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TECHNICAL AND ECONOMIC ASSESSMENT OF SWEPT-WING SPAN-DISTRIBUTED LOAD CONCEPTS FOR CIVIL AND MILITARY AIR CARGO TRANSPORTS

Preliminary Design Department
Boeing Commercial Airplane Company

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16. Abstract

A preliminary design study to assess the technical and economic feasibility of large swept-wing span-distributed load freighter aircraft to be certified in 1995 has been made. The study also assessed the impact of military requirements on the performance, economics, and fuel consumption characteristics.

The study was limited to configurations having net payloads of 272 155 to 544 311 kilograms (600 000 to 1.2 million pounds) contained within swept wings of constant chord. These configurations are of advanced composite construction with controllable winglets and full-span digitally-controlled trailing-edge surfaces. Civil, military, and joint civil/military production programs were considered.

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FOREWORD

This study was performed by the Preliminary Design Department of The Boeing Commercial Airplane Company and monitored by Mr. A. R. Heath, Jr., of NASA, Langley. The following Boeing personnel, under the direction of Mr. Philip C. Whitener, Program Manager, made up the study team and were responsible for the subjects noted.

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TABLES

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FIGURES

FIGURES (Concluded)

1.0 SUMMARY

This document presents analyses and results of a study of the span distributed load design concept as applied to large freighter aircraft. The study concentrates on swept wing configurations of constant chord balanced with wing tip surfaces that provide the proper span airload distribution to permit an efficient payload distribution in the constant chord wings.

The study is based to a large extent on straight wing studies conducted under a NASA contract NAS1-13963 and documented in NASA CR-14463, Reference 1.

A parametric study using a range of distributed load configurations of this general type was conducted to determine the best choice of size and geometry for optimum economics versus payload weight. The first portion of the parametric study explored constant thickness ratio with variable aspect ratio, number of bays, and payload. The net payloads were varied from 272 155 to 816 466 kilograms (600 000 to 1.8 million pounds). Results of this study indicated that an optimum size was reached at 544 311 kilograms (1.2 million pounds) payload using a 19-percent thick (normal to the leading edge) wing section swept to 35 degrees. The study showed that those parameters related to performance continued to improve as the size increased. The increase in transportation cost for airplanes carrying net payloads larger than 544 311 kilograms (1.2 million pounds) was due to increases in airplane cost because a smaller number of very large airplanes is required in the fleet.

Results of the Phase II parametric study indicated that variations of sweepback and thickness ratio at 544 311 kilograms (1.2 million pounds) payload produce relatively small changes in economics. The higher Mach number configurations with thinner wings have higher empty weights, higher prices, and require higher thrust engines, increasing their relative investment and maintenance cost. However, their lower fuel cost and higher productivity compensate for these other costs. Since there is little variation in economics with respect to Mach number, the highest Mach number was chosen. The final civil airplane configuration chosen has a 35-degree sweepback with a 16-percent thickness ratio airfoil (normal to the leading edge). It cruises at Mach 0.85 and carries 544 311 kilograms (1.2 million pounds) net payload at the design range.

Military requirements for this study were formulated by ASD/XR at WPAFB. The military configuration parametric study made maximum use of the civil configuration parametric results. The military payload was specified to be 272 155 kilograms (600 000 pounds) net with a pod installed to permit the carrying of outsized equipment equivalent to M-60 tanks or bridge loaders. In addition, the military configurations have appropriately strengthened floors, ramps, and pressurization. During the study, it was discovered that higher aspect ratios than could be provided by the three-bay baseline configuration were required to meet the military range and field length requirements. A two-bay configuration that provided satisfactory performance was developed.

The final civil configuration was improved over the best parametric configuration by installing the crew compartment in the leading edge of the wing, thus eliminating the body. In view of the airplane's design to operate between a limited number of hubs, where an alternate field is probably not available, the fuel reserve requirements were reconsidered. Standard trip fuel and holding reserves were used, but no alternate field reserves were necessary. Other relatively minor improvements were included.

The swept-wing distributed-load freighter airplane concept shows promising potential, but the optimum size occurs at about triple the payload weight of the conventional civil configuration [544 311 kilograms/181 437 kilograms (1.2 million pounds/400 000 pounds)]. The resulting ton-mile costs are one-half those of present airfreighters, and about 75 percent of the best advanced conventional design incorporating the same technology. This superiority is a result of the DLF configuration's characteristic of continuously improving aerodynamic efficiency with size, while holding or slightly improving the weight fraction. The fuel efficiency is double that of present airfreighters, and 19 percent better than the reference advanced conventional design.

The present study determined the size and shape of the DLF type for the best economics on the basis of an appropriate set of simplifying assumptions. Further studies of greater technical depth (e.g., aeroelastic effects, handling qualities, and high speed aerodynamics) are required to determine the technical limits on size.

The final distributed-load military configuration is slightly better than the conventional configuration in almost every respect. The life-cycle cost of the distributed load configuration is 21 percent less than for the reference configuration, primarily due to the light weight and the low cost associated with the simpler configuration.

2.0 INTRODUCTION

The study was conducted by the Preliminary Design department of the Boeing Commercial Airplane Company. Purpose of the study was to enumerate and quantify the benefits of a swept-wing span-distributed loading concept as applied to future commercial and military air cargo operations. The contractor has conducted the necessary engineering analysis and design studies to make a preliminary evaluation of the technical feasibility, and to demonstrate the potential economic advantages of swept-wing span-distributed loading concepts for air cargo.

This study is an extension of earlier Boeing preliminary design studies and a NASA, Langley Research Center contract, Reference 1, covering straight-wing designs. The parametric study uses the data previously generated at Boeing. These data which have been adjusted to 1990 technology levels (1995 certification), allow selection of the most economic combination of wing geometry and aircraft size as a function of the weight of payload.

The study recognized the desirability of comparing any resulting selected distrubuted-load design with an advanced conventional design at the same technology level. Further, the technology of both types incorporates the best features that can be predicted for commitment to production by 1990. Accordingly, the selected and reference conventional designs were developed for comparison with a common set of technology ground rules.

Data presented in this document for the civil configurations include the parametric study, the sensitivity studies, the engineering analyses of the selected and reference configurations, and the economic comparisons of both configurations. In addition, similar data are presented for several military versions designed for a net payload of 272 155 kilograms (600 000 pounds) and a range of 10 186 kilometres (5500 nautical miles).

The parametric study of the civil configuration covered a range of net payloads from 272 155 to 816 466 kilograms (600 000 to 1 800 000 pounds), and airplane gross weights from 0.75 to 1.95 gigagrams (1.66 to 4.30 million pounds). Wing sweep was varied from 30 to 40 degrees and thickness ratio was varied from 16 to 25 percent. The selected distributed-load configuration was chosen primarily on the basis of good economics and fuel efficiency combined with favorable characteristics relative to such intangibles as growth potential, development risks, and potential improvement.

The parametric study for the military configuration covered sweeps from 30 to 40 degrees and thickness ratios from 19 to 25 percent. The net payload was held constant at 272 155 kilograms (600 000 pounds). Two- and three-bay configurations were covered, each possessing an outsize pod capable of carrying two M-60 tanks or bridge loaders.

3.0 ABBREVIATIONS AND SYMBOLS

A_{wet}	wetted area
A_{π}	frontal area
ACLS	air cushion landing system
A/C	air conditioning
AGE	aerospace ground equipment
AIC	airplane investment cost
ALLRCM	air launched long-range cruise missile
Alt	altitude
A/P	airplane
APU	auxiliary power unit
AR	wing aspect ratio
AR_{eff}	effective aspect ratio
ATA	Air Transport Association
b_f	flap span
b_H	horizontal fin span
$b_{H/2}$	horizontal fin semi-span
b_V	vertical fin span
b_W	wingspan
$b_{W/2}$	wing semi-span
BPR	bypass ratio
BS	body station
$\bar{c}$	mean aerodynamic wing chord
$\bar{c}/4$	mean aerodynamic quarter chord
CACE	cost analysis cost estimating
C_D	drag coefficient
$C_{D_{CREEP}}$	coefficient of compressibility drag at Mach numbers below drag divergence Mach number
C_{D_i}	induced drag coefficient
$C_{D_{lift}}$	coefficient of drag due to lift
C_{D_M}	coefficient of drag due to Mach effects
C_{D_P}	coefficient profile drag due to lift
$C_{D_{P_{min}}}$	coefficient of configuration minimum parasite drag

$C_{D_{TOT}}$	total drag coefficient
$C_{D_{\pi}}$	drag coefficient based on frontal area
$C_{f_{avg}}$	average friction coefficient
C_L	lift coefficient
$C_{L_{trim}}$	trim lift coefficient
$C_{L_{\alpha}}$	lift curve slope
$C_{n_{\beta}}$	side slip yawing moment derivative
c_W	wing chord
cg	center of gravity
cu ft	cubic feet
cu m	cubic metre
DAF	dedicated air freighter
deg	degrees
dia	diameter
DLF	distributed load freighter
DOC	direct operating cost
ECS	environmental control system
EI	bending stiffness
$F_{N_{CL}}$	net thrust-climb
$F_{N_{CR}}$	net thrust-cruise
$F_{N_{T/O}}$	net thrust-takeoff
$F_{Nacelle}$	engine thrust less nacelle drag
F_{PL}	flat-plate drag
FAR	Federal Aviation Regulations
ft	feet
ft^2	square feet
FPR	fan pressure ratio
g	gram
g	acceleration of gravity
G	giga (10^9)
gal	gallons
GJ	torsional stiffness
GR/EP	graphite-epoxy
GTM	gross ton (statute) miles

HP	horsepower
HPX	horsepower extraction
ICAC	initial cruise altitude capability
IDG	integrated drive generator
IFR	inflight refueling
in.	inch
in.–lb	inch–pound
IOC	indirect operating cost
k	kilo (10^3)
K	kelvin
keas	knots equivalent air speed
kg	kilograms
KVA	kilovolt-ampere
L_{ref}	reference length
ℓ_V	vertical fin arm
lb	pound
lb/ft^2	pound per square foot
lb/ft^3	pound per cubic foot
$lb/in.^2$	pound per square inch
$lb/in.^3$	pound per cubic inch
lb/sec	pound per second
L/D	lift-to-drag ratio
L.E.	leading edge
LEMAC	body station of leading edge of the mean aerodynamic chord
LP	low pressure
m	metre
M	Mach number
M	mega (10^6)
M_C	design cruise Mach number
M_{CRIT}	critical Mach number
M_D	design dive Mach number
M_{DD}	Mach number at drag divergence
MAC	mean aerodynamic chord
MAC_V	MAC of vertical tail

MAC_W	MAC of wing
MLF	multimission large freighter
M(L/D)	Mach number times lift-to-drag ratio
M_{SUB}	subcritical Mach number
MTOW	maximum design takeoff weight
MZFW	maximum design zero fuel weight
n	design maneuver load factor
N	newton
NA	not applicable
NASA	National Aeronautics and Space Administration
nmi	nautical mile
OEW	operational empty weight
OPR	overall pressure ratio
P	peta (10^{15})
Pa	pascal (newton per square metre)
PL	payload
POL	petroleum, oil, and lubricants
psi	pounds per square inch
psig	pounds per square inch gauge
q	dynamic pressure
QTY	quantity
R	rankine
R_e	Reynolds number
RF	range factor
ROI	return on investment
RROR	reasonable rate of return
RTM	revenue ton (statute) miles
s	second
S_V	vertical fin area
S_W	wing area
SAS	stability augmentation system
SFC	specific fuel consumption
SLS	sea level static
sq in.	square inch
S.W.	streamwise

t_2	time to double amplitude
t_{30^o}	time to bank of 30^o
t/c	airfoil thickness divided by chord
$t/c \perp$	t/c normal to leading edge
TAI	thermal anti-icing
TIT	turbine inlet temperature
TOFL	takeoff field length
TSLS	thrust, sea-level static
UE	unit equipment
V	velocity, speed
V_A	design maneuvering speed
V_{APP}	approach speed
V_B	design speed for maximum gust intensity
V_C	design cruise speed
V_D	design dive speed
V_{mc_a}	air minimum control speed
V_{mc_g}	ground minimum control speed
$V_{min\ dem}$	minimum demonstration speed
V_{mo}	maximum operating speed
V_S	stall speed
V_{trim}	trim speed
$\overline{V}_V$	vertical fin volume coefficient
V_1	critical engine failure speed
w	watt
W	weight
WBS	work breakdown structure
W/S	wing loading-weight divided by wing area
$\overline{X}_{cg}$	center of gravity fraction MAC
$\overline{y}$	distance between centerline and MAC
yr	year
ZFW	zero fuel weight
$\delta_{ail_{max}}$	aileron maximum deflection
δ_f	flap deflection
δV_{limit}	change in specific parameter

Δ	change in specific parameter
$\ddot{\theta}$	pitch acceleration
Λ_W	wing sweep
ϕ	roll angle

UNITS

Measurement values employed in this document are expressed in the International System of Units (SI) as primary and U.S. customary units as secondary. The U.S. customary system of units was used for principal calculations.

4.0 GUIDELINES

4.1 GENERAL

4.1.1 TIME PERIOD

The configurations generated in this study could be available for service implementation by 1995.

4.1.2 TECHNOLOGY STATUS

The configurations include those elements of advanced technology that may be ready for production by 1990, and that have the potential for improving performance, reducing costs, and solving design or operational problems. However, laminar flow control has not been applied to the configurations.

4.1.3 CONFIGURATION VARIABLES

The configuration variables considered include the following: (1) wing thickness ratio, (2) aspect ratio, (3) wing-sweep angle, and (4) number of cargo bays.

4.1.4 SPEED

The subsonic design Mach number is as high as practical, commensurate with configuration constraints, and economic and fuel consumption considerations.

4.1.5 PROPULSION SYSTEM

The configurations employ advanced technology high-bypass-ratio turbofan engines.

4.2 CIVIL CONFIGURATIONS

4.2.1 DESIGN APPROACH

The design approach consists of a swept flying-wing concept with the payload and fuel distributed within the wing. The wing has constant cross section characteristics and a high-thickness-ratio airfoil section.

4.2.2 THROUGHPUT CAPACITY

The throughput capacity is 167.9 revenue petametre-kilograms (115 billion revenue-ton statute miles) per year.

4.2.3 RANGE

The values of range are from 5556 to 10 186 kilometres (3000 to 5500 nautical miles). The selected design value of 6667 kilometres (3600 nautical miles) was based on the high-density Paris-to-New York route.

4.2.4 PAYLOAD

The net payload weights are 272 155, 408 233, and 544 311 kilograms (600 000, 900 000, and 1 200 000 pounds) and the net payload density is 160 kilograms per cubic metre (10 pounds per cubic foot). The payload is containerized in 2.44 x 2.44 x 6.10 metre (8 x 8 x 20 foot) and/or 2.44 x 2.44 x 12.19 metre (8 x 8 x 40 foot) containers with a tare weight of 24 kilograms per cubic metre (1.5 pounds per cubic foot).

4.2.5 CARGO BAY ENVIRONMENT

The cargo bay does not require pressurization. The temperature control system is capable of maintaining a cargo bay temperature of 283 kelvins (50 degrees Fahrenheit) or greater at maximum cruise altitude.

4.2.6 TERMINAL AREA

The runway lengths are defined by a 3658 metre (12 000 foot) balanced field length requirement. Values of runway width are 61 metres (200 feet) and other higher values as required by the airplane configuration. Width of the taxiways is not a constraining factor.

4.2.7 CERTIFICATION CRITERIA

The configurations meet the requirements of the Federal Aviation Regulations for Transport Category Aircraft.

4.3 MILITARY CONFIGURATIONS

4.3.1 DESIGN APPROACH

The useful load (payload plus fuel) distribution concepts consist of combined loading in the wing and in a fuselage. The wings will have constant cross-section characteristics and a high-thickness-ratio airfoil section.

4.3.2 RANGE

The values of range are from 6482 to 12 038 kilometres (3500 to 6500 nautical miles), with a design value of 10 186 kilometres (5500 nautical miles).

