figure 33, position the two, double-hinged rudder surfaces. Each of the two rudders is moved by two hydraulic actuators that are powered by three hydraulic systems. Neither rudder balance weights nor flutter dampers are used. The hydraulic actuators are controlled from the

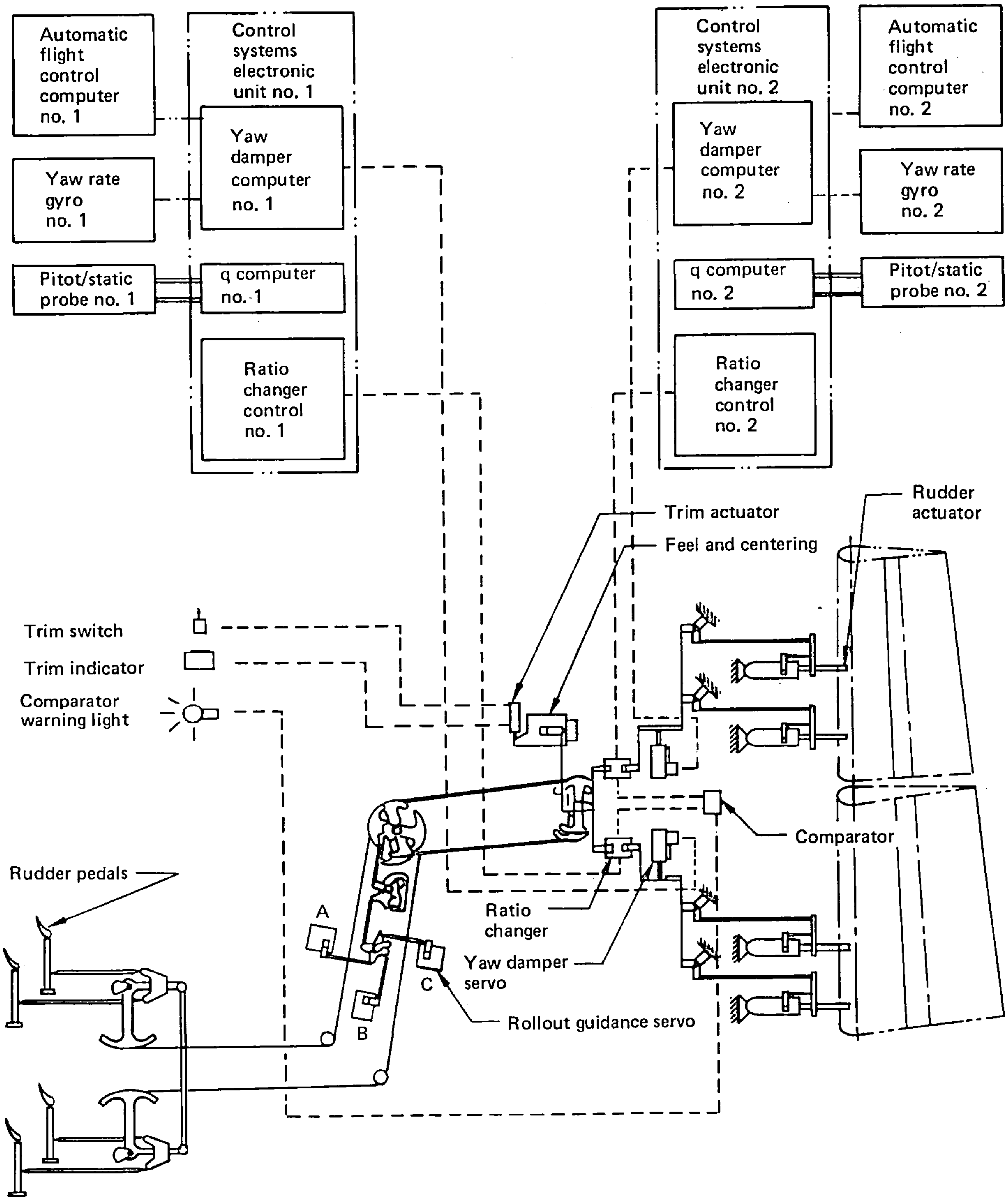


Figure 33. Directional Control System

pilots' rudder pedals by a common cable system and by separate pushrods and linkages in the vertical tail.

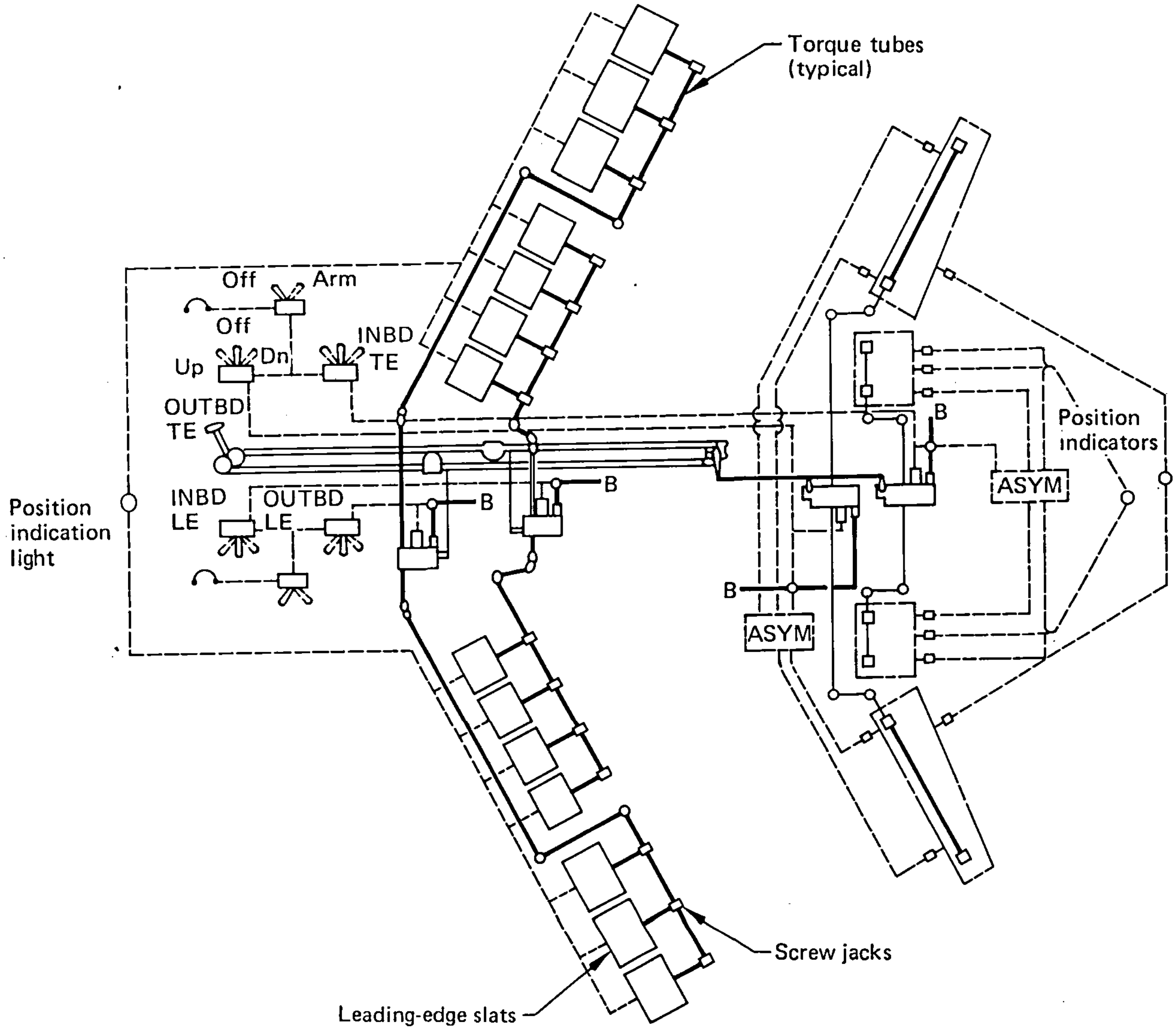
Separate rudder control ratio changers are installed in the upper and lower rudder control linkages. A fixed gradient artificial feel mechanism is connected in parallel to the control cable system common to the upper and lower rudders. The neutral position of the feel mechanism is varied by an electric trim actuator. A trim indicator is on the pilot's aisle stand and controlled by a trim actuator position sensor.

Electronics—The primary flight control electronics provide computation and control of the following functions: yaw damper, stabilizer trim interface, automatic stabilizer trim, spoiler controls, rudder ratio changer, and the outboard aileron lockout. The computational electronics are dual-redundant with a complete functional set contained in each of two control system electronics units (CSEU).

6.2.2.2 Secondary Flight Controls

The secondary flight (high lift) controls provide for the positioning of the wing trailing-edge flaps and the wing leading-edge slats in a coordinated manner. The trailing-edge flaps and the leading-edge slats are controlled to three positions (cruise, takeoff, landing). The wing spoiler panels are controlled in a symmetrical fashion as a secondary flight control function for in-flight drag modulation and landing deceleration.

Wing Trailing-Edge Flap Controls—The wing trailing-edge flap surfaces (see fig. 34) are driven by two centrally located power drive units, one driving the inboard trailing-edge flaps and the other driving the outboard trailing-edge flaps. Each power drive unit is powered by a hydraulic motor with an electric motor alternate. The power drive units receive inputs from the flap control handle by a dual cable system for normal operation. They receive electrical inputs for alternate power operation. The drive units are connected to torque tube systems in each wing to drive mechanical rotary actuators on each inboard and outboard flap segment. A torque limiting brake is provided in each rotary actuator. Each rotary actuator also incorporates a no-back brake. Asymmetry detection systems shut off the respective power drive units in the event the inboard or outboard flaps become asymmetric between left and right wings



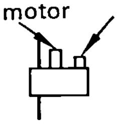
Legend:

Systems:
—— Mechanical
- - - Electric
━━ Hydraulic

⊗ Bypass valve

Power drive unit:

Electric motor / Hydraulic motor



ASYM = asymmetry detection electronics
INBD = inboard
OUTBD = outboard
LE = leading edge
TE = trailing edge
DN = down

Figure 34. Leading- and Trailing-Edge High-Lift System

or between actuators on a flap segment. Flap position indication is provided to the flight deck indicator from each flap panel.

Wing Leading-Edge Slat Controls—The wing leading-edge slat controls shown in Figure 34 use two power drive units located in the inboard wing leading edges; one to drive the inboard leading-edge slats and the other to drive the outboard leading-edge slats.

The controls are programmed to partially extend the leading-edge slats before trailing-edge flap extension to fully retract the slats after trailing-edge flap retraction. Each power drive unit is powered by a hydraulic motor with an electric motor alternate. The drive units are normally controlled by a dual cable system from the flight deck control lever. In the event of loss of hydraulic power, the electric alternate system can be controlled by flight deck switches. The drive units are connected to torque tube systems which run along the front spar of each wing and drive the screw jacks mounted on each slat segment. Each screw jack on each slat segment incorporates a no-back brake and a torque-limiting brake. Indication is provided to the flight deck when the leading-edge slats are in transit or fully extended. Failure of any slat surface to fully extend or fully retract is indicated by the "in transit" lights.

6.2.2.3 Automatic Flight Controls

The automatic flight controls (AFC) consist of the electronics to compute the following functions: autopilot, automatic stabilizer trim, flight director, autothrottle, automatic interlocks, failure warning, system status annunciation, and maintenance functions. The AFC interfaces with airplane and engine sensors, pilot command transducers, the autopilot and yaw damper servos driving the mechanical primary flight controls, and the electric servo driving the thrust levers. Preflight, inflight system status test, and post-flight maintenance test capability are included. Primary electric power is 115V ac, 400 Hz, and the external interlock and system engage power source is 28V dc. The digital intersystem data transmission format will be based upon ARINC 429 (ref 5) for all equipment. Table 5 and Figure 35 show subsystem partitioning and a summary definition of the individual components for the Baseline Configuration.

Table 5. Major Components of Automatic Flight Controls

Description	Quantity
Autopilot controls	
Flight control computer	3
Integrated autopilot/flight director autothrottle/mode control panel	1
Maintenance control and display panel	1
Remote-mounted maintenance panel	1
Thrust management	
Thrust management computer	1
Rating limit select panel	1
Limit display and mode annunciation panel	1
Miscellaneous flight control electronics	
Autopilot interface unit	3
Flight mode annunciator	2
Barometric altitude rate unit	1
Sensor flag warning annunciator	1

The automatic flight controls are shown in Figures 36 and 37 and provide the following major functions which are described below:

- autopilot
- flight director
- turn coordination
- thrust management

Autopilot—The autopilot function is computed in the flight control computer (FCC). Three FCCs are provided for those autopilot modes which require redundancy. The following autopilot modes are provided:

- Control Wheel Steering Mode—Control wheel steering (CWS) is a selectable mode of operation for the autopilot. It provides (1) pitch and roll maneuver control for pilot inputs, and (2) pitch attitude hold and heading hold when there are no pilot inputs.

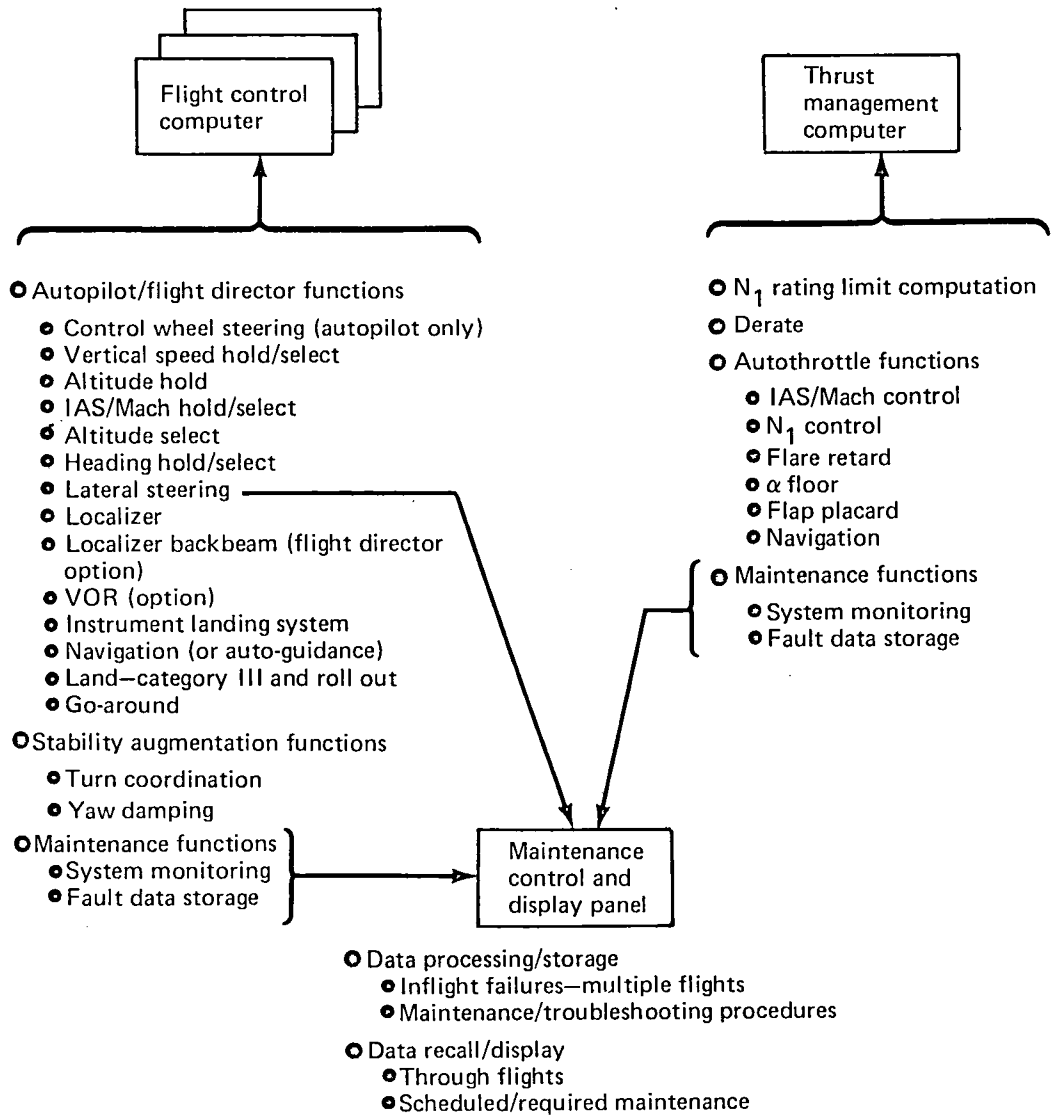


Figure 35. Automatic Flight Controls (Functional Partitioning)

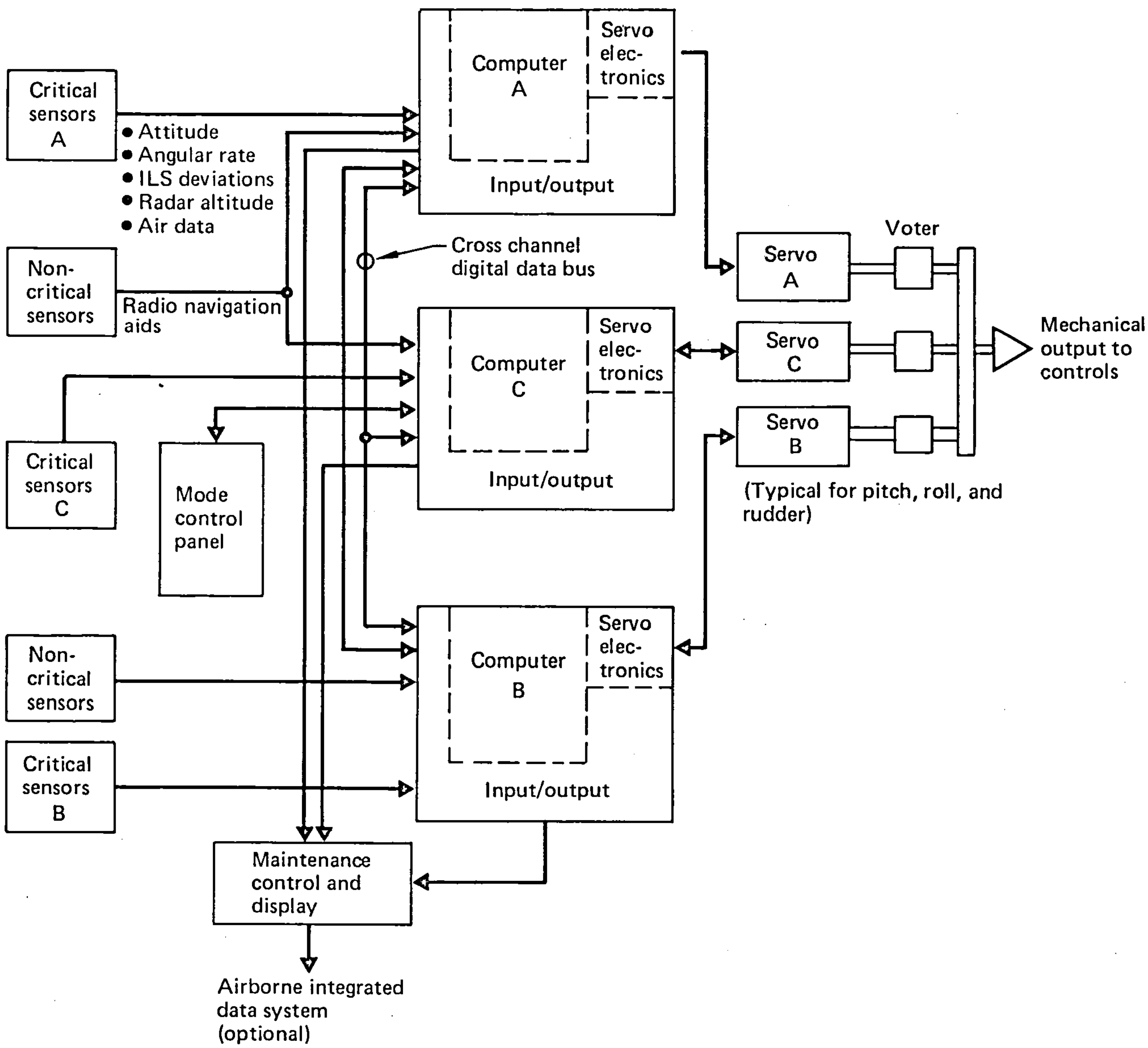


Figure 36. Automatic Flight Controls (Autopilot/Flight Director and Turn Coordination)

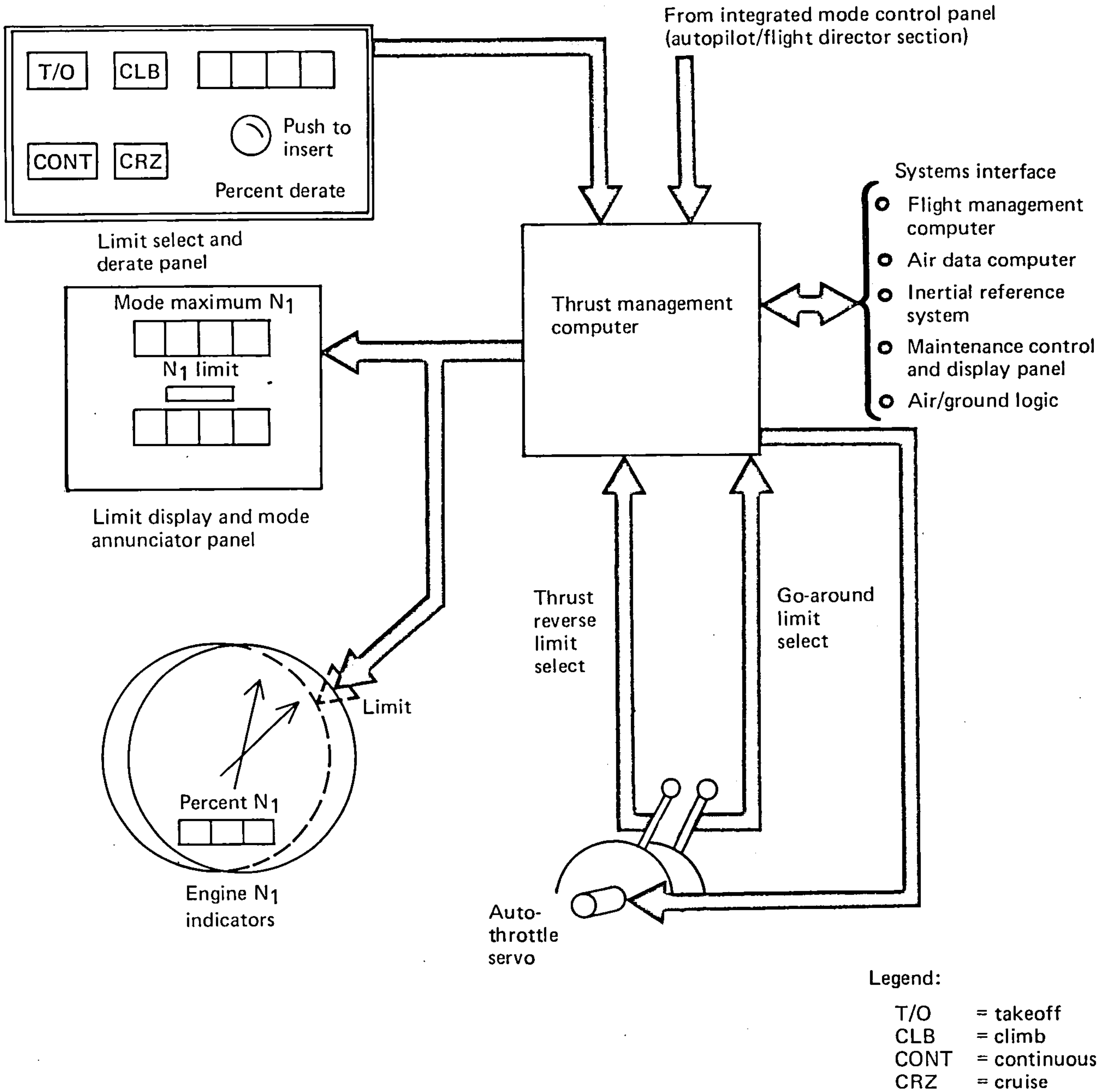


Figure 37. Automatic Flight Controls (Thrust Management)

- Altitude Select Mode—Altitude select is the basic pitch mode and is armed anytime autopilot is placed in the command mode and a glide slope has not been captured.

- Vertical Speed Select/Hold Mode—Vertical speed select/hold is the basic pitch transition mode between selected altitudes. It is engaged in a synchronized fashion anytime the autopilot is initially engaged into the command mode of operation or when the vertical speed mode is selected.

- Verticle Profile Mode—Vertical profile mode is a selectable altitude transition mode which uses combined autothrottle and elevator control. This mode provides airspeed/Mach control through the elevators during altitude transitions coupled with either selected rated (or derated) N_1 (low-pressure compressor speed) control during climb or idle thrust during descent. When maintaining altitude, the airspeed/Mach control is automatically transferred to the autothrottle control section. Two selectable submodes of the vertical profile mode are available. One is the automatic mode in which the target altitudes and speeds are automatically commanded by the flight management computer (FMC). The other mode is manual in which the target altitudes and speeds are manually entered through the mode control panel.

- Heading Select Mode—Heading select is the basic lateral mode and is engaged in a synchronized fashion anytime autopilot is initially engaged into the command mode of operation. Heading preselect capability is available whenever the mode is selected from any other engaged lateral command mode.

- Navigation Mode—Navigation is a selectable lateral mode which uses the FMC to provide lateral steering commands to the autopilot function in the FCC. The FMC commands can be based on VOR information or waypoint guidance.

- Localizer Mode—This selectable lateral mode allows the lateral autopilot function to be coupled to a localizer independently of the glide slope.

- Approach Mode—The approach mode is a selectable mode providing automatic glide slope and localizer control and manually initiated automatic go-around control. This mode can be operated multichannel through touchdown and automatic rollout.

- VOR Mode—The VOR mode is optional if the flight management computer is deleted. In this configuration, the navigation (NAV) mode of autopilot operation couples the VOR receiver outputs into the FCC to provide VOR control. This mode is not part of the basic certification of the airplane.

Automatic longitudinal flight control commands are by three separate parallel autopilot servo actuators. Automatic electric trim commands originate at elevator position sensors when the automatic flight controls are engaged.

Automatic lateral flight control inputs are accepted by three separate autopilot servo actuators; the autopilot servos are parallel-connected to the cable system.

Separate yaw damper servos are series-connected to the upper and lower rudder actuator control linkages. These servos are powered by separate hydraulic systems. For Category III-b autoland and rollout steering capability only, three rollout guidance servos are connected in parallel to the rudder control cable system common to the upper and lower rudders. Each rollout guidance servo is powered by a separate hydraulic system.

Flight Director—The flight director function is also computed in the FCC. The flight director function has the same modes as the autopilot function plus one additional mode, localizer back beam. The flight director command displays can be turned on and off independently of whether or not an autopilot function is selected. Provisions are included to allow the flight director command bars to be automatically biased out of view whenever the autopilot function is engaged in modes other than CWS.

The control mode select functions for autopilot, flight director, and autothrottle are integrated into a single panel. Select knobs are provided for indicated airspeed/Mach, altitude, vertical speed, heading, and course.

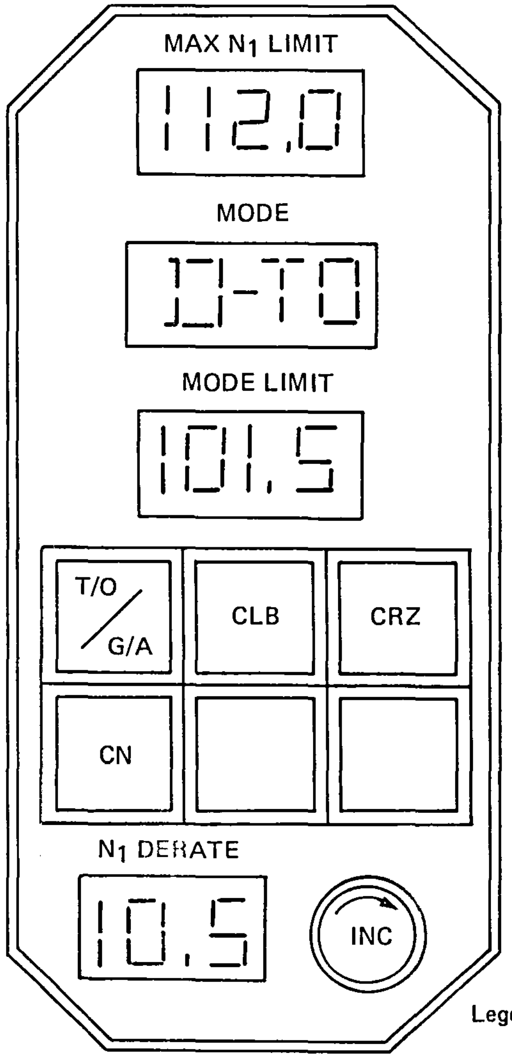
Turn Coordination—The low-speed (flaps down) turn coordination commands are computed in the FCC. These commands are input to the CSEU of the primary flight control system where they are summed with the yaw damper commands.

Thrust Management—The autothrottle function includes the thrust rating limit and thrust control computations. These computations are performed in the thrust management computer (TMC) located in the electronic equipment bay. Space provisions are made for a second (redundant) TMC. The TMC also provides autothrottle engage and mode interlock logic, mode status outputs, self-test, and monitoring functions.

Controls for selecting thrust rating limits and for commanding derated limits are provided on the thrust limit selection panel located near the engine instruments on the flight deck; the limits are displayed on the limit display and mode annunciation panel, shown in Figure 38. Selectable limits include takeoff and go-around, climb, max-continuous, and cruise. Actuation of the auto go-around switch(es) at the thrust levers automatically causes go-around thrust limit to be selected and displayed. Unlocking the thrust reverser automatically causes reverse limit to be selected and displayed.

The TMC interfaces with the engine N_1 instruments to position an indicator to show the value of the selected thrust limit or derated limit. The TMC controls the thrust levers, via the autothrottle servo motor, to provide full-range thrust control (within the thrust rating limits) during climb, cruise, approach, and landing. Autothrottle mode selection controls are provided on the integrated autothrottle, autopilot, and flight director mode control panel. The autothrottle modes include the following:

- airspeed/Mach select/hold
  - alpha floor
  - flap placard ceiling
  - N_1 limit
- vertical profile mode
  - manual submode
  - automatic submode (automatic switchover between low-pressure compression speed [N_1] control works in conjunction with the pitch vertical profile mode)



Legend:

T/O = takeoff
G/A = go-around
CLB = climb
CRZ = cruise
CN = maximum continuous
INC = increase

Figure 38. Thrust Limit Indicator

- control to full or derated N_1 limit
- flare retard

6.2.2.4 Electronics Systems Implementation

This subsection describes the validation of the primary flight control electronics (PFCE), which were functionally described in Subsection 6.2.2.1, and the automatic stabilizer trim which is part of the AFC system described in Subsection 6.2.2.3. The elements of the PFCE and AFC are mechanized in the CSEU. These elements are listed below, and a single-line schematic format is shown in Figure 39.

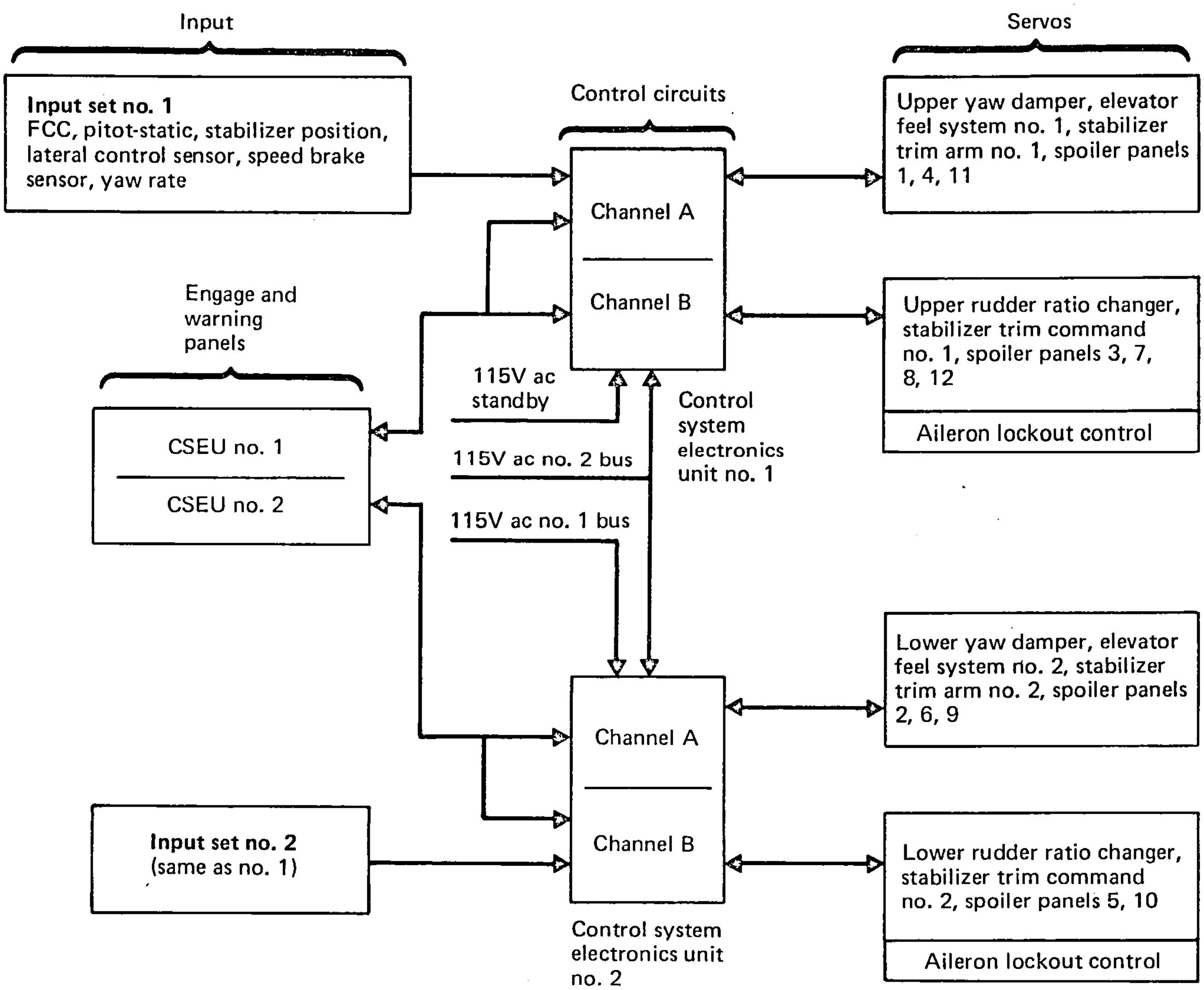


Figure 39. Primary Flight Control Electronics (PFCE)

- Spoiler Controls—The spoiler-controls computers consist of line-replaceable cards and provide both lateral control of the inflight spoilers (via sensors connected to the control wheel) and speedbrake control (via sensors connected to the speedbrake lever). Switching of the speedbrake authority is provided to facilitate ground speedbrakes. Control of the spoiler panel servos is maintained by feeding individual spoiler position commands through the servo electronics to control the electrohydraulic servos located at each spoiler panel. A spoiler control panel, located in the cockpit, allows any of the spoiler control channels to be disengaged.

- Stabilizer Trim Interface—The stabilizer trim interface consists of line-replaceable cards and provides the necessary switching to allow the autopilot function to interface with the automatic stabilizer trim function. In addition, the stab trim interface provides the logic to interrupt either the manual or automatic trim whenever the elevator is positioned to oppose the direction of the stabilizer travel.

- Outboard Aileron Lockout—Outboard aileron lockout is contained on a line replaceable card and controls aileron lockout as a function of dynamic pressure (q).

- Yaw Damper—A control panel, located in the cockpit, allows the yaw damper to be turned off and provides self-test capability. The dual yaw dampers consist of line-replaceable cards.

- Rudder Ratio Changer—The rudder ratio changer consists of line-replaceable cards and provides a computed rudder deflection versus rudder pedal input as a function of q.

- Automatic Stabilizer Trim—The automatic stabilizer trim computers consist of line-replaceable cards that provide automatic stabilizer trim whenever the autopilot is engaged. Trim System 1 provides trim capability when Autopilot A is engaged, and Trim System 2 provides trim when Autopilot B is engaged. Either trim system provides trim capability when Autopilot C is engaged with trim normally operating from Trim System 1.

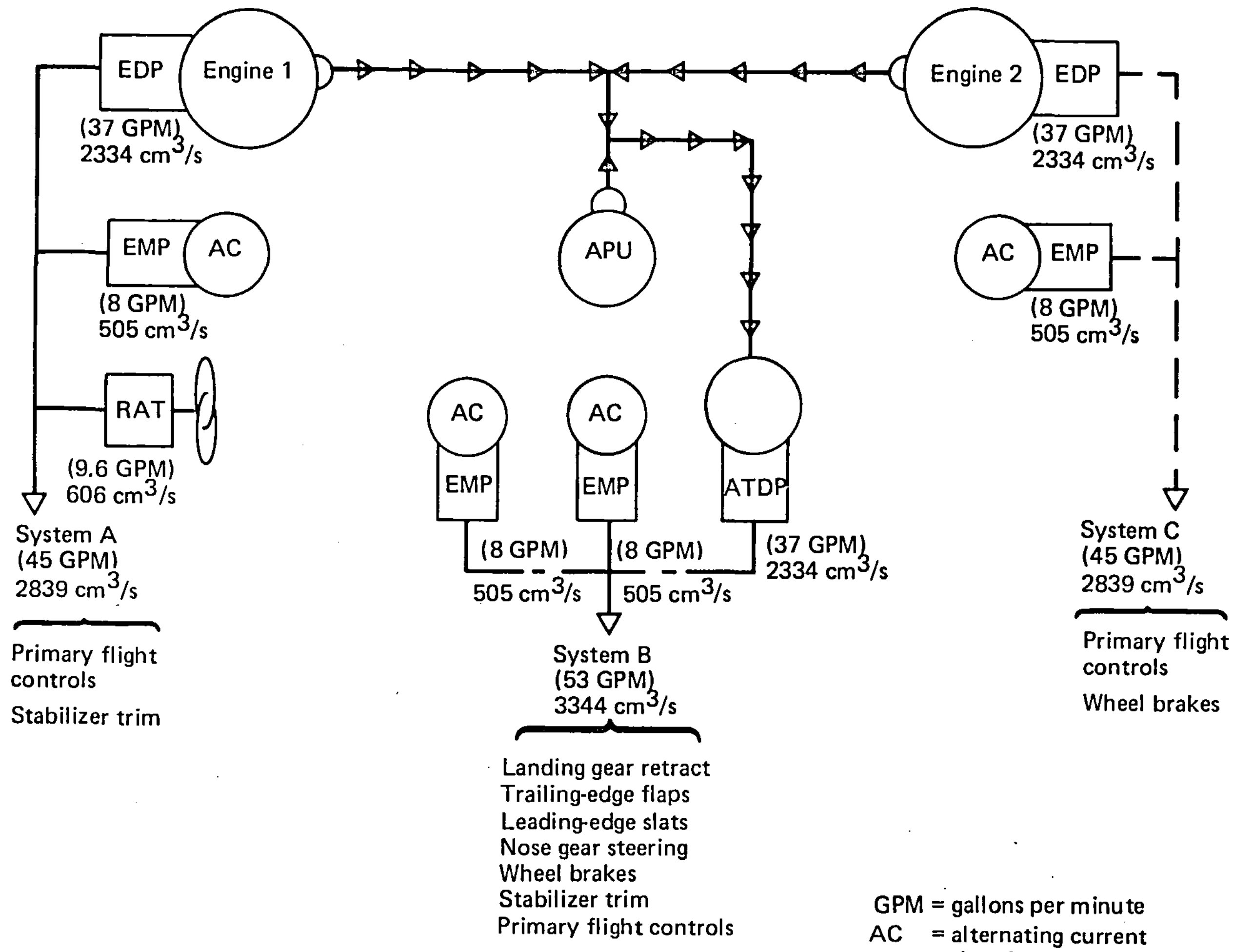
6.2.3 HYDRAULIC POWER SYSTEMS

This section describes:

- hydraulic and landing gear systems
- hydraulic power generation
- hydraulic plumbing and distribution
- landing gear control and actuation system
- landing gear brake control system
- nose wheel steering control and actuation system

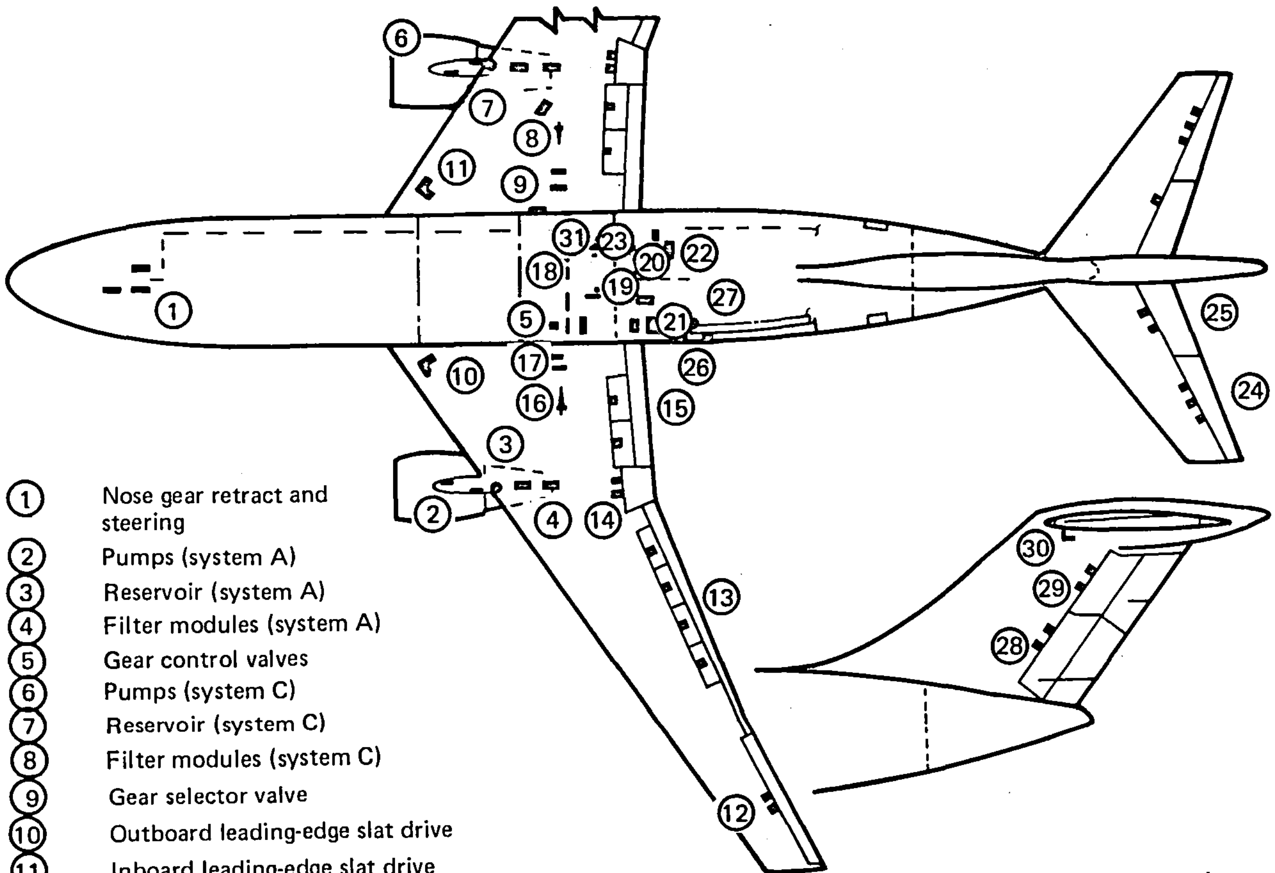
The general arrangement of the hydraulic power system is shown in Figure 40, and the location of major hydraulic and landing gear system components is shown in Figure 41. The hydraulic power generation system consists of three continuous duty 20 680-kPa (3000-psi) systems using Boeing material specification (BMS) 3-11 (phosphate ester) fluid and are identified as Systems A, B, and C as shown in Figures 42 and 43. Hydraulic Systems A and C are functionally similar. Hydraulic power is generated by an engine-driven pump (EDP) installed in parallel with an electric-motor-driven pump (EMP). System B hydraulic power is generated by two ac EMPs and one air-turbine driven pump (ATDP). The bleed air start manifold serves as the pneumatic source and emergency hydraulic power is derived from the wind-milling engines rotating the EDPs. System A is also augmented by a hydraulic pump powered by a ram air turbine (RAT). Ground hydraulic power is available from the ADP, powered by the APU or pneumatic ground cart; the ac pumps are also energized by ground cart, or APU, or an external hydraulic power supply source. Flight deck controls and displays consist of depressurization switches for the EDPs, shutoff switches for the ADPs and EMPs, low-pressure and low-fluid quantity warning lights, and selectable system pressure and fluid quantity readout.