4.3.3 PAYLOAD

Net payload weight is 272 155 kilograms (600 000 pounds). The payload is: (1) containerized in 2.44 x 2.44 x 6.10 metre (8 x 8 x 20 foot) and/or 2.44 x 2.44 x 12.19 metre (8 x 8 x 40 foot) containers, (2) loaded on 463L pallets, (3) military equipment that can roll on and roll off, or (4) any combination of (1), (2), and (3). Payload density for containers and pallets is 160 kilograms per cubic metre (10 pounds per cubic foot) and the tare weights are 24 kilograms per cubic metre (1.5 pounds per cubic foot) for the containers and 113 kilograms (250 pounds) for the 463L pallets.

4.3.4 CARGO BAY

A pod containing two cargo bays each measuring 4.11 metres (13-1/2 feet) high x 5.18 metres (17 feet) wide x 12.19 metres (40 feet) long was designed for carrying outsized cargo. The floor strength of this section is capable of supporting two fully equipped M-60 main battle tanks. The cargo bays have hard points throughout to provide 11 340 kilogram and 4536 kilogram (25 000 pound and 10 000 pound) tie-down points.

4.3.5 CARGO BAY ENVIRONMENT

The cargo bays have the capacity for maintaining pressure at a minimum pressure equivalent to an altitude of 5486 metres (18 000 feet) at maximum cruise altitude. The temperature control system is capable of maintaining a cargo bay temperature of 283 kelvins (50 degrees Fahrenheit) or greater at maximum cruise altitude.

4.3.6 TERMINAL AREA

Runway lengths are defined by critical field length values of 2438 to 3658 metres (8000 to 12 000 feet), with a design value of 3048 metres (10 000 feet). Critical field length is defined as the distance from brake release to the point at which the aircraft attains a height of 15.2 metres (50 feet) using all engines. Values of runway width are 61 metres (200 feet) and other higher values as required by the airplane configuration. Width of the taxiways is not a constraining factor.

4.3.7 ADDITIONAL CONSTRAINTS

The configurations possess the capability for aerial refueling.

4.3.8 CERTIFICATION CRITERIA

The configurations are designed to meet the requirements of the Military Specifications for Transport Aircraft.

4.4 ECONOMICS

4.4.1 DIRECT OPERATING COST

The 1967 Air Transportation Association (ATA) equations with coefficients updated to January 1976 experience are used to calculate direct operating cost (DOC).

Likewise, pricing and other costs are based on January 1976 dollar values. Manufacturing and development costs are estimated by the contractor's in-house methods. An aircraft production run that meets the throughput of 167.9 revenue petametre-kilograms (115 billion revenue-ton statute miles) is used for the civil configurations. This production run plus 125 aircraft is used for the civil version of the military configuration. A production run of 125 aircraft is used for the military configurations. Utilization is 4400 hours per year (available hours: 4649.3 hours for Parametrics; 5683 hours for Finals). Load factor is 85 percent of the gross payload (net payload plus container weight). Fuel price is varied at 97.7, 118.9, and 158.5 dollars per cubic metre (37, 45, and 60 cents per gallon). A crew of two is assumed.

4.4.2 LIFE-CYCLE COSTS

Cost estimates are based on detail cost data sufficient to establish reasonableness, realism, and completeness. The methodology for these cost estimates is contained in Appendix B. Cost estimates include RDT&E, production, and 20 years of steady-state operations and support. AFR 800-2 was used as a guide for the program phases; i.e., validation, full-scale development, and deployment. Military Standard 881, Work Breakdown Structure for Defense Items, was used as a guide. The cost estimates of the aircraft are developed to at least the third level. The OSD-Operations and Cost Development Guide and AFR 173-10 were used as guides. All cost estimates are stated in terms of January 1976 dollars with no allowance for future inflation or escalation. Fuel cost is 97.7 dollars per cubic metre (37 cents per gallon). The validation, full-scale development, and production estimates are based on the following number of aircraft: validation, two; full-scale development, four; and production, 125. The utilization rate for the military configuration is 1000 hours per year.

4.4.3 NONRECURRING AND RECURRING COSTS

Nonrecurring costs include the following: (1) preliminary design encompassing the translation of weapon systems concepts and requirements into specifications for new systems, as well as for major modifications of existing systems; (2) design engineering that entails the specification and preparation of the original set of detailed drawings for new systems, as well as for major modifications of existing systems; (3) tests, test spares, and mockups, regardless of when they occur in the life of the program; (4) all partially completed WBS elements manufactured for tests; (5) costs of all tooling, manufacturing, and procurement specifically incurred by performing tests or initiating developments, except for the manufacture of complete units during the development program; and (6) the initial set of tools and all duplicate tools produced to permit the attainment of a specific rate of production for a program.

Recurring costs include the following: (1) engineering redesign, associated evaluation, and liaison; (2) complete WBS elements produced either for test or operation use; (3) tool maintenance, modification, rework, and replacement; (4) training all service personnel to operate and maintain equipment; and (5) reproduction and updating of technical data and manuals.

4.5 REFERENCE CONFIGURATIONS

4.5.1 CIVIL REFERENCE CONFIGURATION

The civil reference configuration is an advanced fuselage-loaded cargo aircraft. The general and civil configuration guidelines (Sections 4.1 and 4.2) are applied. The reference configuration is capable of operating from runways defined by a 3658 metre (12 000 foot) balanced length. The range of the civil reference configuration is the same as for the civil study configuration.

4.5.2 MILITARY REFERENCE CONFIGURATION

The military reference configuration is an advanced fuselage-loaded military transport aircraft. The general and military guidelines (Sections 4.1 and 4.3) are applied. The range and terminal area performance of the military reference and study configurations are the same.

5.0 CONTRACTOR TASKS

5.1 INTRODUCTION

The parametric study is the first contractor task. It is a wing geometry and sizing exercise to determing the combinations of wing span, chord, sweep, and thickness ratio that result in the most favorable configuration characteristics that warrant further study and refinement. The parametric study approach shows the design background and configuration constraints, chooses a baseline airplane, and defines the configuration matrix for the study. Parametric study results subsequently show the resulting airplane characteristics, performance, and economics.

A configuration was selected and analyzed from results and conclusions of the parametric study. The rationale for the selection is shown first, followed by a detailed definition of the configuration. Next, the 1990 technology is defined and then applied to the selected configuration. The resulting performance of this airplane is described in this document.

The same technical cycle is repeated for a reference conventional airplane but in somewhat less detail than for the selected distributed-load configuration. The configuration and the 1990 technology are then defined and the technology applications analyzed.

Finally, the two concepts are compared. A technical assessment of their relative performance, productivity, and fuel consumption is covered in this publication. Economic comparisons are shown, including sensitivities to economic assumptions and the effect of airplane size.

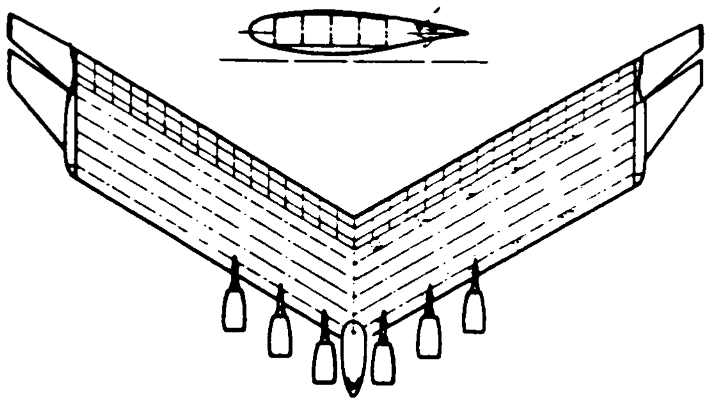
Areas for further refinement and study are discussed and study conclusions are stated.

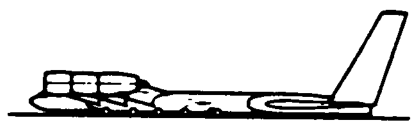
5.2 PARAMETRIC STUDY

The projected air cargo market of 167.9 revenue petametre-kilograms (115 billion RTM) per year in 1995 would support much larger aircraft than in use today. The economics of conventional aircraft with separate wing, body, and tail components improve with size but appear to reach an optimum at a gross weight of around 450 000 kilograms (1 000 000 pounds). Aircraft larger than this have decreasing efficiencies because the slight improvements in aerodynamics with size are more than offset by the progressively increasing wing weights resulting from large wing root bending moments.

It has been recognized for some time that placing all of the payload and fuel in the wing and distributing these loads along the span would result in a much lighter and more efficient airplane. However, the opportunity to exploit this principle for commercial cargo airplanes requires airplanes sufficiently large to accommodate a cargo of standard commercial containers with a cross-section of 2.44 by 2.44 metres (8 x 8 feet) that can be placed entirely within the wing. Boeing studies have indicated that distributed-load commercial freight airplanes of 0.68 gigagrams (1.5 million pounds) gross weight and higher could be configured with the cargo completely in the wings and could compete with large advanced conventional freight airplane designs. Figure 1 indicates the typical configuration that evolved from these design studies. This configuration will serve as the baseline for the parametric study.

The data base for the current swept-wing configuration includes extensive background in straight wing designs using advanced honeycomb construction concepts. The swept-wing design uses the





Range, km(nmi)	6667	(3600)
Net payload, kg (lb)	408 233	(900 000)
MTOW, kg(lb)	1 067 757	(2 354 000)
OEW, kg(lb)	305 767	(674 100)
Wing area, $m^2(ft^2)$	2502	(26 933)
Basic aspect ratio	4.567	
Effective aspect ratio	7.73	
Basic wing span, m(ft)	107	(351)
Total wing span, m(ft)	124	(407)
Sweep, degrees	30.0	
t/c⊥	0.19	
Cruise Mach number	0.78	
Engines BPR	9.5	
TSLS, N(lb)	400 340	(93 000)

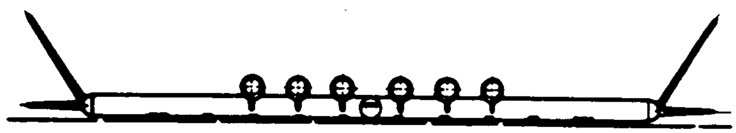


Figure 1 Civil Baseline General Arrangement—759-189-1

same construction method as the straight-wing design, with the exception of a wing root joint for attaching two wing panels at the airplane centerline. The Boeing data base is on a straight wing unpressurized distributed-load airplane. The constant chord design of the wing helps to reduce airplane construction costs by simplifying engineering and tooling, and by promoting commonality of parts throughout the airplane.

Configuration constraints are as follows:

- Fully distributed load—A state in which the entire payload is contained within the wing contours and distributed from tip to tip.

- Container capability—To accommodate standard commercial containers of 2.44 x 2.44 x 6.10 metres (8 x 8 x 20 feet) a 2.54-metre (100-inch) inside height is required at the corners of the bays.

- Advanced-wing section (High t/c)—Baseline 759-189-1 has 0.19 thickness ratio (normal to L.E.) wing section with a cruise speed of M = 0.78.

• Fly-by-wire, hard SAS, active controls—Boeing experience with flight critical stability augmentation systems (SST program) indicated the feasibility of balancing the airplane to a static longitudinal instability level corresponding to unaugmented time to double amplitude as low as two seconds. Active controls modulated by a digital computer will probably be required to properly exploit the low bending moments achievable in level flight when the airplane is upset by gusts or maneuvers.

Performance requirements:

	Civil	Military
Net payload	272 155 to 544 311 kg (600 000 to 1 200 000 lb)	272 155 kg (600 000 lb)
Net payload density	160 kg per cu m (10 lb per cu ft)	160 kg per cu m (10 lb per cu ft)
Payload containers	2.44 x 2.44 x 6.10 m (8 x 8 x 20 ft)	2.44 x 2.44 x 6.10 m (8 x 8 x 20 ft)
Container tare	24 kg per cu m (1.5 lb per cu ft)	24 kg per cu m (1.5 lb per cu ft)
Range	5556 to 10 186 km (3000 to 5500 nmi)	6482 to 12 038 km (3500 to 6500 nmi)
Design range	6667 km (3600 nmi)	10 186 km (5500 nmi)
Takeoff field length	3658 m (12 000 ft)	3048 m (10 000 ft)
Cargo compartment dimensions	8 x 8 containers	4.11 m (13.5 ft) high 5.18 m (27 ft) wide 24.38 m (80 ft) long
Floor strength		Support two M-60 tanks

1990 technology/1995 certification

Baseline Airplane Definition—Using these constraints and background, the airplane chosen as the baseline is shown on the general arrangement drawing, Figure 1. The wing cross-section contains four unpressurized cargo compartments or bays, each large enough to house standard containers. The resulting chord of the 0.19-thickness-ratio airfoils is 23.40 metres (76.79 feet), which, with the 106.91-metre (350.74-feet) span, yields an aspect ratio of 4.57. Engines having a sea-level static thrust of 414 kilonewtons (93 000 pounds) each are used. They are located above the wing to permit short, light landing gear and to keep the cargo floors close to the ground. The tip fins are sized for a static stability level corresponding to an unaugmented divergence time to double amplitude of two seconds. The main landing gear are arranged in pairs, one forward and one aft of the main wing box at spanwise stations. Each gear is steerable for crosswind conditions and has a long oleo stroke to adjust for runway contour variations. Some are powered to provide precise maneuverability.

Parametric Study Geometry Trades—A simplified computer program has been developed that will construct an airfoil cross-section to enclose any arbitrary number of bays while accommodating the structural requirement at the corners of the envelope. The program then sizes the span of the airplane to enclose the number of containers required for the desired payload. The airplane aspect ratio is thus a fallout of the number of bays and the number of containers to be carried. In the next stage, the program locates the required number of engines on a modular schedule accounting for the rib spacing module established for the wing structure. The landing gear is similarly located. Potential fuel volumes are calculated and the geometry data base is constructed to provide input for a weights program that calculates the weight and balance of the airplane. Center of gravity limits are provided. The program then constructs a balance diagram and chooses the fuel tanks required to bring the airplane's cg to the desired location for minimum trim drag. This computer program enabled the development of a parametric series of airplanes with consistent variables throughout the study and at a minimal expenditure of manpower.

5.2.1 TECHNOLOGY DEFINITION AND APPLICATION

5.2.1.1 Aerodynamics

The advanced technology (1990) definitions in the following discussion are projections consistent with other Boeing in-house work.

The advances chosen for this study include improved airfoils, tip fins, fully active control systems, and reductions in drag due to interference, roughness and excrescence. These advances are directly related to those evaluated in the previous DLF study (Reference 1).

The increase in M(L/D) for reduced roughness and excrescence was a result of the increased use of composites in these aircraft. Because of the nature of the construction of the distributed-load freighters (i.e., composite honeycomb) a five percent reduction in roughness and excrescence could be realized. The inherent consistency of the surfaces coupled with the reduction in the number of gaps, joints and extraneous bumps leave very few areas in which drag may be accrued.

Application of advanced aerodynamic configuration analysis tools has already demonstrated that wing-nacelle-strut interference effects can be all but eliminated by proper contouring and fairings. The placement of engines above the wing leading edge is deemed to represent a more difficult installation problem, but a solution is assumed possible.

Thick airfoil studies previously conducted under Boeing IR&D indicated increases of 0.02 in critical Mach number, as was noted in Reference 1. Another 0.02 in Mach number (for 1990) is predicted with further airfoil development. The total Mach improvement of 0.04 is applied to the aircraft studied in this report, but further investigation in this area is required in order to obtain a solid data base.

An increase of two drag counts was assessed for airfoil base drag based on the presence of a clipped trailing edge. This is offset by an approximate 2722 kilogram (6000 pound) reduction in trailing-edge weight.

For the purpose of the parametric study, the factors not considered in the aerodynamic drag estimation were:

1) Pitching moments due to wing camber

2) Pitching moments due to engine thrust

3) Overwing blowing effects due to the engines

Wing camber will generally give rise to higher nose-down pitching moments that must be trimmed out at the expense of trim drag. However, camber shifts the profile drag with lift such that a reduction in wing profile drag with respect to lift coefficient is attained. Since these two effects are compensating, assumption 1) was justified. Similarly, the effects of assumptions 2) and 3) are basically compensating in that the overwing blowing creates a nose-up pitching moment whereas the direct thrust effect has the opposite effect. Net bending moments were not considered in the parametric study since previous work had indicated relatively small drag penalties associated with controlling their levels. For the final configurations it was also assumed that the net bending moment would not be specifically constrained for the drag estimate.