Extension and retraction of the main and nose landing gears, along with the main gear wheel well doors, are accomplished by hydraulic power. The nose gear wheel well doors are slaved to the nose landing gear. The landing gear is actuated by a single control readily accessible to both pilots and connected by cable to a hydraulic selector valve. The retraction and extension operations for the main landing gear are properly phased by gear sequence valves, mechanically connected to monitor and control the



GPM = gallons per minute
AC = alternating current electric motor-driven pump

Figure 40. Hydraulic Power System



1. Nose gear retract and steering
2. Pumps (system A)
3. Reservoir (system A)
4. Filter modules (system A)
5. Gear control valves
6. Pumps (system C)
7. Reservoir (system C)
8. Filter modules (system C)
9. Gear selector valve
10. Outboard leading-edge slat drive
11. Inboard leading-edge slat drive
12. Outboard aileron
13. Outboard spoilers
14. Inboard aileron
15. Inboard spoilers
16. Main gear retract actuators
17. Wheel brake antiskid modules
18. Main gear door actuators
19. Outboard trailing-edge flap drive
20. Inboard trailing-edge flap drive
21. Wheel brake components
22. Wheel brake components
23. Ground servicing center
24. Outboard elevator
25. Inboard elevator
26. Pumps and filters (system B)
27. Reservoir (system B)
28. Lower rudder
29. Upper rudder
30. Stabilizer trim motors
31. Automatic braking module

Figure 41. Hydraulic Component Locations

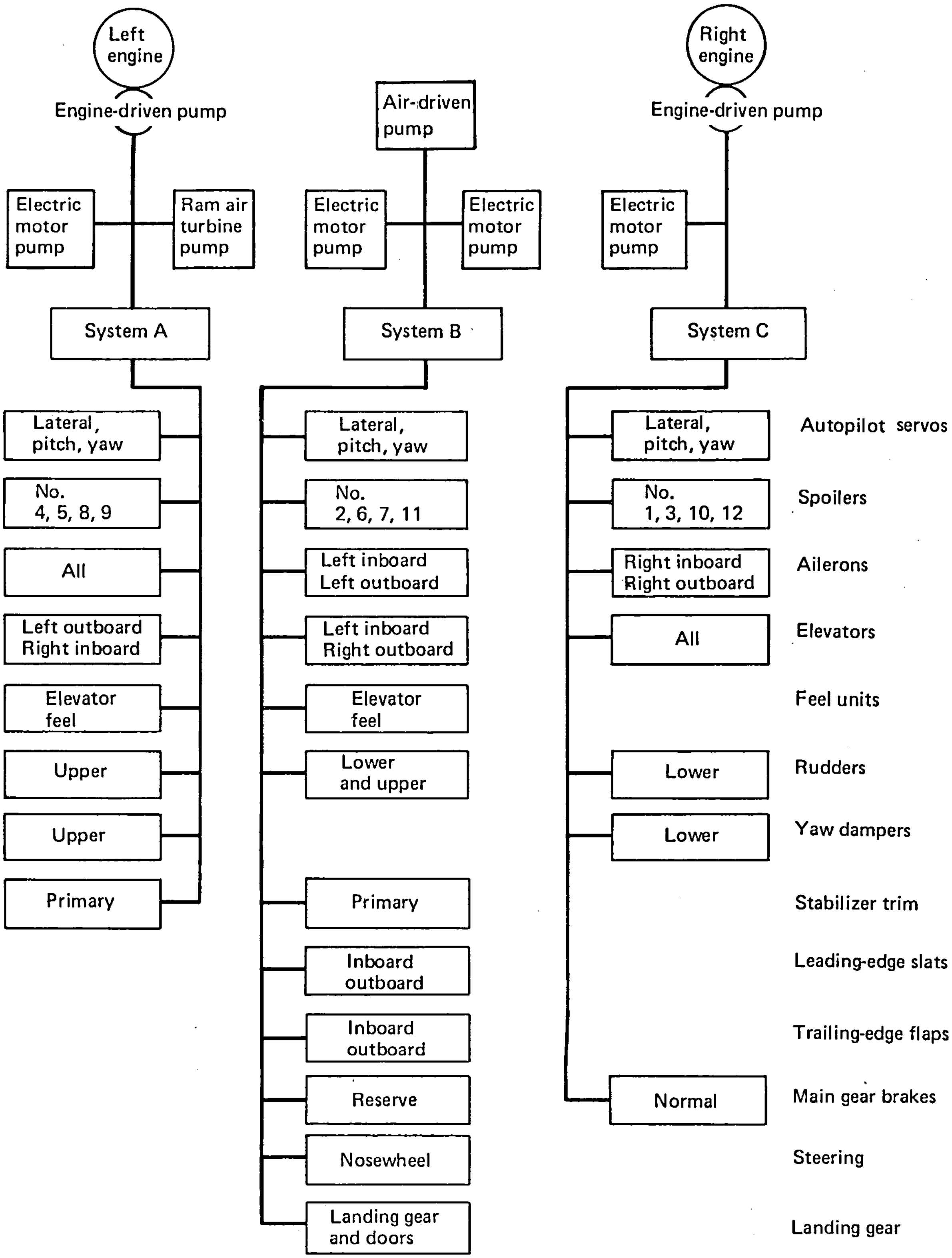


Figure 42. Hydraulic System Block Diagram

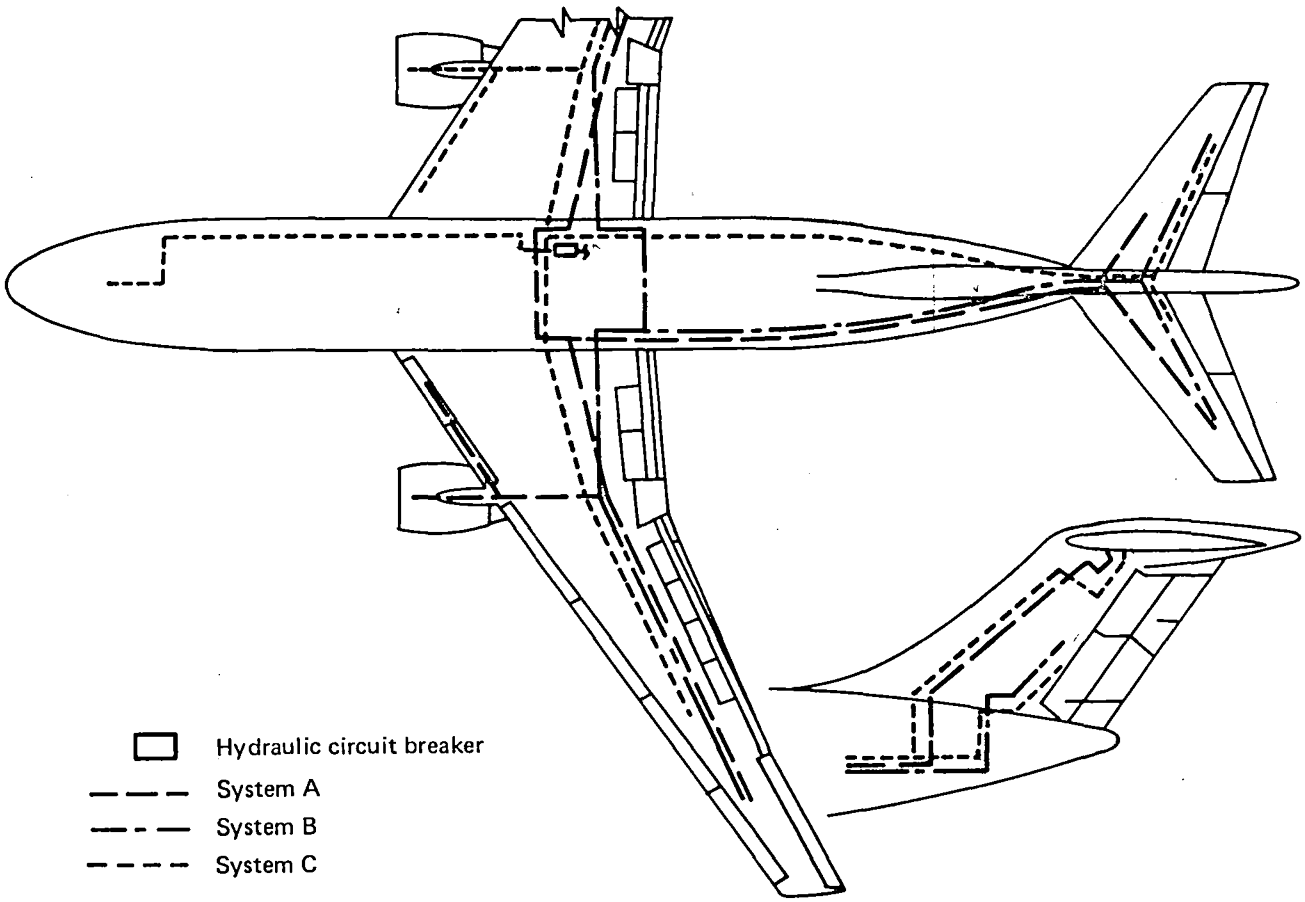


Figure 43. Hydraulic System Routing

door, landing gear, and uplock positions. A completely segregated alternate extension system is provided to unlock the landing gears and doors in the event that primary hydraulic power is unavailable. Two separate and independent landing gear position and warning systems are provided for the flight crew; indication of position for the landing gear, doors, and landing gear truck are provided. A landing gear warning horn provides aural indication if landing gear/flaps/throttles are in any unsafe relationship.

Two hydraulic systems, normal and reserve, energize the brakes. The normal brake system incorporates the automatic braking function and antiskid control of the brakes on an individual wheel basis. The reserve brake system incorporates the parking brake function and antiskid brake control on a paired wheel basis. The reserve brake system will be activated automatically upon depressurization of the normal brake hydraulic system. An automatic braking system operates in conjunction with the antiskid system to obtain automatic (feet-off) braking at selectable deceleration rates during landing. The parking brake system is effective for a minimum 8-hr period. The flight deck indicator provides brake pressure readout.

Nose wheel steering control is accomplished from the flight deck by separate steering controls accessible to each pilot. A steering tiller provides full authority mechanical cable steering control input to the hydraulic steering metering valve. Limited steering authority by each pilot through mechanical connection of rudder pedal displacement into the tiller cable system is provided. The nose wheels are capable of castering within steering limits in the event of loss of hydraulic pressure. Fluid bypass protection is provided to protect the steering system from high pressures developed in the steering cylinders during towing.

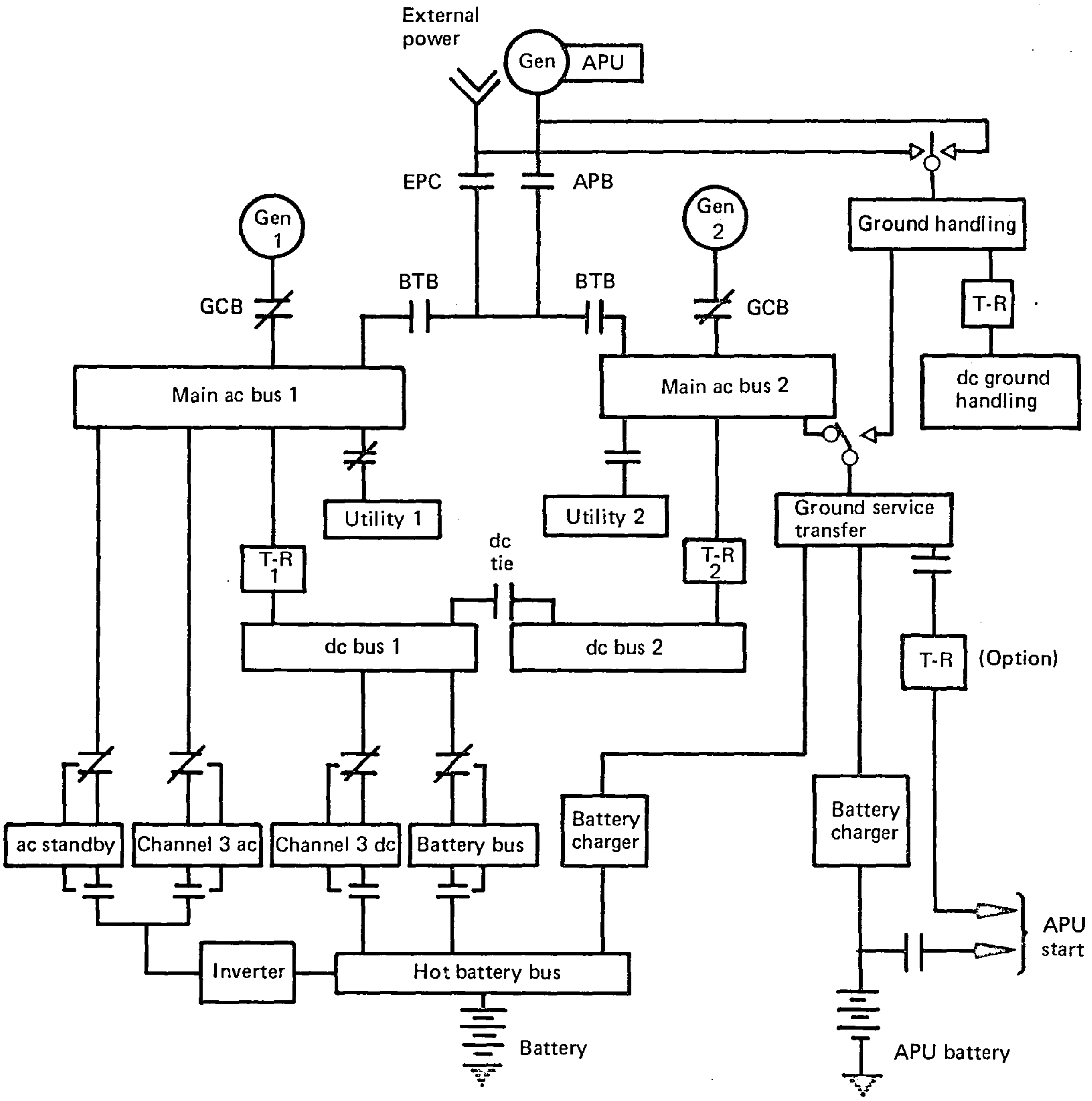
6.2.4 ELECTRIC POWER SYSTEM

The electric power supply and the electric and electronic equipment installation are described and illustrated in this subsection.

6.2.4.1 Power Supply

The electric power system arrangement is shown in Figure 44. Primary three-phase, 115-V, 400-Hz power is supplied by two engine-driven 90-kVA integrated drive generators. The system operates as two separate isolated channels, and paralleling of the generators is not possible. A third 90-kVA APU-driven generator is provided both for ground maintenance operations and as an inflight backup for the two main engine-driven generators. The APU is capable of being started at any altitude up to 7620m (25 000 ft) and can provide full electric power up to 10 670m (35 000 ft). The APU generator control unit is interchangeable with those used for the engine-driven generators; any single generator has sufficient capacity to supply all flight-essential loads. Two of the three generators must be operative for airplane dispatch with no load reduction, or for a Category III landing.

During ground operations, electric power can be provided from either the APU generator or from a ground power cart through the 90-kVA external power receptacle.



Contactors shown in normal flight condition

=	Contactors open
≠	Contactors closed
APB	Auxiliary power breaker
Gen	Generator

Figure 44. Electrical Power System

Ground power can be used to energize all main power buses or only those electric loads required for normal maintenance, servicing, and cargo handling. On the ground or in flight, overload will cause the utility and galley loads to be automatically shed.

Airplane 28V dc power is provided by two 120-amp unregulated TRUs. Each of the two main ac buses supplies its own TRU. The dc system operates isolated only. In the event of a TRU failure, a dc bus tie contactor enables the remaining TRU to supply both main dc buses. During ground operation, a 20-amp TRU provides dc power for ground handling loads.

A 40 ampere-hour (Ah) nickel-cadmium battery and a 1000-VA static inverter are provided to supply backup power to flight critical loads. Typical standby loads are listed below:

- hot battery bus
  - battery protection and control
  - fire extinguishers
  - fuel shutoff control

- battery bus and standby ac
  - generator controls
  - fire detection
  - fuel manifold control
  - hydraulic shutoff control
  - engine start and ignition
  - crew oxygen control
  - emergency light control
  - interphone system
  - VOR/ILS
  - fuel quantity
  - fuel device control
  - engine exhaust gas temperature indication
  - engine anti-ice control
  - N_1 tach indication
  - flight control system electronics unit
  - standby attitude indicator
  - passenger address
  - dc fuel pump

A battery charger provides for controlled recharge of the battery and operates as a TRU to supply the standby loads if the main dc source is lost but ac power is still available. Standby bus transfer is automatic. A TRU is provided for normal starting of the APU when ac power is available from either the main generators or external power. When ac power is not available, a dedicated APU battery is used for APU starting. A dedicated APU battery charger, operating from either the main buses or external power, is also provided.

The third power source for the Category III autoland system is provided by the standby system. During a Category III landing, the third channel autoland dc loads will be supplied by the standby battery and the battery charger (in the transformer-rectifier [T-R] mode). Autoland ac loads will be supplied from the standby inverter. Figure 45 shows the electric power load profile with respect to generator capacity.

6.2.4.2 Installation

The electric and electronic equipment installation locations are shown in Figure 46. The main equipment bay, located between the nose gear wheel well and the forward cargo compartment, includes the area on the left side of the nose gear wheel well (fig. 47). There is a smaller equipment area shown in Figure 48 located in the aft cargo compartment and an equipment area located in the flight deck. Figure 49 also shows the location for the No. 1 weather radar transceiver located aft of the radome, the low-range radio altimeter (LRRA) transceivers installed in the forward cargo compartment (fig. 50), and the high frequency (HF) antenna couplers located in the body beneath the vertical fin. The antenna locations are shown in Figure 51, and the external light locations are shown in Figure 52.

The electric and electronic equipment is installed in two racks which are designed to accommodate equipment with form factors per ARINC 404A (ref 6) short or ARINC 600 (ref 7). Each of the two racks has six shelves. Ten of the 12 shelves are designed to accommodate 32 modular concept units (MCU) of equipment each, and two shelves are designed to accommodate 27 MCUs. MCU is a unit of measure according to ARINC 600 which defines the box width. The three inertial reference units are installed on a common platform which replaces one of the shelves in the rack.

Four power shields are installed in the main equipment bay: one shield for each generator, one shield that includes equipment for APU and external power, and one

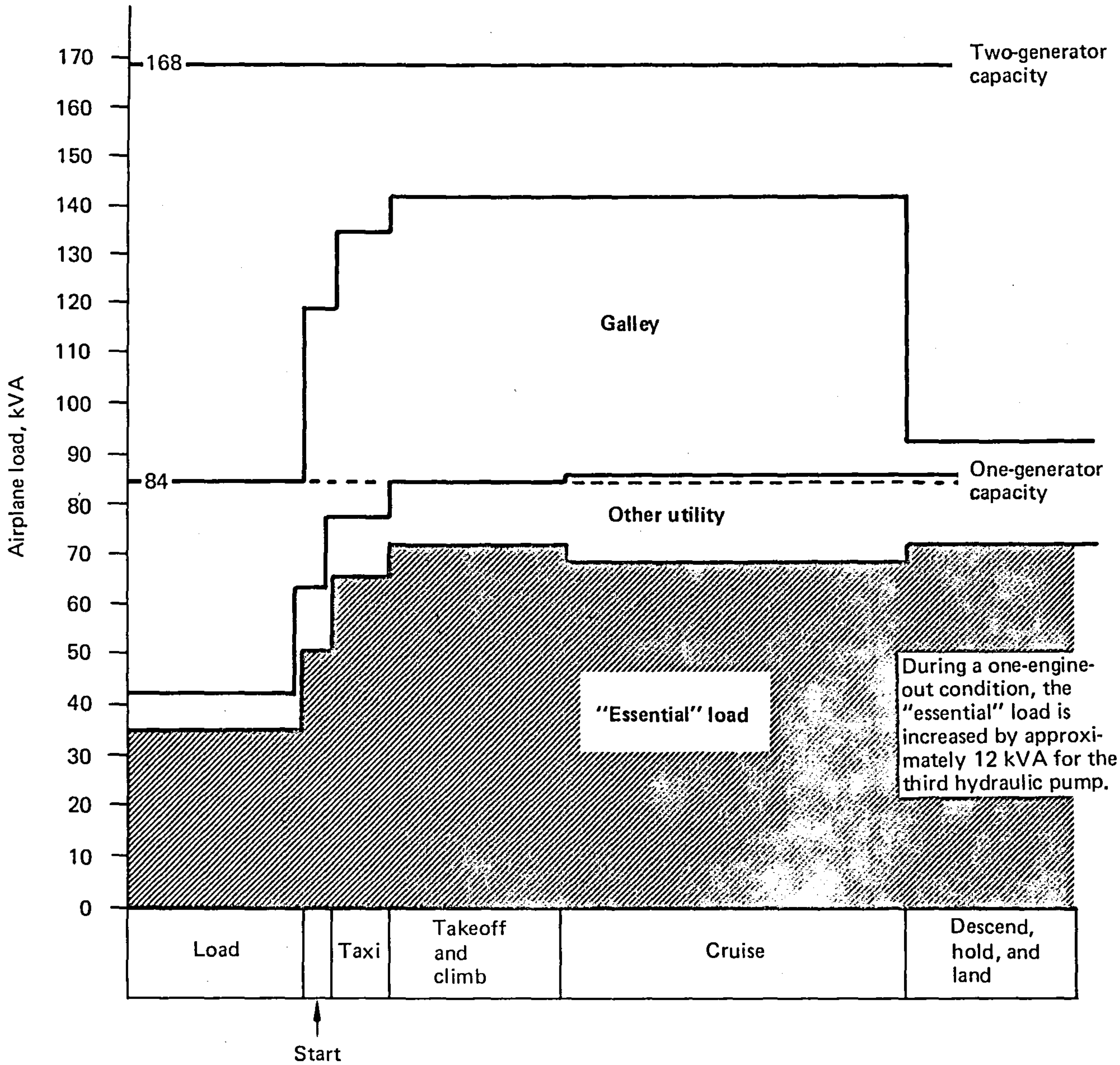


Figure 45. Electric Power Load Profile

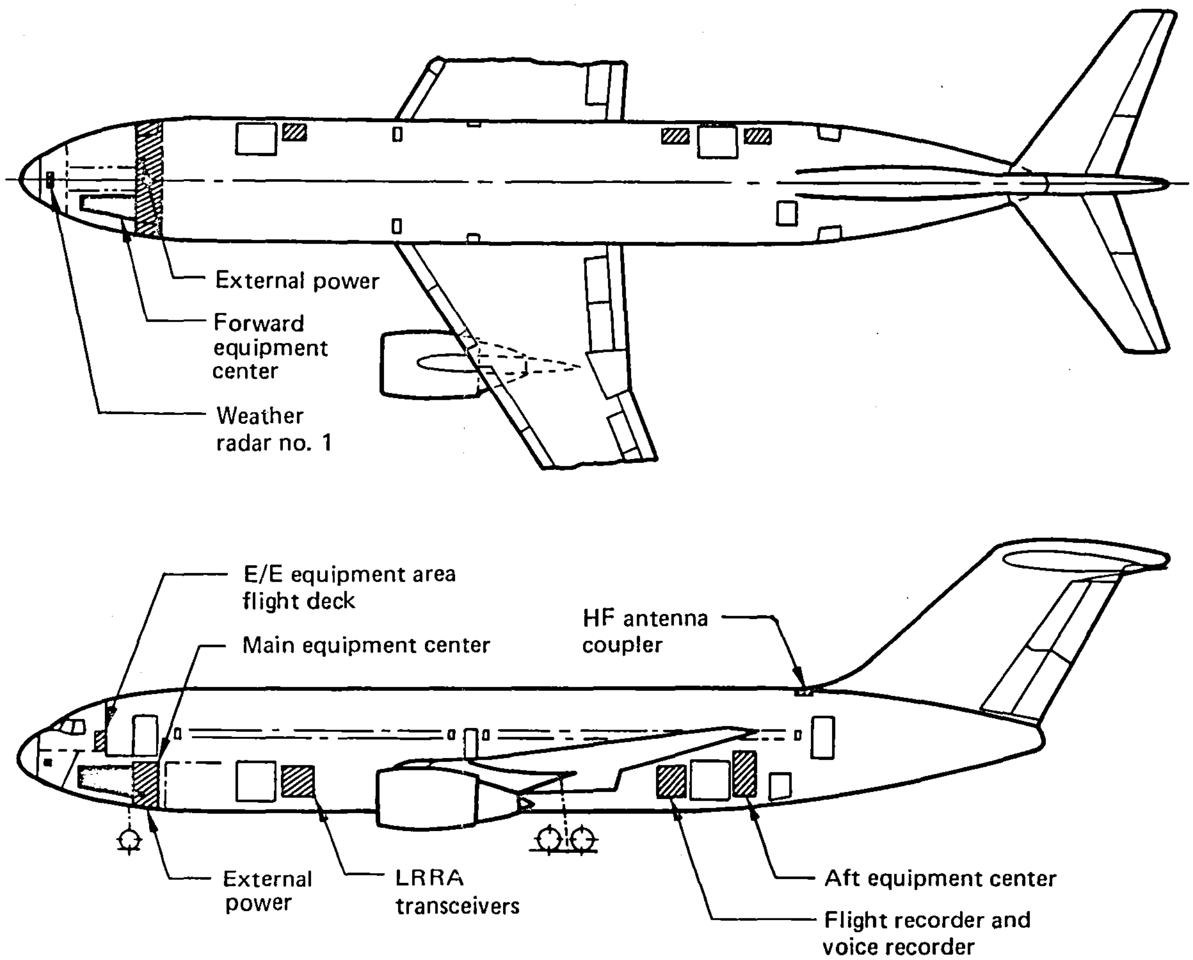


Figure 46. Electric/Electronic (E/E) Equipment Areas

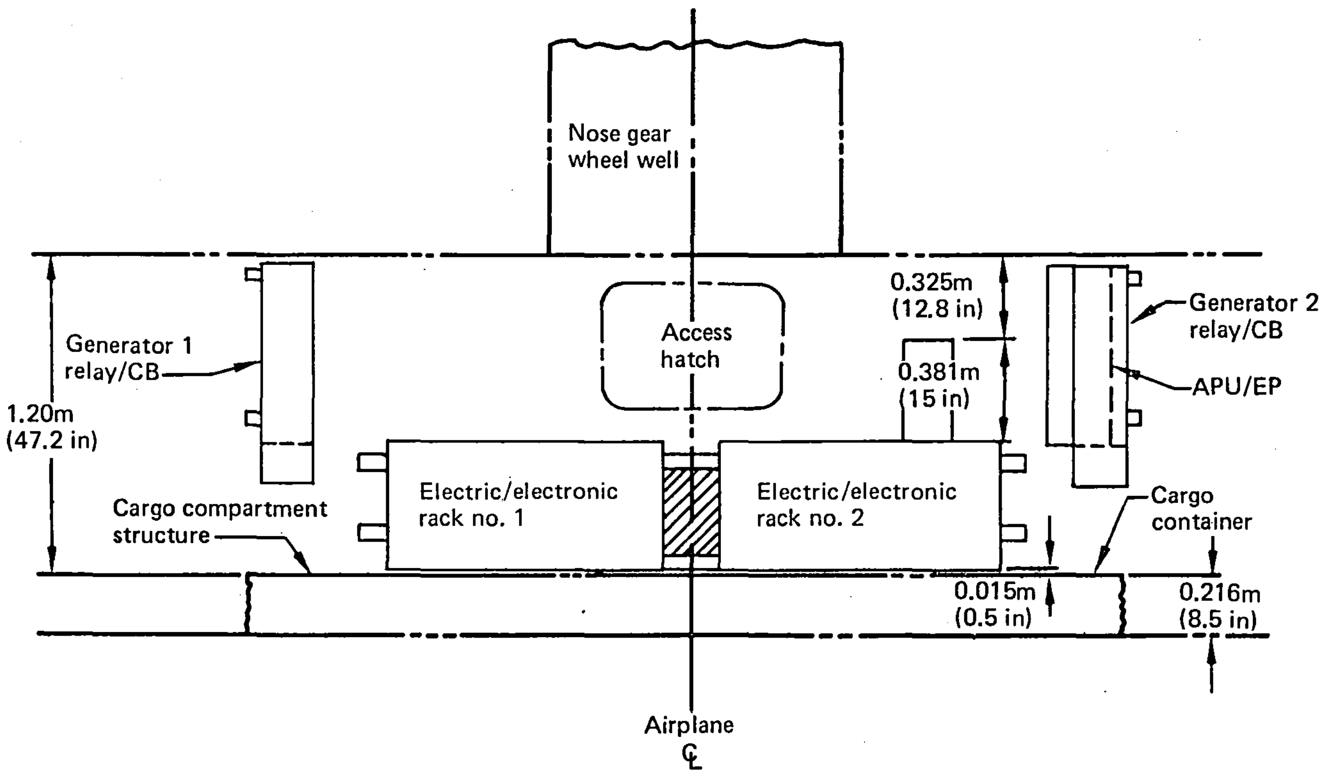


Figure 47. Electric/Electronic Main Equipment Bay

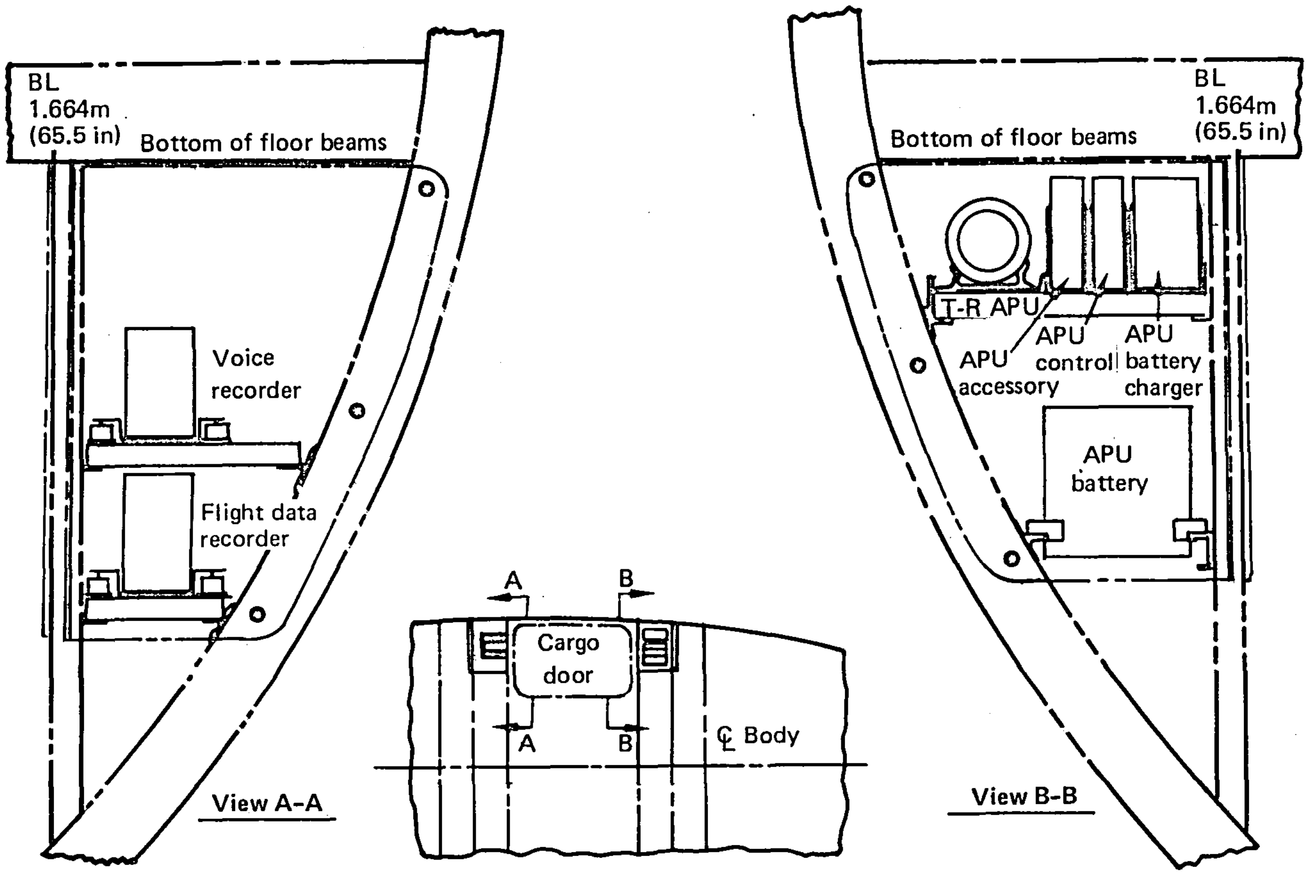


Figure 48. Aft Equipment Center

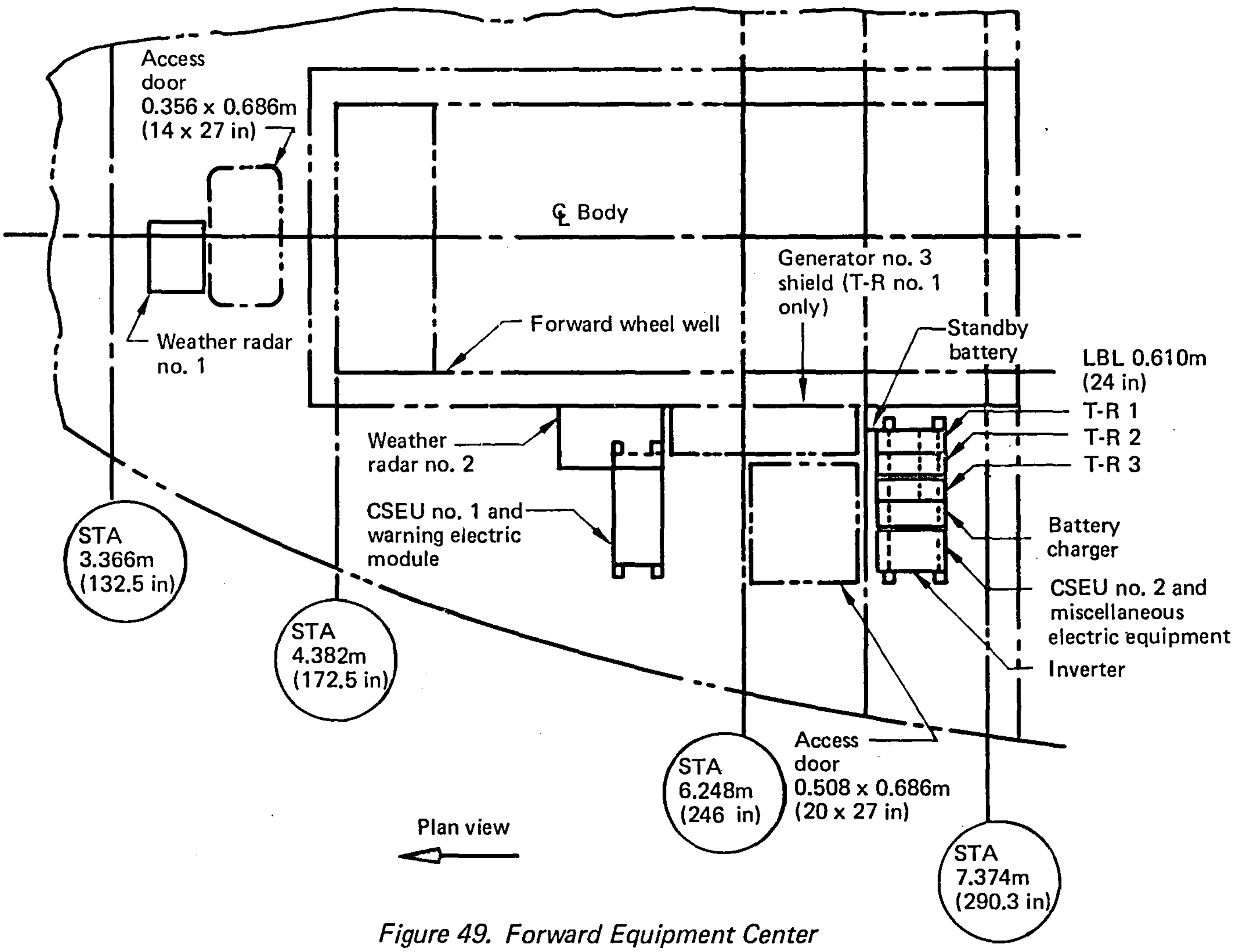


Figure 49. Forward Equipment Center

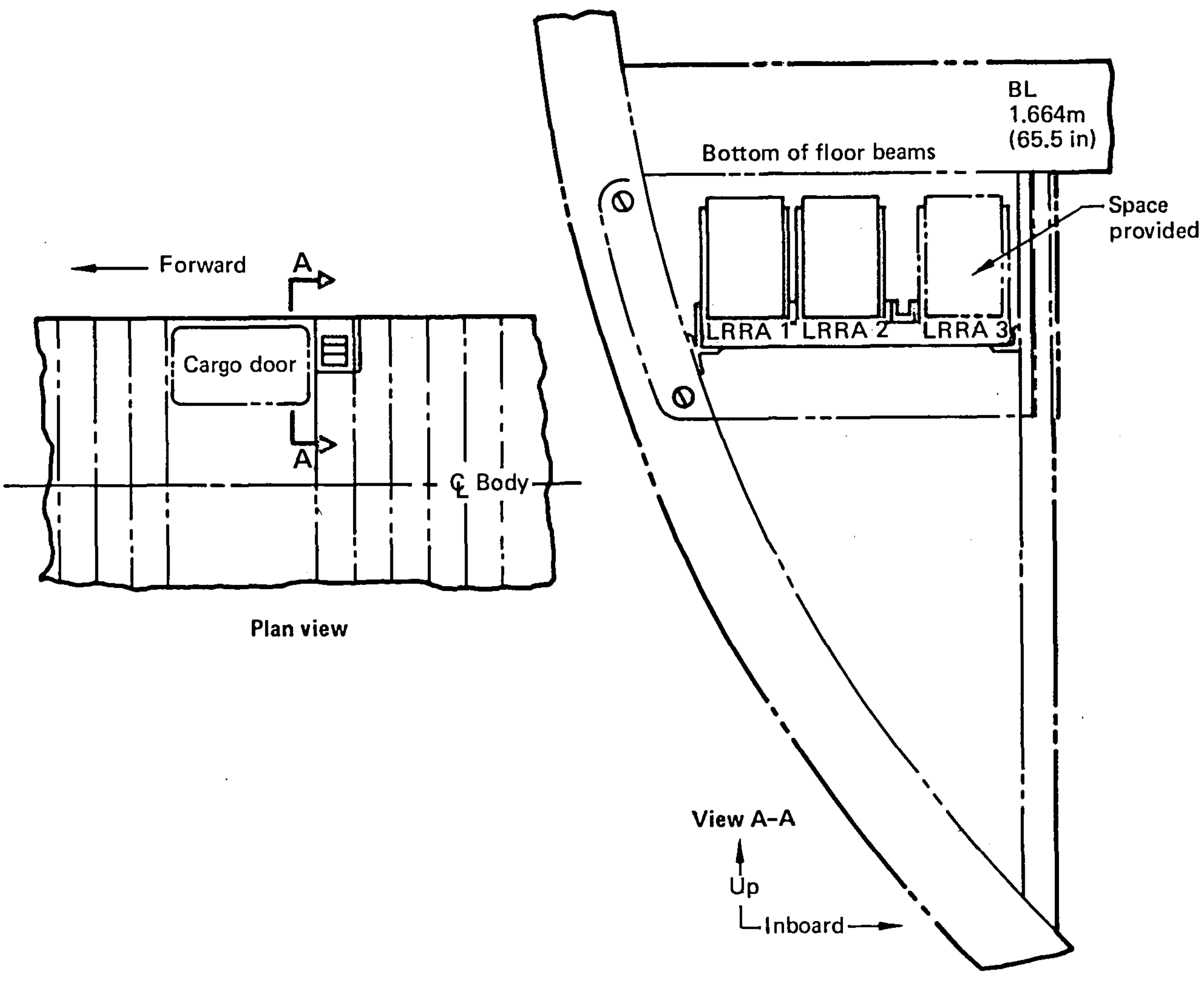


Figure 50. Low-Range Radio Altimeter (LRRA) Receivers

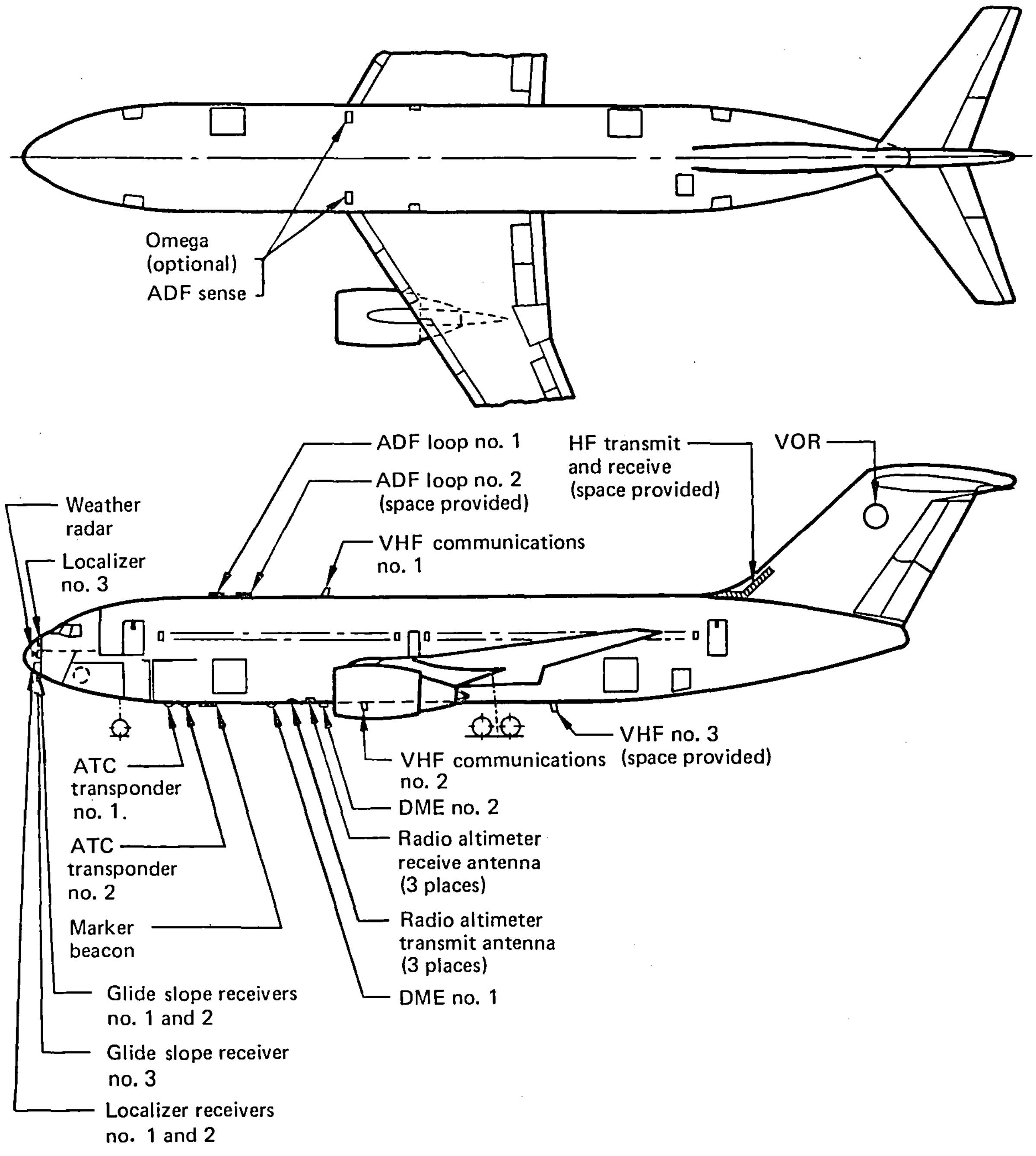


Figure 51. Electronic Systems—Antenna Installations

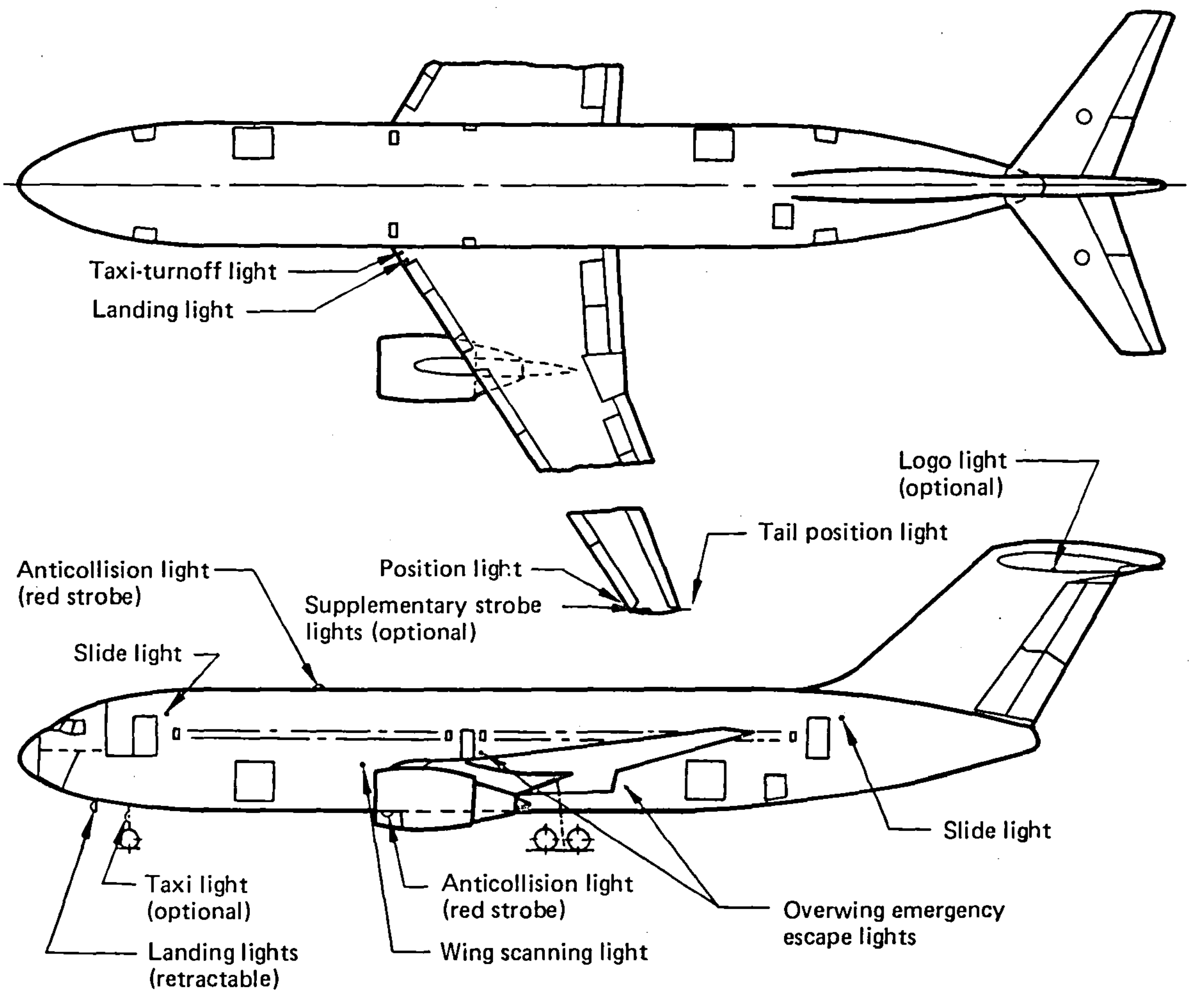


Figure 52. Electric Systems—Light Installations

relay/circuit breaker shield. Two control system electronic unit boxes are also installed.