The areas in which the preceding advanced technology was applied and its relationship to the final output are given as follows (see Appendix A for a detailed methodology):

1) Configuration geometry is received

2) Analysis of aircraft is carried out by respective staffs

- Weights
- Propulsion
- Flight controls
- Aerodynamics
  a) Minimum parasite drag
     - Roughness and excrescence reduction
     - Elimination of wing-nacelle-strut interference
  b) High speed cruise polars
     - 0.04 cruise Mach increase
  c) Low speed polars
  d) Flap/trim/induced drag
     - Fully active control systems to obtain optimum geometry
  e) Mission analysis

3) Thumbprint cycling
   a) Weights and aerodynamic thumbprint output check

4) Final cycled aircraft performance

5.2.1.2 Propulsion System

Criteria and procedures for selecting the engine cycle for the DLF are the same as those reported in Reference 1. In the present study, the engine technology has been extended into the post-1990 entry into service time period and the DLF airplane speed has been increased to Mach 0.85. A review of data presented in Reference 1 indicated that the fan pressure ratio and resulting bypass ratio selected for the Reference 1 engine would be appropriate for the current study at airplane M= 0.85 conditions.

The engine cycle established for the Reference 1 study and used for airplane performance is as follows:

- FPR = 1.6 (geared fan)

- OPR = 40:1

- Standard day critical TIT = 1528 K (2750°R)

- BPR = 9.5

Uninstalled engine design point characteristics for the Reference 1 engine are as follows:

M = 0.74 at 9144 m (30 000 ft)	
Cycle as noted above	
Maximum cruise net thrust	58 672 N (13 190 lb)
Maximum cruise SFC	0.0509 kg/hr/N (0.4988 lb/hr/lb)
Engine weight	3312 kg (7301 lb)
Engine length	2.53 m (99.8 in.)
Fan diameter	2.63 m (103.4 in.)
LP turbine diameter	1.25 m (49.3 in.)
SLS takeoff thrust	226 859 N (51 000 lb)

These estimated 1990 technology data were compared to recent 1990 engine data submitted by the engine manufacturers. Only slight differences were apparent as to the selection of FPR, OPR, and BPR.

Installation correction factors were determined for this engine to account for the effects on performance of the flight installation covering inlet, fan duct, exhaust nozzle, horsepower extraction, and airbleed. Estimated installed engine performance data for takeoff and climb are taken from Reference 1, and Figure 2 presents the cruise data with additional Mach number coverage. These data were used as the basis for the DLF studies and were scaled as appropriate to provide the engine size required for the airplane.

Since the previous study (in Reference 1) for selecting the appropriate fan pressure ratio was conducted at M = 0.78, a question arose as to the applicability of the previously indicated cycle for the M = 0.85 flight condition of the current study. Therefore, utilizing the previously indicated component efficiencies, OPR and TIT, a sensitivity study was conducted to determine the optimum FPR and bypass ratio (BPR) for the Mach 0.85 DLF airplane. Installed performance and pod weight data were calculated for gear-driven and direct-drive turbofan engines. Discussions were held with the engine companies to confirm the validity of the engine performance and pod weight trends. The incremental installed SFC and pod weight trends relative to a 1.6 FPR, 9.5 BPR, gear-driven turbofan are shown in Figure 3. The installed SFC for both gear-driven and direct-drive turbofans tend to increase with increasing FPR. With all engines sized for constant cruise thrust, pod weight decreases with increasing FPR for both engine cycles.

Payload sensitivities of the DLF airplane at constant range for changes in installed SFC and pod weight were developed as follows:

- 1% SFC = 3266 kg (7200 lb) payload

- 1 kg pod weight = 1 kg payload

- Design payload = 0.6332 Mg (1.396 x 10^6 lb)

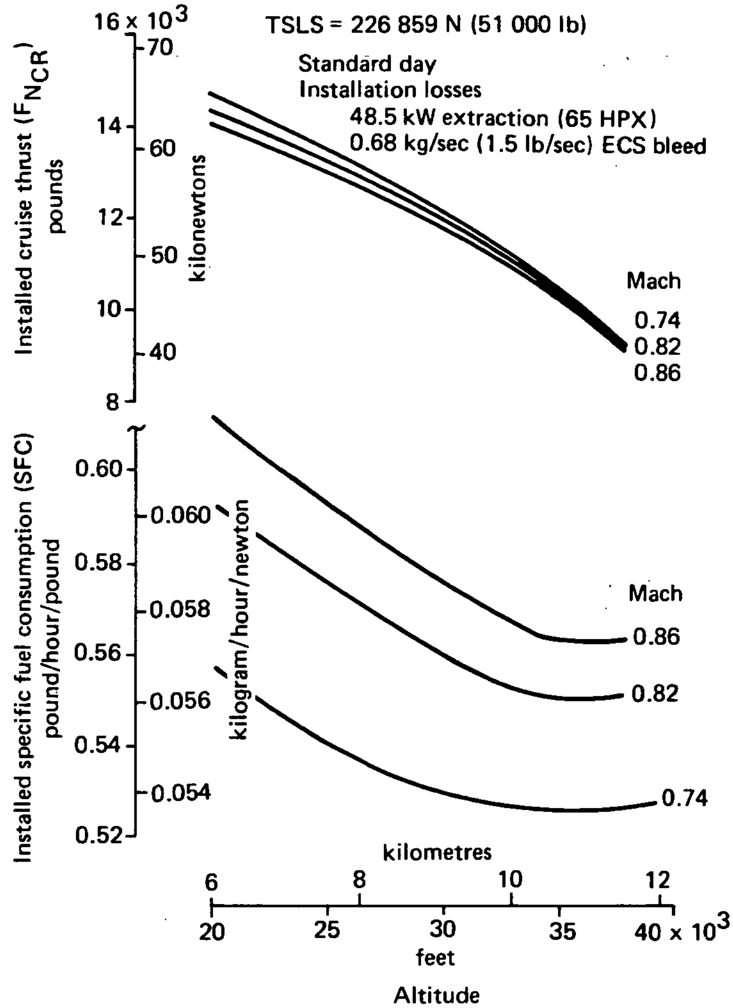


Figure 2   1990 and Post-1990 Technology Engine, Installed Cruise Performance

All engines were sized for a constant cruise thrust at 10 363 metres (34 000 feet) and M = 0.85. The payload sensitivity of increasing FPR including the effects of nacelle drag are indicated in Figure 4. As shown, the highest payload fraction results from a gear-driven turbofan with FPR of 1.6. With this cycle, the reduced pod weight and nacelle drag are offset by increased SFC as FPR increases. Based on these trends, the initial engine cycle selected for DLF studies with a gear-driven fan and the following characteristics is considered the optimum cycle for this airplane study.

• Fan pressure ratio	1.6
• Overall pressure ratio	40
• Bypass ratio	9.5
• Turbine inlet temperature	1528 K (2750°R) at maximum cruise power

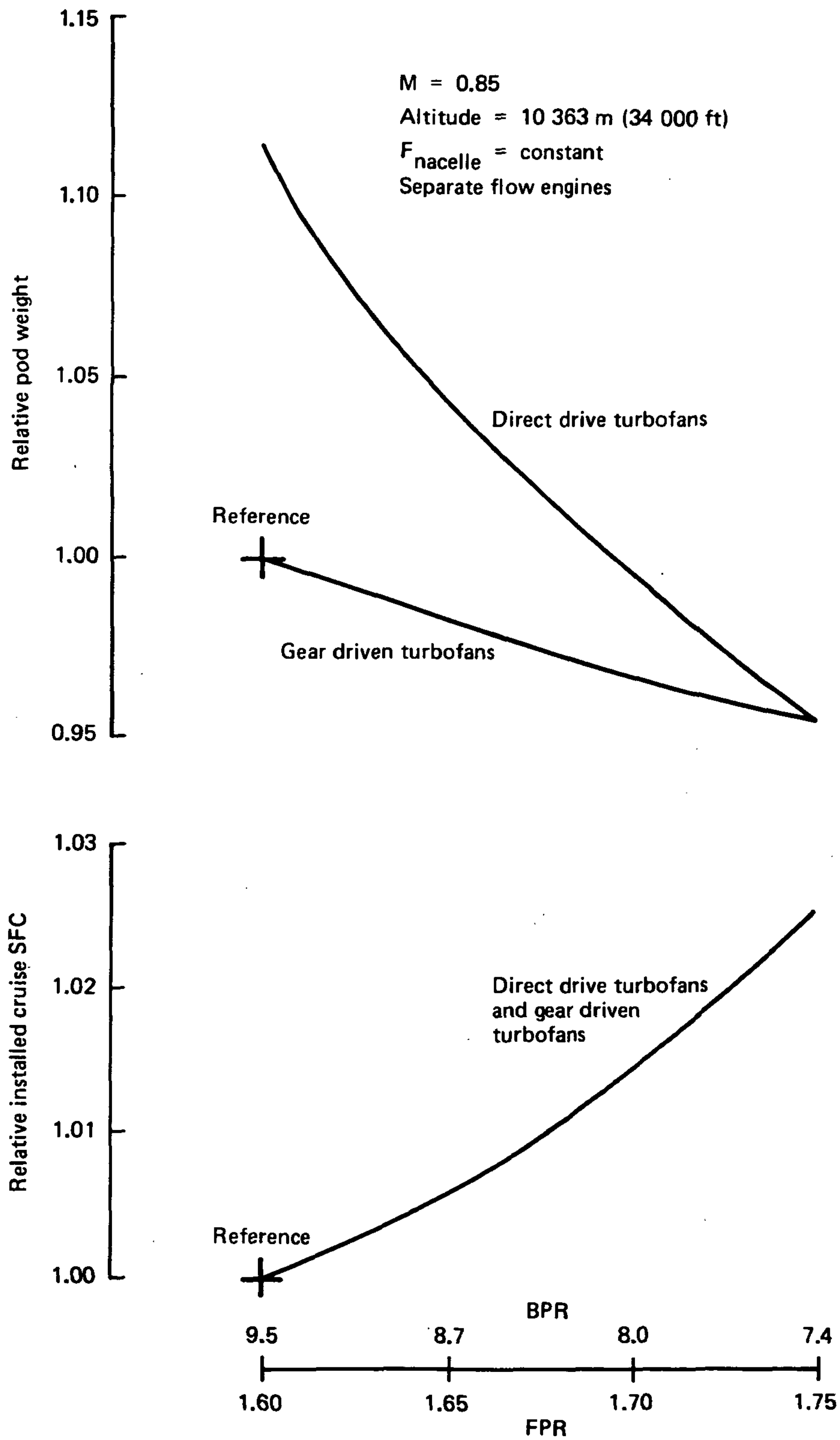


Figure 3 Relative Engine Characteristics—1990 and Post-1990 Technology

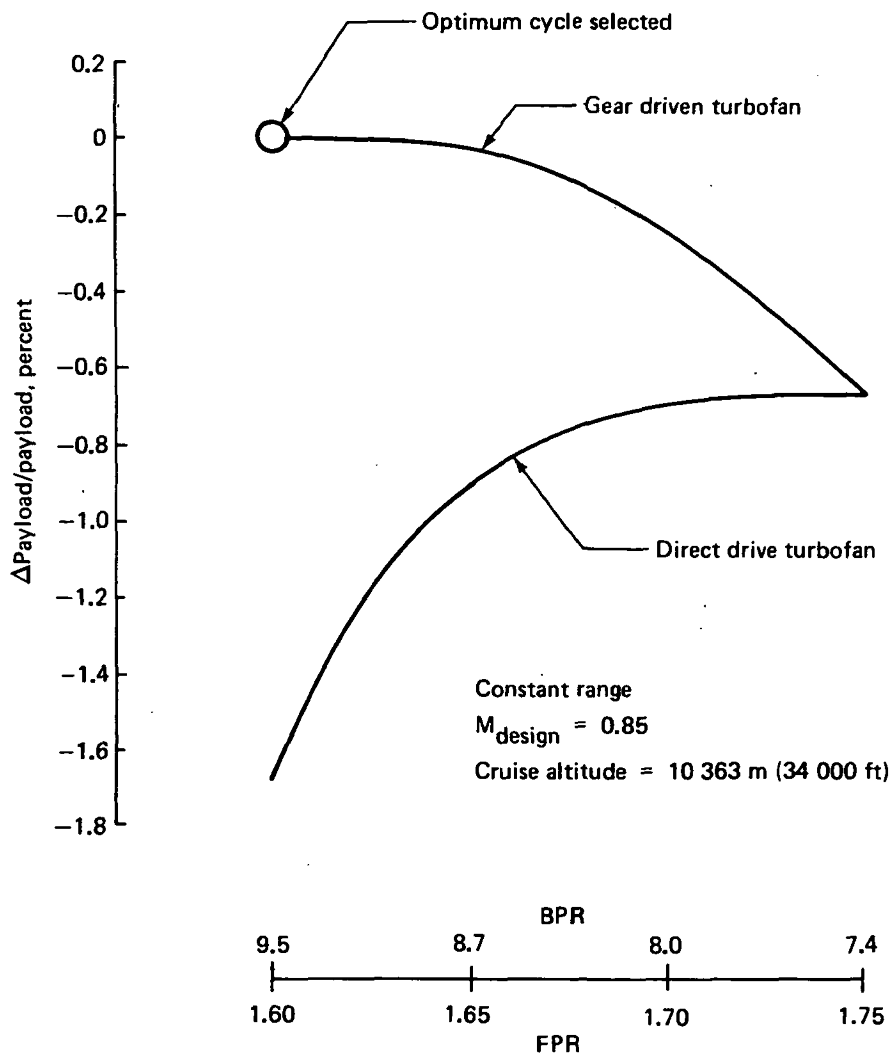


Figure 4 Payload Sensitivity—1990 and Post-1990 Technology

5.2.1.3 Structures

Structural requirements of the distributed load freighter are highly influenced by the degree of control that is available to limit the loads under all possible flight and ground conditions. The ground conditions are adjusted by placement of the landing gear and design of the oleo system to fall within the structural capabilities of the wing as required by flight conditions. The 1-g flight conditions are controlled to provide an optimum balance between minimum induced drag and minimum structural weight by adjusting the span airload distribution. This is covered in more detail in Section 5.3.2.4. The uniform skin gage resulting from this type of optimization is advantageous in producing a low cost lightweight structure. The structure is basically composite honeycomb skin planks bonded to composite honeycomb ribs and spares with appropriate fail-safe provisions between each skin plank.

Aeroelastic effects are considered to be very pronounced, to the extent that a rigid loads analysis conducted on the basis of fixed controls is probably of little value. Analysis of the effect of active full-time, full-span, digitally controlled aerodynamic surfaces on elastic structure was beyond the scope of this study.

The parametric studies were based on advanced structural concepts representative of an airplane designed for certification in 1995. Graphite-epoxy is used for most of the primary structure. Details of the proposed structural design concepts and materials are included in Table 2, Section 5.2.1.4.