Maintainability access to the equipment bay is through a hatch located on the bottom of the body, and one located left of the nose gear wheel well. There is also an inflight access panel located in the ceiling. The racks are accessible from their fronts and backs.

The rear equipment rack is located immediately aft of the main cargo door in the aft cargo compartment. Included in the rack are the main battery, battery charger, APU control, APU start TRU, flight recorder, voice recorder, and a miscellaneous equipment box. The battery, voice recorder, flight recorder, and APU can be serviced through the main cargo door opening with the cargo compartment fully loaded.

Wiring installation is based on centralized wire routing raceways, as shown in Figure 53, and on a plug-to-plug concept. Wire types are compatible with the environment in which they are installed; i.e., wiring installed in inaccessible areas or in a hostile environment is protected by conduit. Bundle separation is used to satisfy safety, redundancy, and electromagnetic interference control criteria. Generator cables are routed separately. Coax cables are installed in raceways with other wiring, provided other criteria are not violated. All equipment shelves and shields are wired in such a way that a shelf or a shield can be removed from the airplane.

6.2.5 FLIGHT MANAGEMENT SYSTEMS

The avionics equipment includes communications, audio equipment, a voice recorder system, a weather radar system, radio navigation equipment, ground proximity warning, and miscellaneous avionics. The communications equipment includes very high frequency (VHF) and operational HF communications, an air traffic control (ATC) transponder, and selective calling (SELCAL). The audio equipment includes the flight interphone system, the service interphone system, the passenger address system, and the passenger service and optional entertainment system. The radio navigation equipment includes the low-range radio altimeter, ILS, VOR, distance measuring equipment (DME), automatic direction finder (ADF), marker beacon, and an optional omega system. Automatic navigation and guidance capability is provided by the FMC

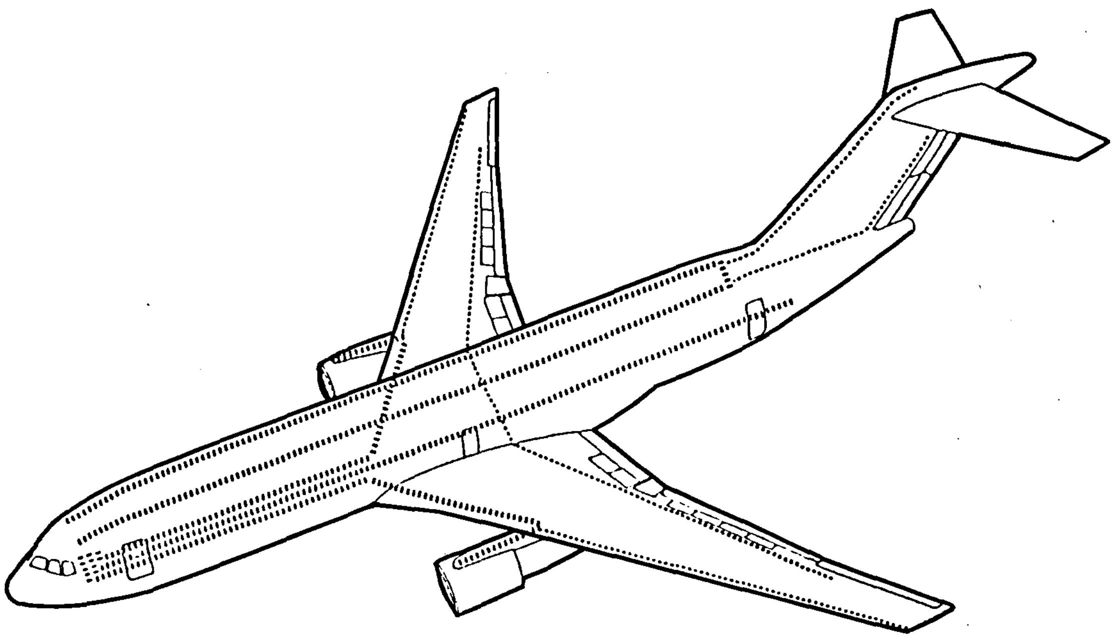


Figure 53. Electric/Electronic Wiring Raceway

subsystem which will conform to the ARINC 702 characteristics (ref 8). This subsystem will provide path guidance and energy management performance guidance in both en route and terminal area flight segments.

Three independent inertial reference systems, ARINC-704 type (ref 9) are installed to provide the necessary attitude, heading, velocity, acceleration, and angular rate data to satisfy the performance requirements specified by the automatic flight control electronics system, weather radar, flight management system, and flight instrument systems.

Two digital air data computers, ARINC-706 type (ref 10), are installed to provide sources for the air data instruments, AFC electronic system, FMC, TMC, inertial reference system (IRS), and advisory, monitoring, and recording systems. The air data computers are installed in the electric/electronic bay. These ARINC-type units are seller-furnished equipment (SFE) for the Baseline Configuration development and certification.

Warning electronics module functions provided include: stall warning, over-rotation warning, altitude alert, unsafe configuration, rudder ratio comparison, and elevator feel comparison. Included within the unsafe configuration/condition function are takeoff warning and unsafe configuration for landing. This module is a SFE "make" item. A stick shaker is provided for each pilot and is activated by the stall warn function in the air and by the over-rotation function on the ground.

Caution/advisory computer functions include: instrument comparison, EDP low pressure and quantity, low hydraulic quantity on second system, autospoiler malfunction, fuel boost pump malfunction, trailing- and leading-edge flap disagree, and generation of messages for the caution/warning displays. The aural warning unit functions are aural signal generation and amplification of these warnings or the voice warning generated by the ground proximity unit. Two caution/warning displays are installed in the flight deck: one in the center panel, plus one at the flight engineer's station. Both displays, which are identical and display the same data, are driven by the caution/advisory computer.

Maintenance monitoring subsystem space provisions are made for an aircraft integrated data system (AIDS).

An ARINC 700 series flight recorder system and a cockpit voice recorder are installed in the Baseline Configuration. The digital flight data recorder and the cockpit voice recorder are located in the aft pressurized section of the airplane and are accessible from the aft cargo door. The flight data acquisition function is provided by a new ARINC flight data acquisition unit (FDAU) compatible with the respective ARINC 700 series digital equipment and an ARINC 573 (ref 11) recorder. The FDAU is installed in the main electric/electronics bay. An ARINC 573 three-axis accelerometer is located within the cg limits of the airplane, and a flight recorder control panel and a flight data entry panel are located at the flight engineer's station.

6.2.5.1 Takeoff Trim Setting Green-Band System

A multiple position, green-band takeoff trim setting indication system is used to reduce the possibility of takeoff with a large amount of mistrim. The green-band system consists of a selector switch, green-band indicator lights on the stabilizer trim scale, aural warning circuits, and an amber light circuit. There are three different takeoff green-band ranges. Airplane gross weight and cg position determine which green-band range should be used for a particular takeoff. The green-band range and associated aural warning are selected by the pilot via a three-position switch on the pilot's control stand and having the notations APL NOSE UP, MID, and APL NOSE DOWN. The warning horn will sound if the stabilizer is set outside the selected green band when the thrust levers are advanced for takeoff in the same way as on present airplanes.

An amber light circuit is incorporated as an added safety measure to guard against a takeoff with the stabilizer mistrimmed to the extreme of an incorrectly selected green band. Two amber lights are used: one is located on the pilot's control stand adjacent to the green-band selector switch, and a second light is on the master annunciator panel. Both lights will illuminate when the selected green band does not correspond to airplane weight and cg, as sensed by nose-gear load (oleo pressure switch). The lights do not illuminate when the MID band is selected.

6.2.6 ENVIRONMENTAL CONTROL SYSTEM

The environmental control system (ECS), shown in Figure 54, conditions and ventilates both the passenger cabin (0.57 m^3/min or 20 cfm per person) and the crew cabin (one air change per minute), cools instruments and electric/electronic equipment, heats the cargo compartment, and maintains cabin pressure at a selected pressure schedule. Thermal anti-icing is also provided for wing leading-edge surfaces.

The air distribution system is sized to deliver 113.3 m^3/min (4000 cfm) to the passenger cabin and 16.3 m^3/min (575 cfm) to the crew cabin at cruise conditions. Of the total 129.6 m^3/min (4575 cfm) of cabin distribution air, 50% is filtered recirculation air, and 50% is conditioned bleed air.

Each of the two cooling packs is sized to deliver 50 kg/min (110 lb/min) airflow during APU operation on the ground. Inflight operation delivers 36.46 m^3/min (1287 cfm) of condition air per pack at cruise, with higher flows up to 5.0 m^3/min (175 cfm) per pack at altitudes below 7620m (2500 ft), which meets cabin conditioning requirements.

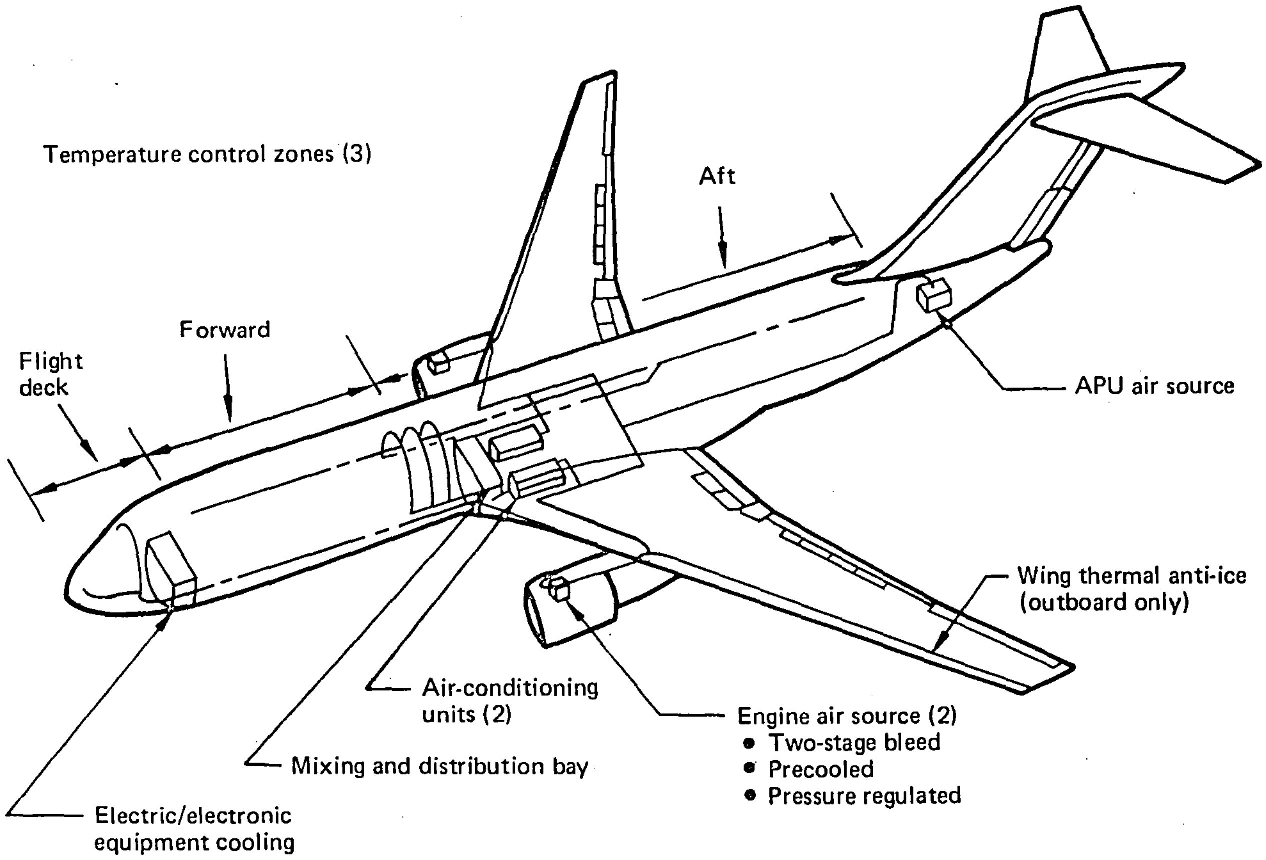


Figure 54. General Arrangement (Environmental Control System)

The air supply system furnishes a total of 12.7 kg/min (28 lb/min) from the engines at takeoff for the two packs. The APU furnishes 75.3 kg/min (210 lb/min) on the ground, engines off.

7.0 ANALYSES AND CHARACTERISTICS

Page

7.0 ANALYSES AND CHARACTERISTICS

Baseline Airplane characteristics and analyses are presented in this section. These data will be used to assess active controls technology (ACT) benefits and will serve as the starting point for integrated application of active controls (IAAC) configuration development.

Stability and control data include trim, control, and stability characteristics about the longitudinal, lateral, and directional axes. Structural design data comprise definitions of critical design conditions and loads, material allowable stresses, flutter and fatigue analyses, and structural member sizing. These data are defined for the wing, body, and horizontal tail. The Aerodynamics section describes lift and drag data for high and low speed. The weight design data include a weight breakdown of operating empty weight, definitions of the major airplane components, weight distribution, and moments of inertia about the three principal axes of the airplane.

7.1 FLYING QUALITIES

This section describes the trim, control, and stability characteristics in Subsections 7.1.1, 7.1.2, and 7.1.3, respectively. Each section begins with the longitudinal axis description followed by that for the lateral-directional, and each section emphasizes the critical flight characteristics relative to limits (e.g., control available) or criteria. These flight characteristics are predicted from static wind tunnel data, damping estimates, and quasi-static aeroelastic (QSAE) correction factors. The prediction and analysis methods and the rigid aerodynamic data were the same as those used later for the Initial ACT Configuration.

Figure 55 shows the high- and low-speed flight envelopes for two gross weights, which represent extremes for flying qualities. The design mission is: takeoff at 122 470 kg (270 000 lb), climb to cruise altitude, and descend and land at 90 720 kg (200 000 lb). The design mission is within the operational flight envelope defined by maximum operating speed (V_{MO}), maximum operating Mach number (M_{MO}), maximum altitude (12 800m/[42 000 ft]), and 1.2 stall speed (1.2 V_S). The design envelope for emergency flight is defined by V_D/M_D flap placard and stall warning. Figure 55

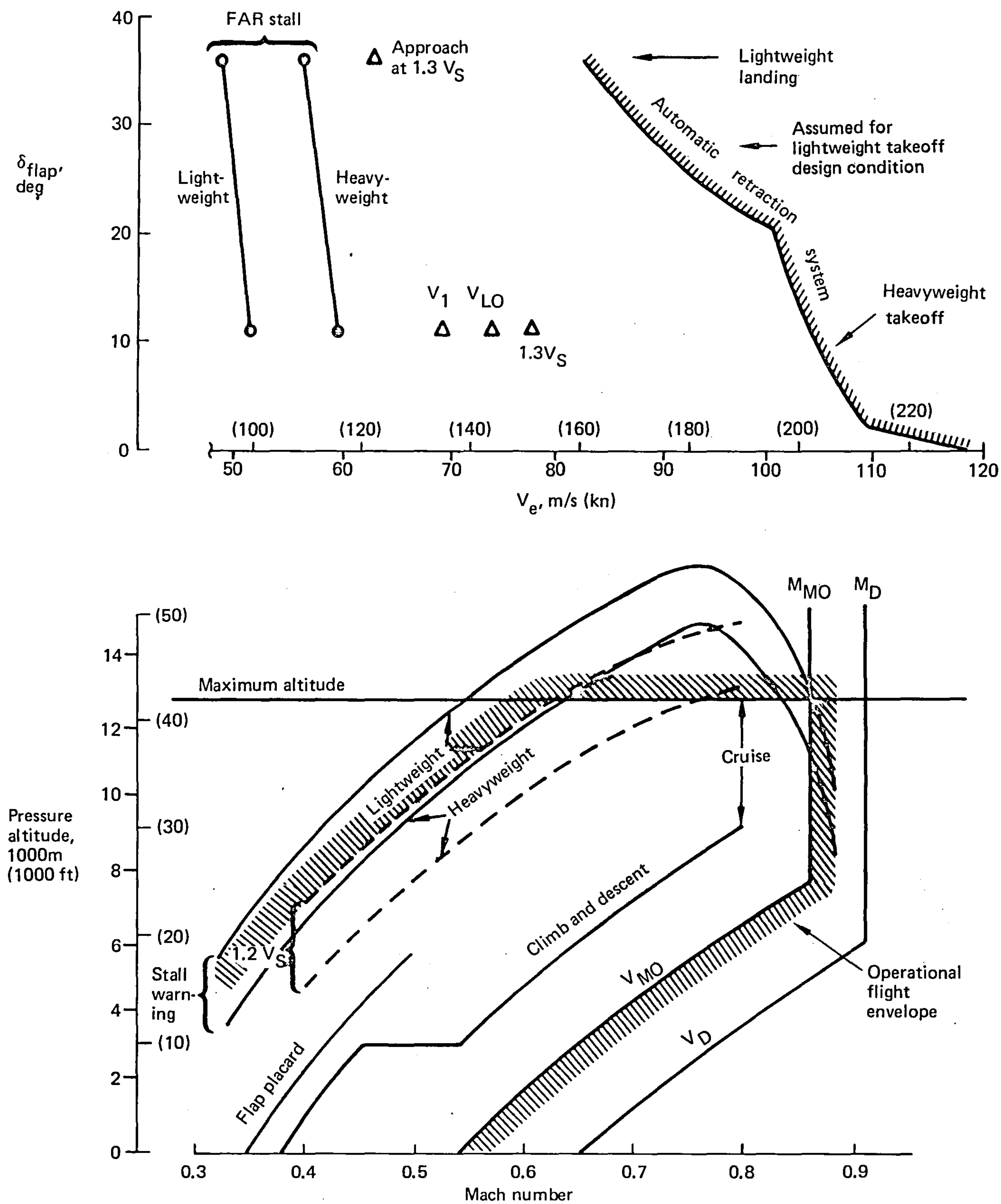


Figure 55. Speed and Altitude Flight Envelopes

illustrates that 1.2 V_S and stall warning depend on weight. The other speed placards, which are independent of weight, are:

- V_D = 221.2 m/s (430 kn) calibrated airspeed
- V_{MO} = 185.2 m/s (360 kn) calibrated airspeed
- flap placard = 118.3 m/s (230 kn) equivalent airspeed (EAS)
- the climb/descent schedule is at 128.6 m/s (250 kn) calibrated airspeed to 3048m (10 000 ft) altitude, then 154.3 m/s (300 kn) calibrated airspeed to 9144m (30 000 ft) altitude

Good (i.e., Level 1) flying qualities are required within the operational flight envelope, and minimum safe (Level 3) flying qualities must be provided at the extremities of the design flight envelope. The Baseline Configuration must meet these flying quality criteria through configuration design and conventional flight control system design.

The flying quality characteristics presented in this section emphasize the extremities of these flight envelopes for heavy or light weight and forward 10% mean aerodynamic chord (MAC) or aft 38% MAC center of gravity (cg) limits. Also, critical moments of inertia are used which may represent unusual, but possible payload-fuel distributions.

Engine out, takeoff mistrim, and hydraulic system failures also affect trim and control characteristics and are presented in a manner to emphasize the critical conditions. Figure 56 shows the reduction of available control power due to hydraulic failures.

The several longitudinal stability and controllability requirements which were considered in establishing the horizontal tail size are shown in Figure 57. The critical aft cg requirement is seen to be stall recovery; that is, the ability to develop recovery pitching moment at a high post-stall angle of attack, thus precluding the development of a stable "lock-in" condition. Coincidentally, minimum stability criteria produce very nearly the same horizontal tail requirement. Also shown is a minimum maneuvering elevator deflection gradient requirement (δ_E/g) which is an objective for o Available control is greater at speeds less than those noted.

Critical for control

Critical for trim

Minimum percent of normal control remaining:

One system out	Failed System: System A	System B	System C
Elevators $V_e > 133.8$ m/s (260 kn)	75	75	50
Speed brake C_M $V_e > 179.0$ m/s (348 kn)	100	50	50
Spoiler/brakes $V_e > 179.0$ m/s (348 kn)	67	67	67
Ailerons $V_e > 87.5$ m/s (170 kn)	50	75	75
Rudders (with ratio changer) $V_e > 72.0$ m/s (140 kn)	90	80	90

Two systems out	Failed Systems: AB	AC	BC
Elevators $V_e > 13.8$ m/s (260 kn)	50	25	25
Speed brake C_M $V_e > 179.0$ m/s (348 kn)	50	50	0
Spoilers/brakes $V_e > 179.0$ m/s (348 kn)	33	33	33
Ailerons $V_e > 87.5$ m/s (170 kn)	25	25	50
Rudders (with ratio changer) $V_e > 72.0$ m/s (140 kn)	40	80	40

Figure 56. Control Power After Hydraulic System Failure

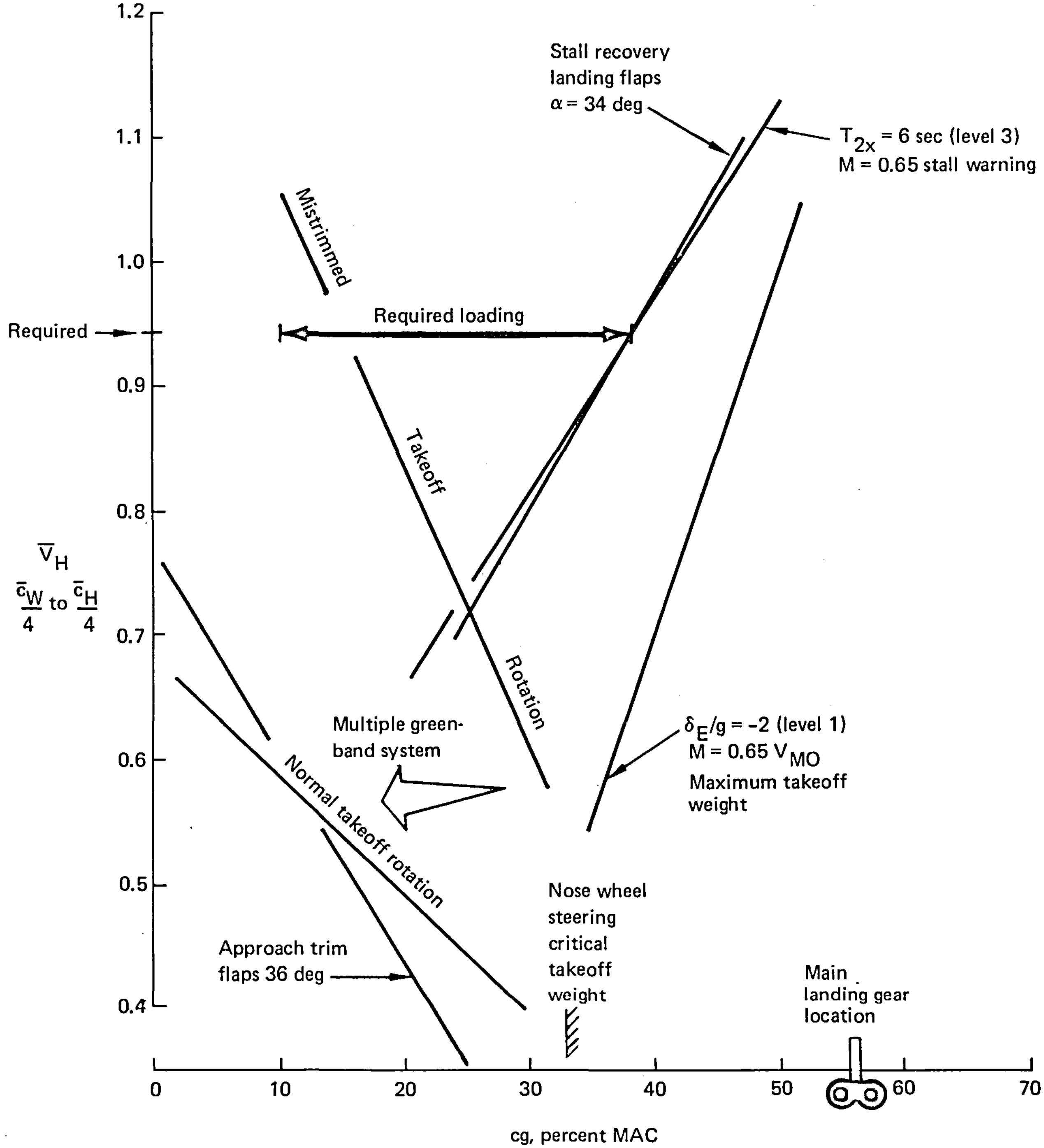


Figure 57. Horizontal Tail Size Requirements

conventional feel system design. As shown in Figure 57, this criterion does not determine tail size. The nose wheel steering limits shown relate to the minimum vertical load on the nose wheel which is required for steering effectiveness. Because the airplane's thrust line is below the cg, the nose wheel load is reduced during ground acceleration, the critical case occurring at brake release.

Figure 57 also illustrates that a smaller horizontal tail could be employed if the cg range were more forward. The forward movement would be limited by normal takeoff rotation and a landing approach trim requirement. Transport airplanes are designed to accommodate some degree of mistrim at takeoff. The permissible takeoff trim range is delineated by the green band on the trim indicator. If trim is set outside the green-band range, the takeoff configuration warning is enabled, and the "takeoff configuration not set" warning will occur when engine power is advanced for takeoff. A multiple position green band, as described in Section 6.2.5.1, is used to further reduce the required mistrim allowance. The cg range was not moved forward on the Baseline Configuration because of other configuration considerations such as trim drag, cargo container loading, reduced tail arm, etc., which are affected by the wing repositioning on the fuselage required to shift the loading range forward relative to its MAC.

The Baseline Configuration employs a plain elevator. Although control power would increase with a double-hinged elevator, stability requirements would be unchanged and, as Figure 57 illustrates, the tail size could not be reduced without major configuration changes. However, mistrimmed takeoff rotation would improve and the multiple green-band system would be eliminated. This trade was not considered attractive enough to incorporate a double-hinged elevator.

The vertical tail is sized by low-speed controllability considerations as shown in Figure 58. As is typical of multiengine airplanes with wing-mounted engines, control of asymmetric power takeoff conditions determines the required directional stability and control power. Three flight conditions are customarily analyzed, any one of which may be limiting. These are ground minimum control speed (V_{MC_G}), air minimum control speed (V_{MC_A}), and the "tameness" requirement which specifies a minimum level of directional stability and lateral control power to permit equilibrium flight

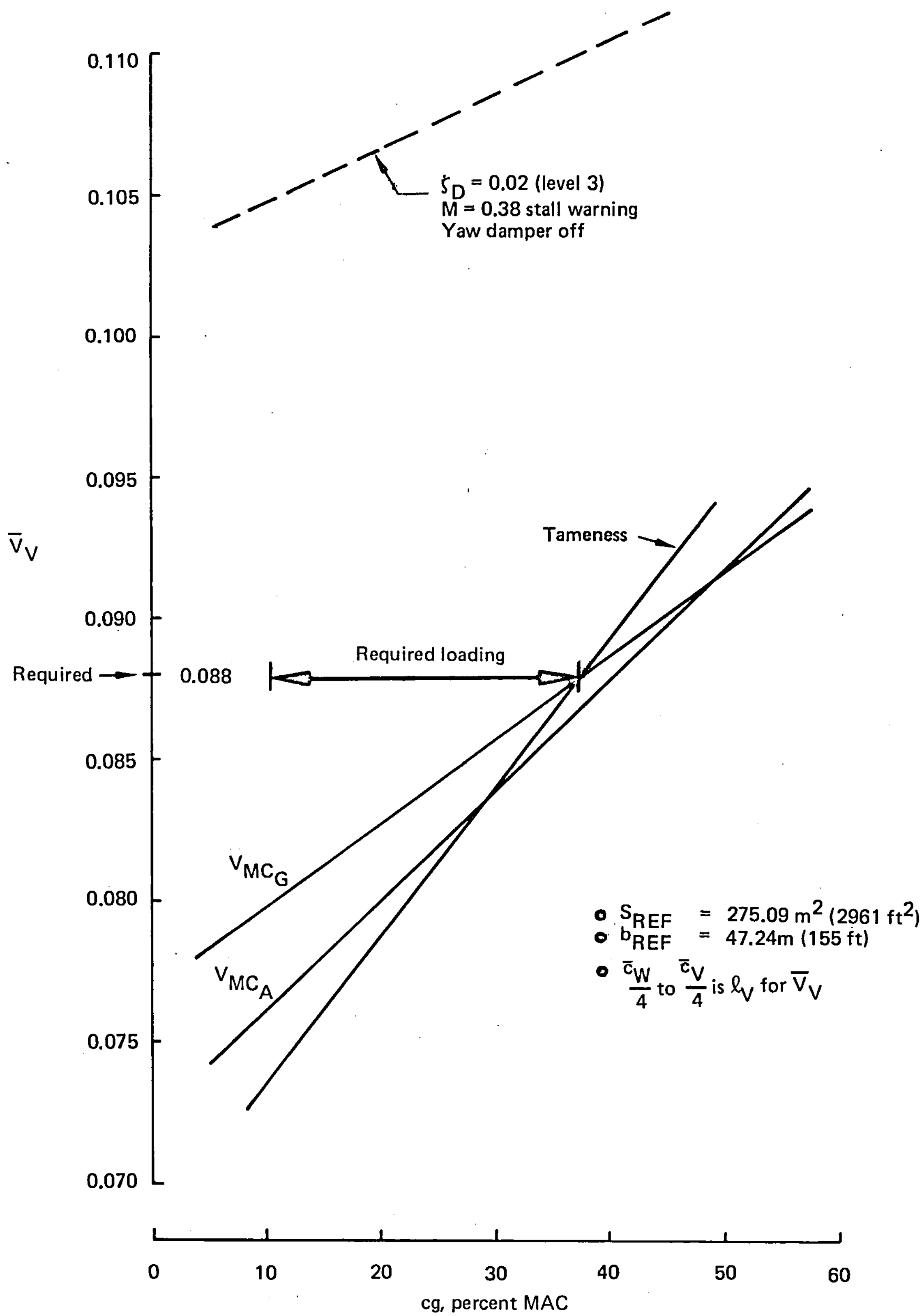


Figure 58. Vertical Tail Size Requirements

with an engine failed and no flight-crew rudder input. For this airplane, the low-speed lateral control power and vertical tail size have been chosen to make V_{MC_G} and "tameness" requirements equal at the aft cg limit. As shown, the V_{MC_A} requirement is slightly less critical.

Also shown on Figure 58 is the vertical tail size for minimum acceptable unaugmented Dutch roll mode damping, which can be seen to be much greater than that to meet the foregoing controllability requirements. Because Dutch roll mode damping requirements will be met by use of a redundant yaw damper, they do not impact the selected vertical tail size.

7.1.1 TRIM

This subsection describes the longitudinal trim characteristics which are important for defining stability, control, and system characteristics. Engine-out and sideslip trim, which are related to lateral-directional control, are also described.

Figure 59 illustrates the extremes of trimmed angle of attack within which flight characteristics are to be presented. The angle-of-attack margin between cruise and stall warning provides about 3/4-g incremental load factor. The corresponding margin between the minimum speed operational limit, defined by 1.2 V_S or 12 800m (42 000 ft), and stall warning is about 1/3 g to M = 0.65, but depends on weight at higher Mach numbers.

Stabilizer for longitudinal trim within the permissible operational flight envelope is shown in Figure 60. The band separates forward and aft cg conditions. Stabilizer trim range required was established by forward cg at landing, under icing conditions, and by aft cg at the V_{MO} placard speed. Note in Figure 60 that cruise trim ranges between +0.5 deg of stabilizer at end of cruise, aft cg, and -0.9 deg at begin cruise, forward cg. If a passive trim failure occurs at cruise (+0.5 deg), then the -8 deg of stabilizer, normally used for landing trim, would require -16 deg of elevator trim. Since there is a deflection of -25 deg of equivalent elevator available at the landing approach speed, even with one critical hydraulic system lost, adequate pitch control for maneuver and landing flare remains (see subsec 7.1.2 and fig. 65).

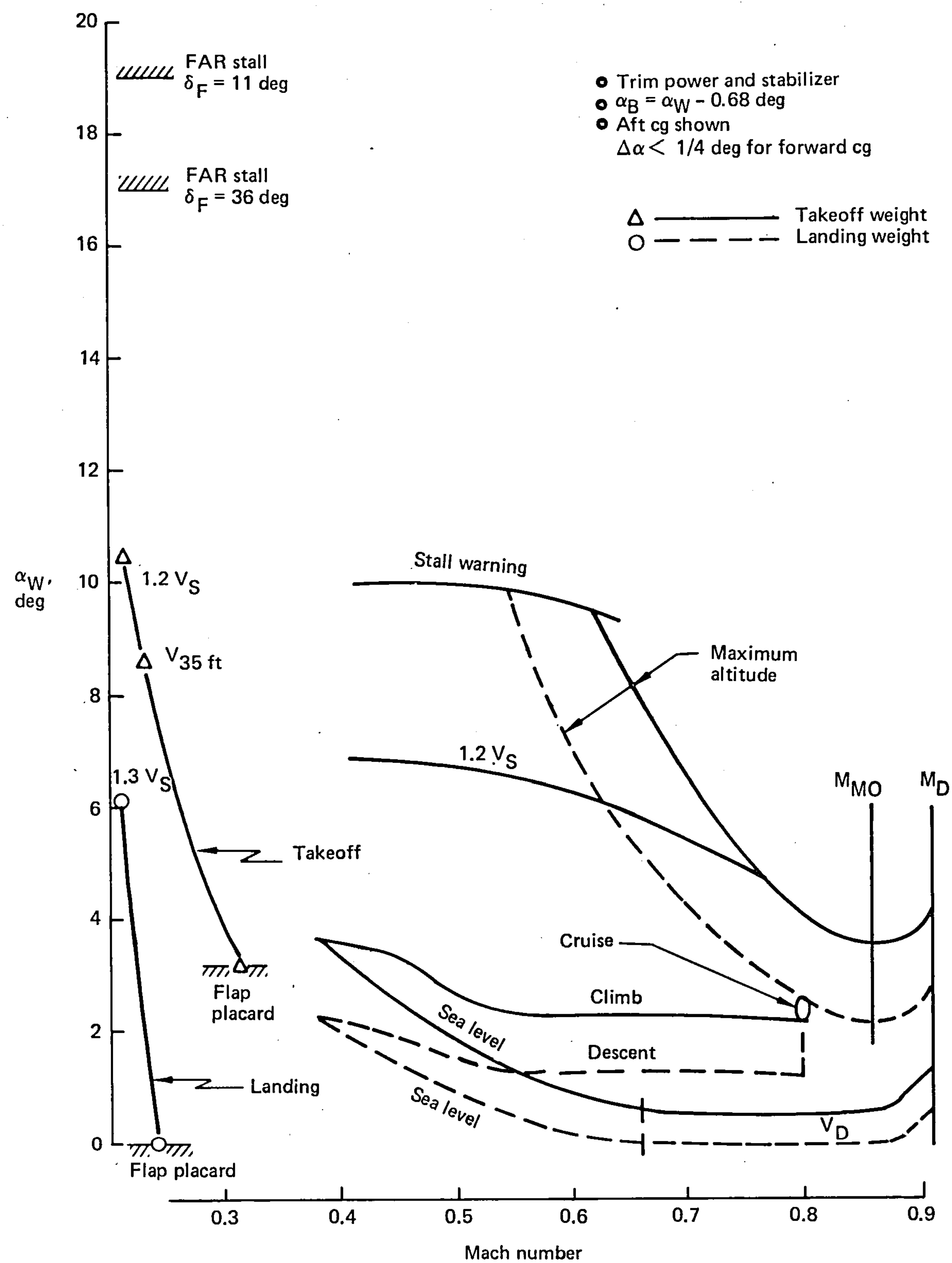


Figure 59. Angle of Attack for Trim

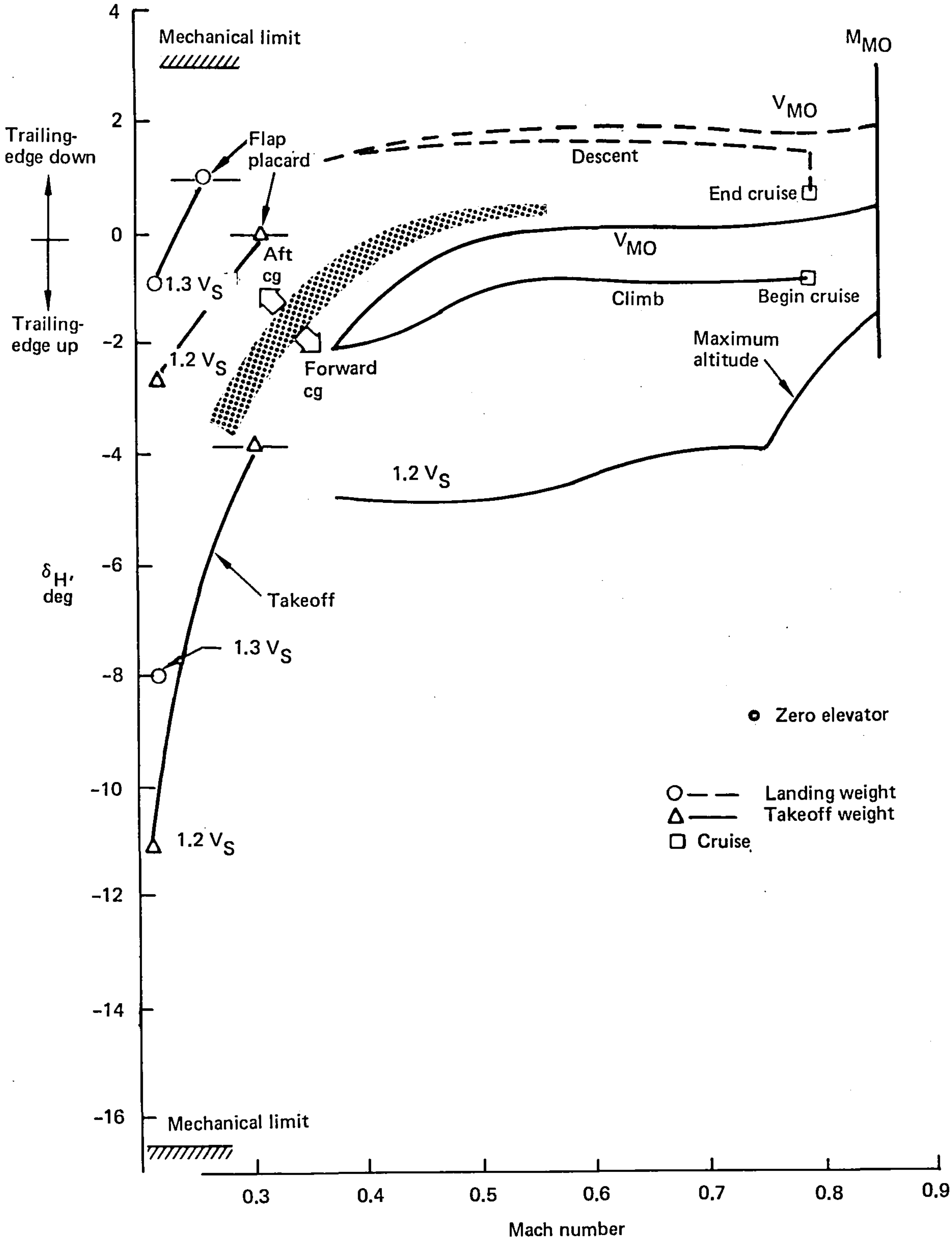


Figure 60. Stabilizer Angle for Trim

The potential takeoff mistrim is illustrated in Figure 60. For example, normal takeoff trim would be -8.3 deg of stabilizer at maximum weight and the forward cg limit. If the trim were inadvertently set at the mechanical limit of +3 deg, takeoff rotation control would be compromised. Subsection 7.1.2 discusses normal and mistrimmed takeoff control.

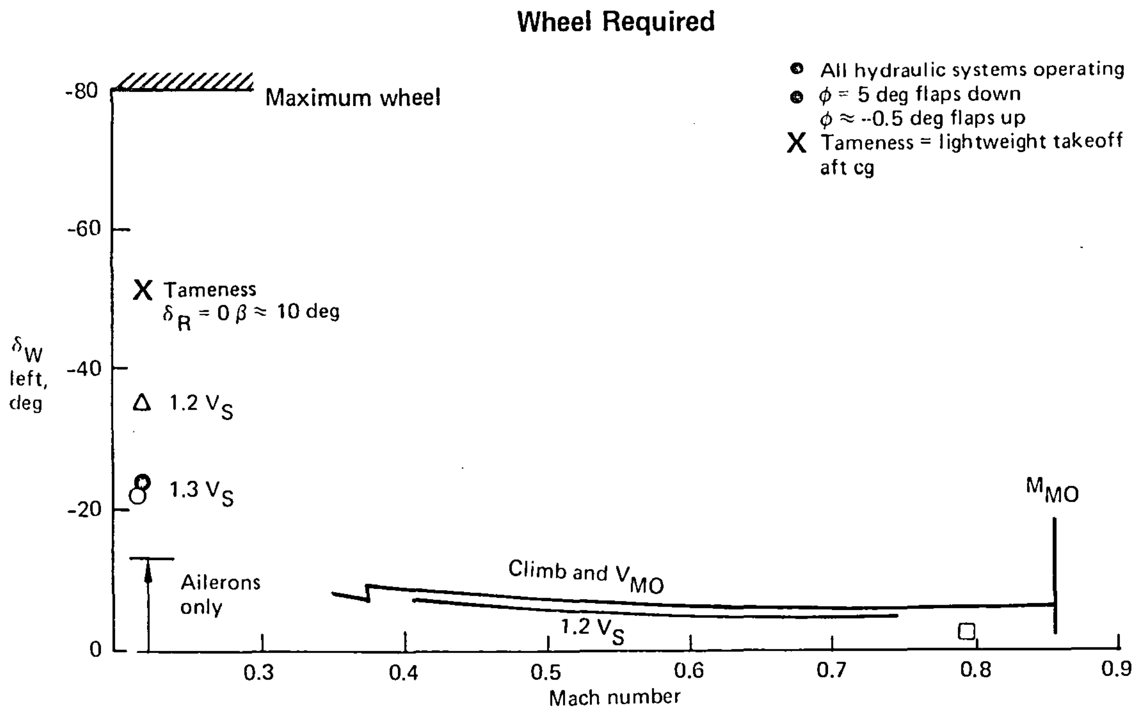
Yaw and roll control are adequate to trim an engine-out throughout the flight envelope as shown on Figure 61. No spoilers are used for engine-out trim at cruise. Another criterion applies during takeoff and landing where only lateral control is used for engine-out trim. This is called "tameness" and is to be statically met with no more than two-thirds of maximum wheel. The remaining one-third is assumed to be sufficient for gusts and/or hydraulic system failures. Figure 61 shows that tameness is just met at the critical aft cg light-weight takeoff condition and is the criterion which sizes the lateral control system.

Figure 62 shows full rudder sideslip trim capability and the required lateral trim. Forward cg leads to the smallest sideslip capability; however, aft cg requires the largest amount of wheel for trim. Figure 62 shows that less than two-thirds of the wheel is required to trim full rudder sideslip throughout the design flight envelope. Landing in a 15.43 m/s (30 kn) cross-wind at normal approach speed with no residual crab angle would require a 13.8-deg sideslip angle; however, with an allowable 4-deg crab, only 9.8 deg of sideslip are required. Figure 62 shows that with rudder power available, even with one hydraulic system out, 12-deg sideslip can be held. Cross-wind landing with the 4-deg crab requires about 38 deg of wheel at the critical aft cg, and the corresponding rudder requirement is 16.8 deg out of 25 deg available.

7.1.2 CONTROL

This section describes control characteristics for takeoff, landing, and en route flight; these data were used for empennage control and actuator sizing.

Takeoff control capability with loss of one critical hydraulic system is shown in Figure 63. Takeoff rotation capability is shown for normal trim set for V_2 and provides control for rotation below the performance rotation speed. However, with



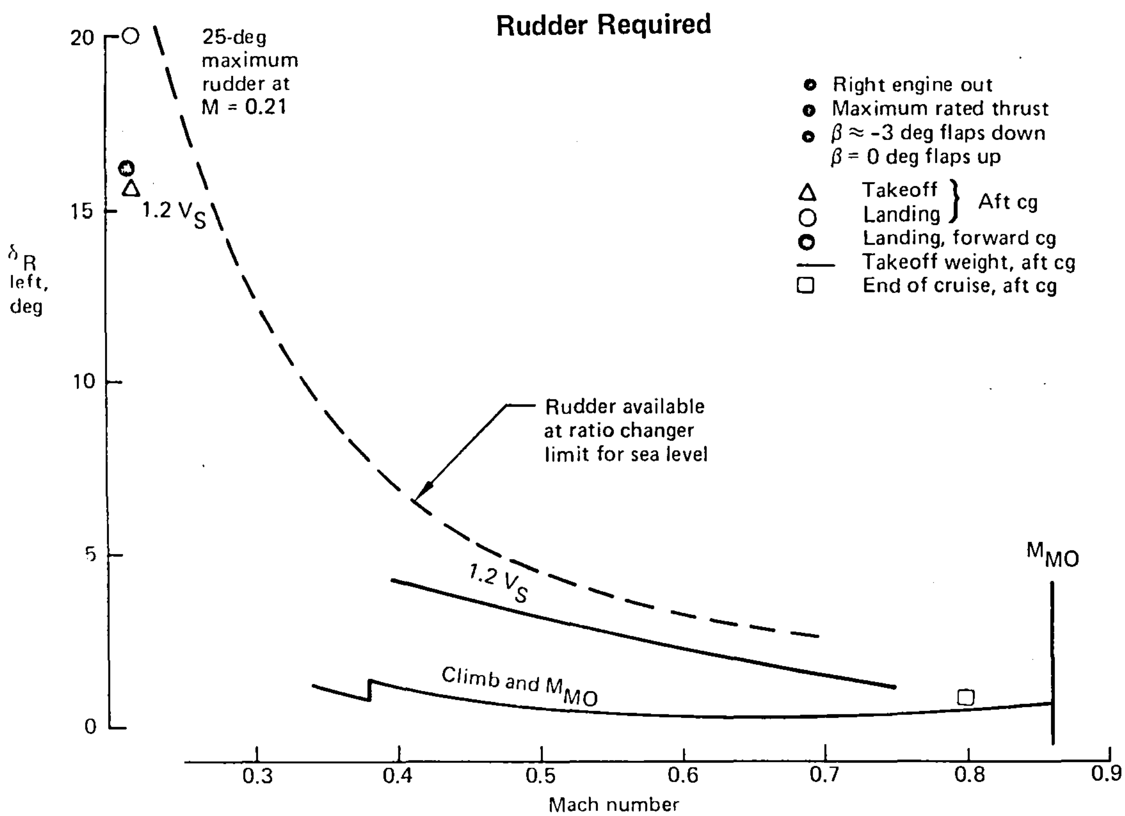


Figure 61. Engine-Out Trim

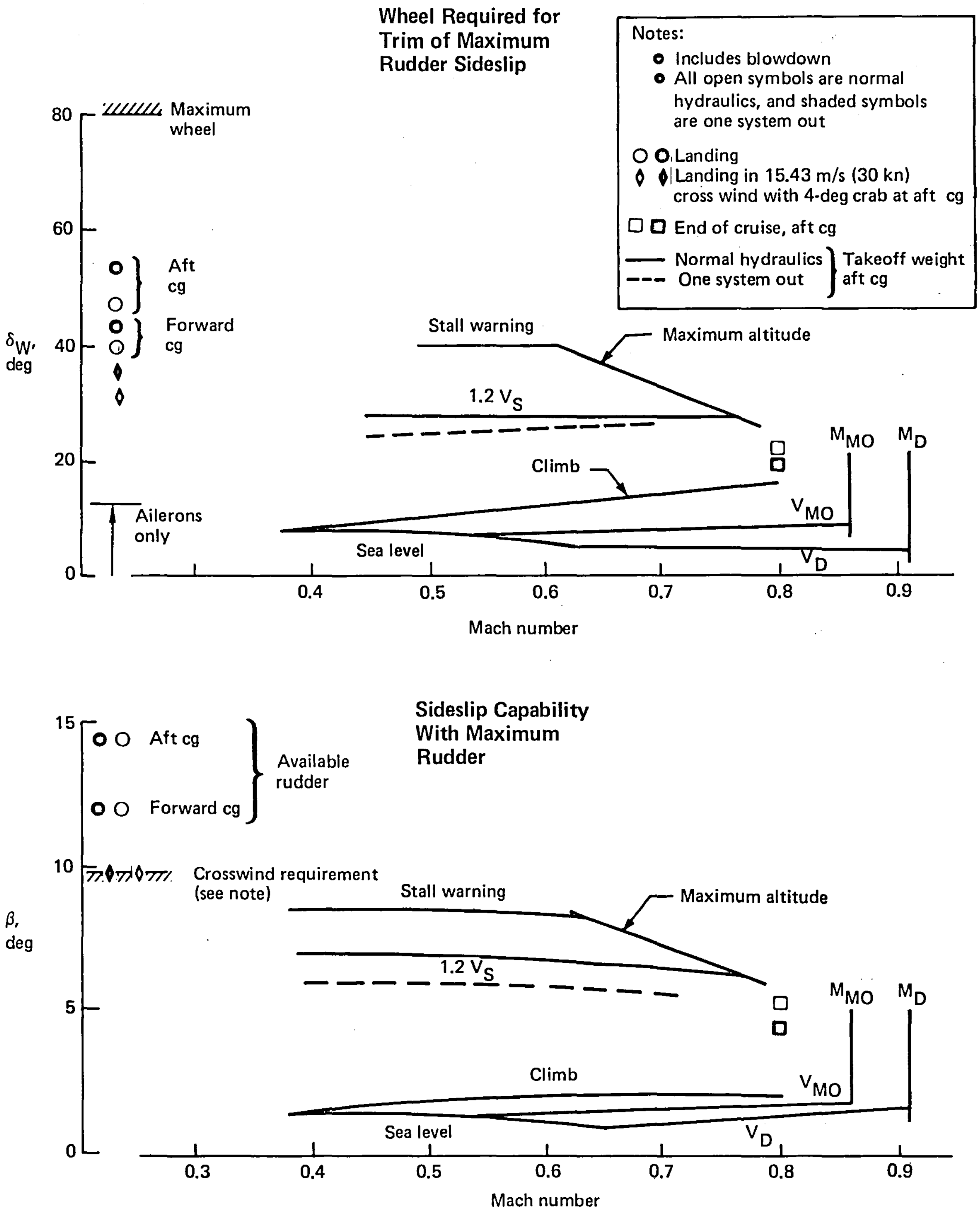


Figure 62. Side-Slip Trim Capability

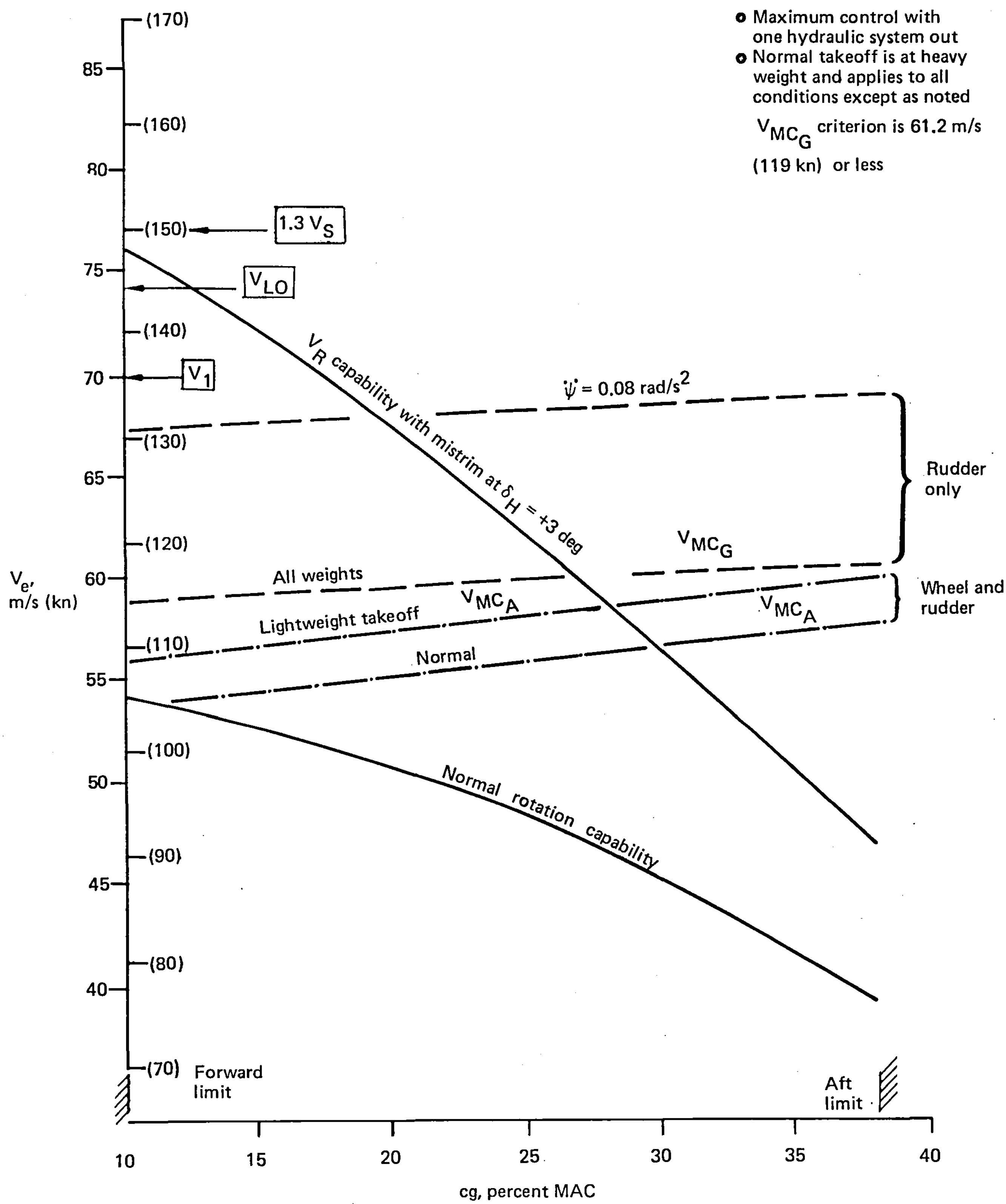


Figure 63. Takeoff Control

full mistrim the elevator cannot meet the performance rotation speed, and a cg sensing multiple green-band system will be incorporated to preclude full mistrim at takeoff.

Stall recovery is critical at landing flaps and is illustrated for aft cg in Figure 64. When trimmed for approach, the airplane develops pitch-up at about 24 deg alpha and does not exhibit natural recovery until about 37 deg; however, full nose-down control does arrest the pitchup. Full control is available at the stall speed even with loss of the critical hydraulic system. The high angle of attack behavior characteristic of high tails and the requirement for nose-down control margin are the critical design conditions which size the horizontal tail and elevators.

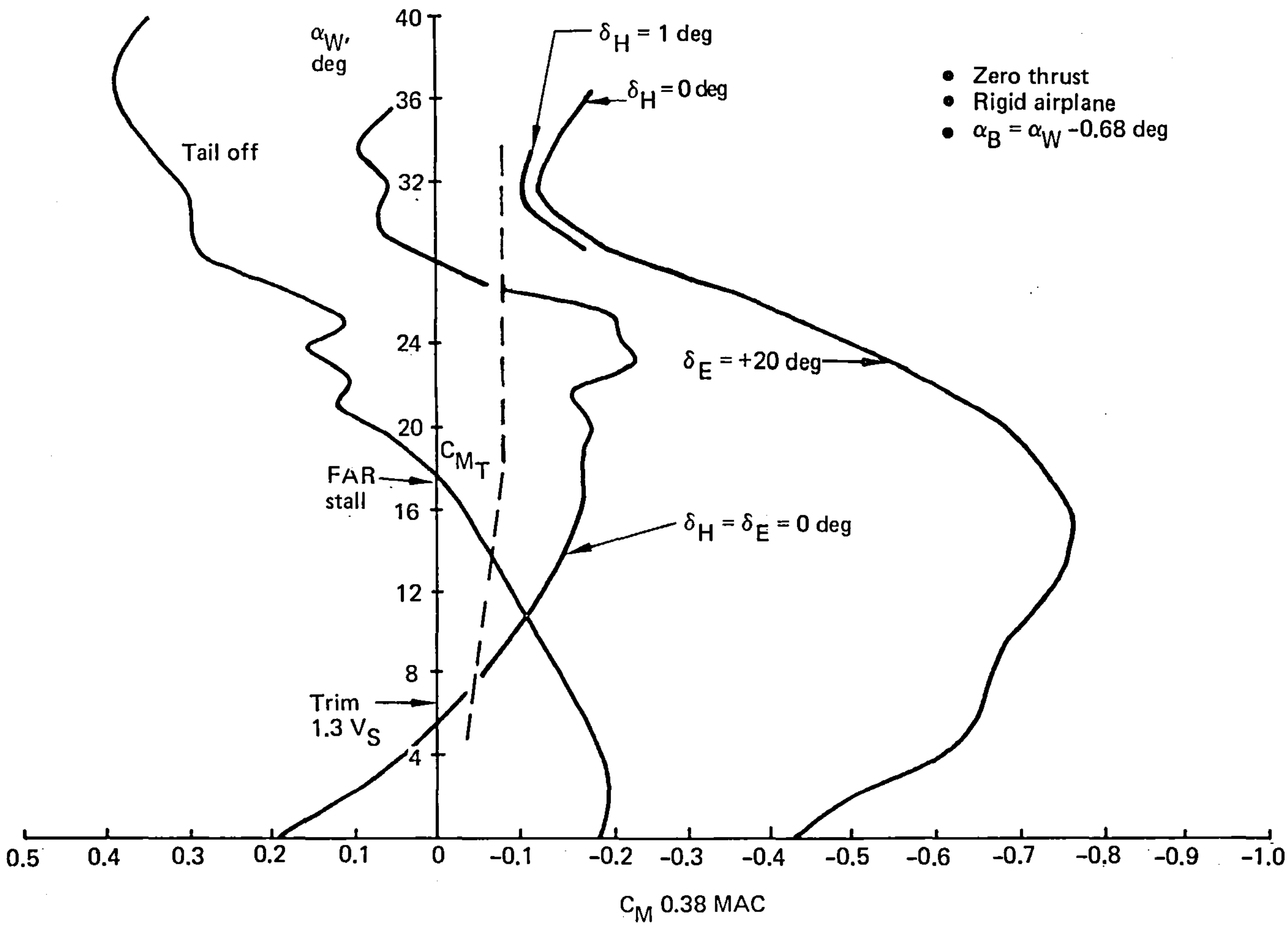


Figure 64. Landing Stall Recovery

Yaw acceleration capability and engine-out control are also shown in Figure 63. Both are critical at aft cg where rudder power is minimum; however, the latter is most critical at light gross weight. Engine-out control on the ground is a dynamic situation involving engine thrust decay, pilot reaction, and rudder deflection rate. Figure 63 shows that this condition just meets the 61.26 m/s (119 kn) criterion and sizes the fin which incorporates a double-hinged rudder. The free air engine-out control is also a dynamic situation. The criterion is 5 deg of bank and 62.3 m/s (121 kn) which is met with a 1.8-m/s (3.5-kn) margin at the aft cg.

Pitch and yaw control power considerations for landing are shown in Figure 65. The speed that provides 0.1 radian per second squared (0.1 rad/s^2) pitch acceleration at landing is shown for both normal approach trim and a mistrim, corresponding to jam at cruise (shown in fig. 60). This pitch control capability is available at speeds well below the normal approach speed of 1.3 V_S. Rudder power capable of producing yaw acceleration of 0.08 rad/s^2 is also illustrated in Figure 65 and is available to speeds well below normal approach. Figure 65 also shows that engine-out control capability exists to speeds well below normal approach.

Roll response capability for takeoff and en route flight are shown in Figure 66 and reflect maximum roll inertia and full control wheel input as a 0.5-sec ramp. Takeoff capability is shown in Figure 66 for all hydraulic systems operating and critical system failures, and it illustrates that normal and two-system failure roll response capabilities are nearest their criteria requirements (Levels 1 and 3, respectively). Normal response at takeoff meets the Level 1 criteria down to about 14% above the stall speed, which is outside the operational flight envelope. The two-system failure case exceeds the 4.5 sec to achieve 30 deg of bank criterion at speeds of less than about 10% above stall and is the critical case. The combined probability of two-system failures and being at stall is remote, and roll response is judged adequate. Roll control is larger at increased flap settings; therefore, landing exhibits better roll response than takeoff. Flaps-up roll response, shown in the upper chart of Figure 66, shows the effect of the aileron lock-out mechanism. All criteria are met, the critical case being normal hydraulic system operating just after roll control commands are eliminated to the outboard aileron by the lockout mechanism (about 250-kn calibrated airspeed).

Figure 67 summarizes speed stability characteristics as the elevator angle required to trim 0.5144 m/sec (1 kn) speed increases. Note that an unstable gradient exists at dive

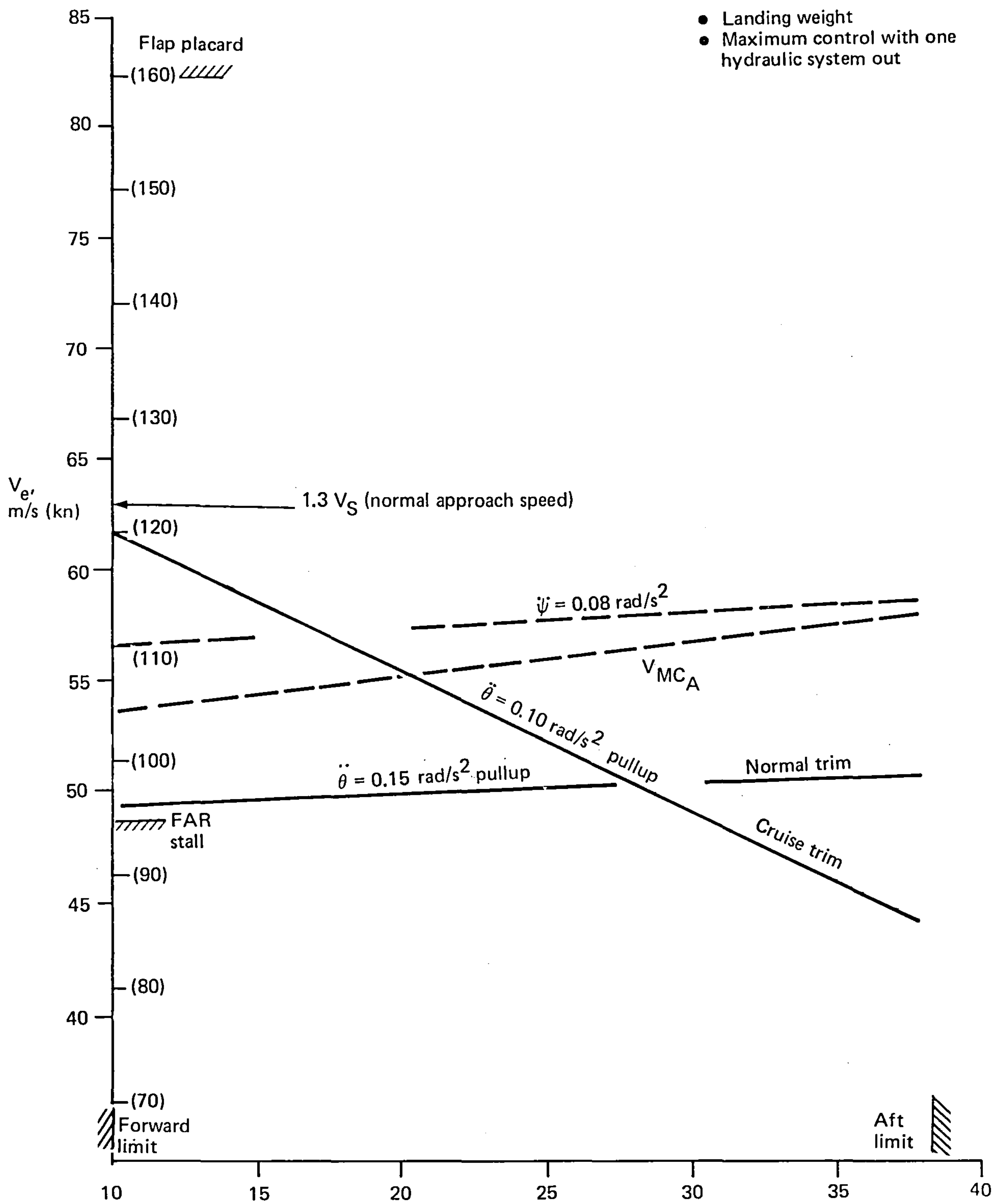


Figure 65. Landing Control

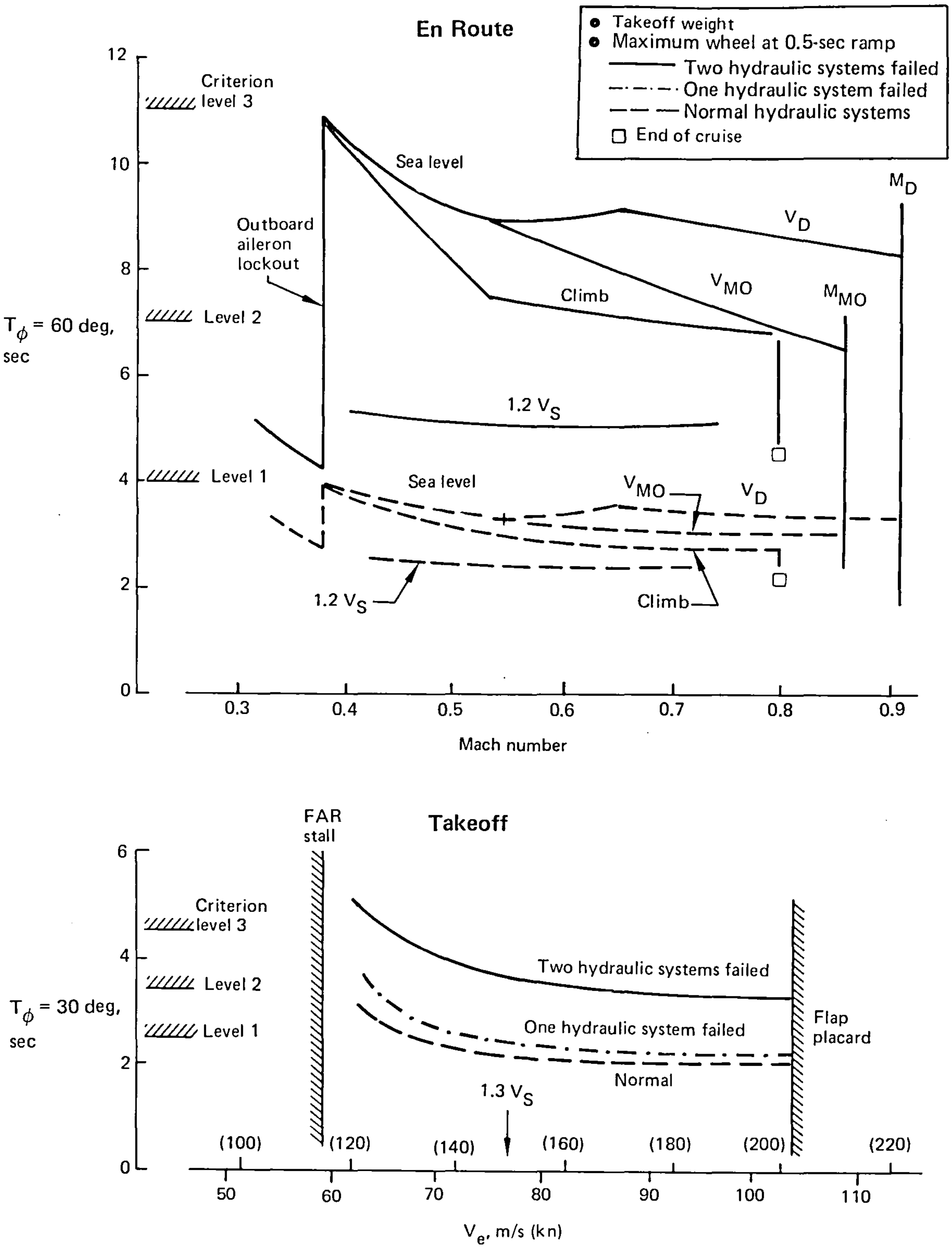


Figure 66. Roll Response Capability

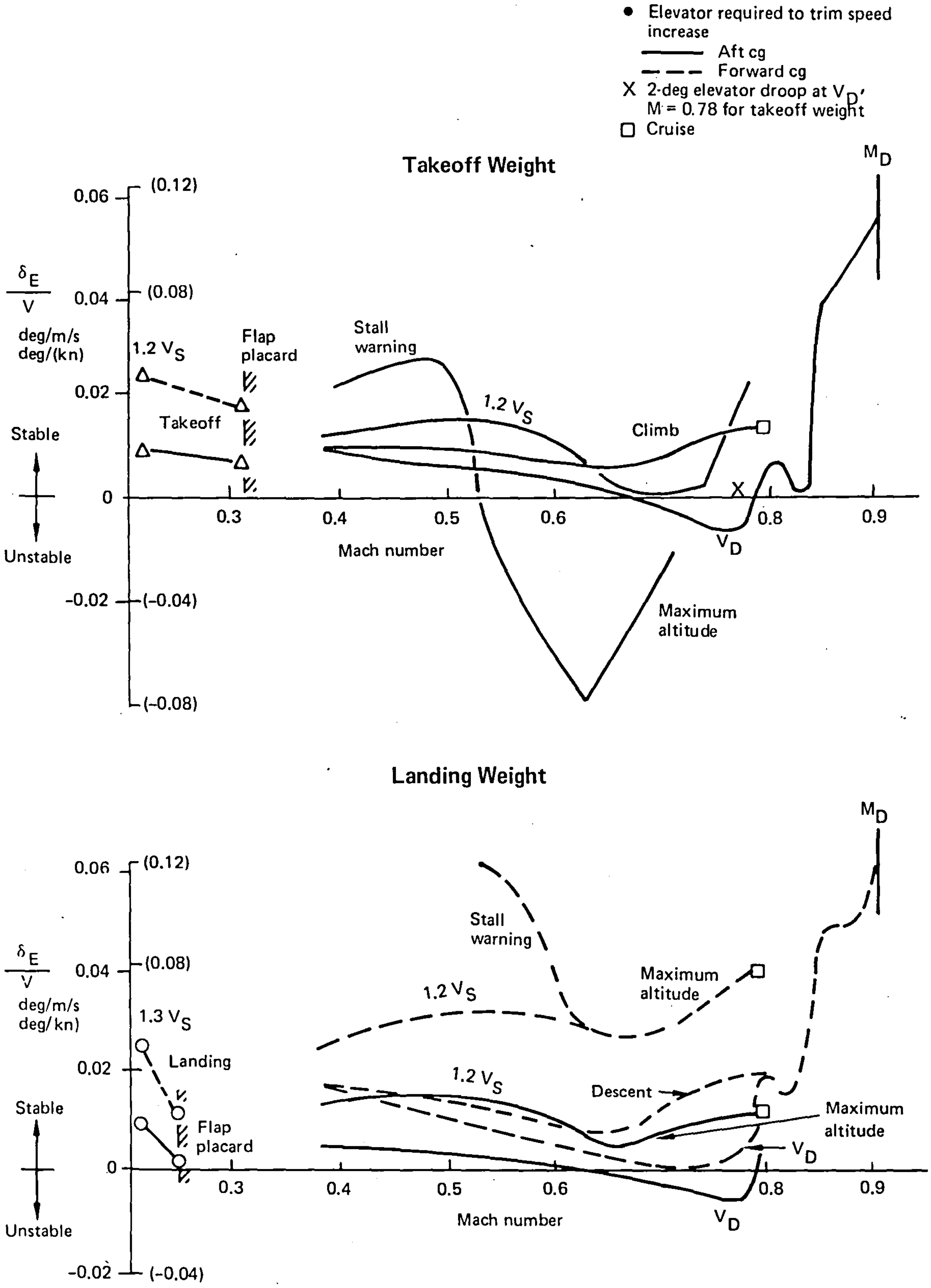


Figure 67. Elevator Angle per m/s (kn)

speed and near the maximum altitude flight envelope. The latter reflects high-speed pitchup at about Mach 0.63 and will be discussed in Subsection 7.1.3. Figure 67 also shows the effect of elevator droop at Mach 0.78 and illustrates that this unstable gradient can be stabilized; however, the high-altitude condition will be little affected by elevator droop. In any event, feel system design will ensure that stick force-speed gradients will meet a 4.45 N (1 lb) per 3.09 m/s (6 kn) criterion.

7.1.3 STABILITY

This section describes aircraft stability in terms of both static and dynamic characteristics. Longitudinal characteristics illustrate that with elevator droop, all stability criteria are met, but that longitudinal feel system augmentation will be needed to compensate for the high-speed pitchup. Unstable Dutch roll damping is shown, which illustrates the design requirements for a yaw damper.

Elevator angle per "g" for a constant speed maneuver is shown in Figure 68 which reflects stable short-period characteristics throughout the design flight envelope. Again the shading band separates forward and aft cg conditions. The minimum value occurs at Mach 0.65 for low altitude. Level 1 flying qualities criteria are met throughout the design flight envelope even though the -2 deg/g need only be met in the operational flight envelope. Figure 68 also illustrates a nearly 10:1 range of elevator gradient which will require a feel system design to provide acceptable range of stick force gradient. There are no plans on the IAAC project to design the feel system.

Longitudinal static stability characteristics are summarized in Figure 69 which illustrates the high-speed pitchup at Mach 0.63. No trim reversal is exhibited; however, the static margin does reduce to 3.5 and 0.5% MAC at dive and maximum flight altitude, respectively.

Rigid high-speed pitching moment characteristics are shown in Figure 70, which illustrates that the curves exhibit a reduced stability (pitchup) tendency at angle of attack between the maximum operational angle of attack and stall warning. As shown in Figure 71, maneuver margin is positive up to stall warning, and data for higher angles of attack (not shown in fig. 70) exhibit increased stability. Therefore, the maneuvering elevator gradient is stable although nonlinear in this region. Feel system design will have to reduce this nonlinearity to an acceptable level.

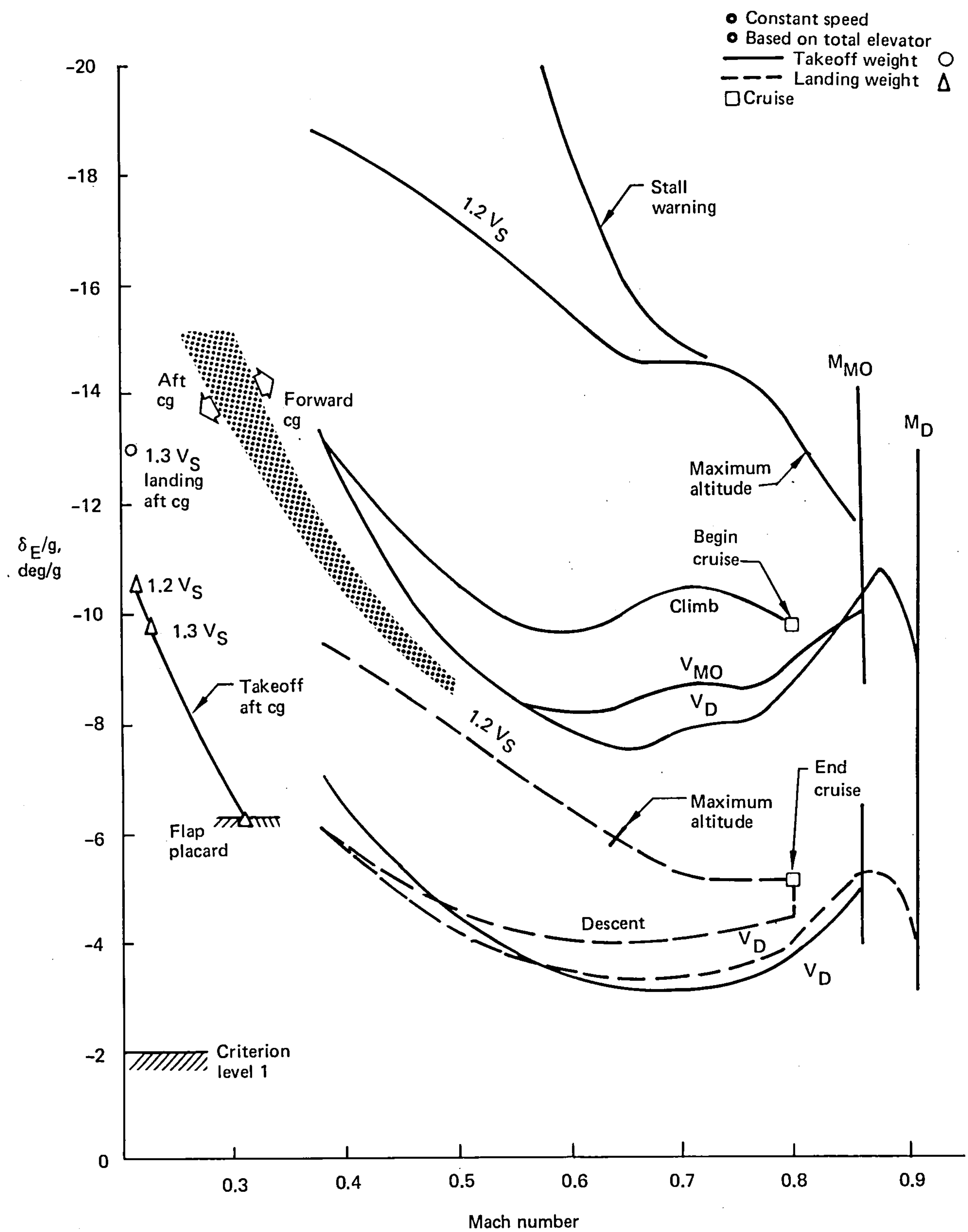


Figure 68. Elevator Angle per g

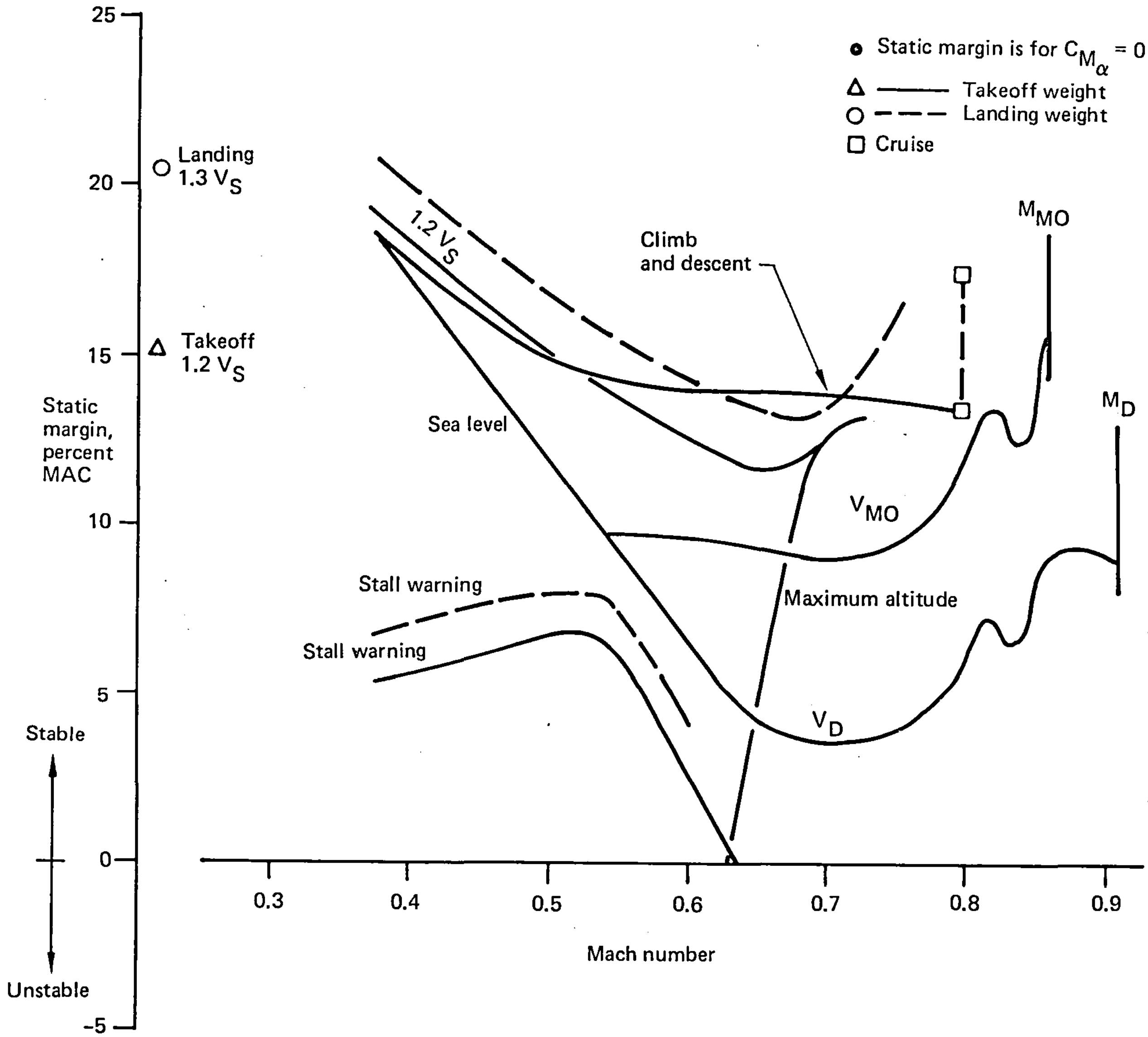


Figure 69. Static Margin at Aft cg

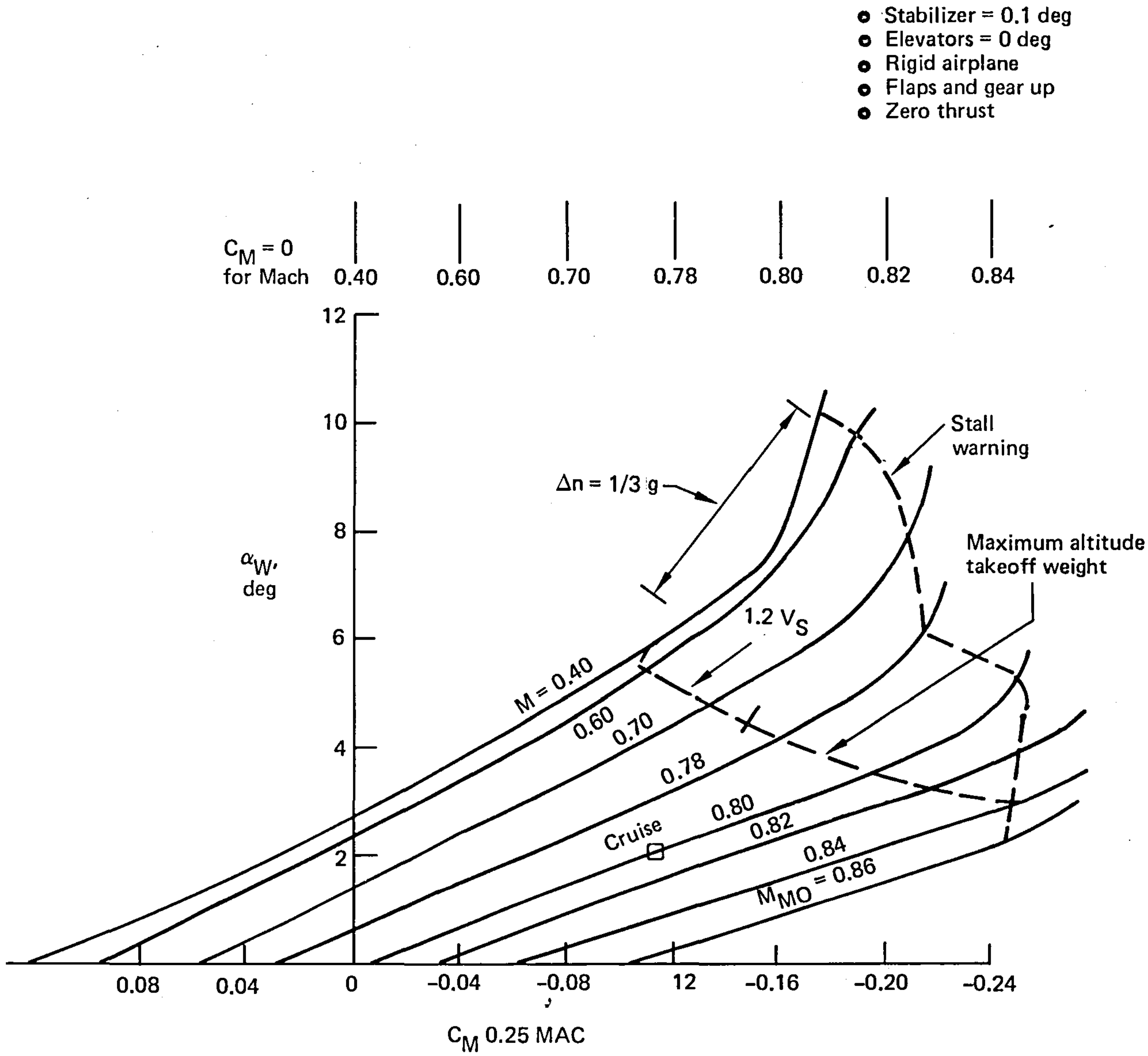


Figure 70. High-Speed Pitchup Characteristics

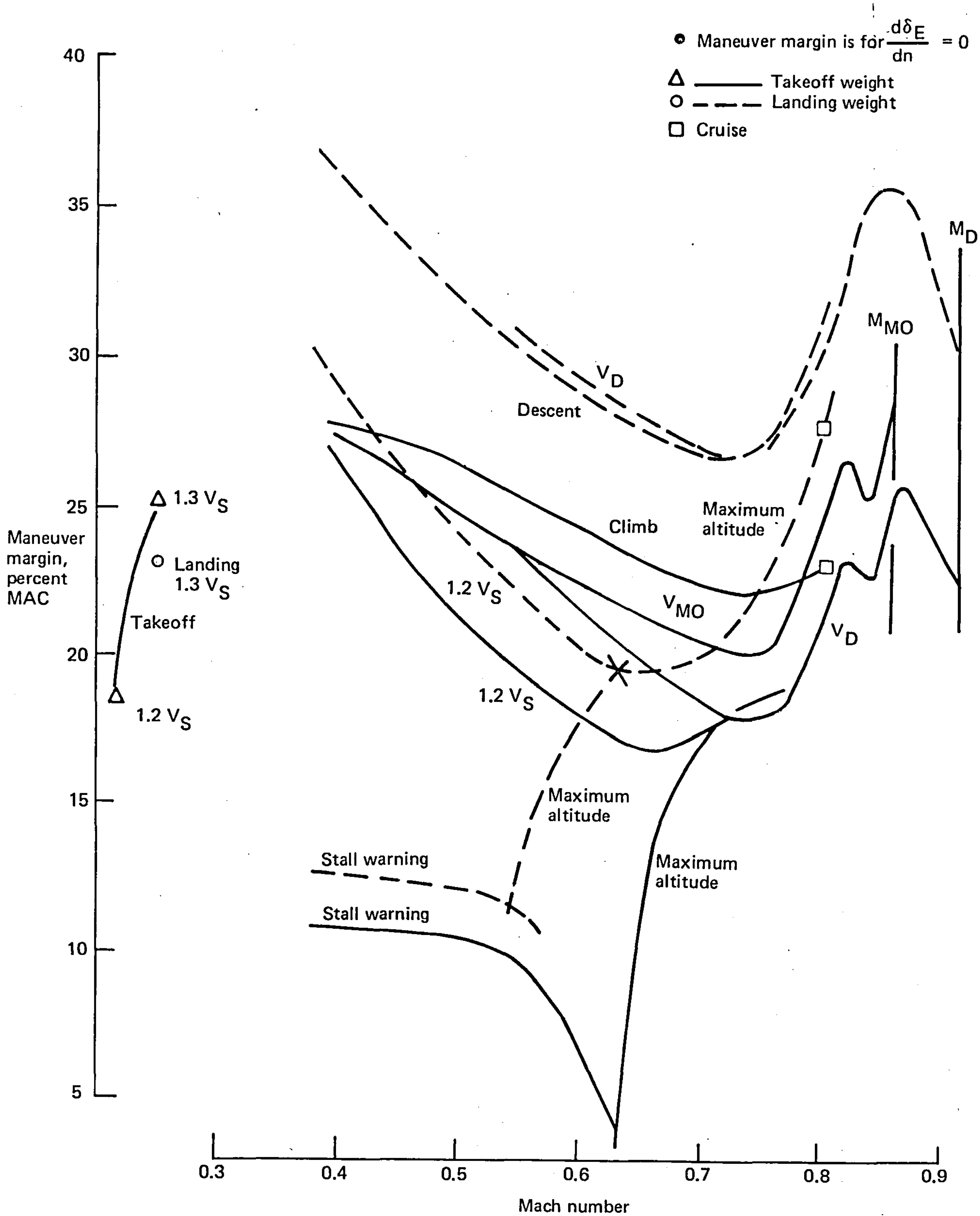


Figure 71. Maneuver Margin at Aft cg

Longitudinal, short-period stability criteria and flight characteristics are summarized in Figures 72 and 73 in terms of the natural frequency and load factor response. These characteristics reflect the critical maneuver margin characteristics shown in Figure 71 and show that most of the criteria are met or exceeded. The high-speed pitchup condition does not quite meet the Level 3 criterion; however, this criterion is treated as an objective for the unaugmented Baseline Configuration.