5.2.1.4 Weight and Balance

Weight data for airplanes in the parametric study phase are presented in Table 1. A typical center of gravity management and loadability diagram is shown in Figure 5 for the 759-189-1 airplane. The figure illustrates the typical degree of management versatility on swept wing DLFs. All cg management between zero fuel weight (ZFW) and maximum design takeoff weight (MTOW) is done by programming fuel usage. Since the DLF carries both fuel and payload in the wing, it is not constrained by a maximum design zero fuel weight (MZFW). Therefore, fuel and payload can be freely

Table 1 Weight Data for Parametric Configurations (SI Units)

Model	OEW*, kg	Cycled (Sized) MTOW, kg	TSLS N/engine	No. of Engines	OEW* per 20-foot Container, kg
759-189-1	305 767	1 067 757	413 685	6	3992
759-189-2	227 431	783 354	493 755	4	4454
759-189-3	385 735	1 361 684	378 099	8	3777
759-190-1	236 231	754 324	444 822	4	4627
759-190-2	330 578	1 060 045	400 340	6	4315
759-190-3					
759-191-1	315 564	1 130 670	462 615	6	4120
759-191-2	391 360	1 414 755	409 236	8	3832
759-192	244 668	772 332	489 304	4	4792
759-193-1	310 847	1 078 189	431 478	6	4058
759-193-2	389 636	1 364 860	386 995	8	3816
759-193-3	471 474	1 661 509	364 754	10	3694
759-193-4	550 208	1 949 540	348 296	12	3592
759-194	412 497	1 368 488	374 985	8	4039
759-195	418 430	1 377 107	391 444	8	4097
759-196	365 958	1 343 994	369 202	8	3594
759-197	401 974	1 392 075	420 357	8	3936
759-198	378 750	1 366 674	404 788	8	3709
759-199	369 315	1 392 075	411 461	8	3616
759-204M	335 046	1 081 137	406 834	8	
759-208M	312 751	985 021	307 684	8	
759-209M	327 906	994 275	296 474	8	
759-210M	362 212	1 004 884	321 829	8	

Note: All values are cycled (sized)

*** 1990 technology**

interchanged within the bounds dictated by the cg management. The wide longitudinal cg range variation is available due to unused space in the leading and trailing edges. Fuel tanks can be located to achieve the cg envelope shown in Figure 5 due to the potential fuel tank space and locations available.

Unit weights based upon the 759-183 airplane analysis (Reference 1) were applied to the swept-wing airplanes in this study. In order to maintain consistency and achieve minimum turn-around time, weight equations for civil airplanes were programmed on an electronic computer. Five equations were developed for weight estimation of the wing, identifying leading edge, box, fixed trailing edge, movable trailing edge, and tip structure. Adjustments were included for thickness ratio, number of cargo bays (rib length), maximum dynamic pressure, and graphite composite material. Allowable stresses and strength capability were not developed for the parametric study, but are representative of those presented in Reference 1. Stiffness requirements have not been defined. No adjustment was made for sweep angle. The vertical tail unit weight is representative of a detailed loads and structural analysis. The body weight was derived by a detailed component analysis (military airplanes). Standard and operational item weight equations were derived for crew and crew

Table 1a Weight Data for Parametric Configurations (Customary Units)

Model	OEW*, lb	Cycled (Sized) MTOW, lb x 10^6	TSLS lb/engine	No. of Engines	OEW* per 20-foot Container, lb
759-189-1	674 100	2.354	93 000	6	8800
759-189-2	501 400	1.727	111 000	4	9820
759-189-3	850 400	3.002	85 000	8	8327
759-190-1	520 800	1.663	100 000	4	10 200
759-190-2	728 800	2.337	90 000	6	9514
759-190-3					
759-191-1	695 700	2.4927	104 000	6	9082
759-191-2	862 800	3.119	92 000	8	8449
759-192	539 400	1.7027	110 000	4	10 564
759-193-1	685 300	2.377	97 000	6	8946
759-193-2	859 000	3.009	87 000	8	8412
759-193-3	1 039 400	3.663	82 600	10	8143
759-193-4	1 213 000	4.298	78 300	12	7919
759-194	909 400	3.017	84 300	8	8905
759-195	922 500	3.036	88 000	8	9033
759-196	806 800	2.963	83 000	8	7901
759-197	886 200	3.069	94 500	8	8678
759-198	835 000	3.013	91 000	8	8177
759-199	814 200	3.067	92 500	8	7973
759-204M	738 650	2.3835	91 460	8	--
759-208M	689 500	2.1716	69 170	8	--
759-209M	722 910	2.192	66 650	8	--
759-210M	798 540	2.2154	72 350	8	--

Note: All values are cycled (sized)

*** 1990 technology**

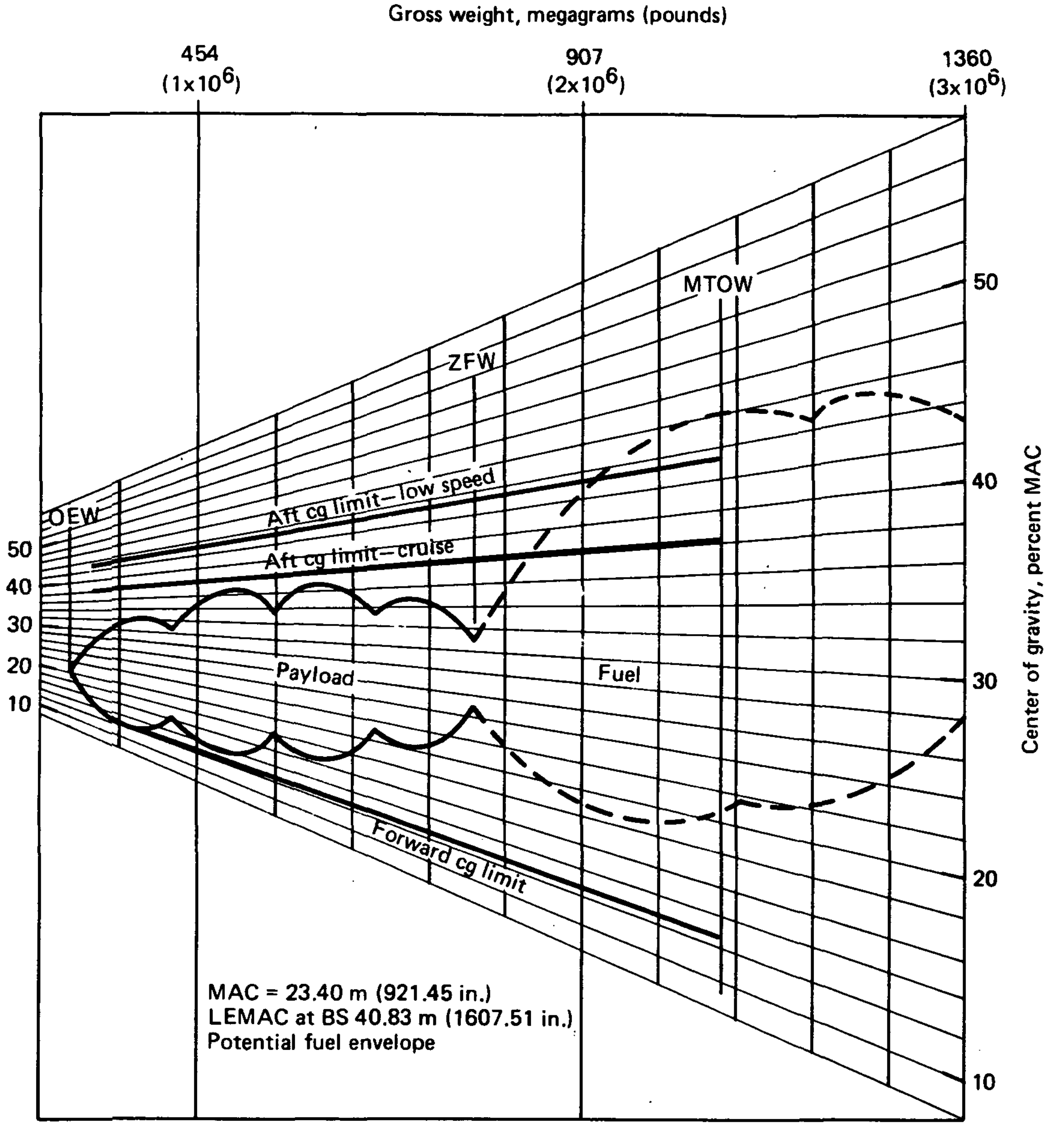


Figure 5 Center of Gravity Management—Model 759-189-1

services, engine oil, and unusable fuel. All other functional group weights were derived from an equation for the entire functional group. Detailed components for military airplanes were identified and analyzed.

A conventional analysis was used for cg management.

Some risk was assumed in establishing an absolute level of weight because the detailed airplane definition is beyond the scope of this study contract. Analyses would be required to establish a baseline airplane that definitely meets minimum design requirements and criteria. Differences between configurations were accurately represented.

Ground rules, airplane definitions, design concepts, structural materials, design requirements, and design criteria assumed as the basis for weight data are shown in Table 2. Advanced technology definitions are expanded in Table 3 with corresponding weight increments from current technology.

Table 2 Weight Definition Assumptions for Parametric Study Configurations

Item	Definition
Configuration/interior arrangement	
Airplane Geometry	Per computer printouts; military airplane data per general arrangement (see Figure 32)
Fuel capacity	Entire leading and trailing edge bays provided except for (1) APU dry bay, (2) Environmental controls dry bay, (3) Landing gear bays, and (4) Outboard dry bay
Design concepts/materials	
Structural and systems concepts for performance baseline	1990 technology (1995 certification)
a. Wing, horizontal tail, vertical tail	Graphite-epoxyhoneycomb skins. Graphite-epoxy (pultrusion) chords, tubes and fittings where feasible. Multi-spar slab horizontal and vertical tails.
b. Body	Conventional aluminum semi-monocoque
c. Landing gear	Conventional steel, two wheel truck geometrically similar to 747 nose gear, modified for DLF loads and brakes added. No powered wheels.
d. Brakes	Conventional disc, carbon
e. Nacelle and strut	Conventional aluminum, stiffened web
f. Hydraulic actuator	Conventional, steel
g. Thrust reversers	Fan stage only
h. Wing leading edge	Aluminum honeycomb sandwich for bird strike protection
i. Engine burst protection	Kevlar membrane on all engines (for interchange-ability considerations)
j. Fuel system	Scavenger pumps, integral tanks (no sealant required)
k. Hydraulic system	Conventional, 20.7 MPa (3000 psi)
l. Anti-icing	Conventional, engine inlet only
m. APU system	Rubberized L-1011 system with PT 6 engine, 746 kW (1000 HP)
n. Flight controls system	Same as 747, except full time flight critical stability augmentation system
o. Signal wires	Conventional (not fiber optics)

*Selected straight-wing configuration from Reference 1.

Table 2 Weight Definition Assumptions for Parametric Study Configurations (Continued)

Item	Definition
p. Cargo compartment floor	
Commercial airplane	None (container handling system only)
Military airplane	None in wing; permanent in pod
q. Cargo lane width	
Commercial airplane	2.62m (103 in.) (tension ties used in wing)
Military airplane	2.92m (115 in.)
Criteria/requirements	
Commercial airplane compliance	FAR
Military airplane compliance	FAR and MIL specifications
Design criteria	Assumed same criteria as 759-183* configuration
Static loads	Not evaluated. Assumed equivalent to 759-183.
Dynamic/aeroelastic loads	Not evaluated. Assumed not critical.
Minimum gage requirements	Same as 759-183
Bending/torsional stiffness (EI/GJ) requirements	Not evaluated. Assumed not critical.
Kinetic energy in fan stage for engine burst	4.4MNm (39×10^6 in-lb) for TSLS = 266 893 N (60 000 lb)
Maximum dynamic pressure (q)	Proportional to V_{mo} of 759-183
Cargo compartment pressure	
Differential—commercial airplane	Zero
—military airplane	31 kPa (4.5 psi) -5486m (18 000 ft) equivalent cabin altitude
Data source	
Weight data baseline	759-183 (1985 technology)
Balance criteria	
Cargo centroid variation	±5% container length and ±10% container width
OEW variation	±1% MAC
Center of gravity management	Fuel use during cruise to achieve cg for maximum aerodynamic efficiency
Crew comfort level and safety	
Number of flight crew	Two
Services	Food warming and beverage provisions included
Overwater equipment	Included

*Selected straight-wing configuration from Reference 1

Table 3 Advanced (1990) Technology Definition and Weight Improvement for Parametric Study Airplanes

Component	Definition	Weight Improvement ~% [1]
Wing upper and lower surfaces, ribs, spars	Graphite-epoxy honeycomb sandwich. Bonded fastening	-8
Wing leading edge	Aluminum honeycomb sandwich	0
Wing trailing edge	Graphite-epoxy honeycomb sandwich	0
Control surfaces	Graphite-epoxy honeycomb sandwich	-25
Vertical tail box	Graphite-epoxy honeycomb sandwich	-25
Horizontal tail box (DLF wing tip fin)	Graphite-epoxy honeycomb	-20
Landing gears	Conventional steel	0
Brake assembly	Conventional disc, carbon	-40
Body	Conventional aluminum semi-monocoque	0
Nacelle	Conventional aluminum stiffened web	0
Strut	Conventional aluminum stiffened web	0
Lift augmentation	None	—
Maneuver load control	Yes	[2]

[1] Weight increment from current technology - percent of component affected.

[2] Included in baseline.

5.2.1.5 Stability and Control

Longitudinal balance, stability, and control studies were performed with a constant wing sweep of 30 degrees with variables of wing aspect ratio and wing tip tail span. Forward and aft cg locations were determined for low-speed trim limit and both high-speed and low-speed stability limits. Studies of cg range and nosedown pitch recovery at stall attitude, were conducted at a base configuration only.

Ground Rules—The pitch requirements for longitudinal analyses used the following ground rules:

- Flight critical stability augmentation system
- Aeroelastic effects not considered
- Full span trailing edge control surfaces
  - Inboard 50 percent span used for high lift and trim
  - Outboard 50 percent span used as elevons
- Preliminary balance, including sizing of horizontal tail at wing tip, used an aft cg limit for unaugmented airplane where the unstable pitch divergence time to double amplitude was not less than two seconds ($t_2 \geqslant 2$)—flight critical SAS is used to stabilize the airplane.
- Stall recovery $\ddot{\theta} \geqslant -0.08$ radians/second^2 at V_S

Vertical tail sizing was conducted for a single baseline configuration only with the following ground rules:

- Directional stability (C_{n_β}) no less than 0.0010 per degree for the unaugmented airplane
- Engine-out trim limit at low speed using all moving vertical tail with maximum travel of $\pm 15^\circ$ for the following conditions:

  $V_{mc} \leqslant V_1$ where V_1 is taken at 75 percent MTOW
  $V_{mc_a} \leqslant 1.1\ V_S$ where V_S is taken at 1.25 OEW

- Crosswind landing [15.43 m/s (30 knots) wind at 90°] use crosswind gear

Longitudinal Stability and Control—Longitudinal dynamic disturbance time histories were calculated for the unaugmented rigid airplane at low-speed and cruise-speed conditions. Figure 6 shows the low-speed data for the 759-189-1 configuration and indicates the choice of the aft cg limit with the unaugmented time to double amplitude of two seconds ($t_2 = 2$).

Stall recovery capability at aft cg is shown in Figure 7 to understand the influence of the horizontal wing tip extensions on allowable aft cg. This analysis was conducted on the 759-189-2 configuration, which is similar to the 759-189-1 but carries only 272 155 kilograms (600 000 pounds) of net payload. The complete longitudinal balance for the 759-189-1 rigid configuration is shown in Figure 8 as a function of allowable cg range versus size of horizontal wing tip extension. The aft limit is high-speed stability for the unaugmented rigid airplane (t_2 = 2 seconds) and the forward limit is climbout trim at maximum gross weight. Takeoff condition is highly influenced by ground effect. It is highly advantageous to take off at the far aft cg location to permit the airplane to rise off the ground in a trimmed condition with large deflections of the trailing-edge surfaces.

The climbout trim limit used for the forward cg limit assumes that the airplane has already rotated to climbout altitude following the runway lift-off. No detailed analyses of this transition from lift-off in ground effect to climbout in free air have been made; however, wind tunnel tests have been conducted to confirm a favorable ground effect.