Figure 74, which shows phugoid stability characteristics, illustrates moderately unstable aperiodic roots at pitchup and Mach 0.78 and reflects the elevator angle per knot characteristics shown in Figure 68. The high-speed condition is characteristic of "Mach tuck" and will be corrected with elevator "droop" of 1 or 2 deg, depending on compromises with the feel and actuation systems. For example, 2 deg of elevator droop will stabilize the critical Mach 0.78, V_{MO} condition to oscillatory roots with 0.157 damping ratio. Again, the Mach 0.63 condition will be unaffected by elevator droop, but does meet the Level 3 stability criteria.

Lateral-directional static stability is summarized in Figure 75, which illustrates positive stability throughout the flight envelope for the critical heavy gross weight with an aft cg. While these characteristics ensure conventional control deflection for trim and maneuver, the level of stability to Mach 0.83 requires a yaw damper to increase Dutch roll damping at high altitude.

Figure 76 shows unaugmented Dutch roll damping characteristics and illustrates the design requirements for a yaw damper. The largest Dutch roll damping augmentation required is at end of cruise where the damping ratio must be increased from -0.008 to at least 0.08 and, preferably, 0.20 for Level 1 flying qualities. The unaugmented characteristics do not meet Level 3 criteria (minimum safe). Therefore, the yaw damper should be triply redundant, and flight altitude would be restricted to about 7620m (25 000 ft) after two failures. This yaw damper would provide good flying qualities but has not been designed. The additional damping required is about the same as that required for the Model 727 airplane, so there is no reason to believe that the yaw damper would not be a standard design concept.

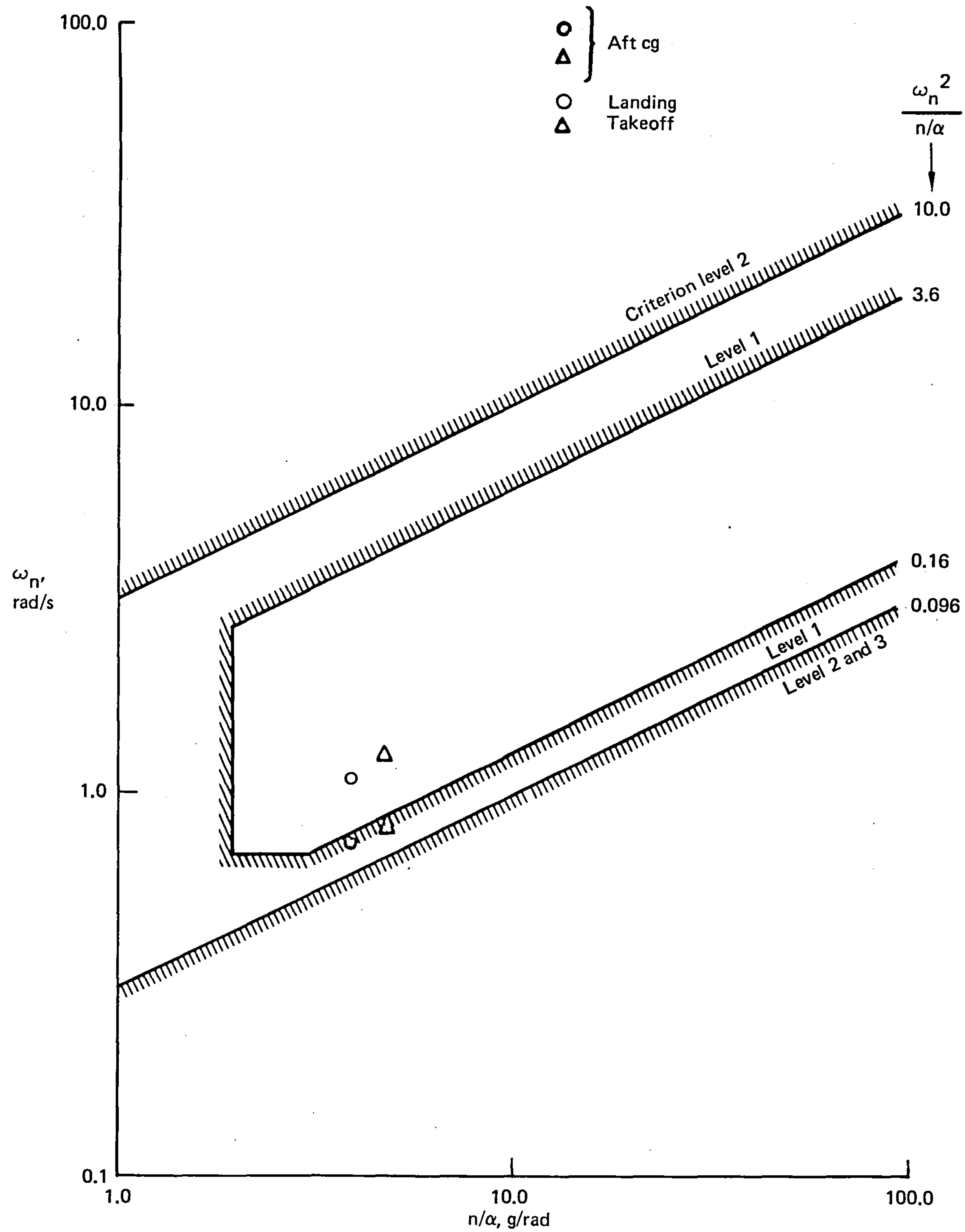


Figure 72. Terminal Short Period Stability

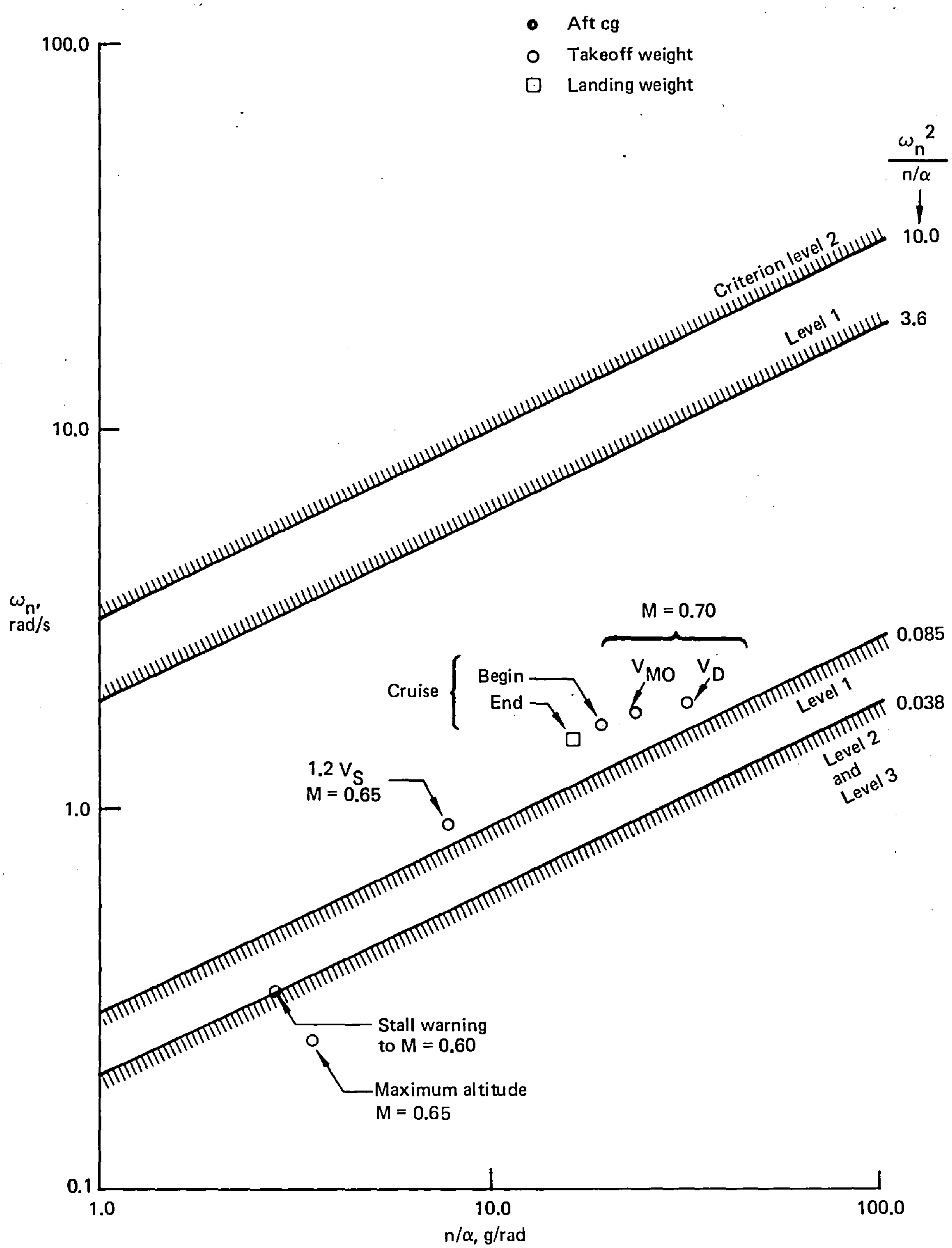


Figure 73. En Route Short Period Stability

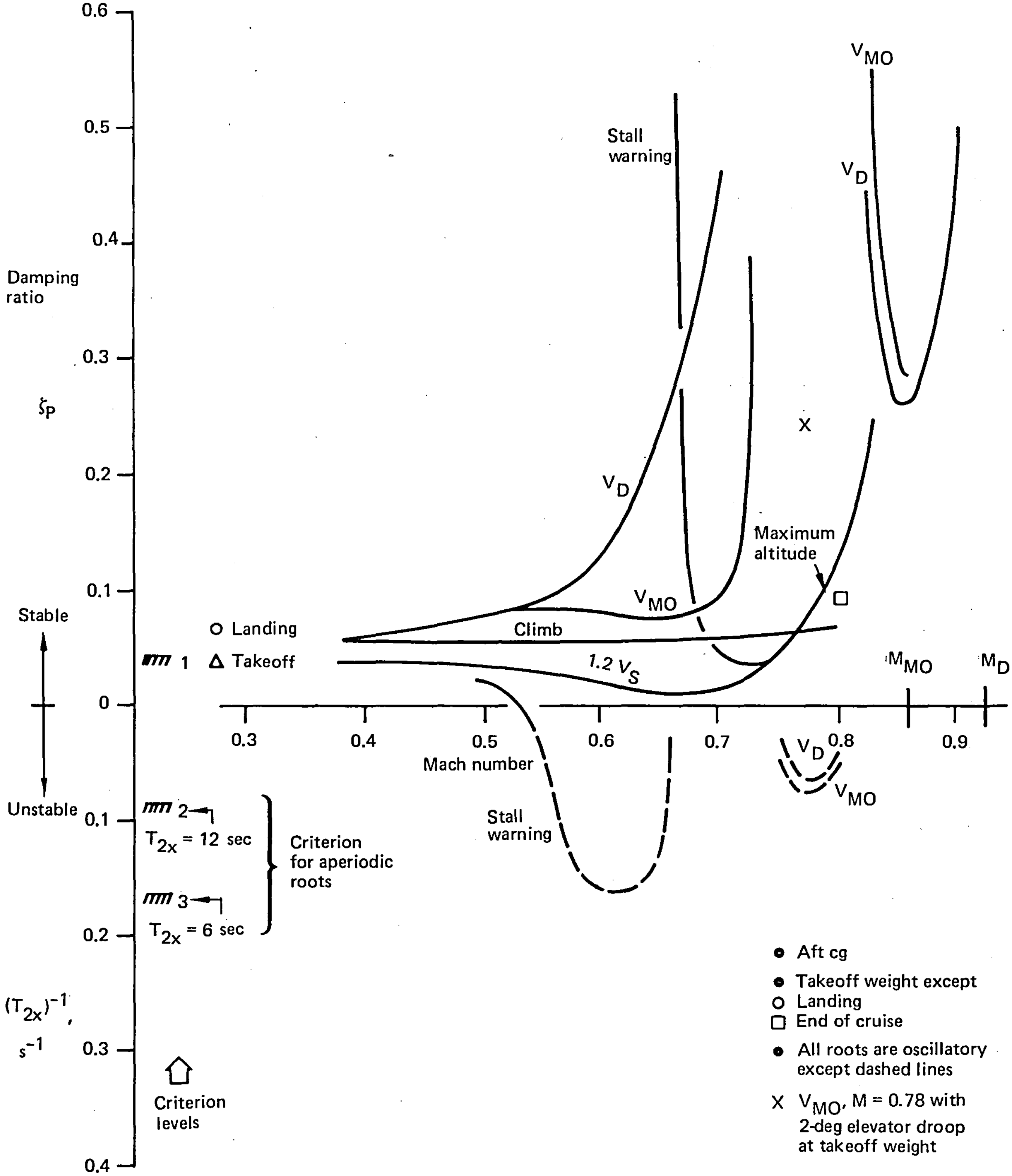


Figure 74. Phugoid Stability

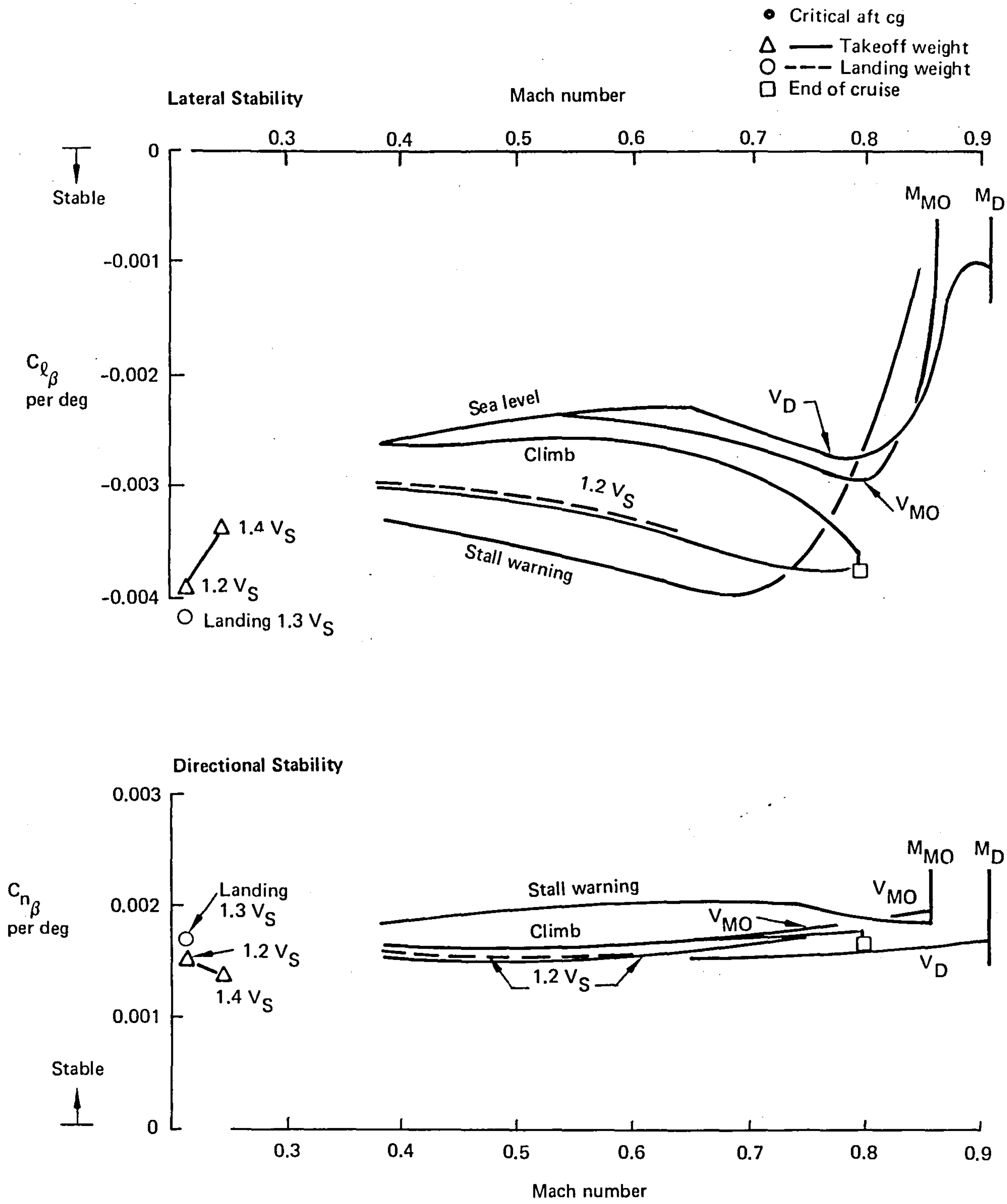


Figure 75. Lateral-Directional Static Stability

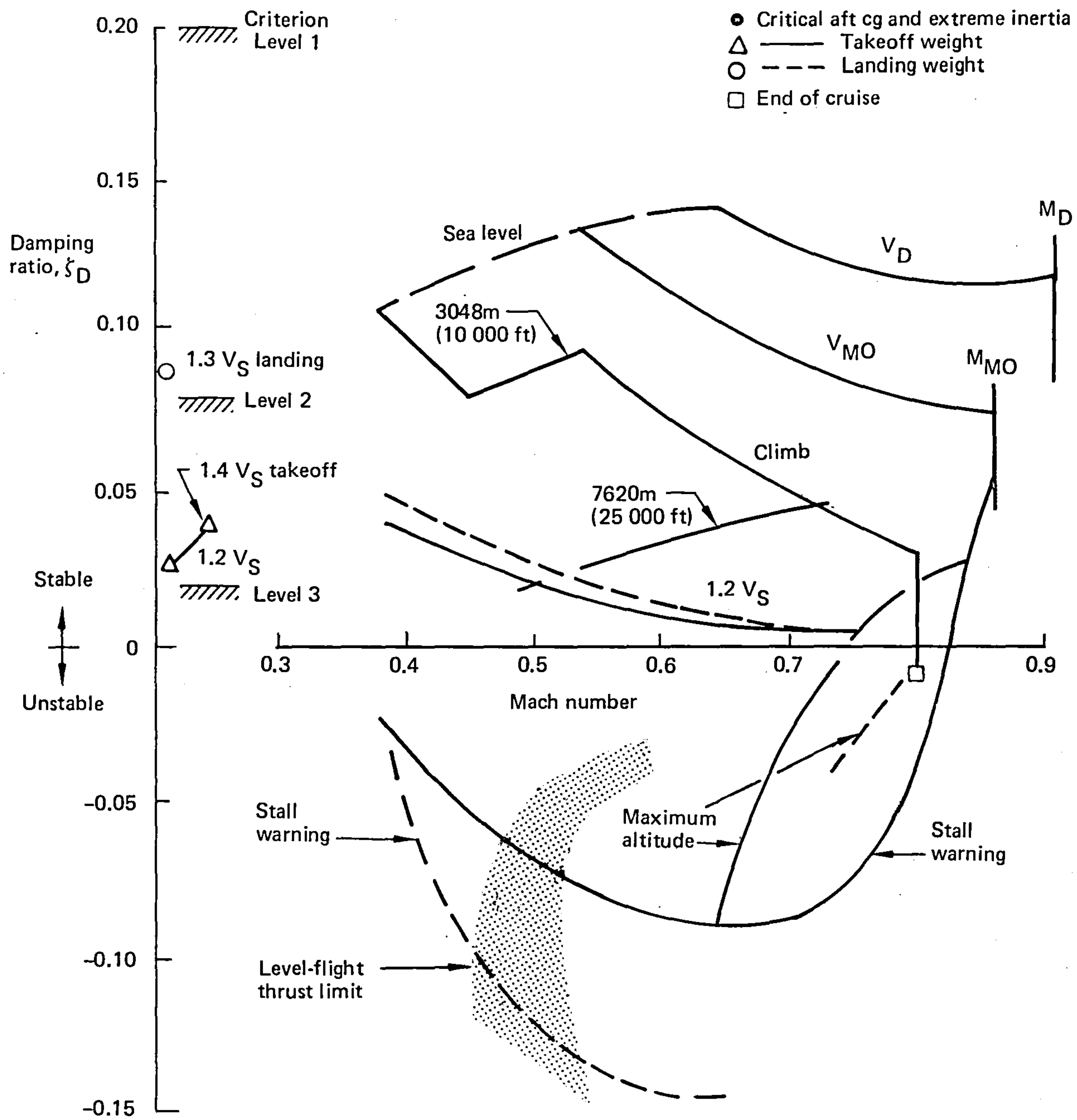


Figure 76. Dutch Roll Damping (Unaugmented)

Figures 77 and 78 show basic roll and spiral mode characteristics. These characteristics illustrate that Level 1 flying qualities are exhibited throughout the operational flight envelope and that lateral augmentation is not needed. However, the roll mode does deteriorate near stall warning at high Mach number and lightly couples with the spiral mode at this extremity of the design flight envelope.

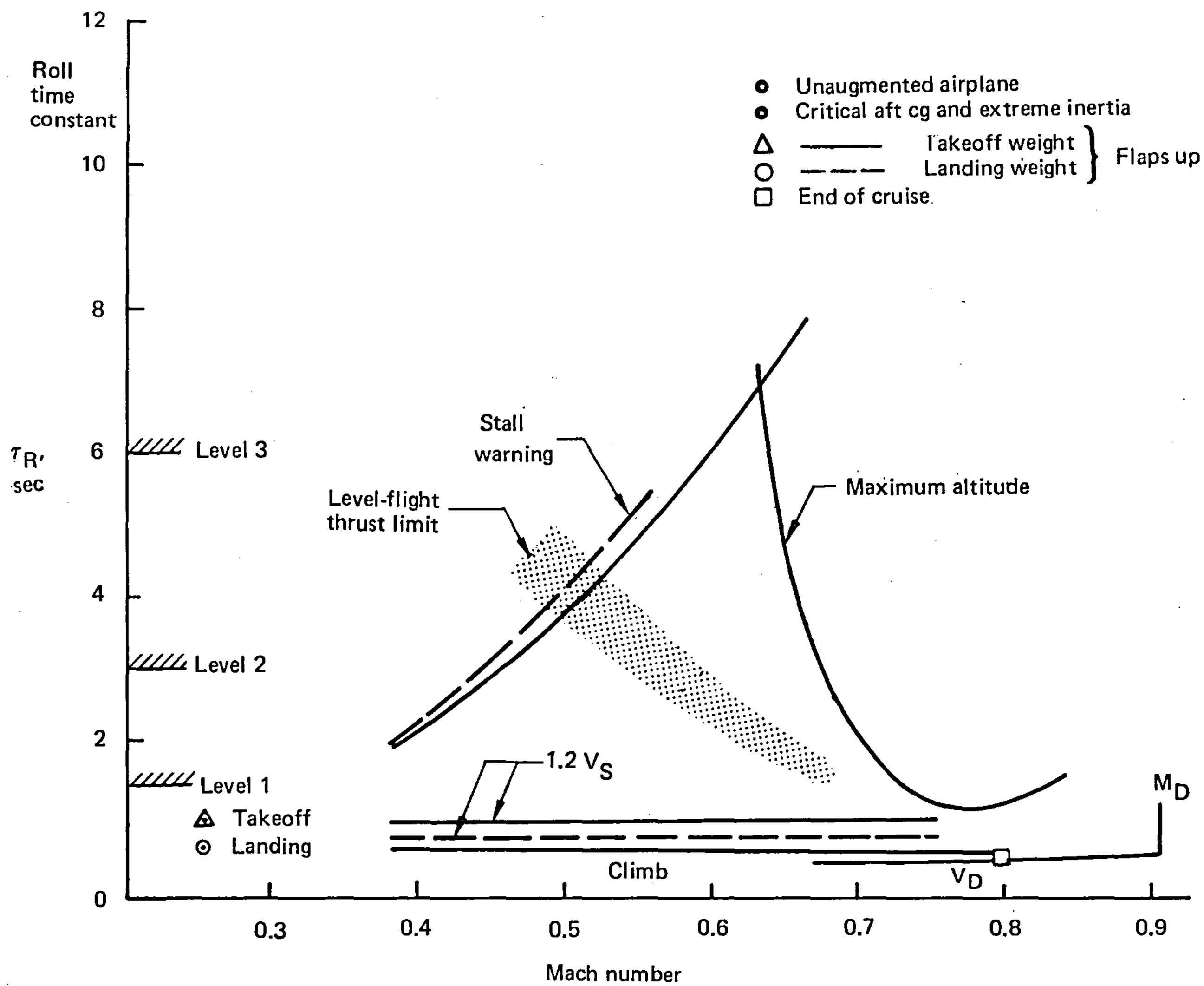


Figure 77. Roll Mode Time Constant

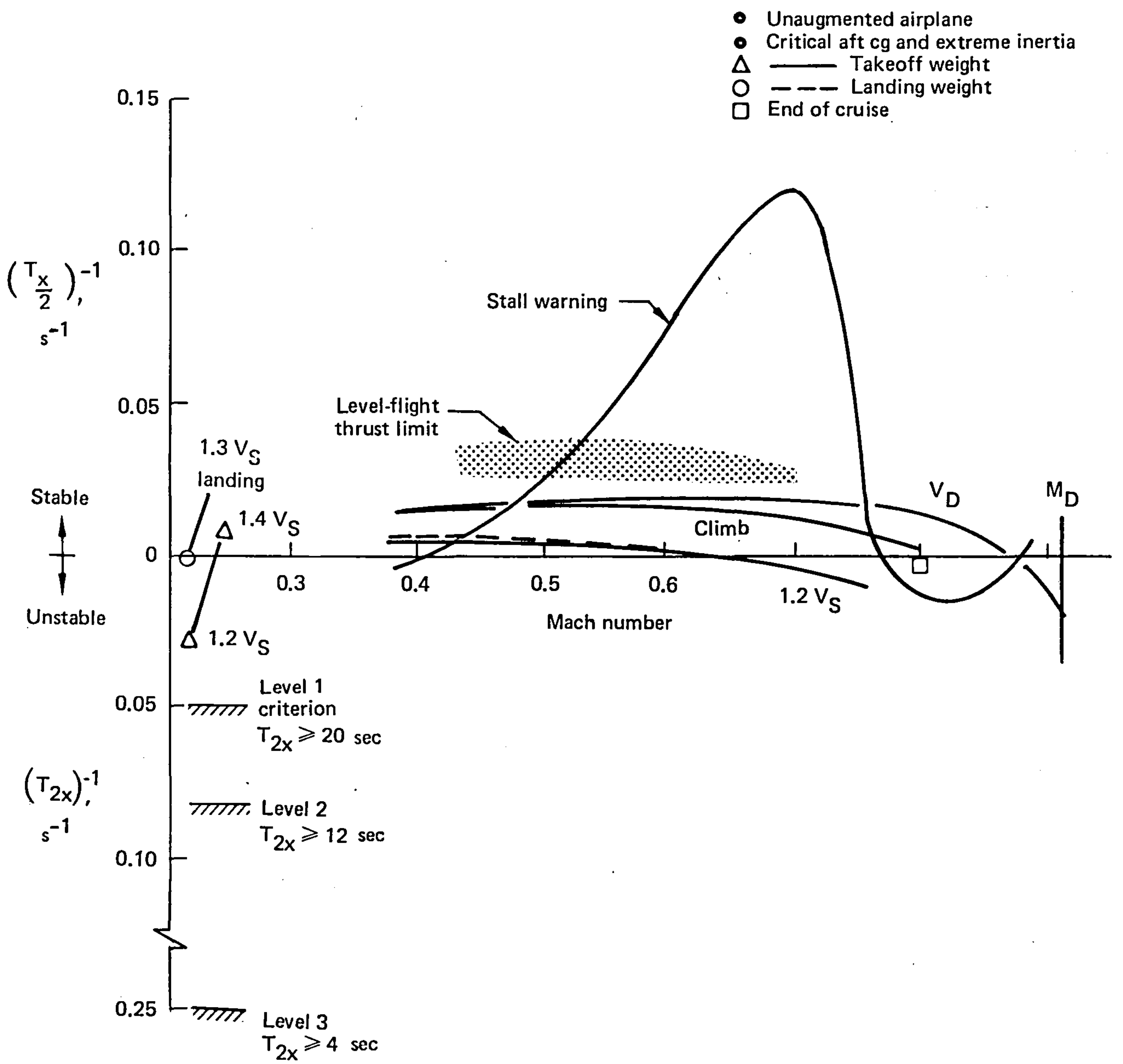


Figure 78. Spiral Mode

7.2 STRUCTURAL ANALYSIS

7.2 STRUCTURAL ANALYSIS

This section presents the structural analysis results for the Baseline Configuration. These results establish a structural design base for use in evaluating alternate configurations which employ active controls to reduce structural material requirements.

The analysis is based on design criteria and methods established by Boeing which comply with the requirements of Federal Aviation Regulations (FAR) Part 25 (ref 12).

Wind tunnel test data used in the analysis were obtained and analyzed during the preliminary design phase of the Boeing New Airplane program. These data were also used to analyze the Initial ACT Configuration.

7.2.1 WING

Results presented in this section include the ultimate load, flutter, fatigue, and structural sizing data required for design of the primary wing structure. These results provide a base for defining the weight of the wing box (presented in subsec 7.4).

7.2.1.1 Maneuver and Gust Formula Loads

Ultimate wing beam bending moment envelopes for critical maneuver and gust formula conditions are compared in Figure 79. The analysis conditions are summarized in Figure 80 and Table 6.

Wing box strength requirements are almost entirely determined by gust formula conditions; however, maneuver conditions are almost as critical.

Maneuver and gust formula loads, wing sizing, wing stiffness, and jig twist were computed in an iterative cycle using the ORACLE system, a Boeing-developed program for the preliminary design of high-aspect ratio wings. Final maneuver and gust formula loads were computed using final stiffness for the strength-design wing. Jig twist was optimized using a required wing airload distribution for cruise flight performance. Wing aerodynamics were based on lifting line theory adjusted to match wind tunnel pressure data, and wing structure was modeled using beam theory. Gust

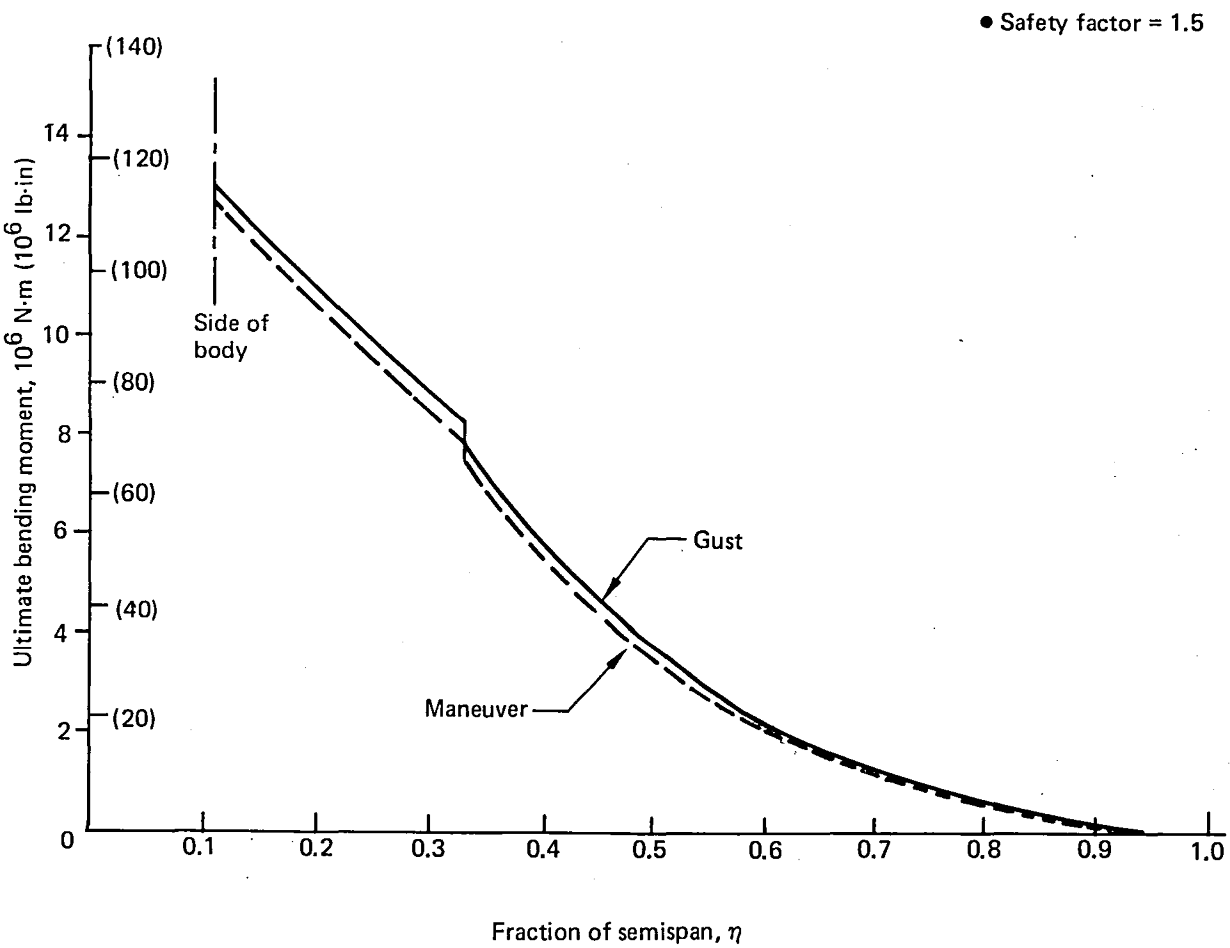


Figure 79. Design Wing Bending Moment Envelope

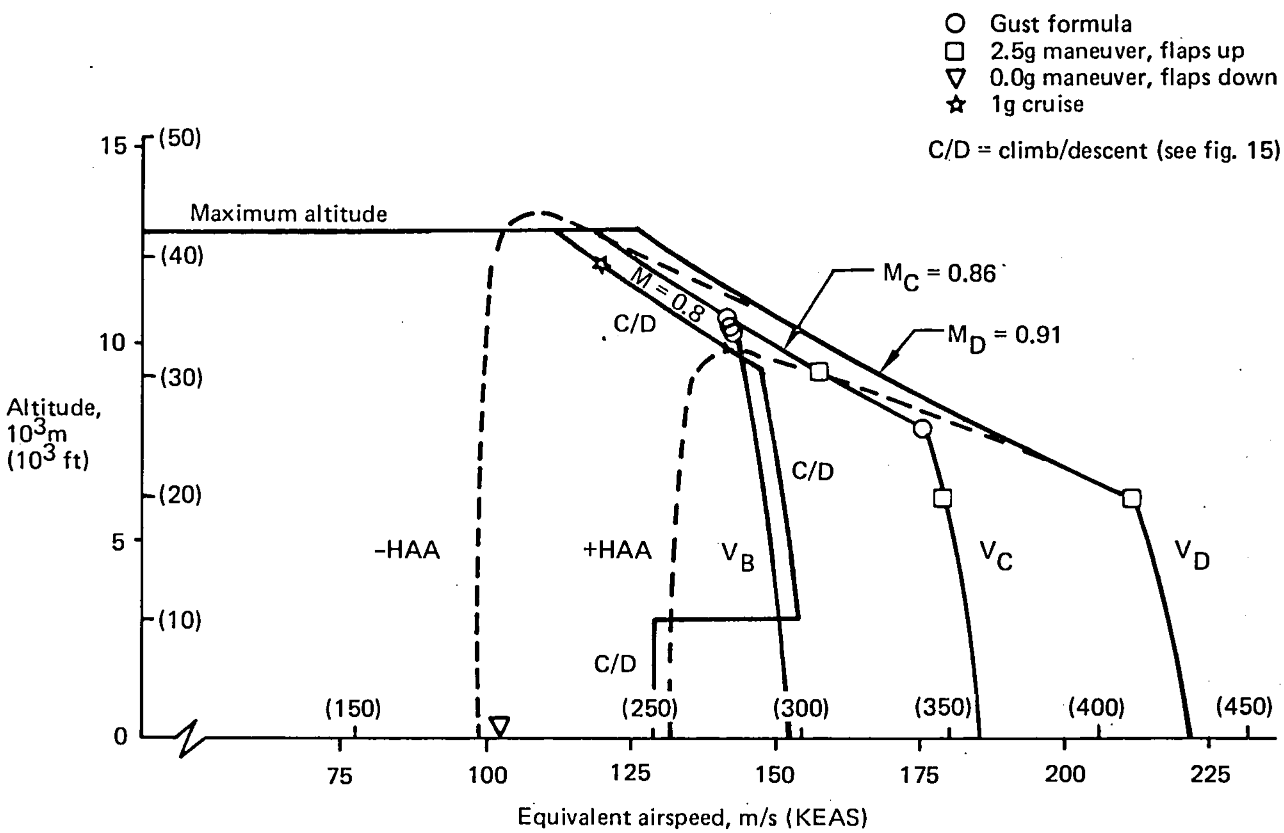


Figure 80. Design Envelope

Table 6. Selected Design Conditions and Cruise Condition

Condition number	Type	Altitude, m (ft)	V_e, m/s (KEAS)	M	Gross weight, kg (lb)	Fuel, kg (lb)	n_z	cg, MAC	U_{de}, m/s (fps)	Tail load, N (lb)
1	Gust formula	10 668 (35 000)	142 (276)	0.86	120 202 (265 000)	30 590 (67 440)	2.646	0.09	15.85 (52)	127 575 (28 680)
2	Gust formula	10 668 (35 000)	142 (276)	0.86	120 202 (265 000)	16 497 (36 370)	2.610	0.09	15.85 (52)	120 013 (26 980)
3	Gust formula	10 485 (34 400)	142 (276)	0.85	120 202 (265 000)	16 552 (36 490)	2.635	0.09	16.03 (52.6)	130 199 (29 270)
4	Gust formula	10 668 (35 000)	142 (276)	0.86	120 202 (265 000)	16 497 (36 370)	2.610	0.09	15.85 (52)	16 369 (3680)
5	Gust formula	10 485 (34 400)	142 (276)	0.85	120 202 (265 000)	16 552 (36 490)	2.635	0.09	16.03 (52.6)	22 419 (5040)
6	Gust formula	10 302 (33 800)	143 (277)	0.84	120 202 (265 000)	30 649 (67 570)	2.677	0.09	16.18 (53.1)	135 137 (30 380)
7	Gust formula	7650 (25 100)	176 (342)	0.85	121 109 (267 000)	31 457 (69 350)	-0.658	0.39	-13.93 (-45.7)	-281 128 (-63 200)
8	+Maneuver	9235 (30 300)	157 (306)	0.86	120 656 (266 000)	16 618 (36 636)	2.5	0.09	–	-142 761 (-32 094)
9	+Maneuver	6035 (19 800)	210 (409)	0.91	121 563 (268 000)	17 912 (39 490)	2.5	0.09	–	-248 558 (-55 878)
10	+Maneuver	6096 (20 000)	179 (347)	0.78	121 563 (268 000)	17 912 (39 490)	2.5	0.09	–	-27 419 (-6164)
11	+Maneuver	6096 (20 000)	210 (408)	0.91	121 563 (268 000)	17 894 (39 450)	2.5	0.09	–	-248 113 (-55 778)
12	-Maneuver	12 863 (42 200)	118 (229)	0.85	119 295 (263 000)	15 826 (34 890)	-1.0	0.09	–	-23 126 (-5199)
13	-Maneuver	12 863 (42 200)	118 (229)	0.85	119 295 (263 000)	29 865 (65 840)	-1.0	0.09	–	-21 485 (-4830)
14	Zero g	305 (1000)	102 (199)	(20° flaps down)	122 470 (270 000)	18 824 (41 500)	0	0.09	–	-132 771 (-29 848)
15	Taxi	0	–	–	122 924 (271 000)	33 316 (73 450)	2.0	0.18	–	–
16	1g cruise	11 887 (39 000)	120 (233.1)	0.80	110 608 (243 850)	15 059 (33 200)	1.0	0.30	–	–

loads were computed using the FAR gust formula and include dynamic magnification factors based on discrete gust and continuous turbulence dynamic load results.

Critical wing design conditions were determined from a survey of selected maneuver, gust, and ground conditions within the boundaries of the design envelope. The airplane cg locations for wing design purposes were conservatively selected as the extreme positions shown in Figure 19, Section 5.3, plus a 1% MAC tolerance to account for fuel shift with airplane attitude or other unknowns.

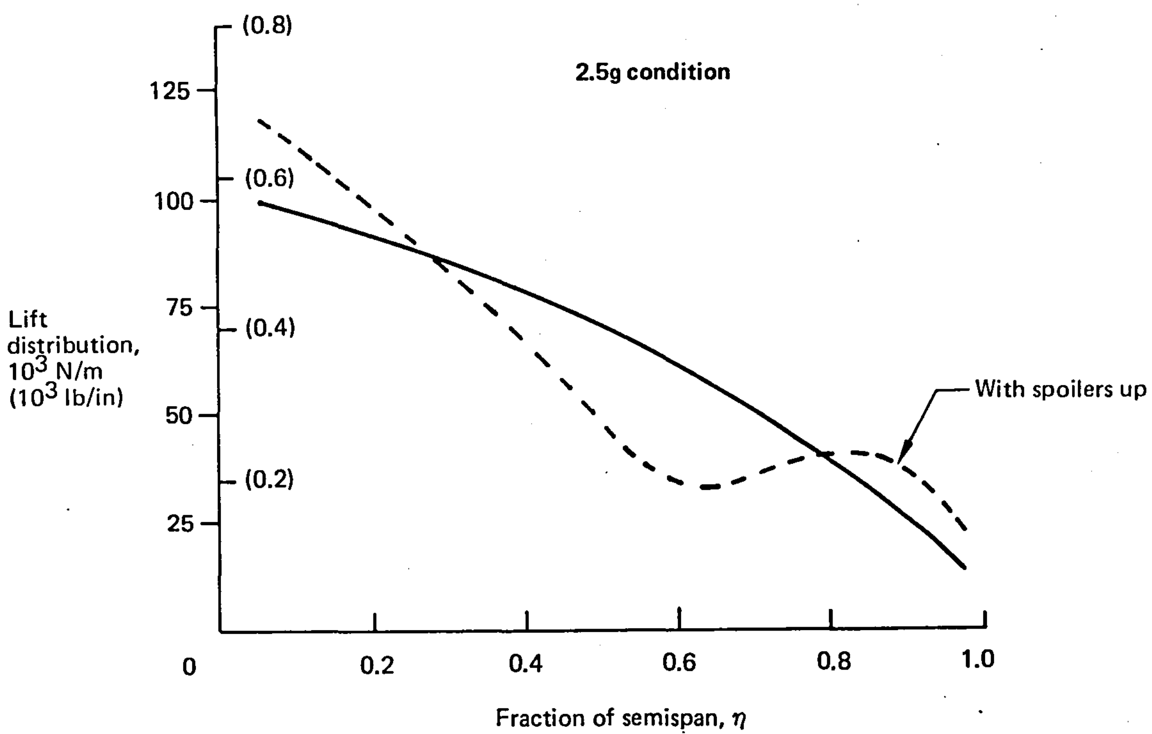
A description of selected conditions is presented in Table 6. The tabulation includes the positive and negative gust and maneuver conditions which produce maximum wing loads, a 2.0-g taxi condition which is critical for the inboard rear spar, and the 1.0-g cruise condition which was used to define the desired cruise span load for the optimization of wing jig twist. The most critical conditions of each type are identified in Figure 80.

Wing span airload distributions for two typical maneuver and gust conditions are shown in Figure 81. The effect of spoilers, used as speed brakes, is significant in redistributing the span load and results in critical design loads for a wing region between 60 and 90% of the semispan.

A diagram of the wing used in the ORACLE analysis is shown in Figure 82. The wing was divided into 12 streamwise aerodynamic panels, conveniently grouped to provide a good representation of regions near the control surfaces. The structural analysis of the wing box was performed at the indicated analysis stations for sections perpendicular to the load reference axis. The analysis stations are located at the center of the aerodynamic panels.

7.2.1.2 Dynamic Gust Analysis

Dynamic loads due to vertical gusts were computed by a design envelope power spectral density procedure. The analysis was done for cruise speed, V_C, and at 7833m (25 700 ft) altitude. Choice of this condition was based on analytical experience from the Boeing New Airplane program. Figure 83 shows maximum wing bending moments from the dynamic loads analysis, compared with design wing bending



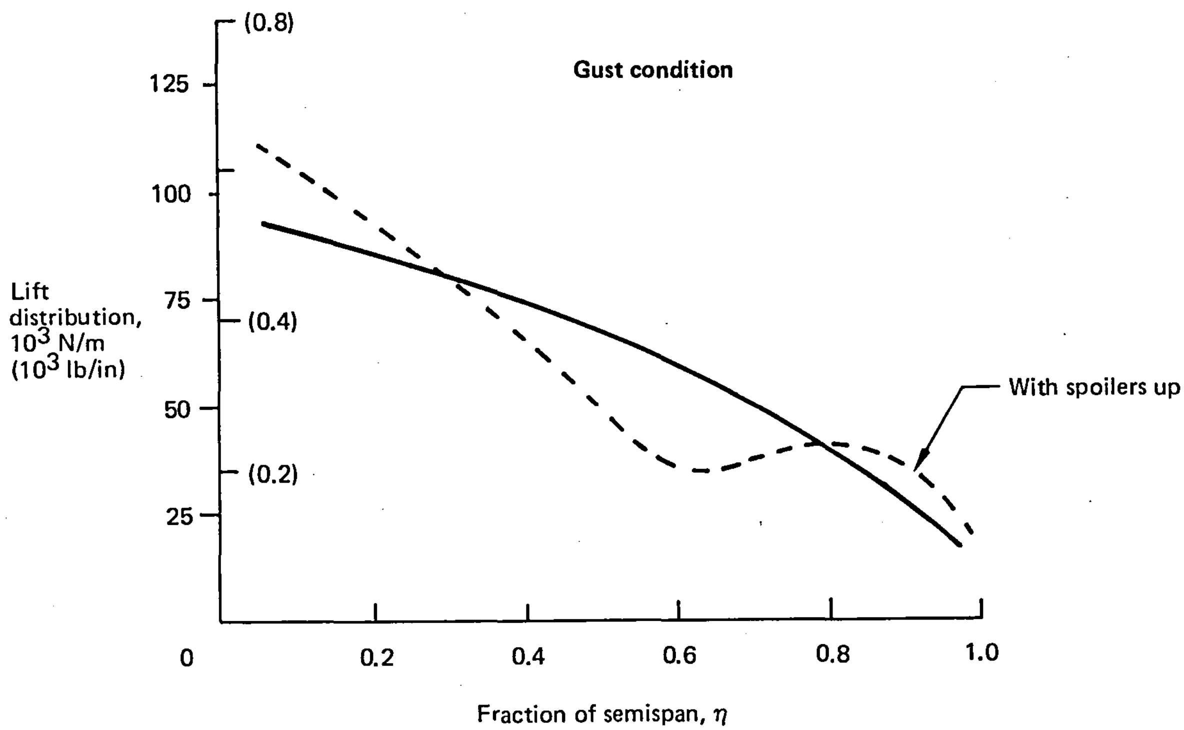


Figure 81. Typical Span Load Distributions

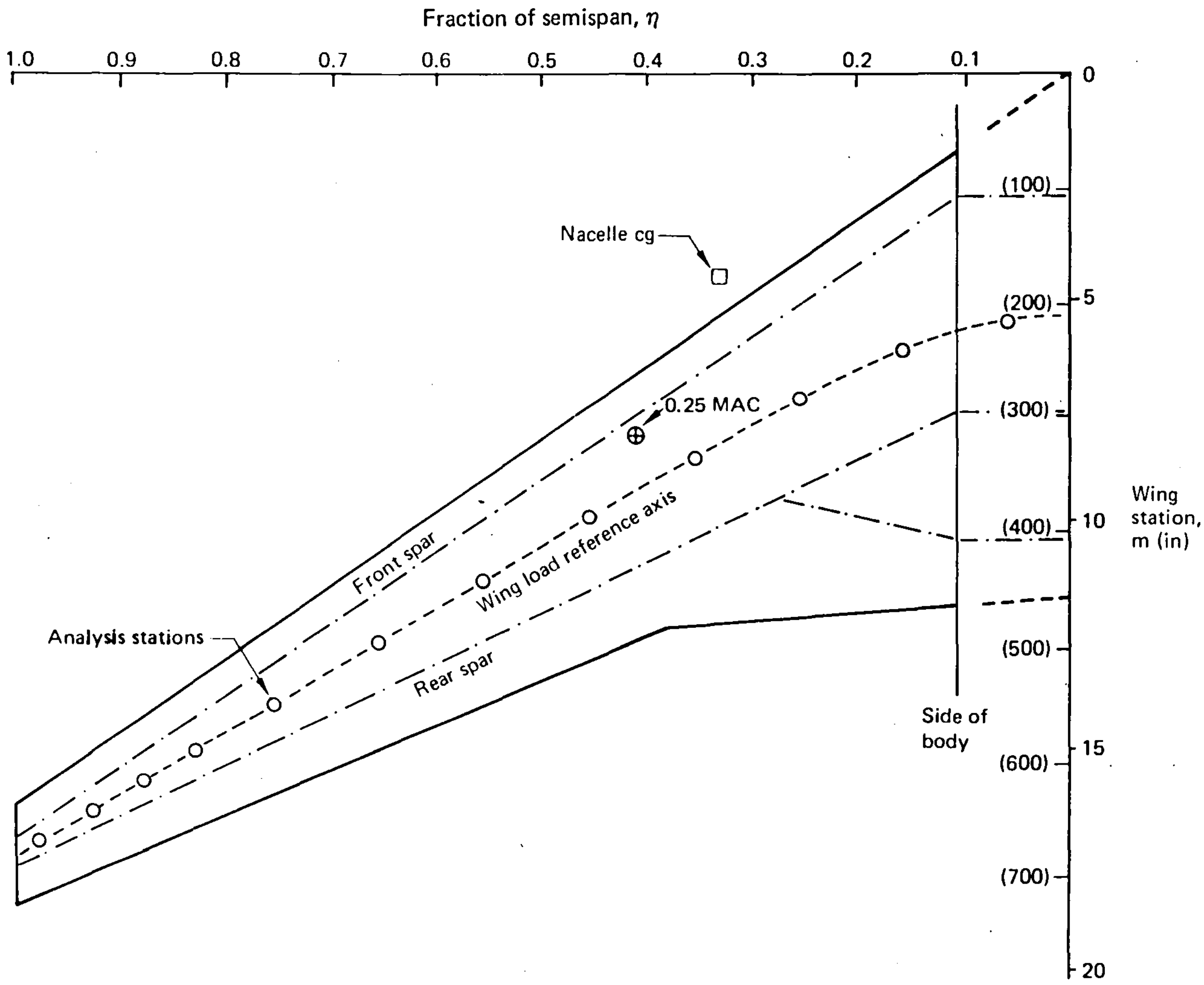


Figure 82. Wing Diagram for Structural Analysis

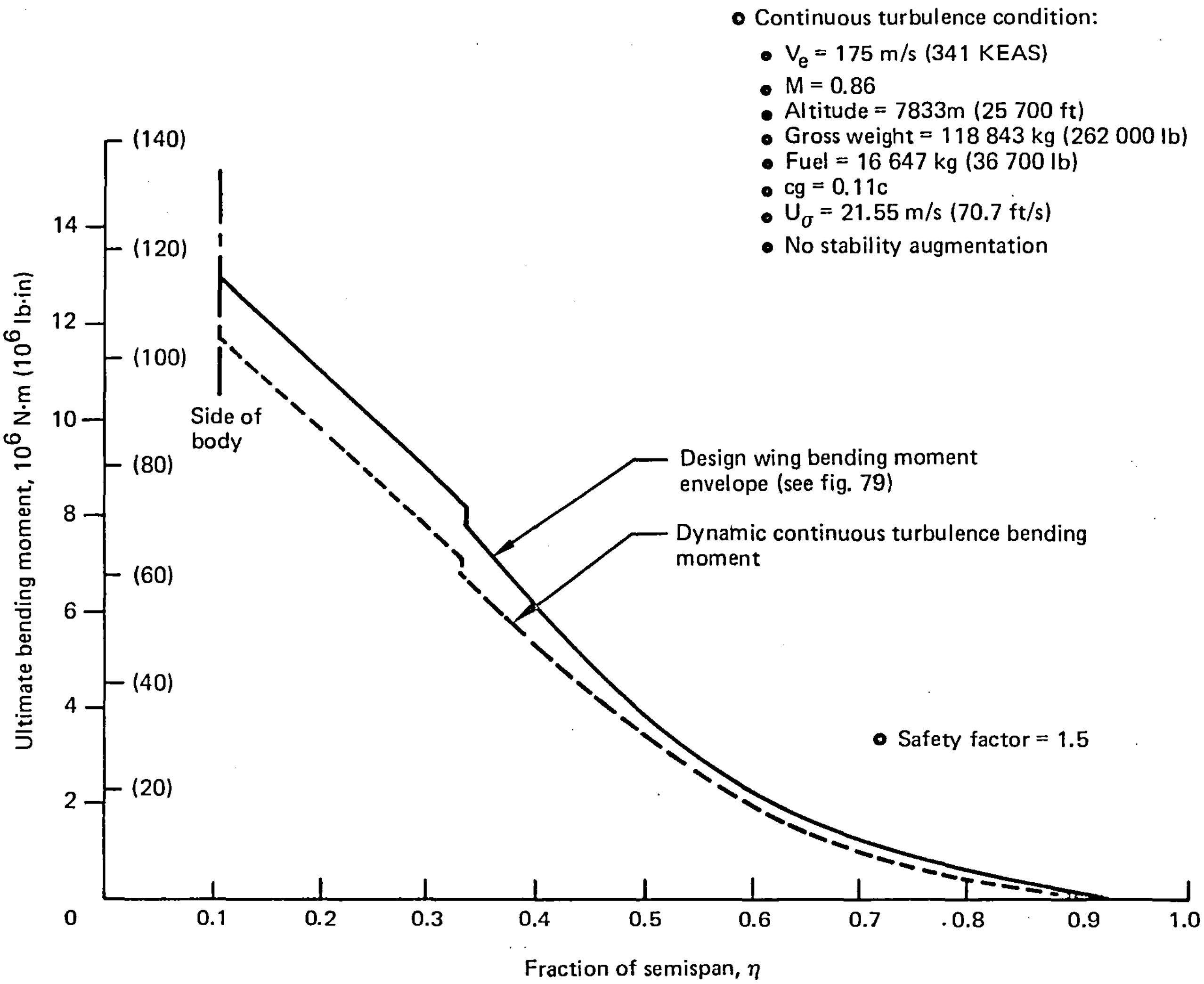


Figure 83. Wing Bending Moment, Vertical Gust Analysis

moments from the maneuver and gust formula analyses. The dynamic loads are not critical for structural design, as indicated in this figure.

The airplane structure was idealized as a system of elastic beams with lumped masses. Vibration modes were calculated using stiffness data for the strength-design wing. Fifteen, coupled, free-free modes were calculated and used as generalized coordinates based on the uncoupled branch modes shown in Table 7. The uncoupled modes were selected to best model the significant flexible responses of the structure to vertical gust excitation and control surface deflection. Higher frequency mode shapes were

Table 7. Branch Modes Used to Calculate Free-Free Modes for Dynamic Gust Analysis

Branch	Modal description
Airplane	Rigid airplane plunge Rigid airplane pitch
Forebody	First vertical bending Second vertical bending
Aft body (rigid empennage)	First vertical bending Second vertical bending Third vertical bending
Vertical tail (rigid horizontal tail)	First fore/aft bending (in plane) Second fore/aft bending (in plane) Third fore/aft bending (in plane)
Horizontal tail	First vertical bending Second vertical bending First torsion Rigid stabilizer pitch
Wing (rigid nacelle strut)	First vertical bending Second vertical bending Third vertical bending Fourth vertical bending Fifth vertical bending First torsion Second torsion Third torsion
Nacelle	Side bending Vertical bending Roll/side bending

retained to adequately model the aeroelastic characteristics of the wing and stabilizer and to provide satisfactory short-period response characteristics. Structural damping was conservatively estimated to be 0.02g based on previous experience with configurations of similar construction.

Aerodynamic loading was calculated using strip theory. Wing aerodynamic section data were derived from wind tunnel model pressure data, and calculated airplane stability derivatives were adjusted to match wind tunnel force balance data. The Von Karman spectrum of atmospheric turbulence was used with a design gust intensity recommended by the Aerospace Industries Association. Lift lag effects were included using Wagner and Kussner functions for a wing of infinite aspect ratio. Gradual gust penetration effects were not included.

A bending moment gust transfer function and an associated bending moment output spectrum for a typical wing station at 45% semispan are shown in Figures 84 and 85 for reference. The output spectrum shows that most of the gust load response occurs below 2 Hz. This is the expected result and is due to the attenuation of atmospheric gust velocities at higher frequency.

7.2.1.3 Flutter Analysis

A detailed flutter analysis was performed for the Baseline Configuration to determine the structural requirements for flutter stability. The results show that flutter requirements are not met by the strength-design wing of the Baseline Configuration. A wing stiffening scheme was developed to increase the wing flutter speed to provide the required flutter stability. The structural weight increment required is 404 kg (890 lb).

Conventional V-g flutter solutions were obtained using flutter equations referred to branch mode coordinates and using doublet lattice unsteady aerodynamics with corrections based on steady-state wind tunnel pressure data. Flutter boundaries were established by applying compressibility corrections to calculated flutter speeds for incompressible flow using correction factors based on measured values of wing lift curve slope.

The sensitivity of flutter speed to variations in wing fuel load, nacelle strut flexibility, and aft body vertical bending frequency were evaluated. Both symmetric and antisymmetric cases were analyzed.

Flutter Criteria and Results—The flutter speed boundaries for the stiffness-design wing are shown in Figure 86 and the associated incompressible V-g curves are shown in Figures 87 and 88. Results for zero wing fuel, 80% wing fuel and full wing fuel, are shown. The flutter speeds in Figure 86 are for neutral stability with 0.0g and 0.03g structural damping. Criteria established by Boeing require that flutter speed with zero structural damping must not be less than V_D and flutter speed with 0.03g

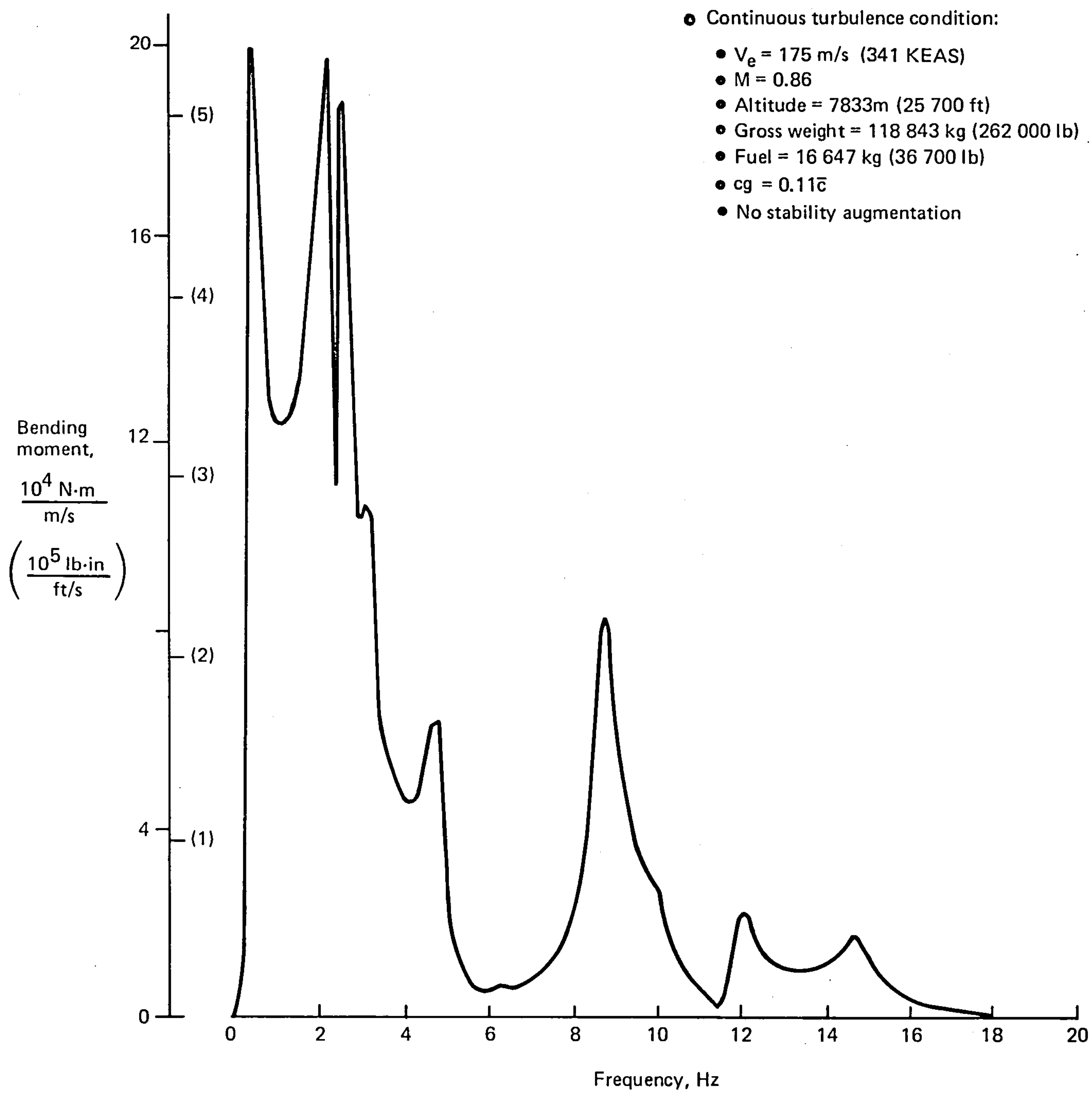


Figure 84. Wing Bending Moment Transfer Function at 45% Semispan

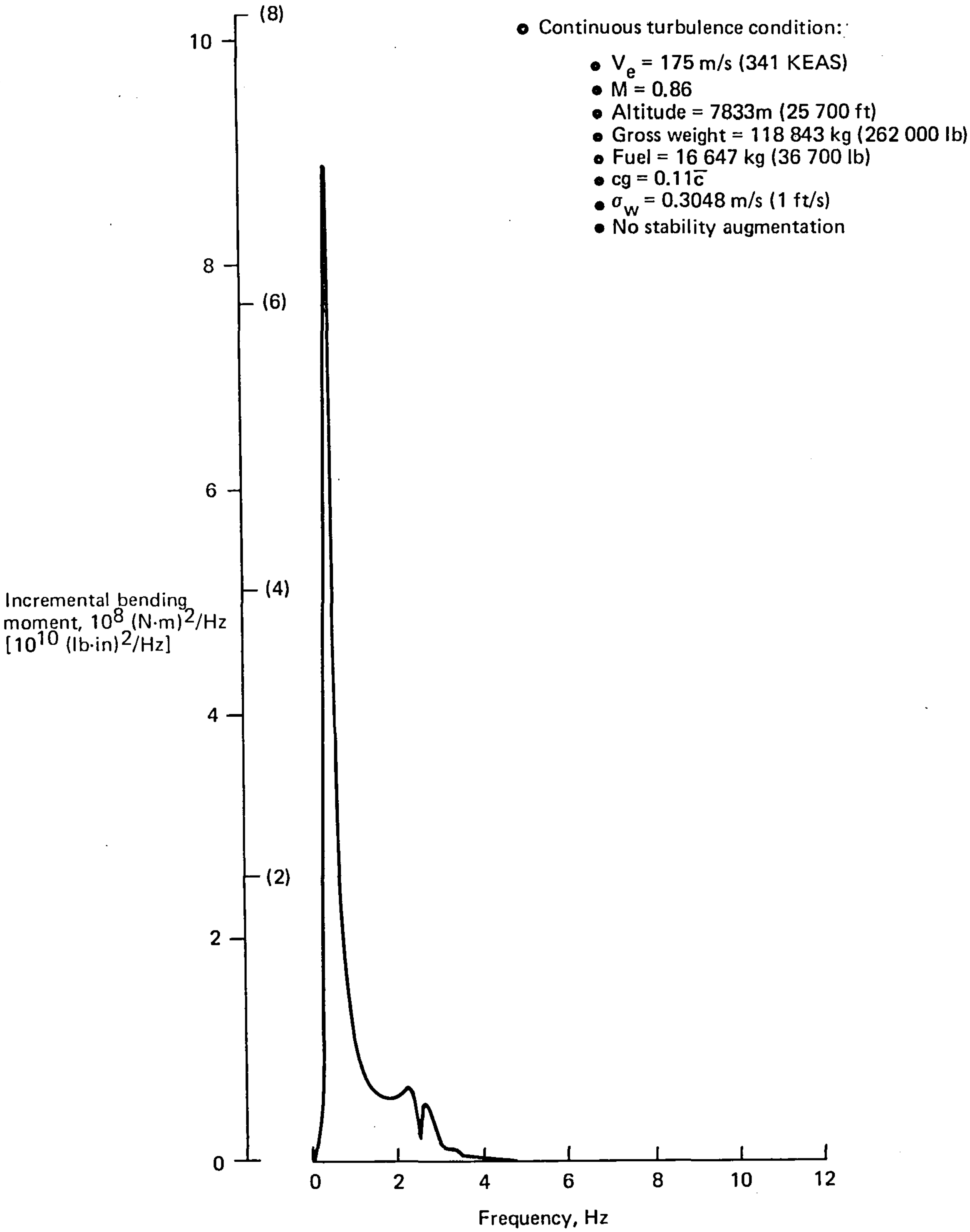


Figure 85. Output Spectrum for Wing Bending Moment at 45% Semispan

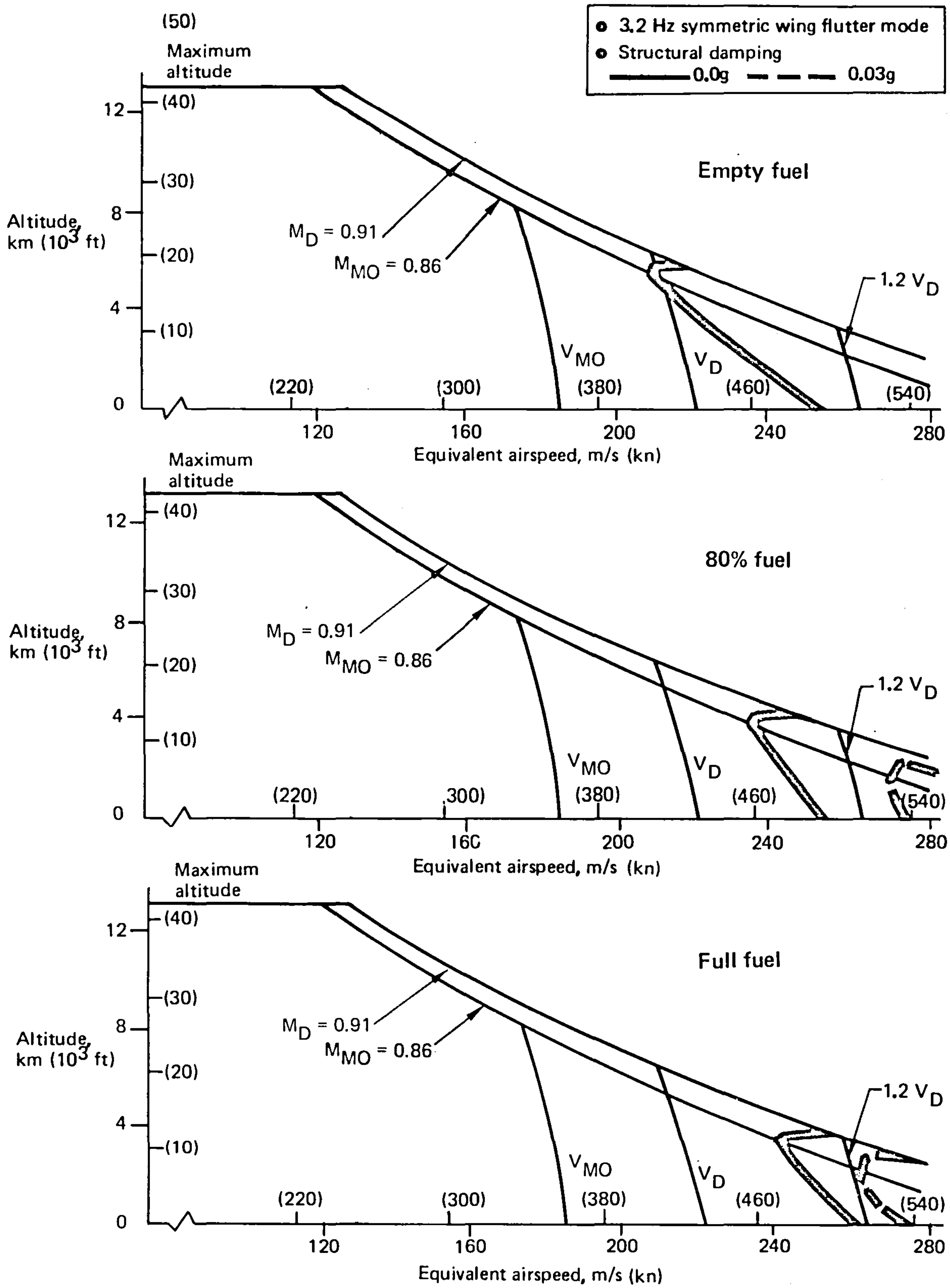


Figure 86. Flutter Boundary for Stiffness-Design Wing

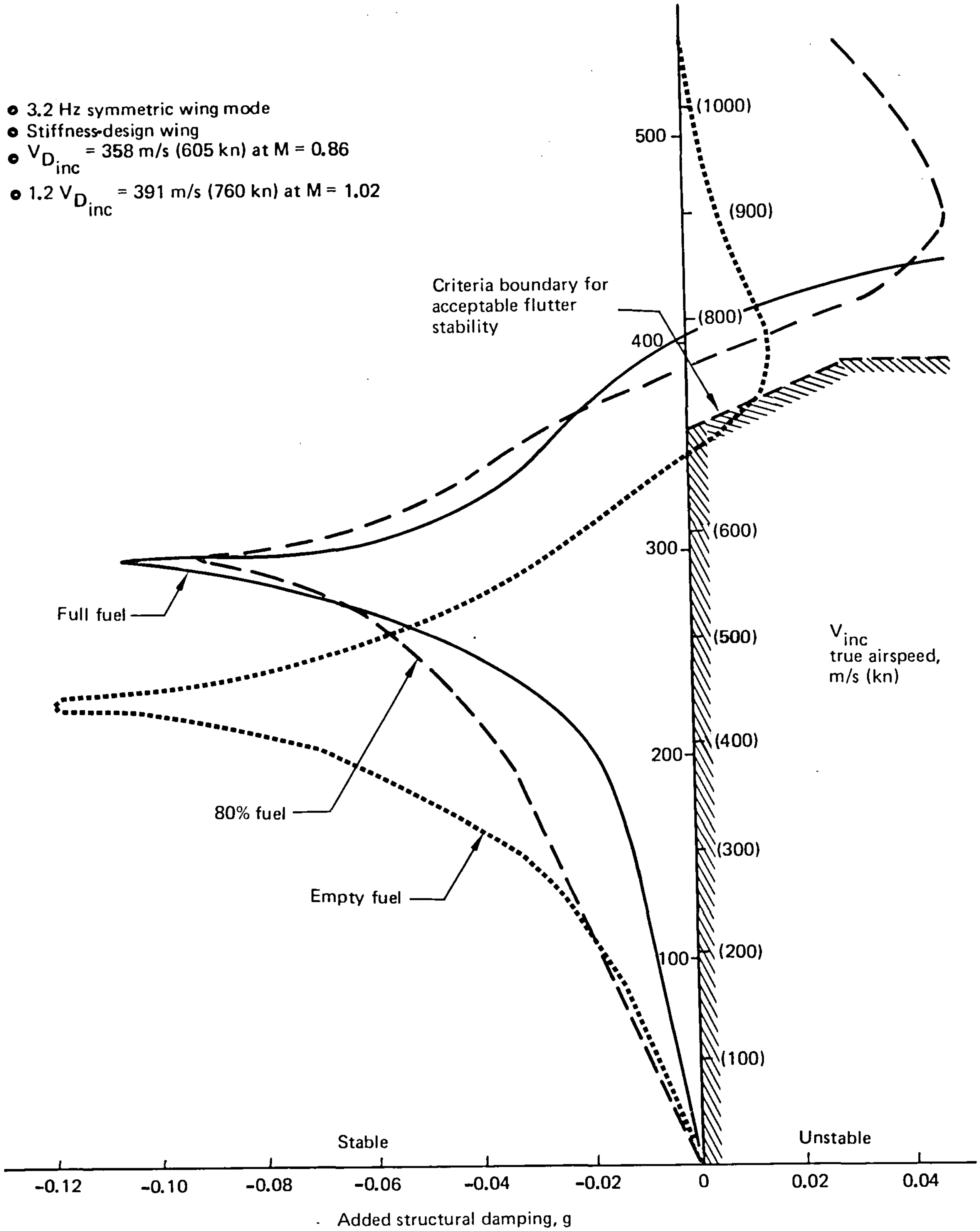


Figure 87. Critical V-g Trends, Altitude = 4968m (16 300 ft)

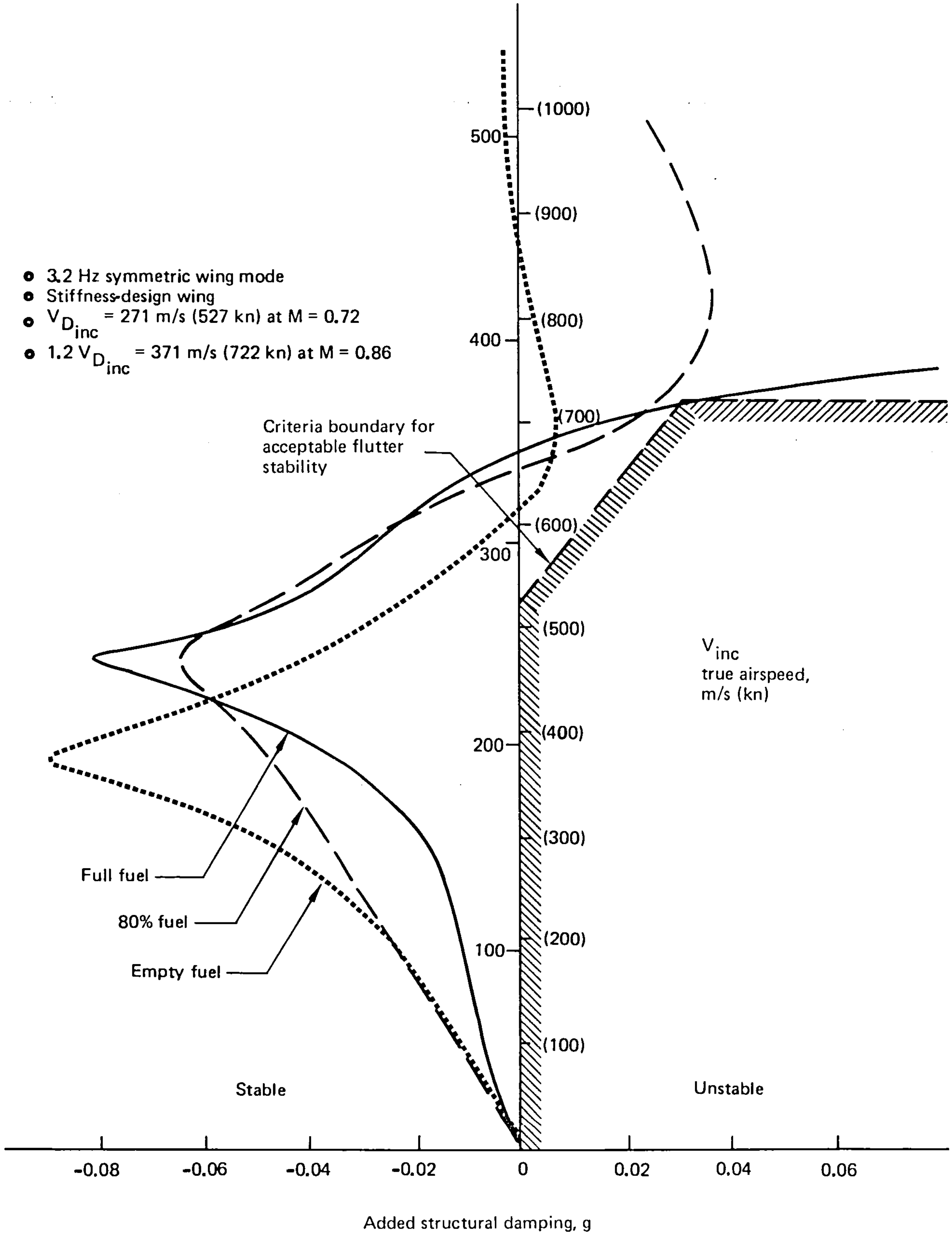


Figure 88. Critical V-g Trends, Altitude = 1890m (6200 ft)

structural damping must not be less than 1.2 V_D. These criteria are based on past experience with configurations of similar construction. The boundaries indicated in Figures 87 and 88 were determined using an inverse compressibility correction to calculate the effective incompressible V_D and 1.2 V_D speeds. These boundaries demonstrate the flutter stability criteria.

The critical flutter mode is a 3.2 Hz symmetric mode which includes significant nacelle pylon vertical bending and inboard wing vertical bending and torsion. This mode is very mild with zero wing fuel and is progressively "hardened" with increasing wing fuel as indicated in Figures 87 and 88. The full wing fuel condition is critical at 1.2 V_D; the empty wing fuel condition is critical at V_D and the 80% wing fuel condition is nearly critical at both V_D and 1.2 V_D. This was judged to be within the accuracy of the analysis since any flutter stiffener refinements would not result in significant wing weight.

Stiffness material was added to the strength-design wing to achieve the indicated flutter stability. The stiffness increase was achieved by increasing the spar web gage to equal the smaller of the upper or lower skin gage.

As indicated in Figure 86 the flutter boundaries for the stiffness-design wing satisfy the required flutter criteria for all conditions except the zero wing fuel condition. The flutter speed for this condition with zero structural damping is slightly less than V_D in violation of flutter requirements. Further stiffening to remove this small negative flutter margin was considered unnecessary because the flutter mode is very mild, as shown in Figure 87, with structural damping of only 0.01g needed to achieve the required stability at V_D.

Flutter Backup Data—Table 8 shows the generalized coordinates used for the flutter equations, and Table 9 lists the weight conditions analyzed. Figure 89 shows the wing elastic axis and lumped inertia modeling, and Figure 90 shows the aerodynamic paneling. Flutter speed compressibility corrections are indicated in Figure 91. Wing stiffness for the strength and stiffness-design wings are compared in Figure 92.