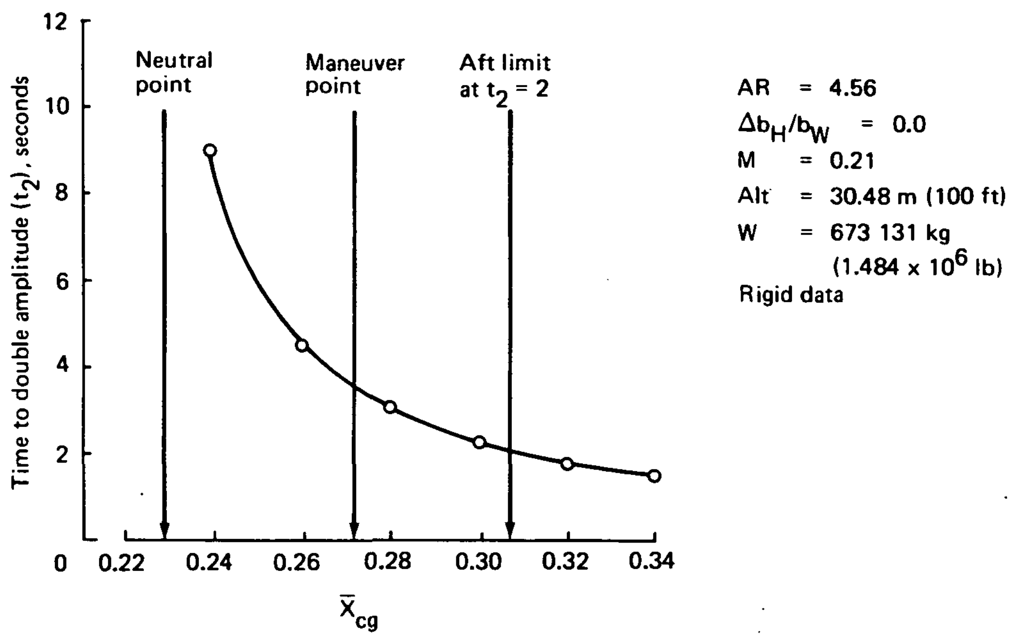


Figure 6 Unaugmented Longitudinal Stability—Model 759-189-1

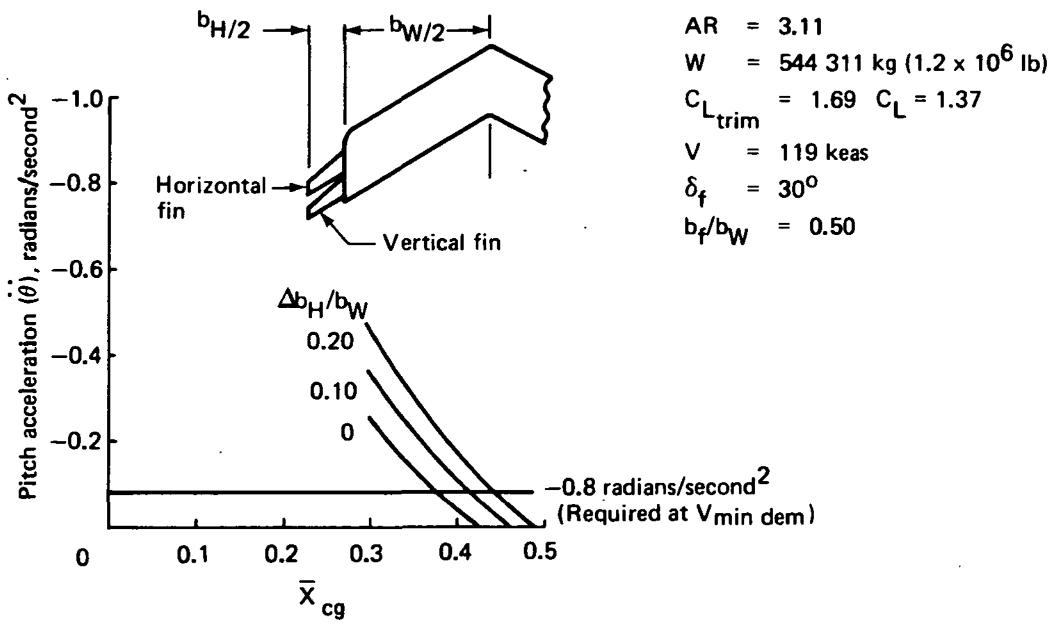


Figure 7 Nose-down Pitch Recovery at Stall—Model 759-189-2

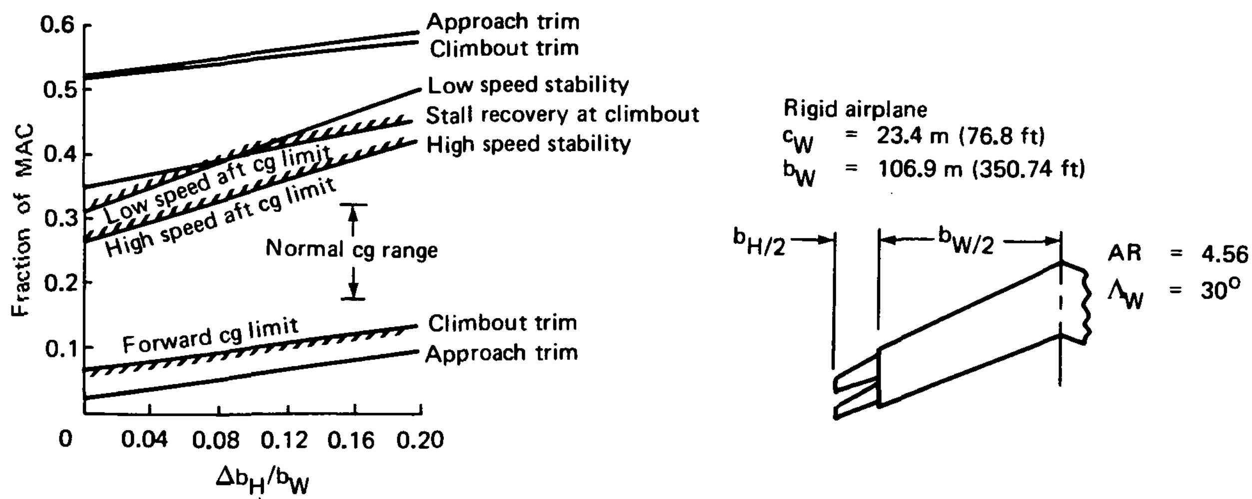


Figure 8 Longitudinal Balance—Model 759-189-1

Lateral-Directional Stability and Control–The parametric study criteria were used for vertical tail sizing of the baseline airplanes. Figure 9 shows the application of the criteria for the 759-189-3 configuration and indicates that, with the wing tip extensions selected, both directional stability ($C_{n_\beta} > 0.0010$) and engine-out trim ($V_{trim} = V_{mc_a} = 1.1V_S$) are satisfied.

Roll response was not specified as a criterion in the parametric study and, until simulator studies can be undertaken, no firm criteria can be proposed for this class of large airplanes. Figure 10 shows the roll response of the 759-189-3 airplane at a landing weight of 907 185 kilograms (2 million pounds), clearly not meeting the current Boeing goal and MIL F specification of 30-degree

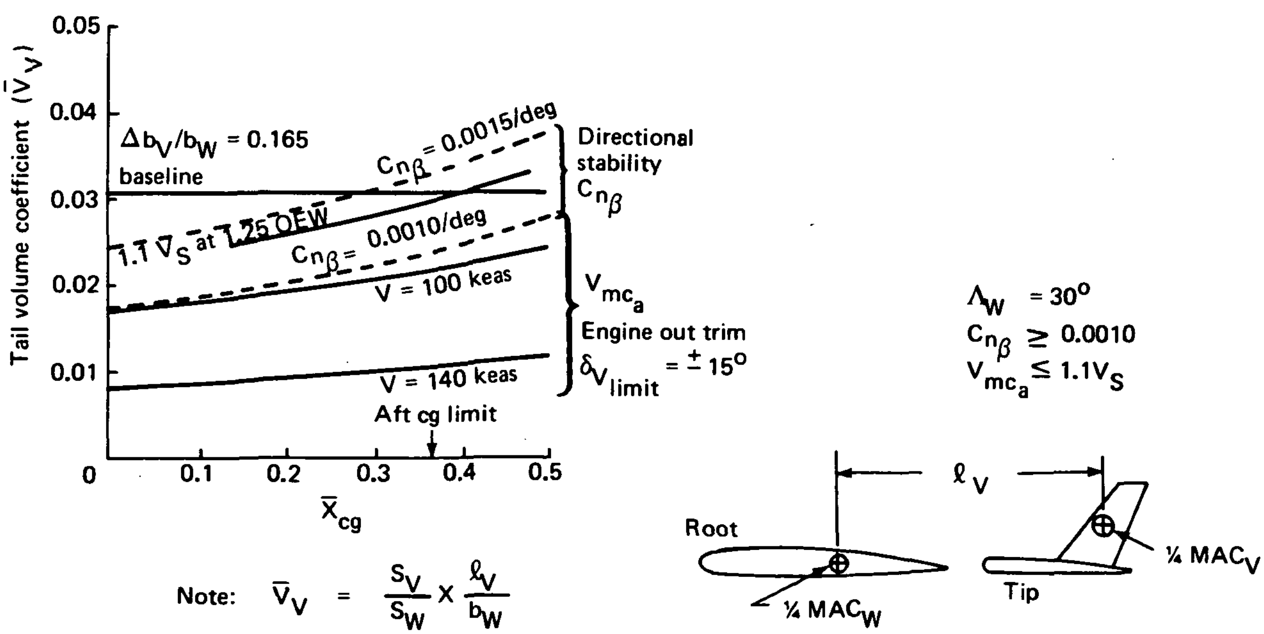


Figure 9 Vertical Tail Sizing

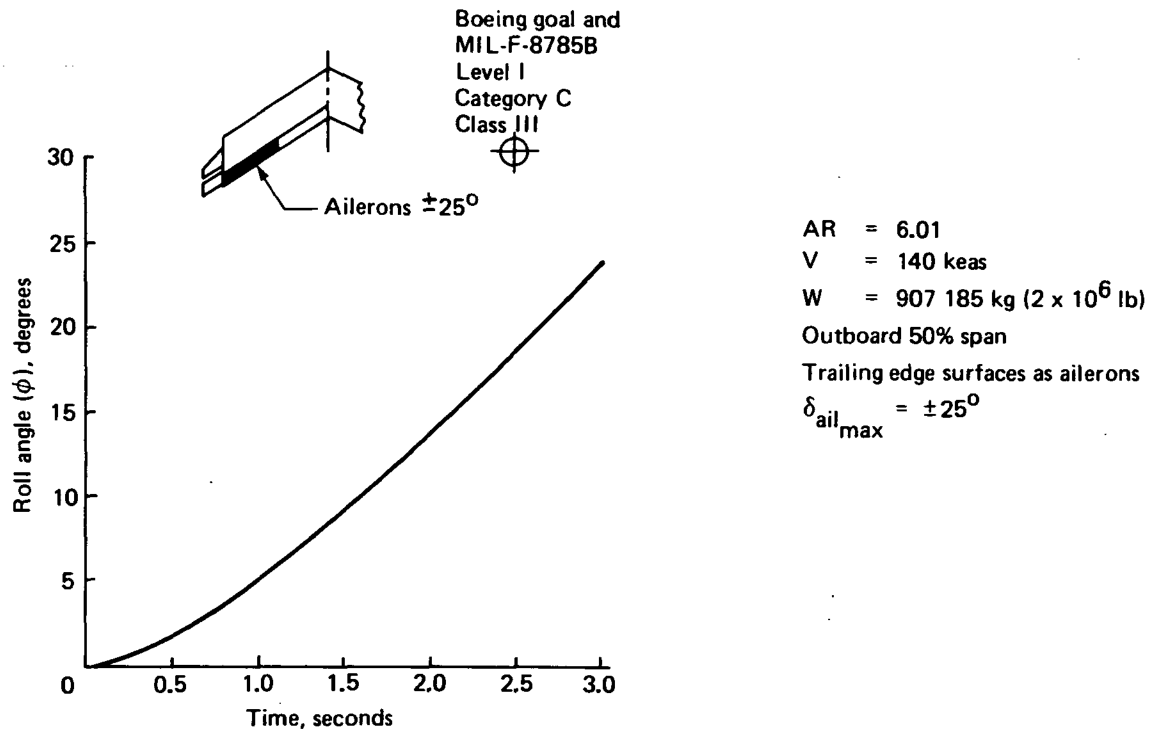


Figure 10 Roll Response, Approach Condition —Model 759-189-3

bank in 2.5 seconds. However, when shown in comparison with other large airplanes (Figure 11), the DLF airplane does not appear to have inadequate roll response.

Data Base—All analyses conducted on the 759-189 configurations employed data derived from potential flow programs. The airframe was assumed to have rigid characteristics pertaining to a feedback control logic, whereby spanwise structural deformation induced by nondistributed air loads would be compensated by local offsetting trailing-edge control settings.

5.2.2 CIVIL CONFIGURATION TRADES

The parametric study plan for the civil configurations was conducted in two phases. A process of elimination was performed on the major study variables, resulting in continually narrowing the number of potential configurations and arriving at a final configuration. This parametric study approach permitted flexibility in the choice of point designs as trends were established.

5.2.2.1 Phase I

In Phase I (see Figure 12), the effect of size (and, indirectly, aspect ratio) was explored by holding thickness ratio constant (t/c normal = 0.19) and varying the wing span by increasing the payload at a constant payload density of 160 kilograms per cubic metre (10 pounds per cubic foot). The cross section was varied from three to five bays and the sweep was set at 30 degrees and 35 degrees. A sufficient number of combinations was studied to allow making a preliminary choice (Configuration X) of size, sweepback, and number of cargo bays. The baseline configuration, Model 759-189-1, for Phase I is shown in Figure 1.

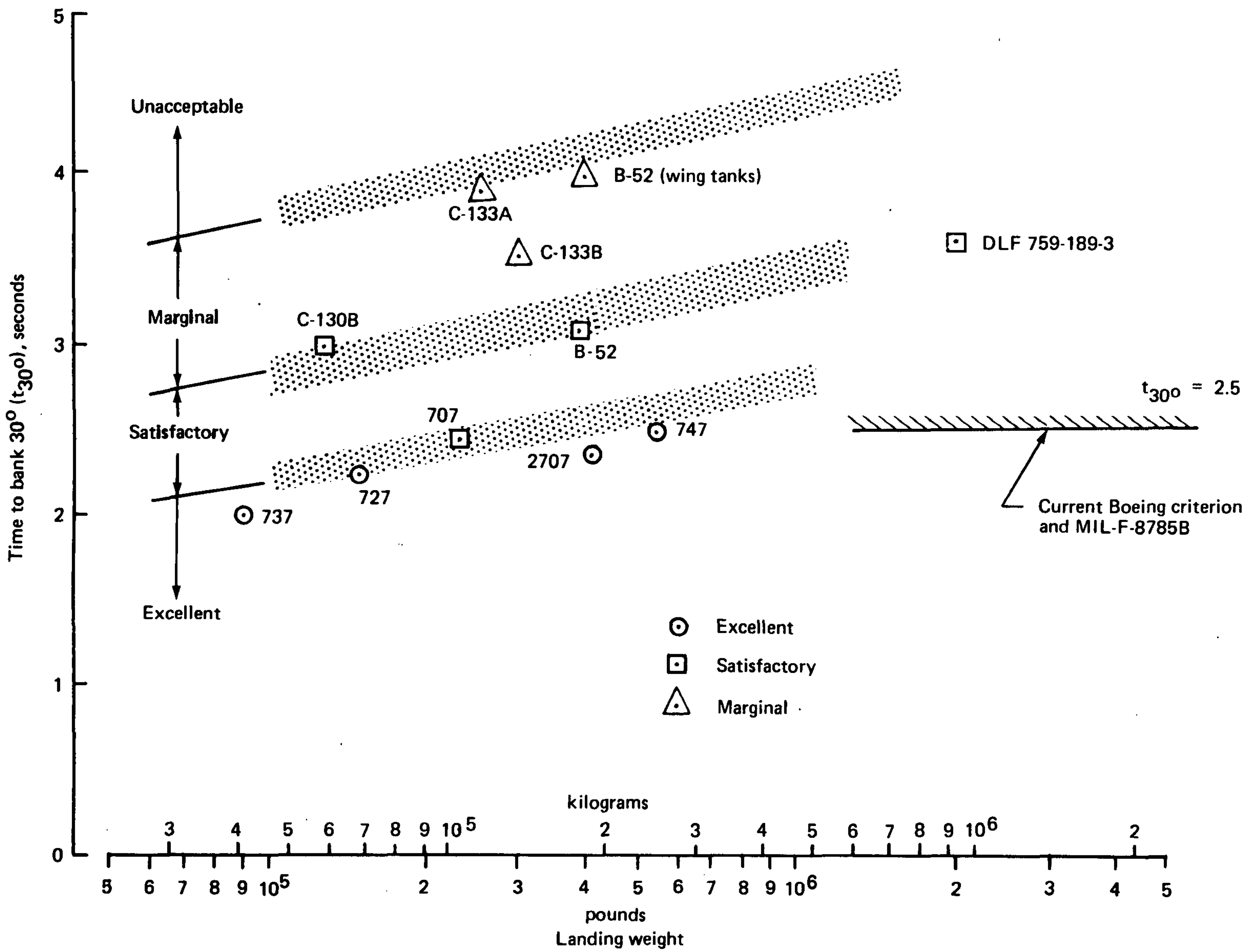


Figure 11 Roll Response at Landing Approach

t/c ⊥ = 0.19	Payload kilograms (pounds)	Sweep = 30° (M = 0.78)	Sweep = 35° (M = 0.82)
$\bar{c}$ = 18.44 m (726 in.)	272 155 (600 000)	–190-1 AR = 4.489	–192 AR = 4.046
	408 233 (900 000)	–190-2 AR = 6.613	
$\bar{c}$ = 20.27 m (798 in.)	272 155 (600 000)	–189-2 AR = 3.118	
	408 233 (900 000)	189-1 AR = 4.567 Baseline	–193-1 AR = 4.113
	544 311 (1 200 000)	–189-3 AR = 6.015	–193-2 AR = 5.408 Selected for phase II
	680 389 (1 500 000)		–193-3 AR = 6.704
	816 466 (1 800 000)		–193-4 AR = 8.001
$\bar{c}$ = 23.32 m (918 in.)	408 233 (900 000)	–191-1 AR = 3.214	
	544 311 (1 200 000)	–191-2 AR = 4.221	

Figure 12 Parametric Study Plan — Phase I

Figure 13 compares the aerodynamic efficiency and, as expected, the L/D and M(L/D) improves with increased aspect ratio. Structural efficiency is shown in Figure 14, favoring the four- and five-bay configurations and their lower aspect ratios. The effect of number of bays on fuel efficiency is shown in Figure 15. It is the aerodynamic efficiency that is predominant. While the four- and five-bay configurations exhibit good structural efficiency, they are less efficient than the three-bay in M(L/D). The effect of range on fuel efficiency is shown in Figure 16.