Table 8. Generalized Coordinates for Flutter Analysis

Branch	Antisymmetric analysis	Symmetric analysis
	Dominant Motion	Dominant motion
Airplane	Rigid airplane lateral translation Rigid airplane yaw Rigid airplane roll	Rigid airplane fore/aft Rigid airplane plunge Rigid airplane pitch
Forebody	First lateral bending	First vertical bending
Aftbody (rigid empennage)	First lateral bending First torsion Second lateral bending (torsion)	First vertical bending Second vertical bending
Vertical tail (rigid horizontal tail)	First lateral bending First torsion Second lateral bending (torsion)	First fore/aft bending (in plane) Second fore/aft bending (in plane)
Horizontal tail	First vertical bending Second vertical bending First torsion bending (coupled)	First vertical bending Second vertical bending First torsion bending (coupled)
Wing (rigid nacelle strut)	First vertical bending Second vertical bending First fore/aft bending (in plane) First torsion Third vertical bending Second fore/aft bending (in plane) Fourth vertical bending Second torsion	First vertical bending Second vertical bending First fore/aft bending (in plane) First torsion Third vertical bending Second fore/aft bending (in plane) Fourth vertical bending Second torsion
Nacelle	Side bending Vertical bending Roll/side bending	Side bending Vertical bending Roll/side bending

Table 9. Flutter Analysis Weight Conditions

Fuel. condition	Payload loading	Analysis gross weight, kg (lb)	Airplane cg
Empty	End loaded[a]	95 708 (211 000)	Aft limit
80%	End loaded	122 470 (270 000)	Aft limit
Full	End loaded	129 270 (285 000)	Aft limit

[a]Payload is loaded from most forward and most aft body locations (selected from available strength-design load conditions)

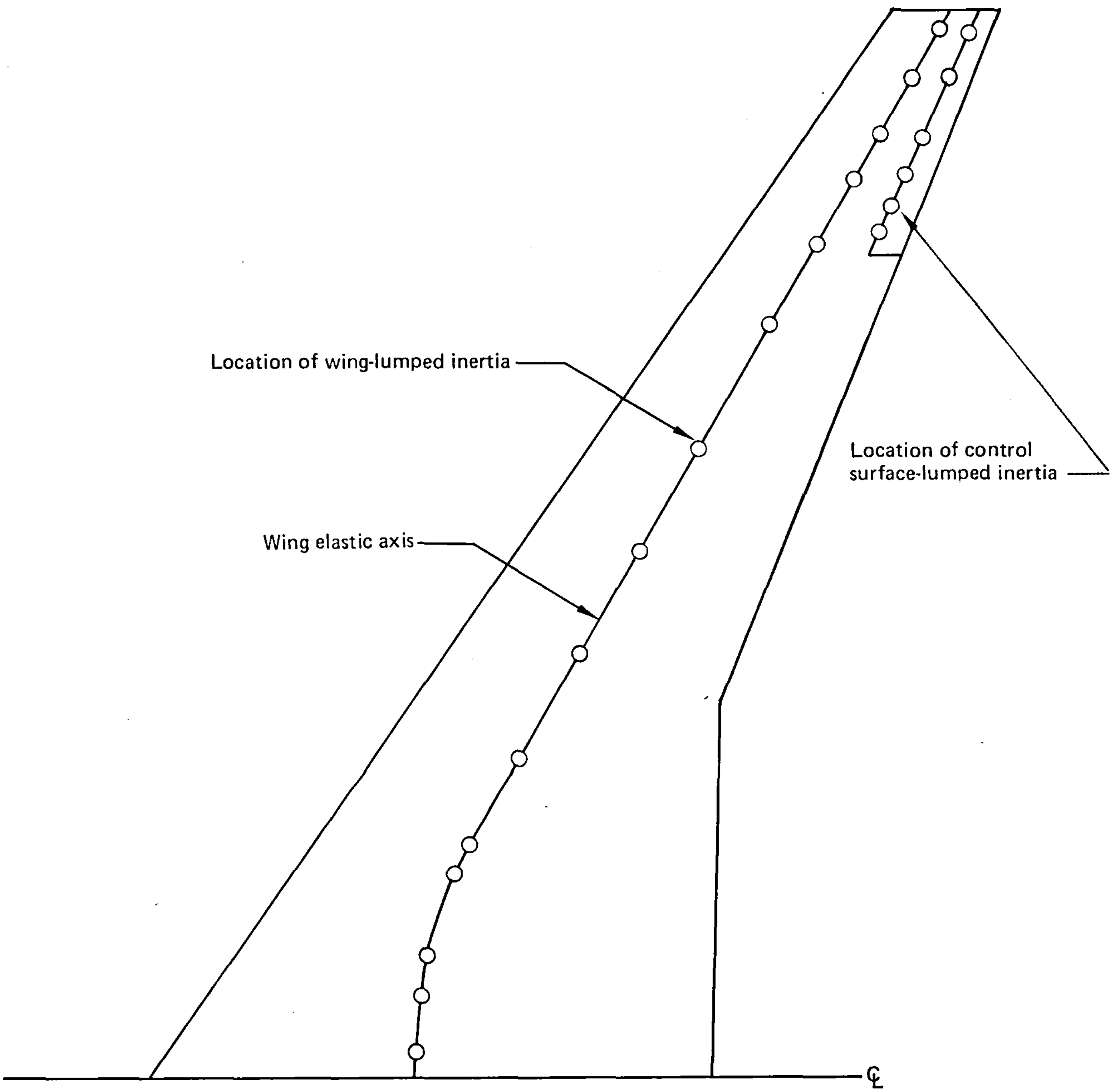


Figure 89. Location of Wing Elastic Axis and Lumped Inertias

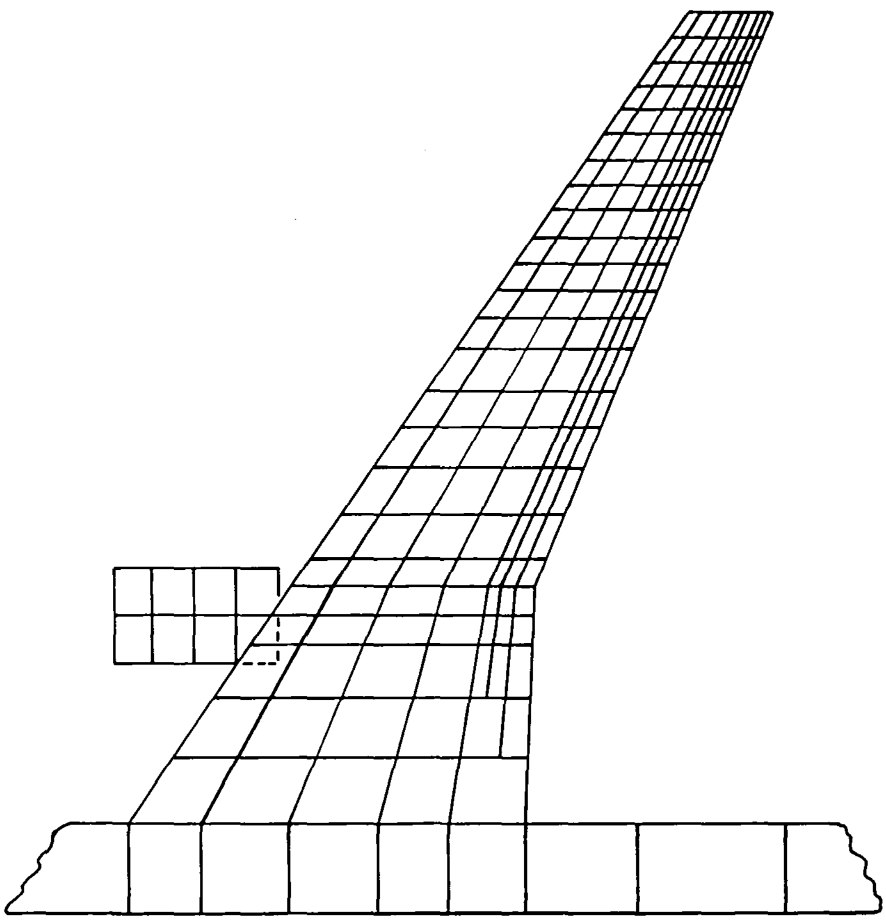


Figure 90. Doublet Lattice Aerodynamic Panels for Wing

Wing lift curve slope based on wind tunnel test	
M	C_{L_α}, deg^{-1}
0.4	0.0685
0.5	0.0716
0.6	0.0765
0.65	0.0805
0.7	0.0860
0.75	0.0945
0.775	0.0995
0.8	0.1065
0.86	0.1150
0.91	0.1010

Flutter speed, $V = C_c V_{inc}$

where V_{inc} = incompressible flutter speed

$$C_c = \left(\frac{C_{L_\alpha} \text{ at } M = 0.4}{C_{L_\alpha} \text{ at } M} \right)^{1/2}$$

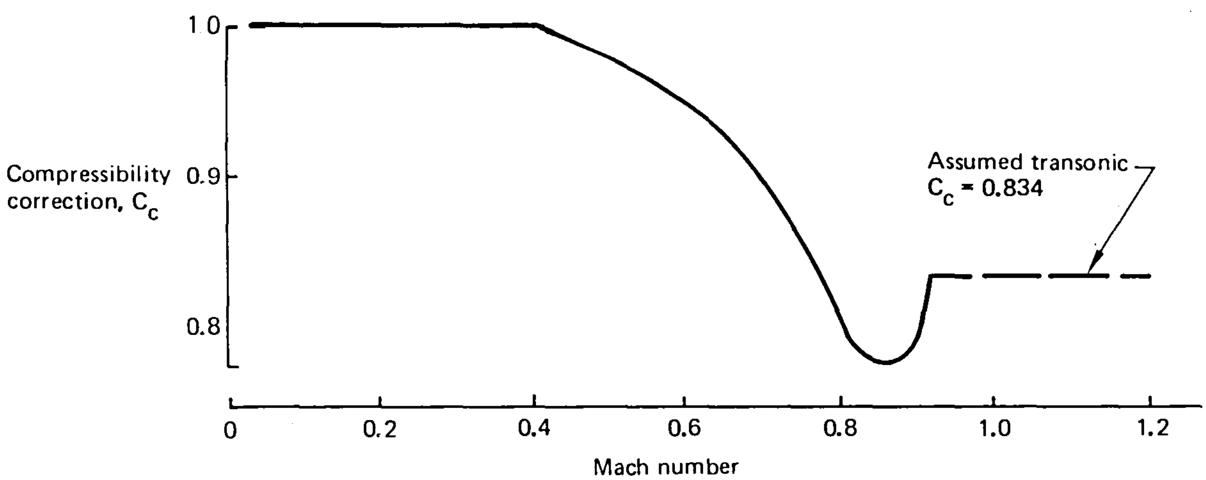


Figure 91. Compressibility Corrections to Incompressible Flutter Speeds

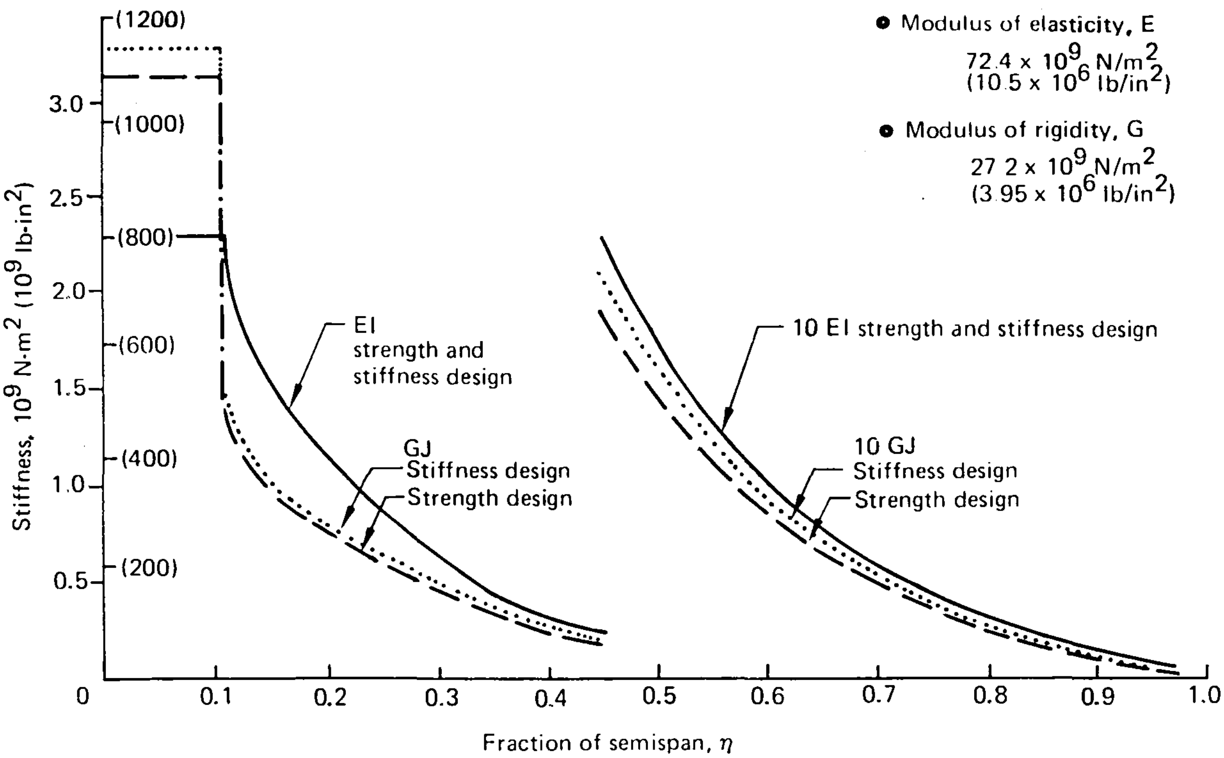


Figure 92. Wing Box Stiffness

7.2.1.4 Fatigue Analysis

The fatigue analysis for the wing box was performed for an airplane design life goal of 20 years. The flight segment distributions considered for design included the following:

- 62 000 short flights--567 km (306 nmi)
- 40 500 medium flights--954 km (515 nmi)
- 18 000 long flights--3369 km (1819 nmi)

The fatigue analysis was most critical for the short flight segment, based on the flight profile shown in Figure 93. For the fatigue calculations, the profile was simplified to delete the less critical conditions. The simplified profile, applied cycles, and load increments are listed in Table 10.

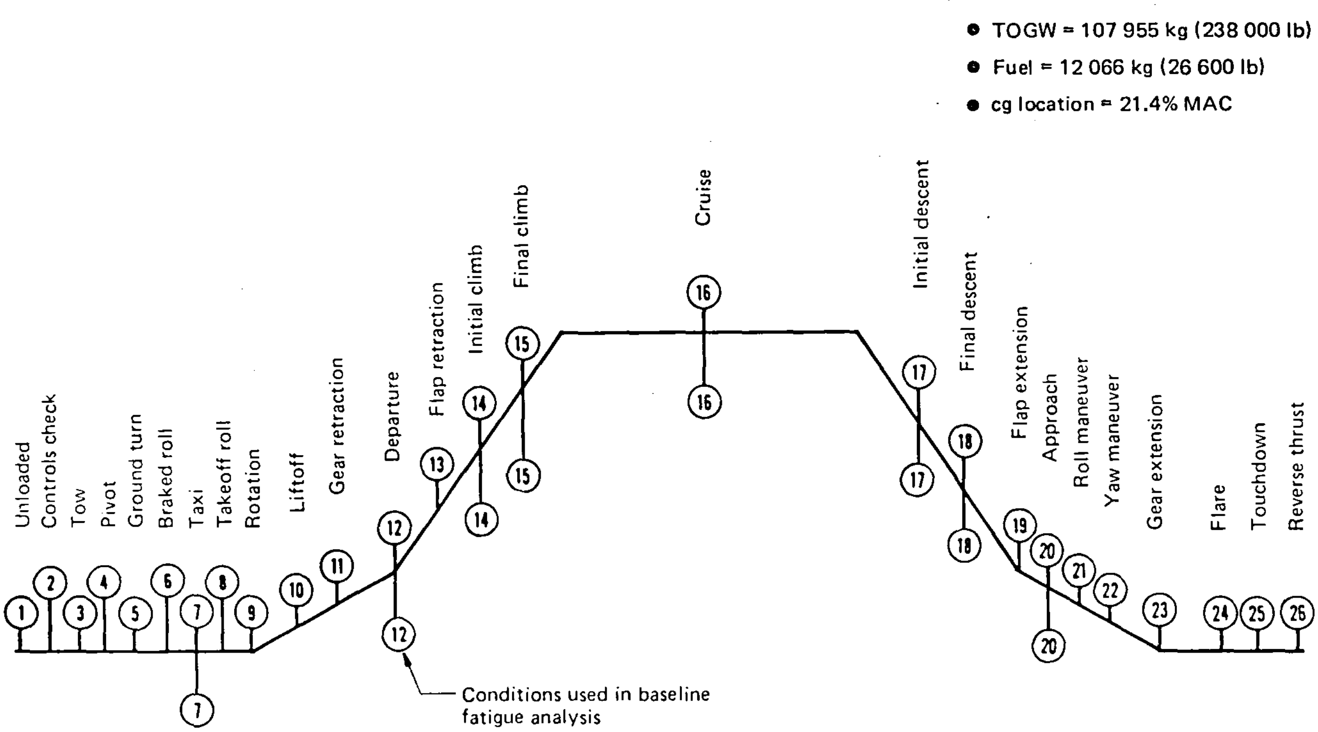


Figure 93. Flight Profile for Fatigue Analysis

Table 10. Fatigue Segment Distribution Short-Flight 567 km (306 nmi) Mission Summary Calculation

Condition number	Segment	Length		Cycles/ flight	g or gust velocity
		km	(nmi)		
7	Taxi	0	0	8	1 ± 0.3g
12	Depart	0	0	2	1 ± 0.3g
14	Initial climb	15	(8)	2	± 3.05 m/s (± 10 ft/s)
15	Final climb	100	(54)	2	± 2.74 m/s (± 9 ft/s)
16	Cruise	331	(179)		
	Gust			3	± 3.05 m/s (± 10 ft/s)
	Maneuver			2	1 ± 0.3g
17	Initial descent	104	(56)	2	± 2.74 m/s (± 9 ft/s)
18	Final descent	17	(9)	2	± 3.05 m/s (± 10 ft/s)
20	Flaps down approach	0	0	2	1 ± 0.3g

The results of the fatigue analysis are summarized in Figure 94. The wing box is not critical for fatigue. However, the inboard 50% of the lower wing box surface has low fatigue margins. The major part of the damage, 68%, is caused by the ground-air-ground cycle, and 11% of the damage is caused by gust.

7.2.1.5 Structural Sizing

The stress analysis and structural sizing of the wing box were performed using the ORACLE program. Typical results of the analysis are presented in Figure 95. The upper and lower surface cross-sectional areas represent the theoretical material requirements of the combined skin, stringers, and spar caps. The weight of nonoptimum structure (joints, fasteners, ribs, pads, etc.) was added to the weight of theoretical material in the calculation of total wing box structural weight. The incremental surface weights identified represent gust requirements relative to maneuver. Structural material requirements for the spars are also indicated in Figure 95. Increments are shown for maneuver, gust, and flutter requirements.

The fraction of surface material contained in the wing skins is indicated in Figure 96. This distribution is based on past experience with configurations of similar construction and provides a satisfactory balance between fail-safe design practice and high strength allowables.

The material properties used for structural sizing are presented in Figures 97 through 99. The upper surface skin and stringer properties are for 7150-T65 material. The lower surface skin properties are for 2024-T3 material with stringers of 2224-T3 material. The spar properties are for 7075-T6 material.

7.2.2 FUSELAGE

The fuselage design loads were determined from a survey of flight and ground conditions. All critical combinations of gross weight, payload distribution, and cg were analyzed. The combined effects of inertia, airload, and tail load were accounted for in all cases.

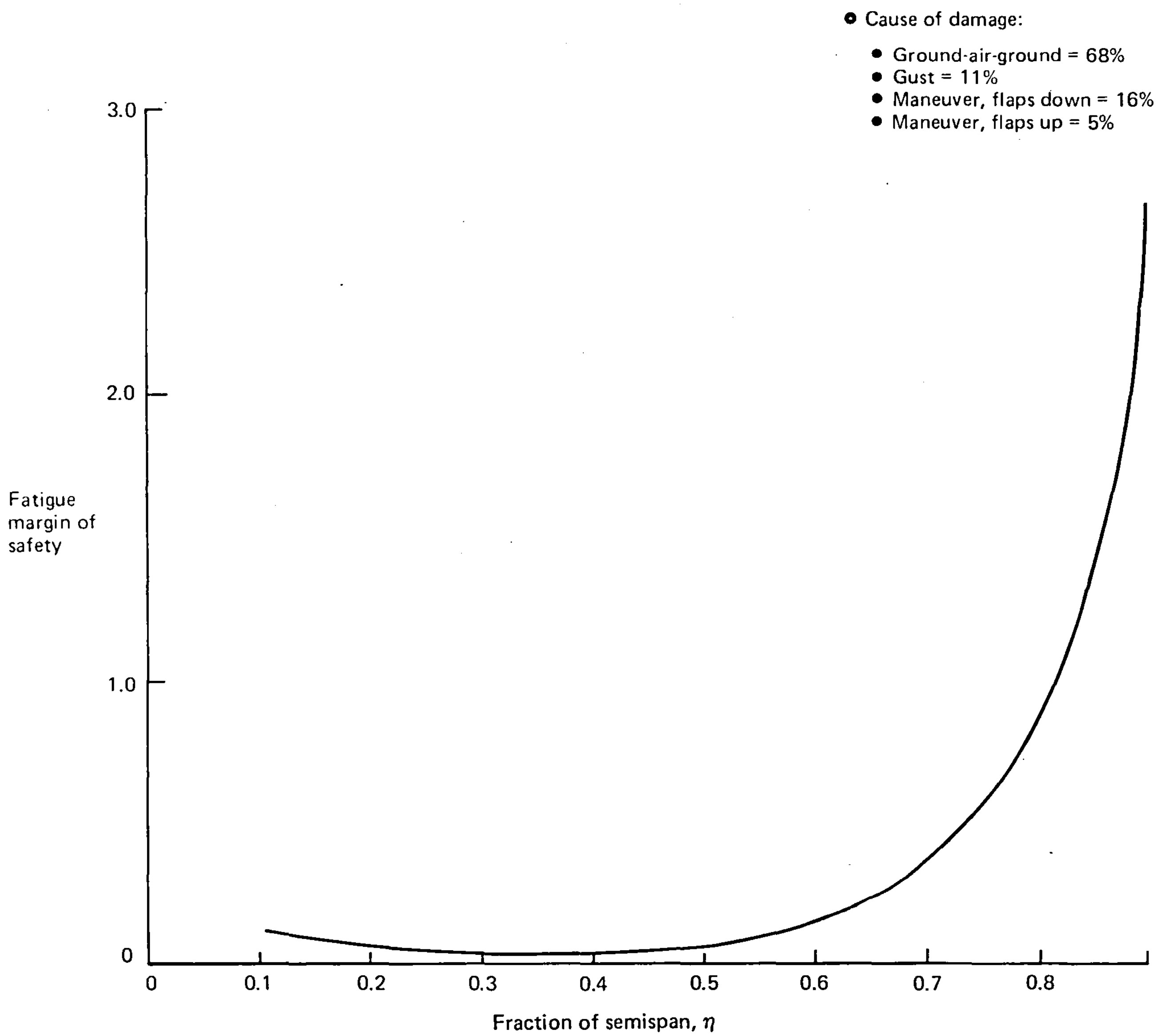


Figure 94. Baseline Fatigue Analysis, Wing Lower Surface

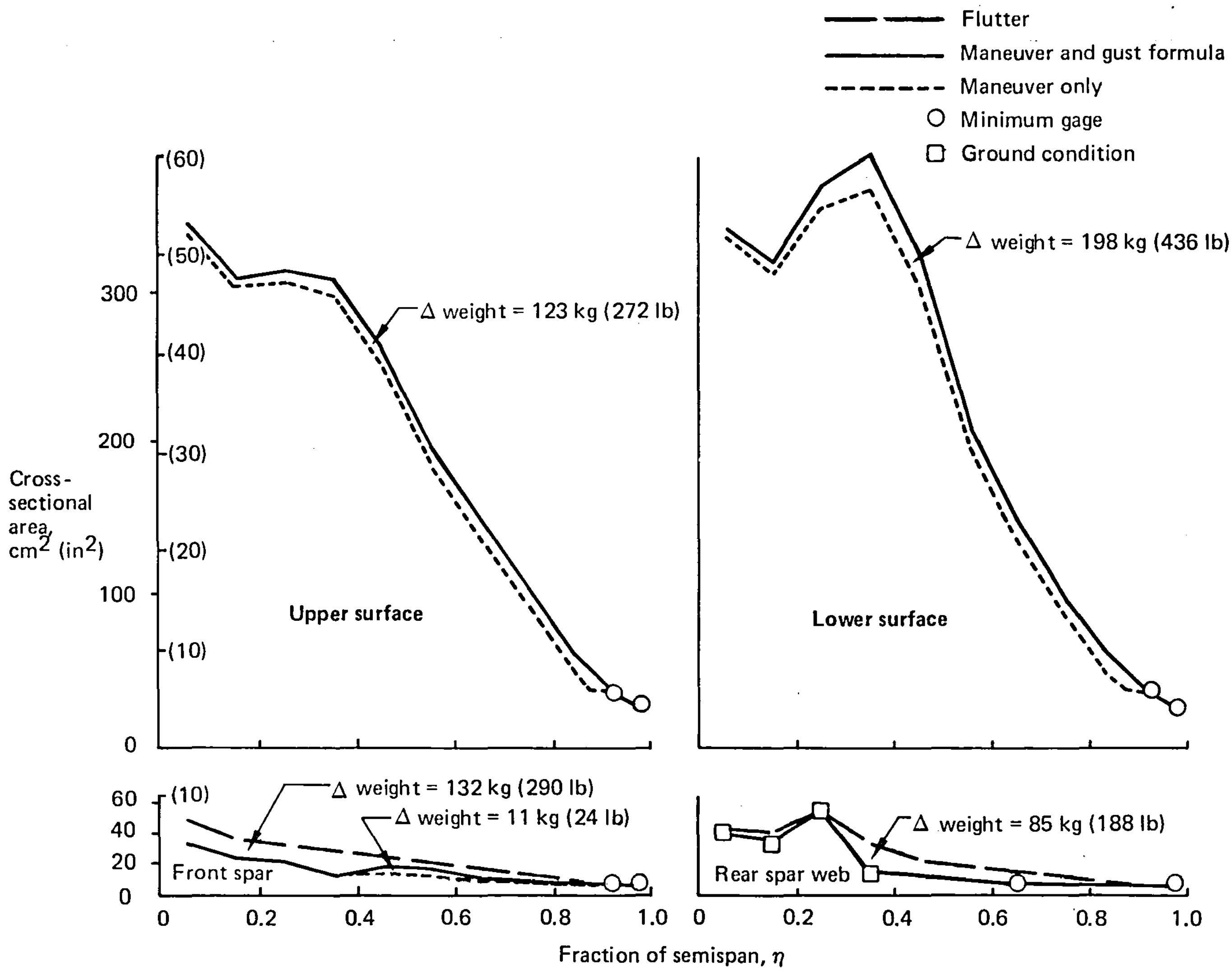


Figure 95. Theoretical Wing Box Material Requirements

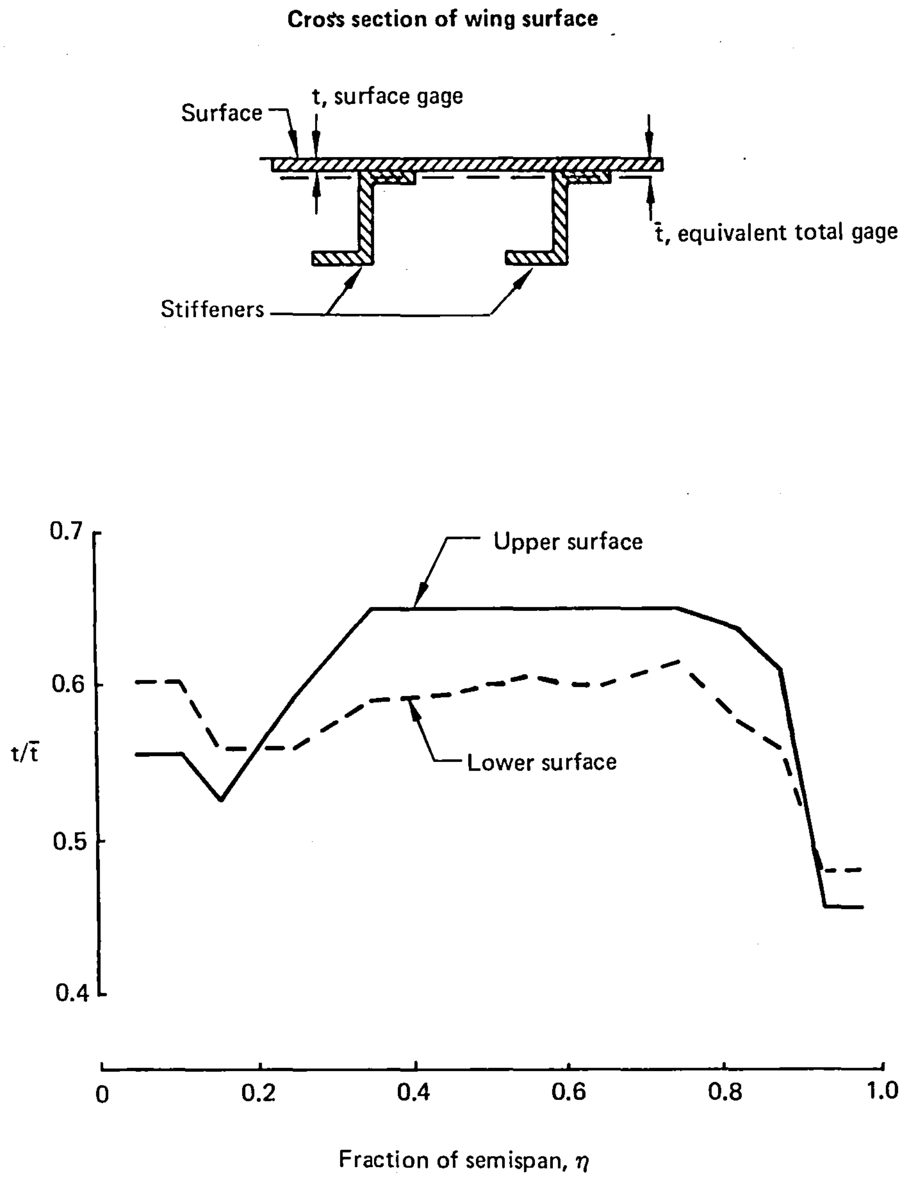


Figure 96. Wing Box Surface $t/\bar{t}$ Ratios

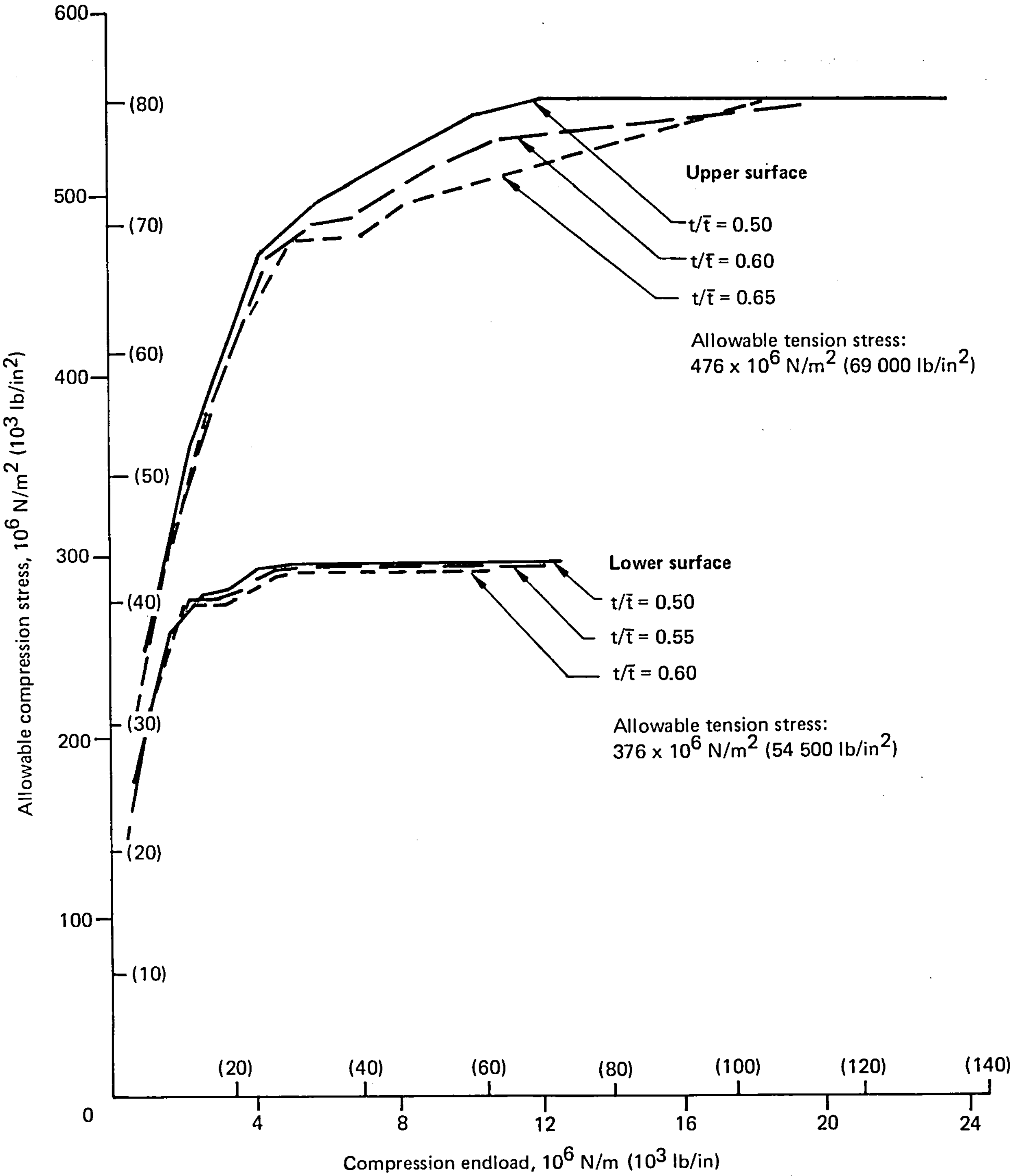


Figure 97. Allowable Compression and Tension Stress for Wing Surfaces

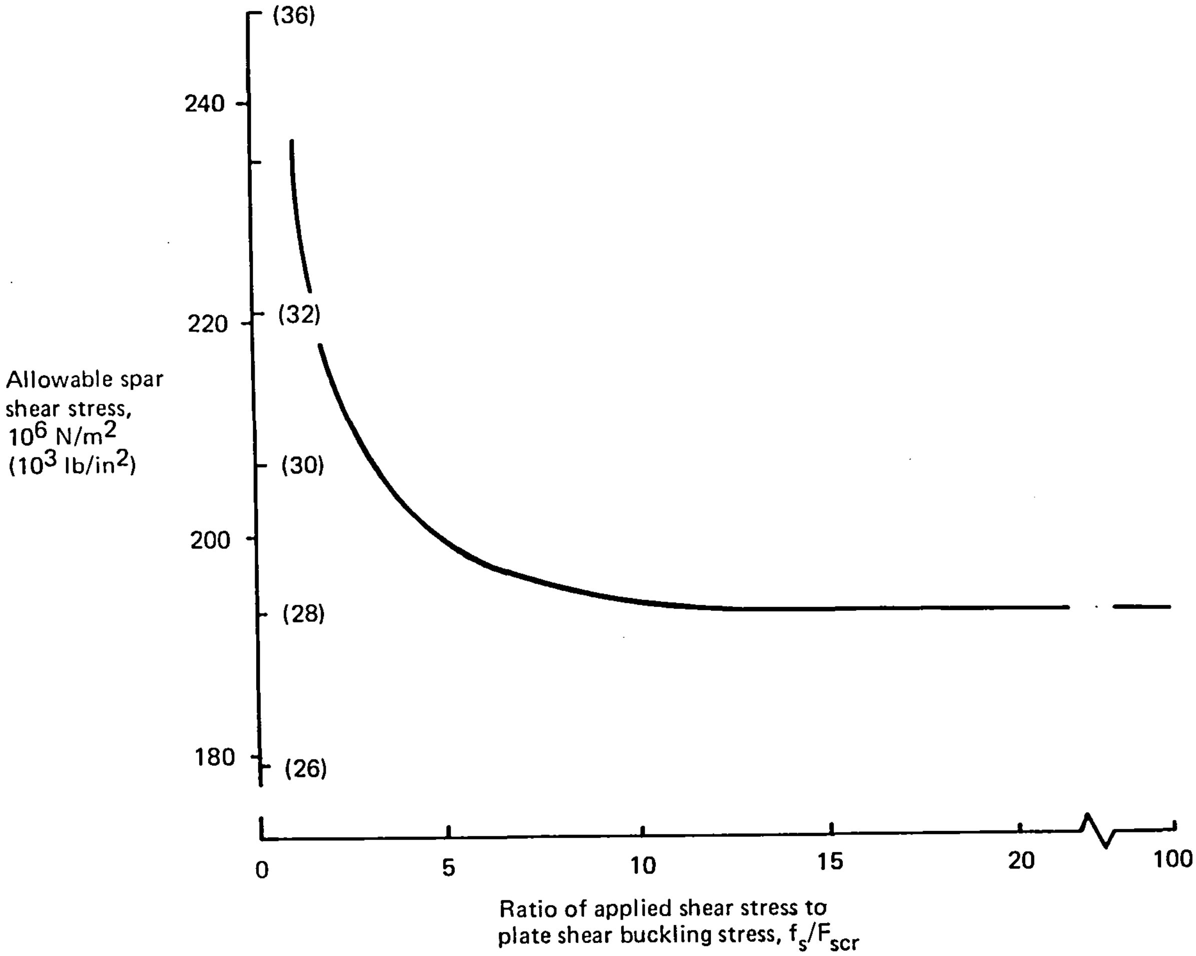


Figure 98. Allowable Shear Stress for Spars

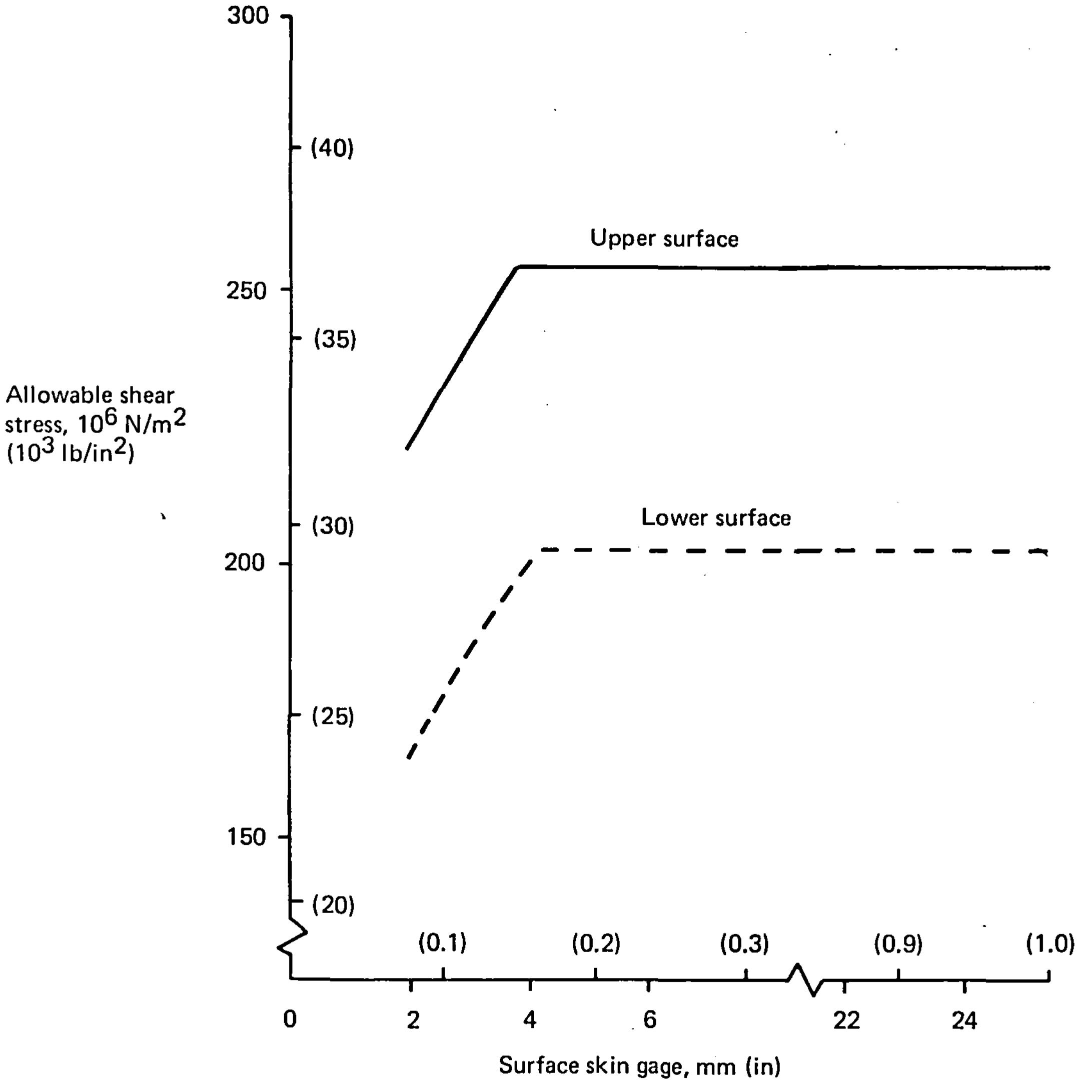


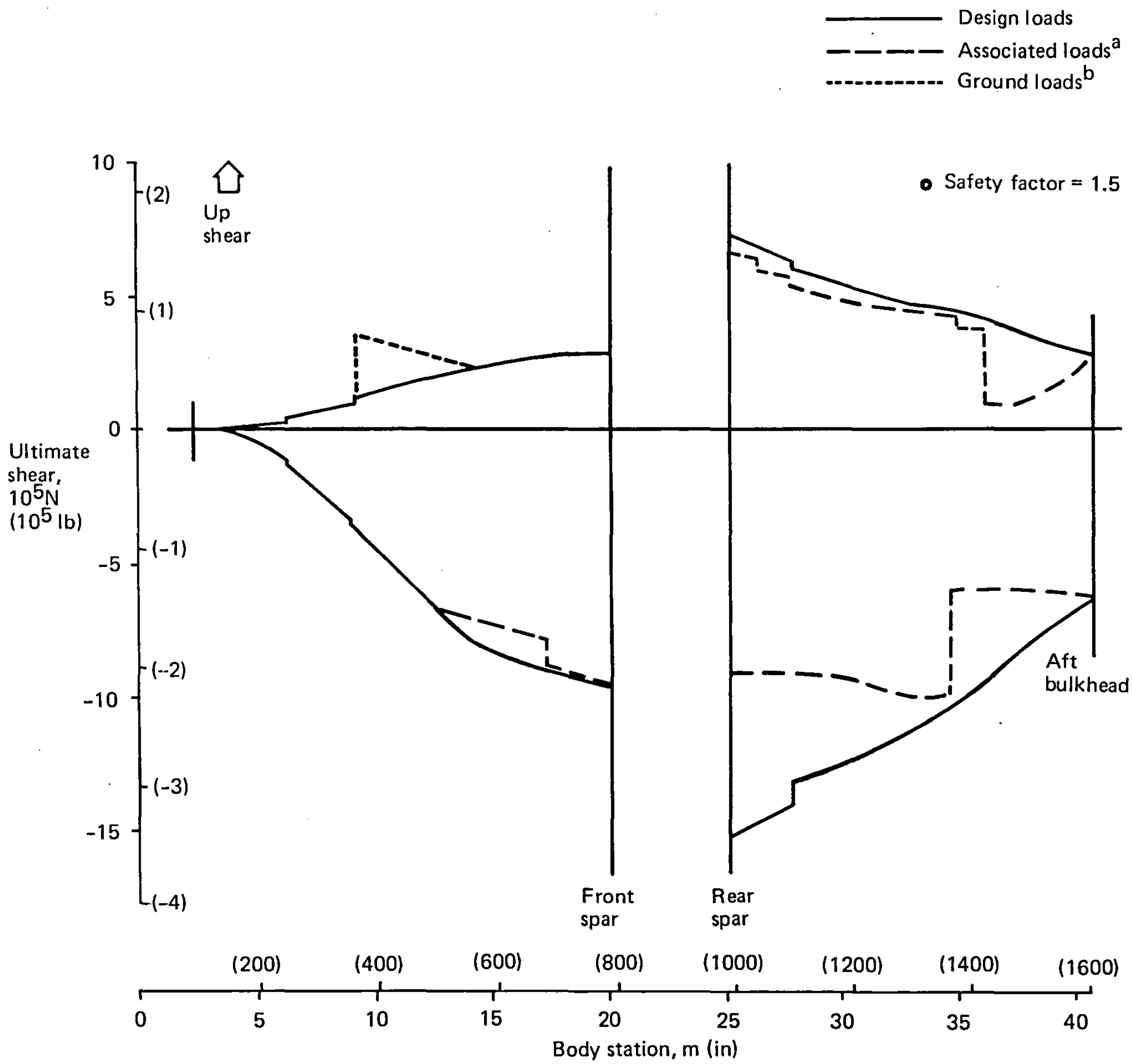
Figure 99. Allowable Shear Stress for Wing Surfaces

Envelopes of ultimate vertical shears and moments, used for structural sizing of the fuselage, are presented in Figures 100 and 101. An overview of significant design conditions for specific sections of the fuselage is presented in Figures 102 and 103. The combined external loads and cabin pressure differential loads are critical in several aft fuselage sections and in the forward fuselage.

7.2.3 HORIZONTAL TAIL

The horizontal tail loads were determined from a survey of critical maneuver and gust conditions. The horizontal tail design airload envelope is presented in Figure 104. The critical condition for positive load is a check-back maneuver from positive design load factor ($n = 2.5$). Critical conditions for negative loads are an abrupt up-elevator condition, which combines large negative shear with positive torsion, and a negative check-back maneuver from the negative design load factor ($n = -1.0$).