The three-bay configuration has the best fuel efficiency in the 408 233 kilogram (900 000 pound) payload group of aircraft. These aircraft are relatively insensitive to off-design operation down to approximately half the design range [3334 kilometres (1800 nautical miles)] ; but for ranges above the design point, fuel efficiency drops off rapidly due to the off-loading of payload.

For the four-bay configuration, it is necessary for the payload to increase to 544 311 kilograms (1.2 million pounds) in order to have it compare favorably with a three-bay, 408 233 kilogram (0.9 million pound) net payload aircraft. Payload/range curves are shown in Figure 17.

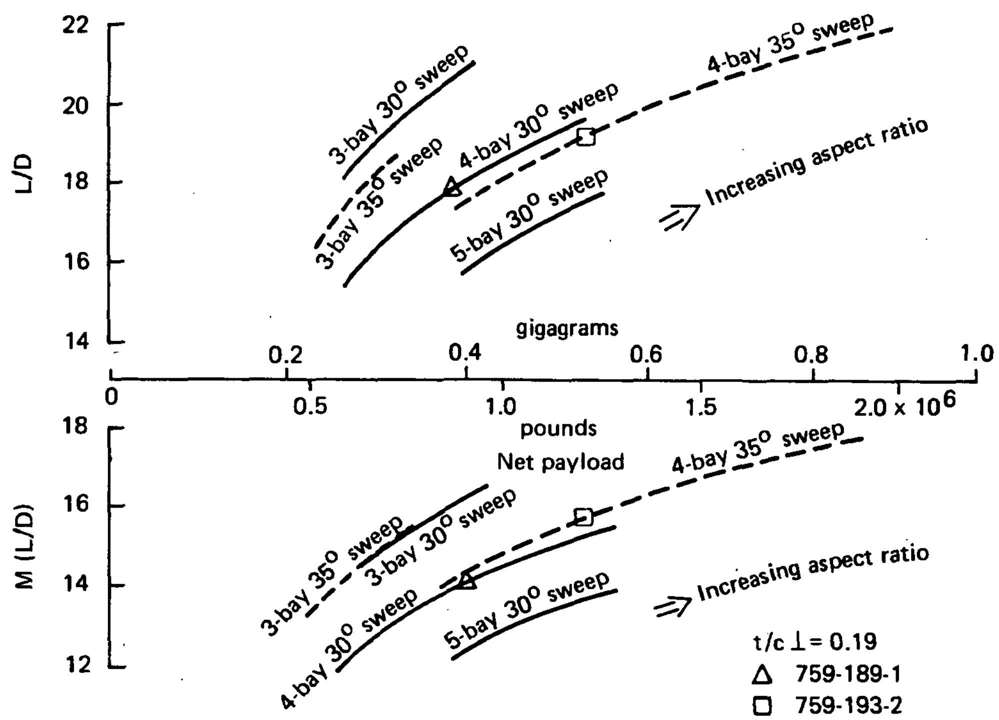


Figure 13 Aerodynamics Efficiency—Phase I

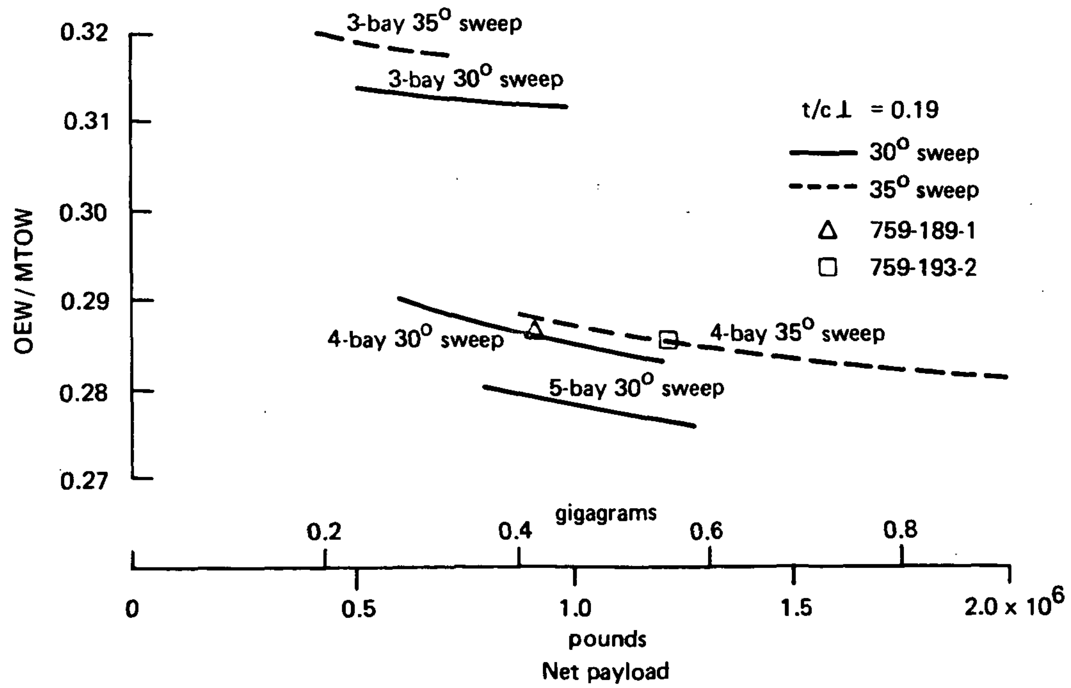


Figure 14 Structural Efficiency—Phase I

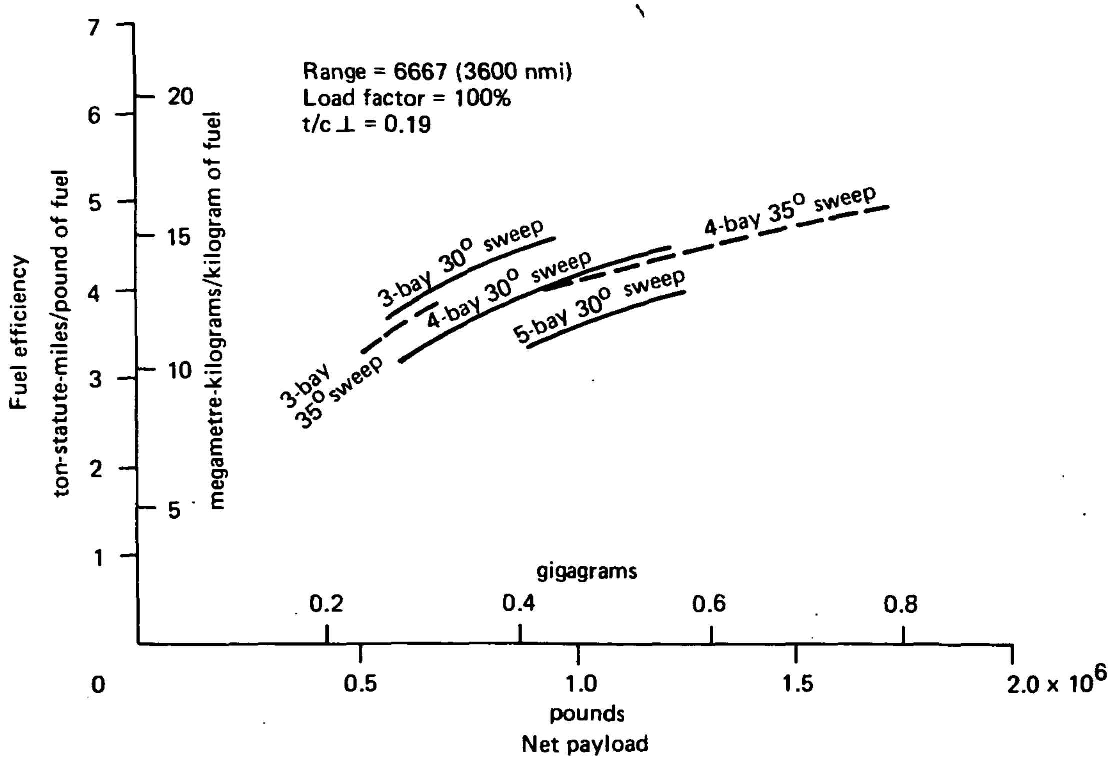


Figure 15 Fuel Efficiency—Phase I

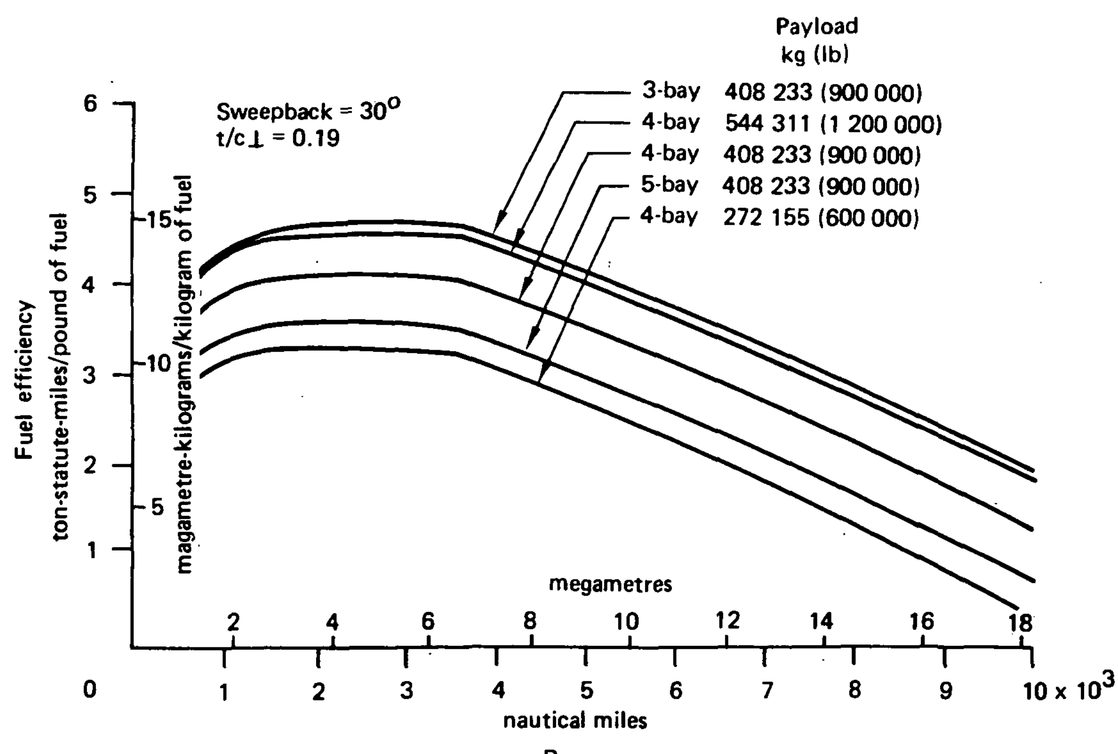


Figure 16 Effect of Range on Fuel Efficiency—Phase I

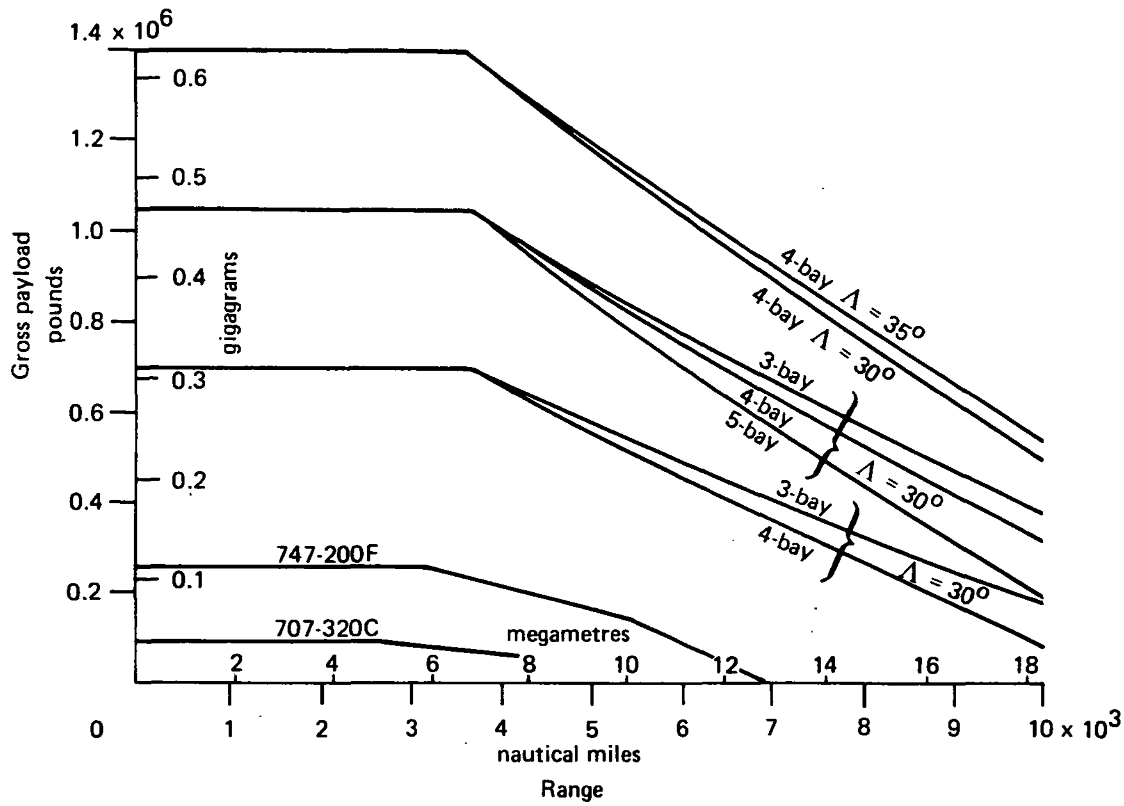


Figure 17 Payload/Range—Phase I

The effect of airplane size and range on economics is shown in Figure 18. The DOC and DOC + AIC were based on a constant annual fleet throughput of 167.9 revenue petametre-kilograms (115 billion revenue ton-miles). Fleet size vs. payload is shown in Figure 19. The effect of airplane size on economics is shown in Figure 20 and the effect of fuel price on economics is shown in Figure 21. Figure 21 points out that for the most economic configuration [4-bay, 35° sweep, 544 311 kilogram (1.2 million pound) payload], an increase of 62 percent in fuel price [97.7 to 158.5 $\$/m^3$ (37 to 60 cents/gal)] produced only an increase of 17 percent in DOC + AIC. One of the most encouraging results of the study was the outstanding fuel economy of the larger configurations that deliver over twice the ton-miles of payload per pound of fuel used compared with a 747.

Operating cost breakdown as a function of net payload is shown in Figure 22. The DOC shows a slight improvement as payload increases; however, the DOC + AIC reaches a minimum at the 544 311 kilogram (1.2 million pound) net payload. It should be noted that the component of transportation cost that is airplane price sensitive is much larger than the portion that is airplane performance sensitive. The importance of low cost methods in the design and manufacture of these large aircraft is significant. The increase in transportation cost for airplanes larger than 544 311 kilograms (1.2 million pounds) is due to the high nonrecurring costs associated with the larger airplanes distributed over a small number of airplanes.

In addition to the size trends, Phase I results also indicated some effects from the wing cross-section variations that have guided the choice of Phase II configurations. Although the three-bay configurations exhibited better economics at the 272 155 kilogram (600 000 pound) payload size, the four-

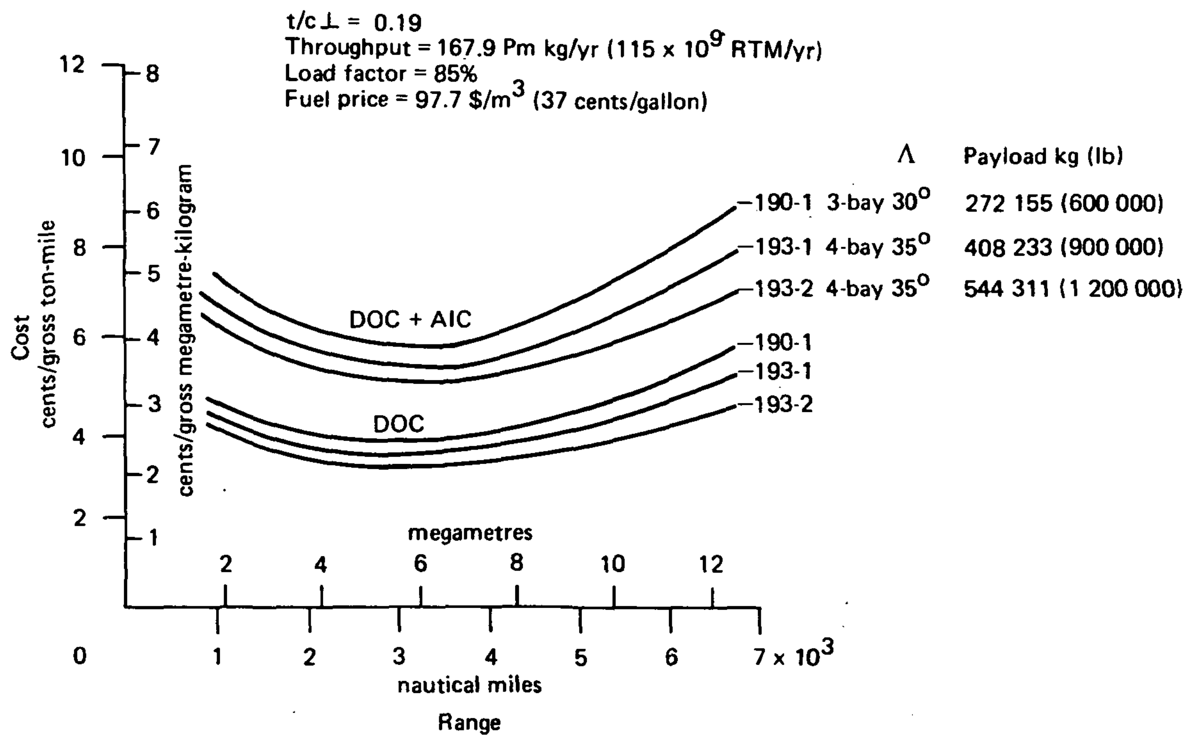


Figure 18 Effect of Size and Range on Economics

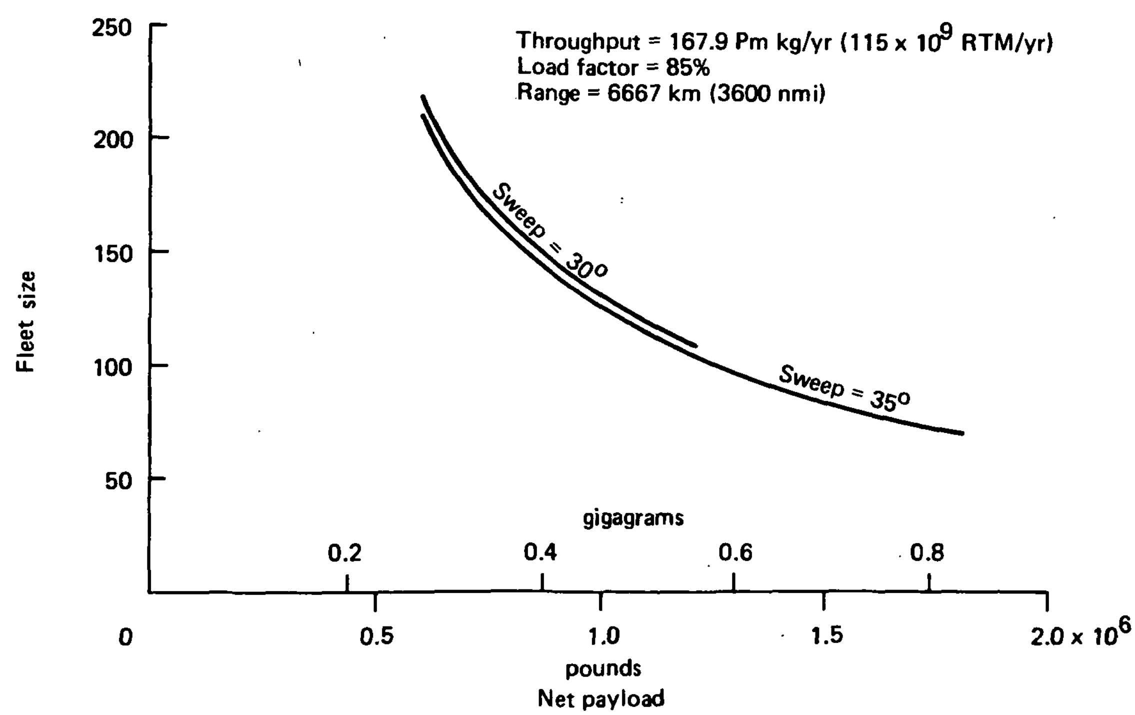


Figure 19 Fleet Size—Phase I

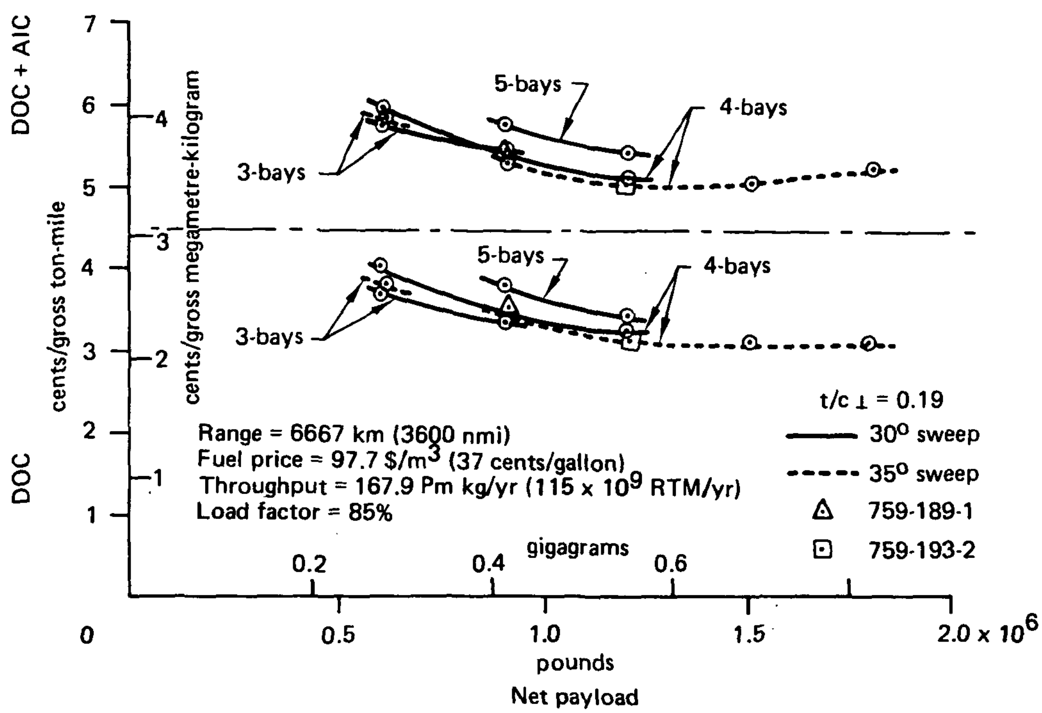


Figure 20 Effects of Airplane Size on Economics—Phase I

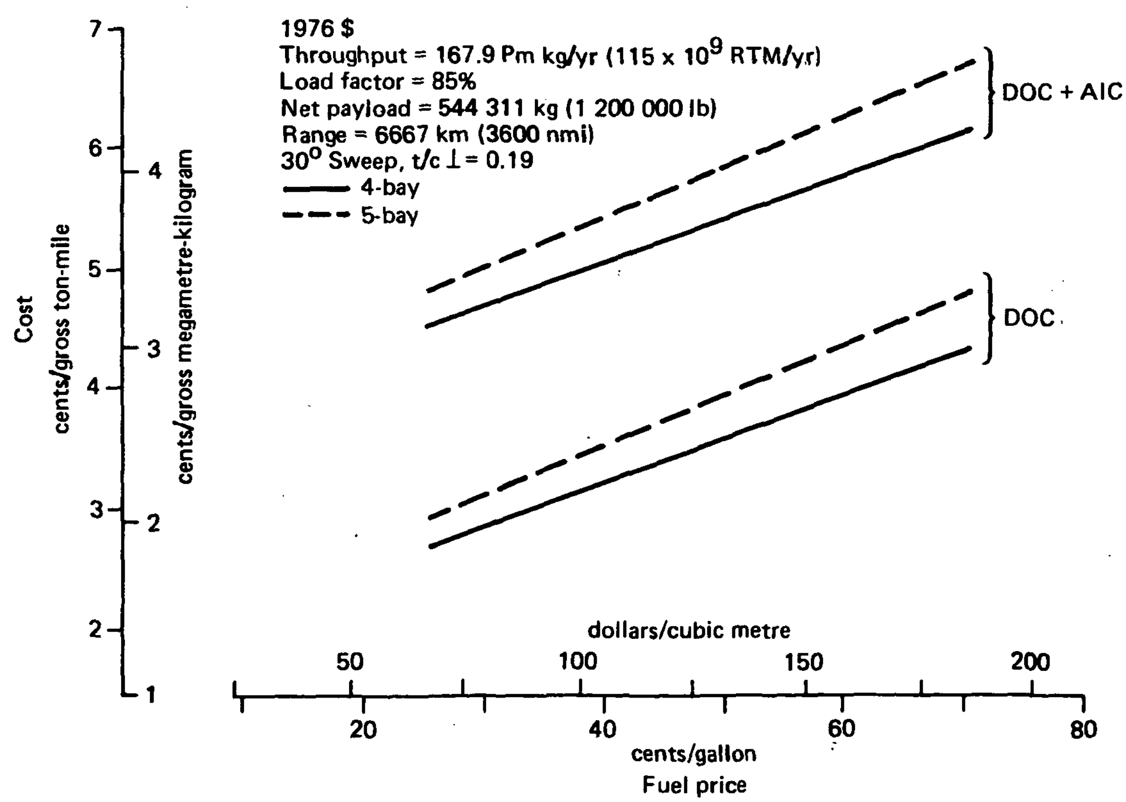


Figure 21 Economic Sensitivity to Fuel Price—Phase I

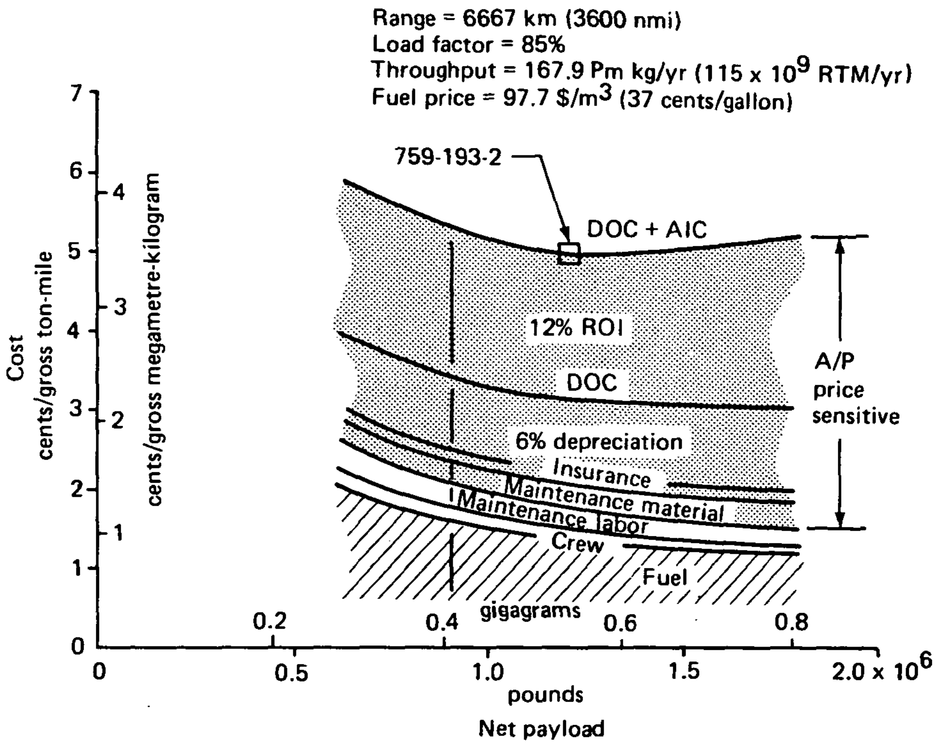


Figure 22 Operating Cost Breakdown—Phase I

bay configuration economics improved more rapidly with increase in payload and were superior at the larger sizes. Therefore, the four-bay 544 311 kilogram (1.2 million pound) payload configurations were selected for analysis in Phase II of the parametric study. Similarly, the 30-degree sweepback was superior at the 272 155 kilogram (600 000 pound) payload, but the 35-degree sweepback was better at the 408 233 kilogram (900 000 pound) payload and further improved with size. Since economics improved with the 35-degree sweepback for the large payloads, the sweepback study was expanded in Phase II to include 30 to 40 degrees.

Only one airfoil thickness ratio (t/c = 0.19) was analyzed in Phase I; therefore, thickness ratios above and below this value were selected for analysis in the Phase II study. Model 759-193-2 [t/c = 0.19; 35^{o} sweep; 544 311 kilogram (1.2 million pound) payload] was chosen as the best Phase I configuration to carry into the Phase II parametric (Figure 22).