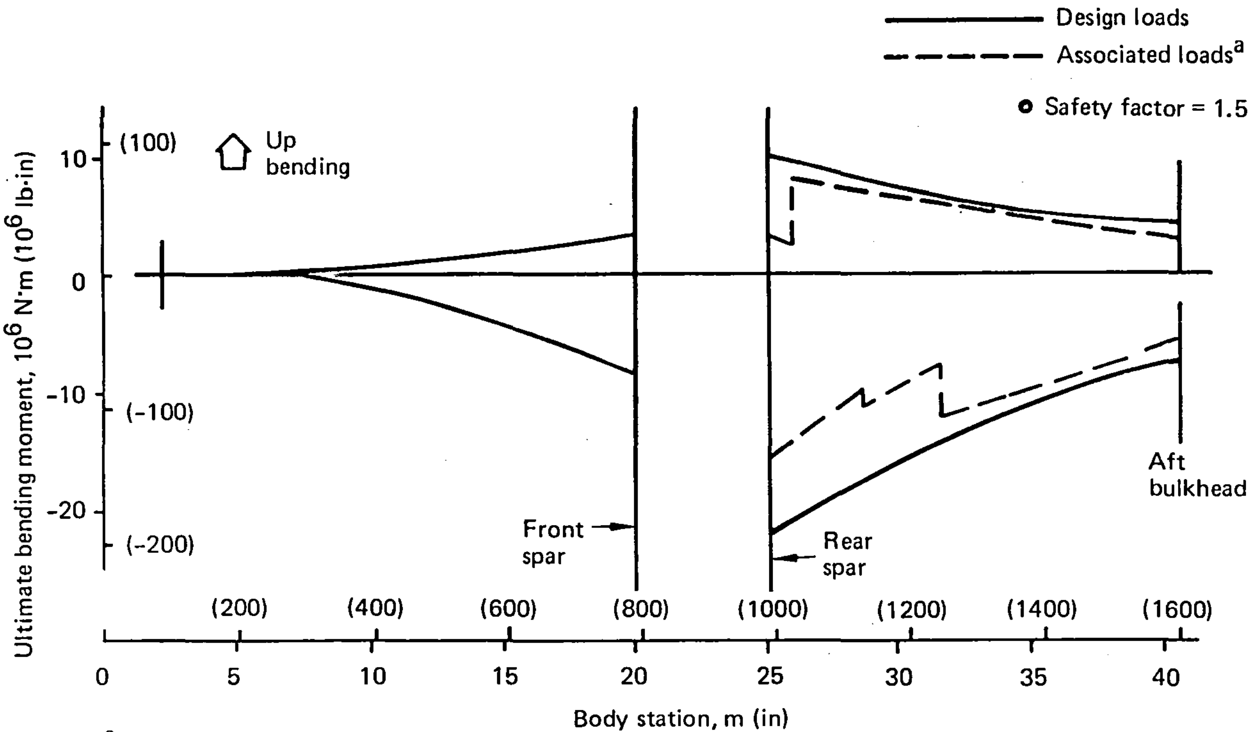
Principal airplane design data for the envelope conditions are summarized in Table 11.



[a]Shears associated with design moments

[b]Flight loads more critical due to cabin pressure

Figure 100. Ultimate Vertical Shear, Fuselage



[a]Moments associated with design shears

Figure 101. Ultimate Vertical Bending Moment, Fuselage

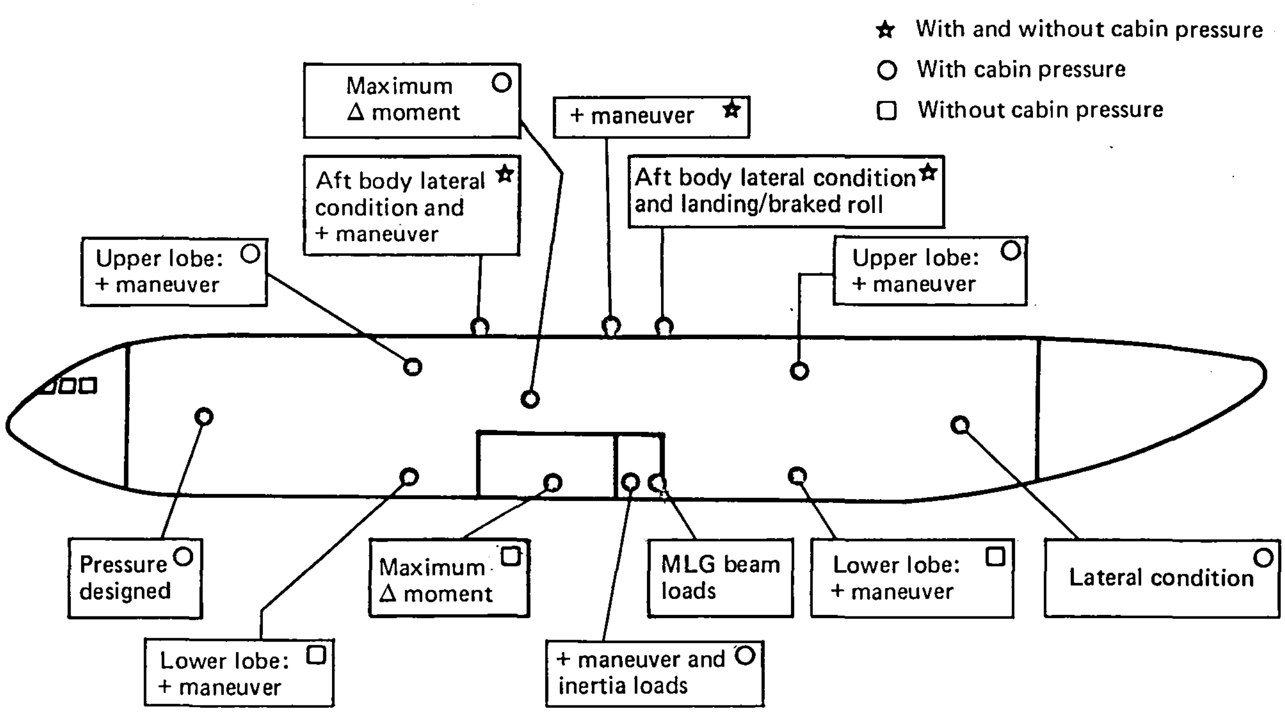
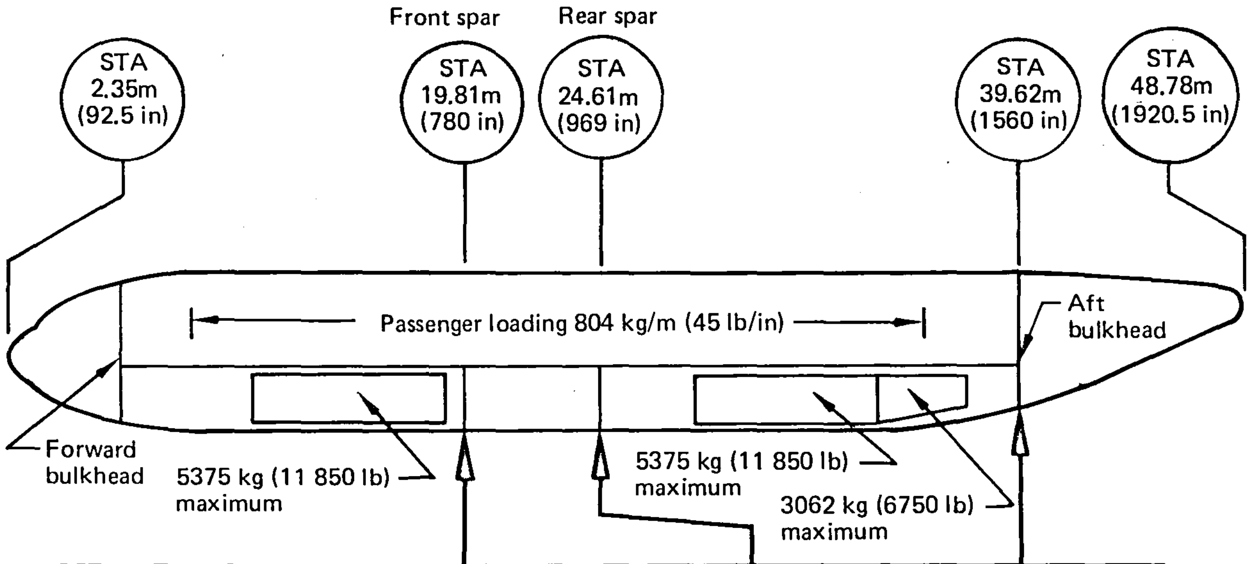


Figure 102. Body Locations, Significant Design Conditions



Load	Direction	Critical condition	Critical condition	Critical condition
Vertical shear	+ Up	-HAA maneuver, with spoilers, 3048m (10 000 ft)	$-1g$ maneuver at V_C, with spoilers, M = 0.84	Abrupt down elevator, with spoilers, sea level at V_A
	- Down	2.5g maneuver at V_D, M = 0.91	Vertical gust at V_B, M = 0.84	Abrupt up elevator, sea level at V_D
Bending moment	+ Crown compression	-HAA maneuver, with spoilers, 3048m (10 000 ft)	-HAA maneuver, with spoilers, at sea level	Abrupt down elevator, with spoilers, sea level at V_A
	- Crown tension	Vertical gust at V_C, M = 0.8	2.5g maneuver at V_D, M = 0.91	Abrupt up elevator, sea level at V_D
Lateral shear	+ Right	Nose gear yaw and rudder maneuver	Lateral gust at V_B	Rudder maneuver
Lateral moment	+ Right side compression	Nose gear yaw and rudder maneuver	Lateral gust at V_B	Rudder maneuver
Torsion	+ Right wingtip down	Nose gear yaw	Rudder maneuver	Rudder maneuver

Figure 103. Critical Body Load Summary

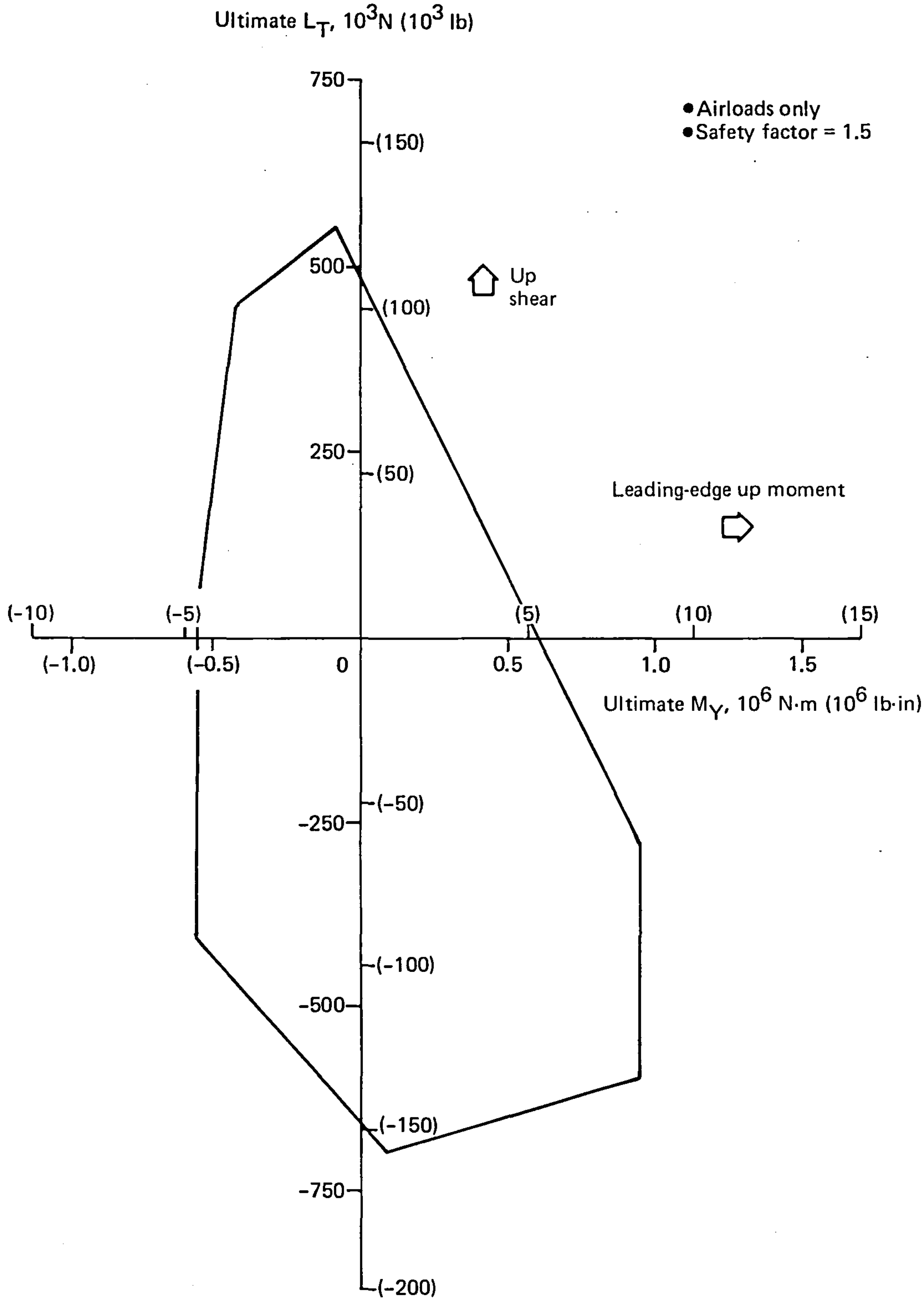


Figure 104. Horizontal Tail Ultimate Load Design Envelope

Table 11. Summary of Horizontal Tail Design Load Conditions

Condition	Direction	Altitude, m (ft)	V_e, m/s (KEAS)	Mach no.	Weight, kg (kip)	cg, MAC	n_z[a], g	$\ddot{\theta}$, rad/s^2
Balanced maneuver	Positive n_z	6065 (19 900)	211 (410)	0.91	119.8 (264.1)	0.09	2.5	0
	Zero n_z	6065 (19 900)	211 (410)	0.91	119.8 (264.1)	0.09	0	0
	Negative n_z	12 191 (40 000)	127 (246)	0.86	117.9 (260.0)	0.09	−1.0	0
Abrupt elevator	Up	6065 (19 900)	211 (410)	0.91	119.8 (264.1)	0.09	0.14	0.31
	Down	0 (0)	152 (295)	0.45	120.7 (266.1)	0.39	2.33	−0.51
Checkback	Positive	0 (0)	131 (255)	0.39	120.7 (266.1)	0.39	3.24	−0.28
	Negative	0 (0)	185 (360)	0.55	120.7 (266.1)	0.13	−2.5	0.54

[a]Load factor at horizontal tail $\bar{c}/4$

7.3 AERODYNAMIC CHARACTERISTICS

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7.3 AERODYNAMIC CHARACTERISTICS

Lift and drag characteristics have been estimated for the Baseline Configuration using methods and data bases selected from contemporary Boeing commercial airplane studies.

7.3.1 HIGH-SPEED LIFT AND DRAG

The Baseline Configuration is representative of current commercial airplane design practice and technology. The high-speed aerodynamic characteristics have been established from a data base developed with advanced aerodynamic technology correlated to and verified by data from an extensive series of high-speed wind tunnel tests. Methods for predicting flight characteristics of the Baseline Configuration have been confirmed to give results that are compatible with the more detailed method commonly used for building up aerodynamic characteristics from wind tunnel test data for the airplane configuration.

Complete wing design data and estimated high-speed lift and drag characteristics are considered by Boeing to be proprietary data. However, nominal values of lift and drag coefficients for long-range cruise are specified in Table 12 to establish a reference to which the estimated characteristics of other IAAC configurations may be compared.

Table 12. Nominal Lift and Drag Coefficients

Flight condition	C_L	C_D
Typical cruise	0.45	0.02525
Takeoff climbout (1.2 V_{stall}) (one engine inoperative)	1.35	0.1250
Landing approach (1.3 V_{stall})	1.33	0.1645

7.3.2 LOW-SPEED LIFT AND DRAG

The high-lift system for the Baseline Configuration consists of single-slotted, Fowler trailing-edge flaps with 22.5% chord, in combination with leading-edge slats. The slats extend to sealed or slotted positions as programmed by the flaps. Low-speed lift and drag characteristics have been estimated using an empirical method developed on the basis of extensive wind tunnel and flight data applicable to commercial transport aircraft. The same method will be used to estimate low-speed lift and drag characteristics for other IAAC configurations.

Complete low-speed lift and drag characteristics are considered by Boeing to be proprietary data and, therefore, are not included here. However, nominal values of lift and drag coefficients for takeoff climb and landing approach conditions are included in Table 12.

7.4 WEIGHT ANALYSIS

Page

7.4 WEIGHT ANALYSIS

This section presents a weight and moment of inertia definition of the Baseline Configuration which is more detailed than the data provided in Subsection 5.3. Weight and balance analyses of airplane components are discussed in Subsection 7.4.1, and mass distribution and panel moments of inertia are provided in Subsection 7.4.2.

7.4.1 WEIGHT AND BALANCE ANALYSIS

7.4.1.1 Wing

The wing structure weight of 15 695 kg (34 600 lb) was analyzed by using a computerized beam analysis program (ORACLE) which sizes "theoretical" box structure. Included are upper and lower surface skins and stringers and front and rear spar webs. Additional design considerations required for an "installed" weight were applied, based on Boeing's extensive production experience. These consist of manufacturing tolerance, feather material, pads, fasteners, and spar web stiffeners. Ribs and side-of-body joint weights were developed, consistent with the Reference Configuration. The relationship of the Reference Configuration to the Boeing New Airplane program and the IAAC project is described in Section 2.0 and illustrated in Figure 3.

Wing secondary structure (leading and trailing edges) weight is identical to the Reference Configuration weight.

The wing structure longitudinal cg at body station 26.07m (1027 in) was derived by an analysis of detailed components. Reference Configuration cg data were used as a basis, and modifications were applied where necessary.

The wing structure weight and longitudinal cg are shown in Table 13, along with all other functional groups in the airplane. The definition of subcomponents which comprise each functional group is consistent with aerospace industry practice as defined in Reference 13. The operational empty weight has been subcategorized into four parts: structure, propulsion system, fixed equipment (systems), and standard and operational items. This division provides a functional distribution. Component distributions must be extracted from the subcategories. An example is "propulsion

Table 13. Weight and Balance Statement

Functional group	Weight		Longitudinal cg body station	
	kg	(lb)	m	(in)
Wing	15 695	(34 600)	26.09	(1 027)
Horizontal tail	1 597	(3 480)	52.88	(2 082)
Vertical tail	2 009	(4 430)	47.70	(1 878)
Body	15 545	(34 270)	24.10	(949)
Main landing gear	6 237	(13 750)	25.98/26.49[a]	(1 023/1 043)[a]
Nose landing gear	875	(1 930)	6.17/6.76[a]	(243/266)[a]
Nacelle and strut	2 545	(5 610)	21.44	(844)
Total structure	44 485	(98 070)	26.64/26.72[a]	(1 049/1 052)[a]
Engine	7 951	(17 530)	21.72	(855)
Engine accessories	100	(220)	18.26	(719)
Engine controls	82	(180)	17.55	(691)
Starting system	77	(170)	20.60	(811)
Fuel system	599	(1 320)	25.88	(1 019)
Thrust reverser	1 638	(3 610)	21.84	(860)
Total propulsion system	10 446	(23 030)	21.89	(862)
Instruments	472	(1 040)	10.92	(430)
Surface controls	2 182	(4 810)	32.74	(1 289)
Hydraulics	1 021	(2 250)	26.09	(1 027)
Pneumatics	354	(780)	21.64	(852)
Electric	853	(1 880)	14.50	(571)
Electronic	721	(1 590)	12.14	(478)
Flight provisions	417	(920)	4.90	(193)
Passenger accommodations	6 681	(14 730)	22.30	(878)
Cargo handling	1 234	(2 720)	22.81	(898)
Emergency equipment	422	(930)	20.02	(788)
Air conditioning	975	(2 150)	18.82	(741)
Anti-icing	186	(410)	21.41	(843)
Auxiliary power unit	676	(1 490)	43.00	(1 693)
Total fixed equipment	16 194	(35 700)	22.91	(902)
Exterior paint	68	(150)	23.04	(907)
Options	907	(2 000)	25.17	(991)
Manufacturer's empty weight	72 100	(158 950)	25.10/25.15[a]	(988/990)[a]
Standard and operational items	6 196	(13 660)	25.86	(1 018)
Operational empty weight	78 296	(172 610)	25.17/25.22[a]	(991/993)[a]

[a]Gear up/gear down

pod" which includes: nacelle and strut under structures; engine accessories, engine controls, starting system, and thrust reverser under propulsion system; and portions of hydraulics and electrical under fixed equipment.

7.4.1.2 Empennage

Empennage weights are identical to the Reference Configuration weights.

7.4.1.3 Body (Fuselage)

Body structure weight of 15 545 kg (34 270 lb) is identical to the Reference Configuration.

7.4.1.4 Landing Gears

Main and nose landing gear weights were developed by GEARS, a Boeing developed program which is sensitive to design loads and configuration geometry.

7.4.1.5 Propulsion Pod and Systems

The propulsion pod and systems weight, 10 446 kg (23 030 lb), is identical to the Reference Configuration weight.

7.4.1.6 Fixed Equipment and Systems

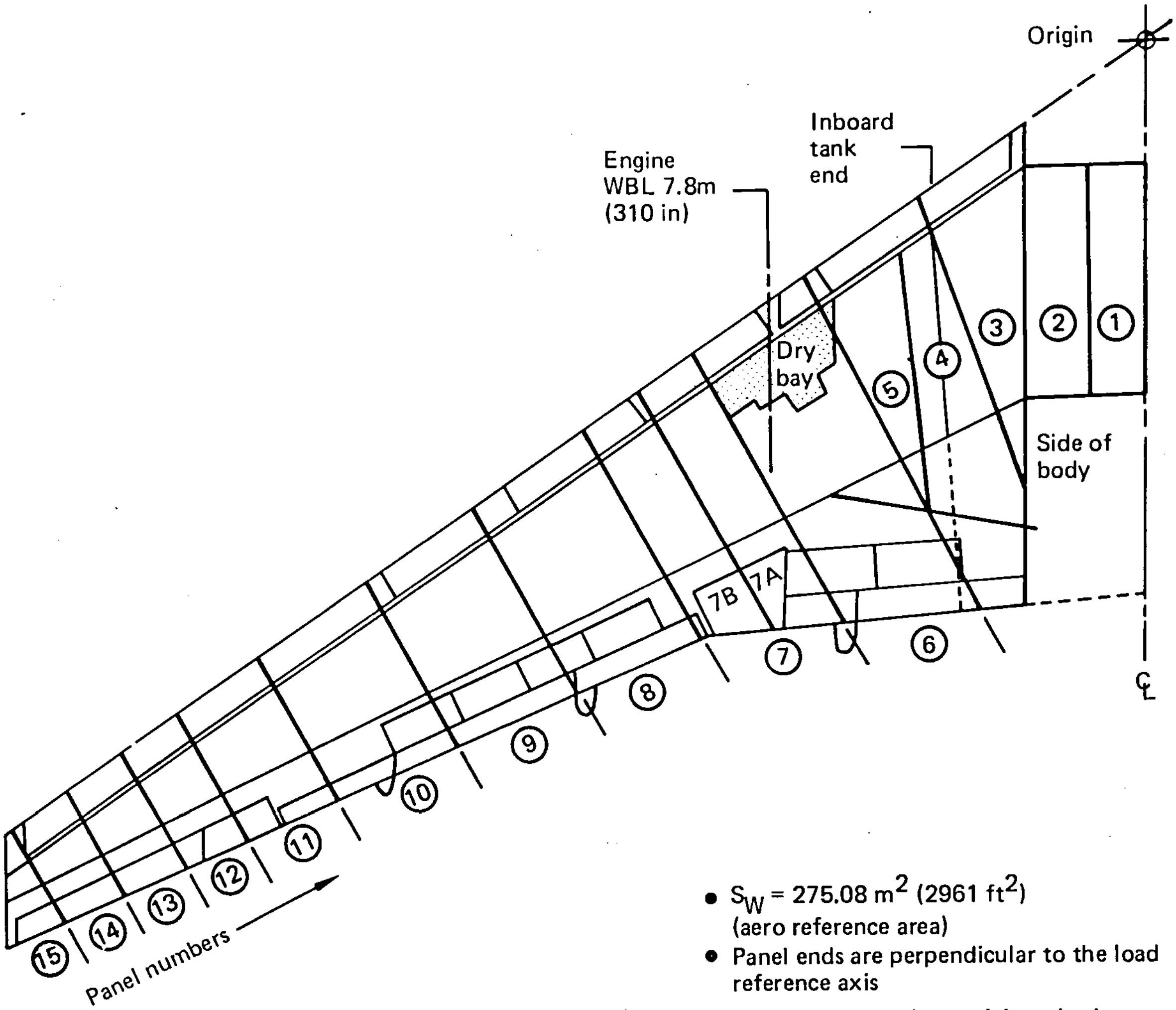
Fixed equipment and system weight of 16 194 kg (35 700 lb) is identical to the Reference Configuration weight. Centers of gravity were based upon the Reference Configuration and modified where necessary.

7.4.1.7 Standard and Operational Items

The standard and operational items weight of 6 196 kg (13 600 lb) is identical to the Reference Configuration.

7.4.2 MASS DISTRIBUTION AND PANEL MOMENTS OF INERTIA

In support of the airplane mathematical model for structural loads analyses, mass distribution and moments of inertia of detailed components were analyzed. Panel geometry definition for each of the major airplane components (wing, horizontal tail, vertical tail, body, nacelle, and strut) is shown in Figures 105 through 108. The Reference Airplane mass data were adjusted to establish the Baseline Configuration data. Detailed component data were subtotaled for the entire wing, body, horizontal tail, vertical tail, landing gear, and propulsion pod as shown in Table 14. The resulting mass distribution and moments of inertia are shown in Table 14.



- S_W = 275.08 m^2 (2961 ft^2) (aero reference area)
- Panel ends are perpendicular to the load reference axis
- Wing reference plane origin at body station 15.46m (608.6873 in), body buttock line 0m (0 in), body waterline 4.10m (161.6103 in)
- Intersection of side of body and wing reference plane at body buttock line 2.47m (97.42 in), body waterline 4.10m (161.6103 in)

Figure 105. Wing Mass Panel Definition

- Panel ends are perpendicular to the rear spar
- Horizontal tail area = 57.60 m^2 (620 ft^2)

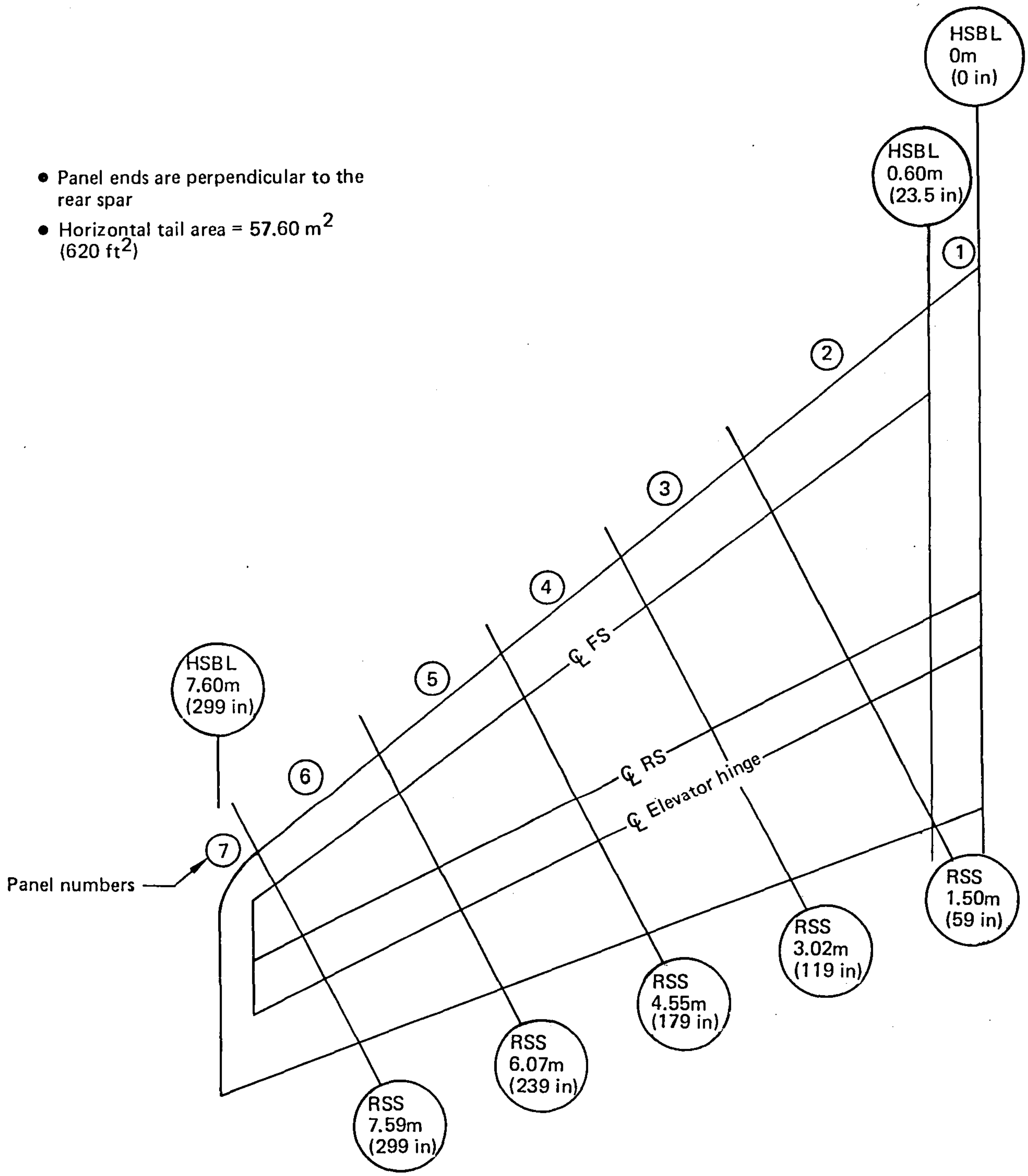


Figure 106. Horizontal Tail Mass Panel Definition

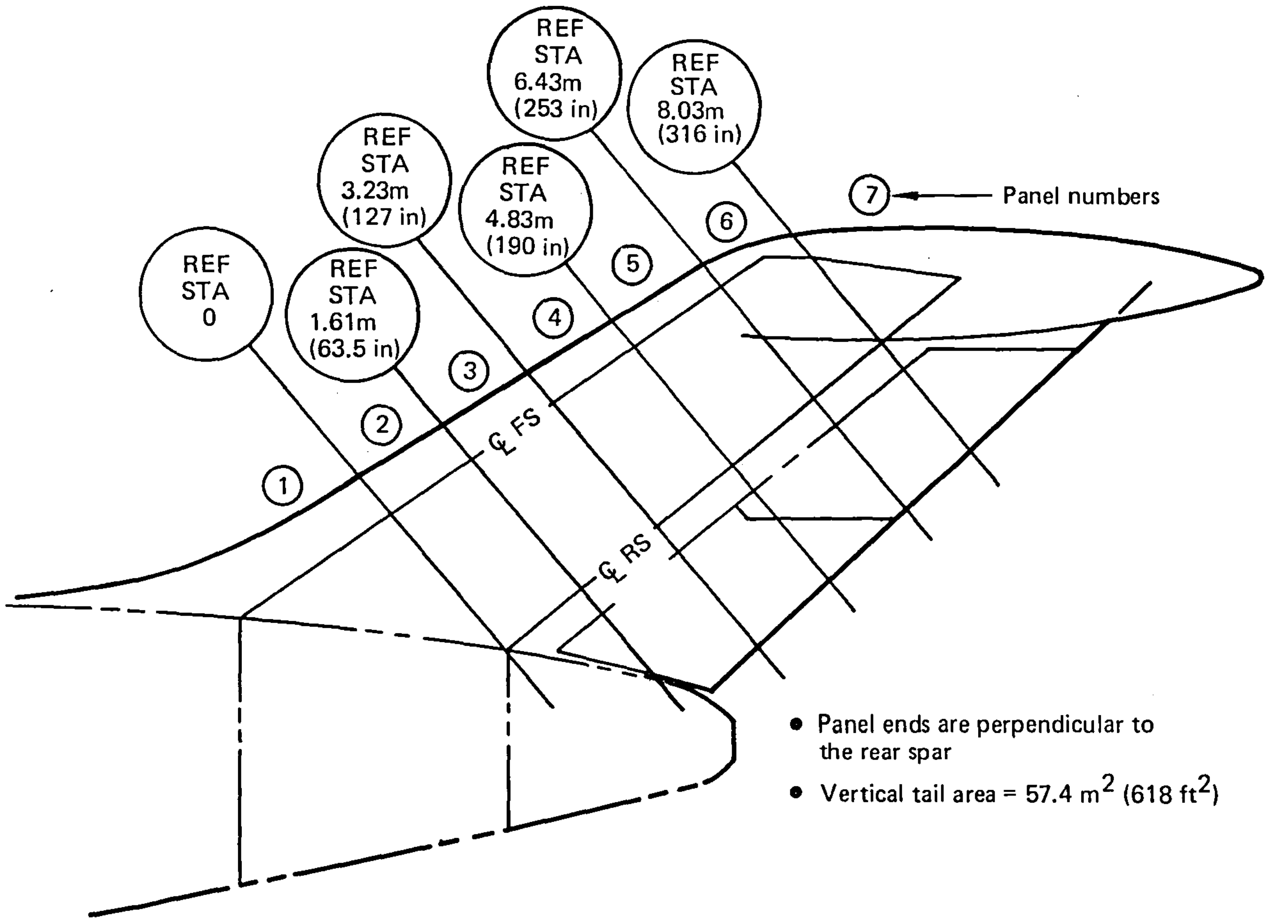


Figure 107. Vertical Tail Mass Panel Definition

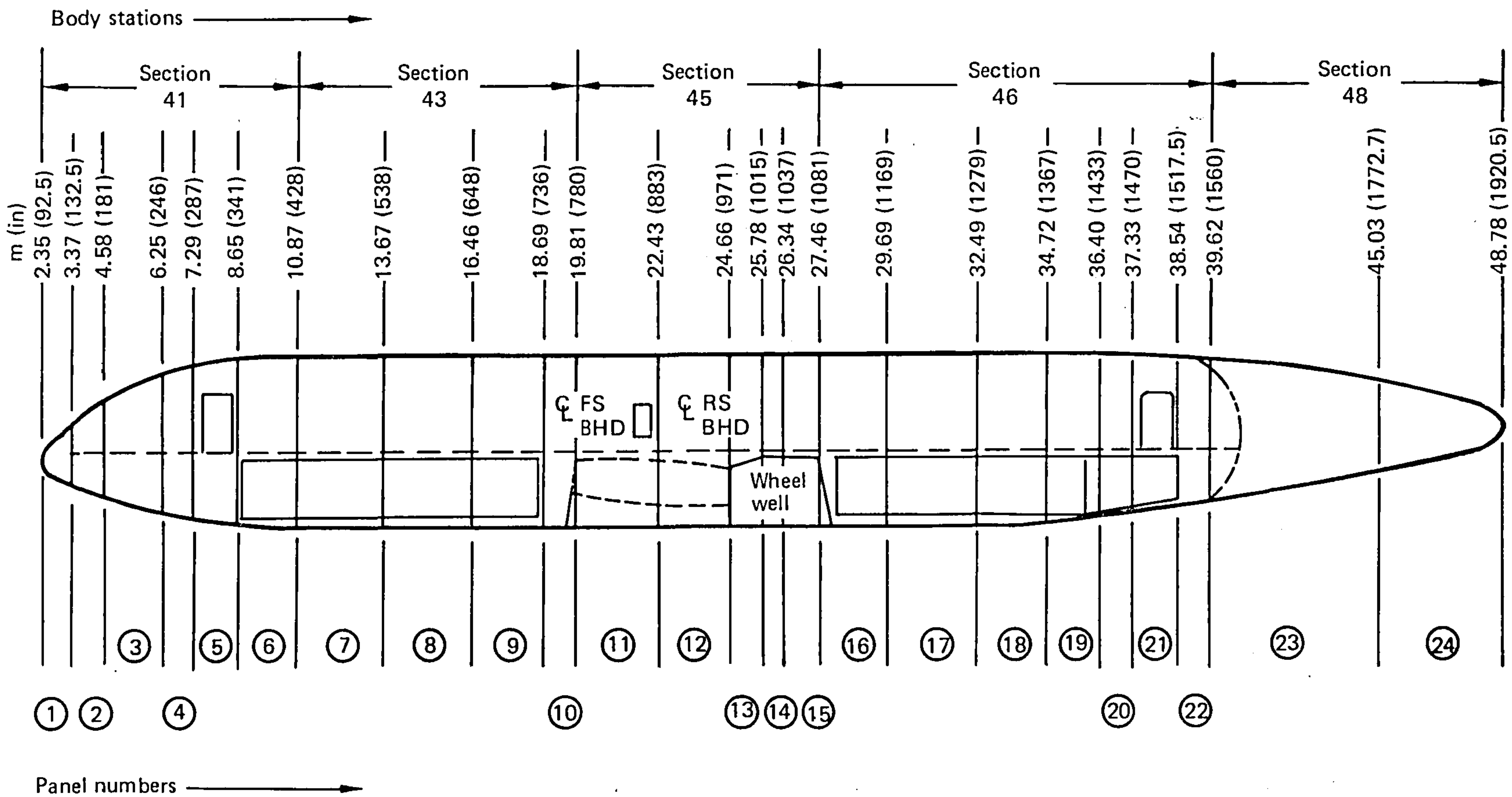


Figure 108. Body Mass Panel Definition

Table 14. Airplane Mass Distribution and Moments of Inertia Summary

Note: This is an example of mass properties buildup to MTOW—80% fuel, endloaded, aft MZFW cg condition.
(These data were used for flutter analysis.)

Item		Weight		Global cg					
				$\bar{X}$		$\bar{Y}$		$\bar{Z}$	
		kg	(lb)	m	(in)	m	(in)	m	(in)
Wing	OEW	18 947	41 770	26.31	1 036	0.00	0	4.87	192
Horizontal tail	OEW	1 864	4 110	53.13	2 092	0.00	0	13.45	529
Vertical tail	OEW	2 505	5 522	47.77	1 881	0.00	0	10.76	424
Body	OEW	36 604	80 698	23.00	906	0.06	2	5.23	206
Main gear (up)	OEW	4 434	9 776	26.25	1 033	0.00	0	3.80	150
Nose gear (up)	OEW	599	1 320	6.05	238	0.00	0	3.95	155
Nacelle/strut	OEW	13 248	29 208	21.67	853	0.00	0	3.20	126
Airplane	OEW	78 201	172 404	25.14	990	0.03	1	5.08	200
Main deck payload		7 362	16 230	29.52	1 162	0.00	0	5.54	218
Maximum forward cargo		2 123	4 680	9.96	392	0.00	0	4.06	160
Aft cargo		5 375	11 850	33.46	1 317	0.00	0	4.06	160
Maximum bulk cargo		2 721	6 000	37.65	1 482	0.00	0	4.06	160
Payload		17 581	38 760	29.62	1 166	0.00	0	4.68	184
Maximum design zero fuel weight		95 782	211 164	25.96	1 022	0.02	1	5.01	197
Mission fuel		26 688	58 836	25.16	990	0.00	0	4.78	188
Maximum design takeoff wt		122 470	270 000	25.79	1 015	0.02	1	4.96	195

Item		Moment of inertia about cg											
		I_{XX}		I_{YY}		I_{ZZ}		I_{XY}		I_{XZ}		I_{YZ}	
		kg-m^2	(lb-in^2 x 10^3)	kg-m^2	(lb-in^2 x 10^3)	kg-m^2	(lb-in^2 x 10^3)	kg-m^2	(lb-in^2 x 10^3)	kg-m^2	(lb-in^2 x 10^3)	kg-m^2	(lb-in^2 x 10^3)
Wing	OEW	1 965 109	6 715 109	229 436	784 021	2 175 449	7 433 873	0	0	+31 787	+108 621	0	0
Horizontal tail	OEW	26 736	91 361	4 421	15 107	31 045	106 085	0	0	- 258	- 882	0	0
Vertical tail	OEW	10 543	36 026	34 689	118,538	25 264	86 332	0	0	+12 073	+ 41 256	0	0
Body	OEW	151 130	516 435	5 539 799	18 930 424	5 522 645	18 871 804	-16 553	-56 566	+22 939	+ 78 385	-298	-1 017
Main gear (up)	OEW	23 543	80 451	2 156	7 368	24 530	83 843	0	0	+ 94	+ 322	0	0
Nose gear (up)	OEW	73	248	331	1 132	288	984	0	0	- 66	- 227	0	0
Nacelle/strut	OEW	821 175	2 806 093	32 437	110 844	842 419	2 878 688	0	0	+ 2 056	+ 7 027	0	0
Airplane	OEW	3 266 117	11 160 869	9 430 976	32 227 228	11 941 699	40 806 790	-21 512	-73 511	+903 655	+3 087 940	+ 49	+ 166
Main deck payload		15 077	51 520	471 810	1 612 254	485 012	1 657 368	0	0	- 1 229	- 4 201	0	0
Maximum forward cargo		2 311	7 897	1 222	4 175	2,478	8 467	0	0	0	0	0	0
Aft cargo		5 852	19 996	12 601	43 060	15 780	53 923	0	0	0	0	0	0
Maximum bulk cargo		2 963	10 124	2 532	8 651	4 141	14 152	0	0	0	0	0	0
Payload		35 584	121 598	1 573 092	5 375 518	1 582 956	5 409 228	0	(0)	- 2 328	- 7 956	0	0
Maximum design zero fuel weight		3 304 029	11 290 420	11 294 080	38 593 768	13 812 383	47 199 232	-23 415	-80 014	+875 519	+2 991 795	+219	+749
Mission fuel		1 740 979	5 949 216	87 086	297 586	1 821 837	6 225 524	0	0	+10 606	+36 244	0	0
Maximum design takeoff wt		5 046 189	17 243 674	11 396 008	38 942 073	15 647 906	53 471 521	-23 007	-78 619	+890 124	+3 041 703	+339	+1 158

8.0 CONCLUDING REMARKS AND RECOMMENDATIONS

Page

8.0 CONCLUDING REMARKS AND RECOMMENDATIONS

The purpose of the IAAC project is to assess the effects of the integrated application of active controls technology (ACT) to a medium-range subsonic transport airplane. As a prerequisite in that study, the Baseline Configuration described in this report was developed, and a comprehensive data base was established in the traffic segment where most fuel is used.

The technical level of this configuration consists of conventional construction with advanced aluminum alloys and some graphite epoxy for the secondary structure. Modern systems emphasize application of digital electronics and advanced displays.

A substantial amount of design, test, and analysis data were taken from an early stage of the Boeing New Airplane program. Refinements included more indepth studies in areas which will benefit from the application of active controls and a supplementary wing flutter analysis and corresponding update.

It is concluded that, at the end of the task described in this document, the Baseline Configuration is adequate to serve as a data base for the assessment of the benefits of active controls.

It is recommended that the ACT Configuration development be continued according to the IAAC project plan with the Initial ACT Configuration and the Wing Planform Study leading to development of a resized Final ACT Configuration.

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9.0 REFERENCES

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16. Abstract

This document concludes the first task of the Integrated Application of Active Controls (IAAC) Technology project within the Energy Efficient Transport (EET) program. A comprehensive data base was developed and assembled for a modern Mach 0.8 passenger airplane in the traffic segment where most fuel is used. The technology level of this configuration consists of conventional construction with advanced aluminum alloys and some graphite epoxy for the secondary structure. Modern systems emphasize application of digital electronics and advanced displays. A substantial amount of design, test, and analysis data was taken from an early stage of the Boeing New Airplane program. Refinements included more indepth studies in areas that will benefit from the application of active controls and of a supplementary wing flutter analysis and corresponding update. This data base will be used as a point of departure, during later activities on the long range IAAC project, for development of additional configurations that use active controls technology (ACT) for reduced weight and improved aerodynamics. The technical characteristics of the future ACT configurations will be measured against this Baseline Configuration.

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