5.2.2.2 **Phase II Civil**

The Phase II civil study was made at a constant net payload of 544 311 kilograms (1 200 000 pounds) and with only four-bay cross-sections as a result of the Phase I study. In Phase II (see Figure 23), the effect of change in cruise speed was studied by varying thickness ratio and sweepback, holding payload constant. The increments chosen, ±5 degrees of sweepback and ±0.03 change in thickness ratio, allow assessing independently the effect of a change in cruise speed, since each change varies cruise Mach number by approximately $\Delta M = 0.03$. The study encompassed 30- to 40-degree sweepback, t/c from 0.16 to 0.25, and cruise Mach numbers from M = 0.78 to M = 0.86.

Payload = 544 311 kg (1 200 000 lb)	t/c ⊥	Sweep = 30°	Sweep = 35°	Sweep = 40°
$\bar{c}$ = 22.76 m (896 in.)	0.16	M = 0.81 AR = 5.357 −194	M = 0.85 AR = 4.817 −195 Selected	
$\bar{c}$ = 20.27 m (798 in.)	0.19	M = 0.78 AR = 6.015 −189-3	M = 0.82 AR = 5.408 −193-2 Baseline	M = 0.86 AR = 4.757 −197
$\bar{c}$ = 18.69 m (736 in.)	0.22		M = 0.79 AR = 5.864 −196	M = 0.835 AR = 5.158 −198
$\bar{c}$ = 17.45 m (687 in.)	0.25			M = 0.805 AR = 5.510 −199

Figure 23 Parametric Study Plan – Phase II

The data for each of the parametric configurations were generated. With these data, trades between thickness ratio and sweepback were established and the resulting effect on the economics were determined.

Figure 24 shows that the highest productivity airplanes have the lowest t/c. This figure points up the importance of thick airfoil studies to improve aerodynamic efficiency at higher thickness ratios by means of new shapes such as blunt base airfoil or possibly some boundary layer suction.

Figure 25 shows the relative structural efficiency and indicates a fairly strong trend toward the thicker airfoil section. This again points up the need to find thick airfoils with good aerodynamic efficiency.

Figure 26 shows the fuel use efficiency comparison. Fuel efficiency is influenced predominately by aerodynamic efficiency as was determined in Phase I. The best cruise Mach number is proportional to sweep and inversely proportional to the thickness ratio; therefore, fuel use efficiency is expected to peak at higher Mach numbers as sweep increases and the thickness ratio decreases. At the lower Mach numbers, high sweepback becomes detrimental with respect to the lift-to-drag ratio, thereby reducing the level of fuel use efficiency. It is significant that fuel use is not very strongly affected by Mach number, but the thickness ratio trends are very strong.

Figure 27 shows the effect of range on fuel efficiency. The sensitivity of all configurations was about the same with the 16-percent t/c configuration showing the best fuel efficiency with essentially no difference between the 30-and 35-degree sweep.

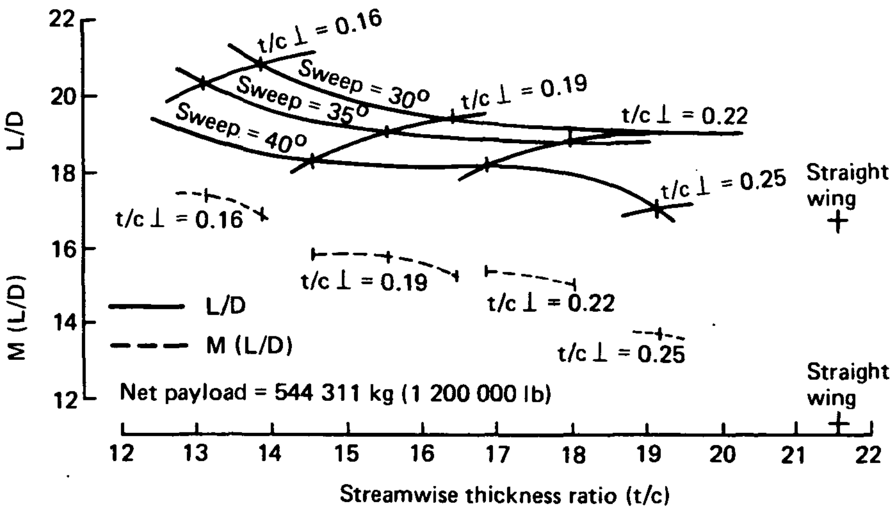


Figure 24 Aerodynamic Efficiency—Phase II

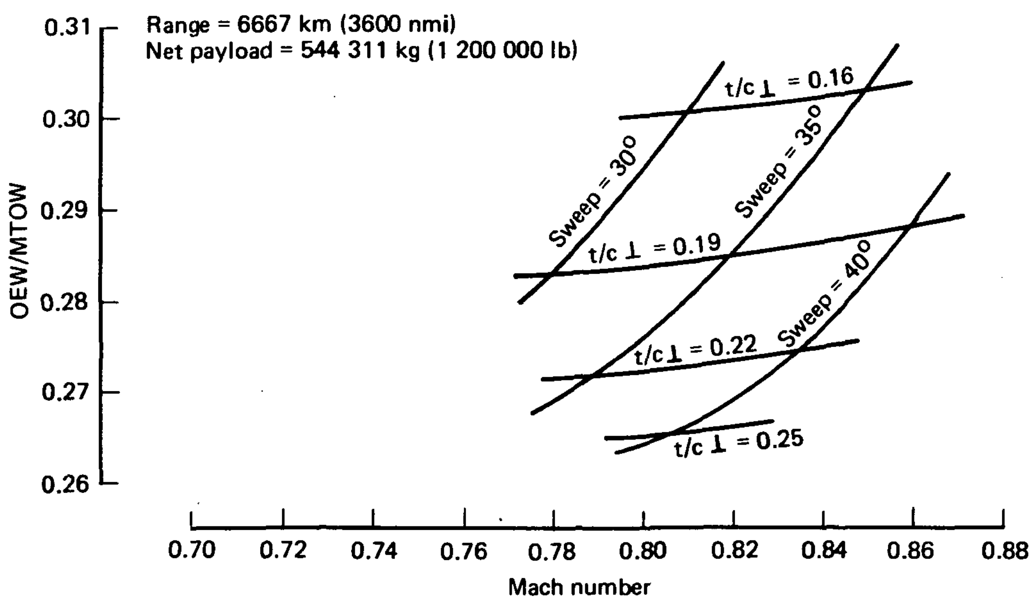


Figure 25 Structural Efficiency—Phase II

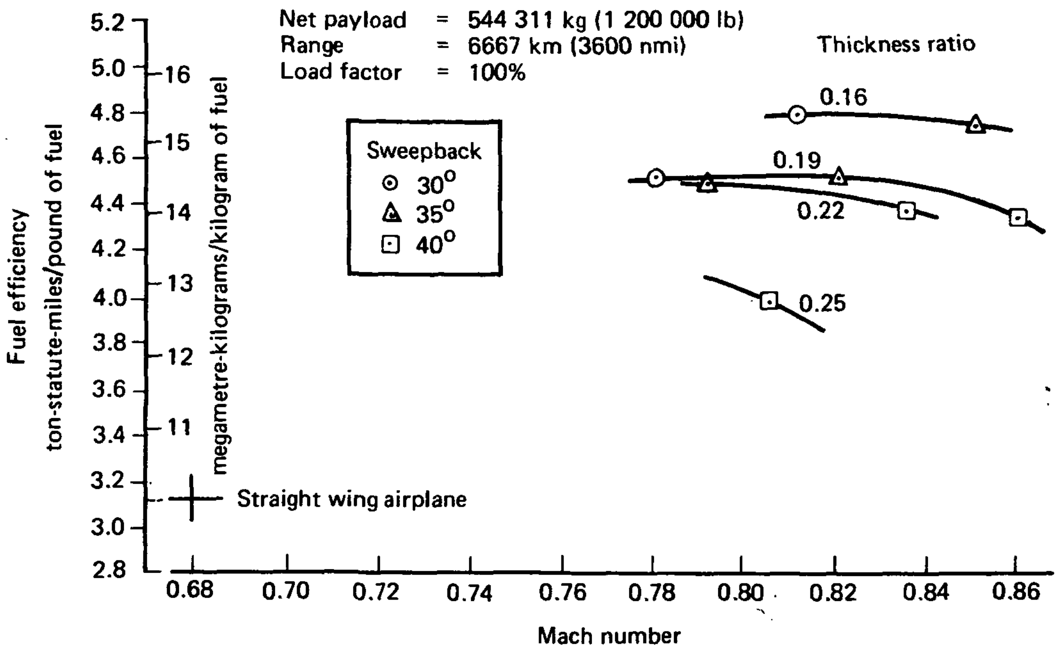


Figure 26 Fuel Use Efficiency Comparison—Phase II

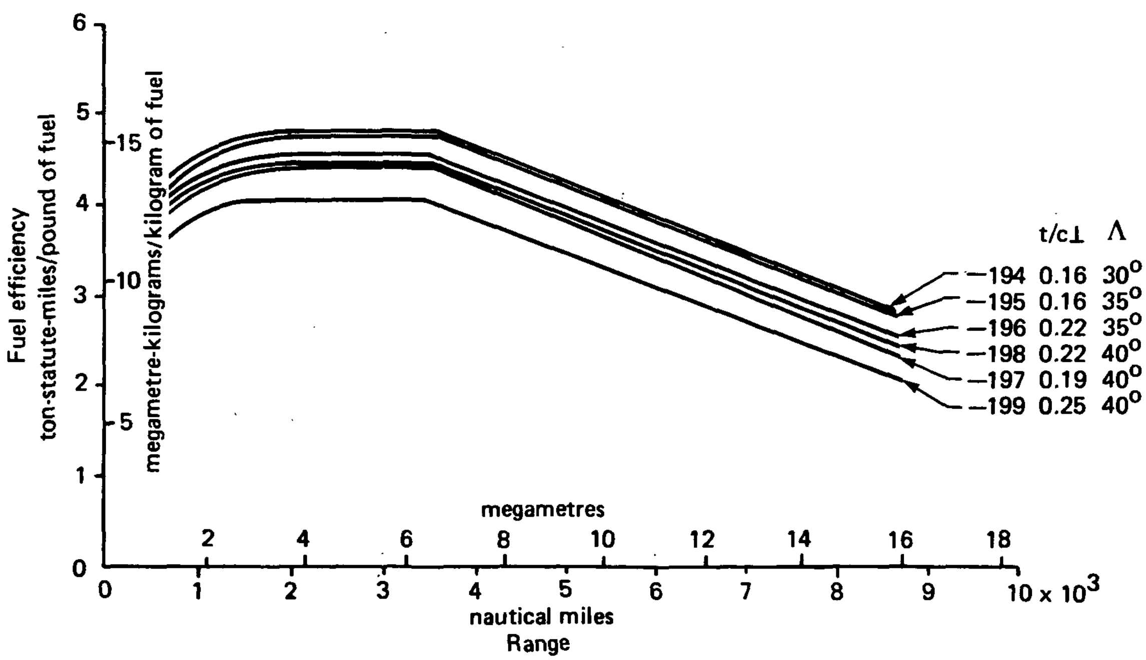


Figure 27 Effect of Range on Fuel Efficiency—Phase II

Figure 28 shows the airplane study price. It shows that the airplane price decreases as wing thickness is increased up to thickness ratios of at least 25 percent. It also shows that it is more costly to get speed by a reduction in thickness rather than sweep. This again points up the importance of airfoil studies.

Figure 29 shows the fleet study investment cost. It shows the effect of airplane productivity on fleet quantity. The slowest airplanes, of course, require the largest number of airplanes in the fleet, and the numbers vary from 110 at Mach 0.78 to 102 at Mach 0.86. The potential cost reduction available for airplanes configured with both higher Mach number and greater thickness ratio is significant.

Figure 30 shows the effect of airplane configuration on economics. Based upon economic considerations, the compensating trade of airplane cost, fuel efficiency, and speed results in very little difference between configurations.

Results of Phase II indicate that variations of sweepback and thickness ratio at a 544 311 kilogram (1.2 million pound) payload produce relatively small changes in the economics (see Figure 30). The higher Mach number configurations with thinner wings have higher empty weights, higher prices, and require higher thrust engines, which increase their relative investment and maintenance costs. However, their lower fuel costs and higher productivity compensate for these other costs. Since there is little variation in economics with Mach number, the highest Mach number will be chosen since greater route flexibility and better utilization would be gained from this choice. The 35-degree sweepback with 16-percent thickness ratio airfoil cruising at M = 0.85 and carrying 544 311 kilograms (1.2 million pounds) net payload appears to be the most desirable configuration (Model 759-195).

The chosen 16-percent thickness ratio, having the largest chord, had the lowest takeoff wing loading and hence the shortest takeoff distances and/or the greatest takeoff gross weight growth potential; i.e., MTOW can be increased to the point that the TOFL just meets the requirement. The cross-section has the most flexibility since the chord is so wide that a fifth bay of reduced height can be

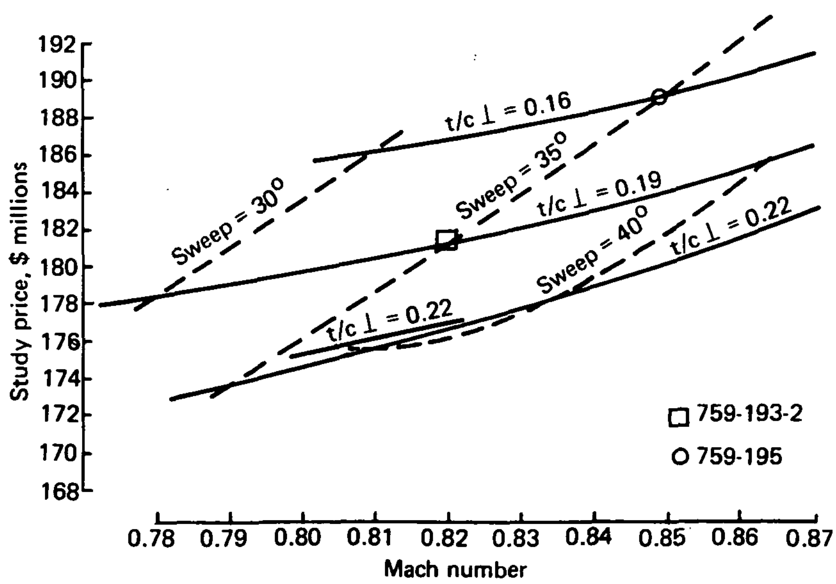


Figure 28 Airplane Study Price—Phase II (1976 $)

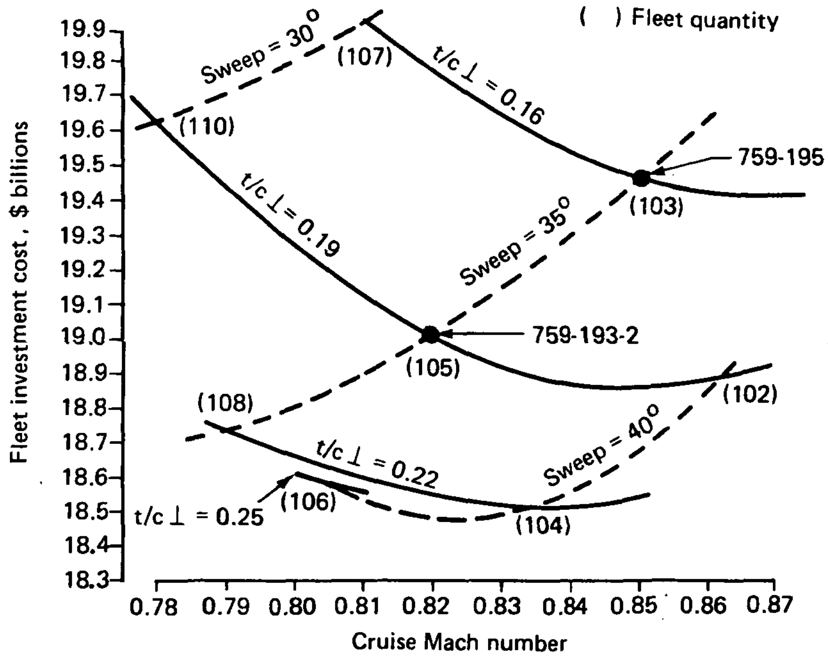


Figure 29 Fleet Study Investment Cost

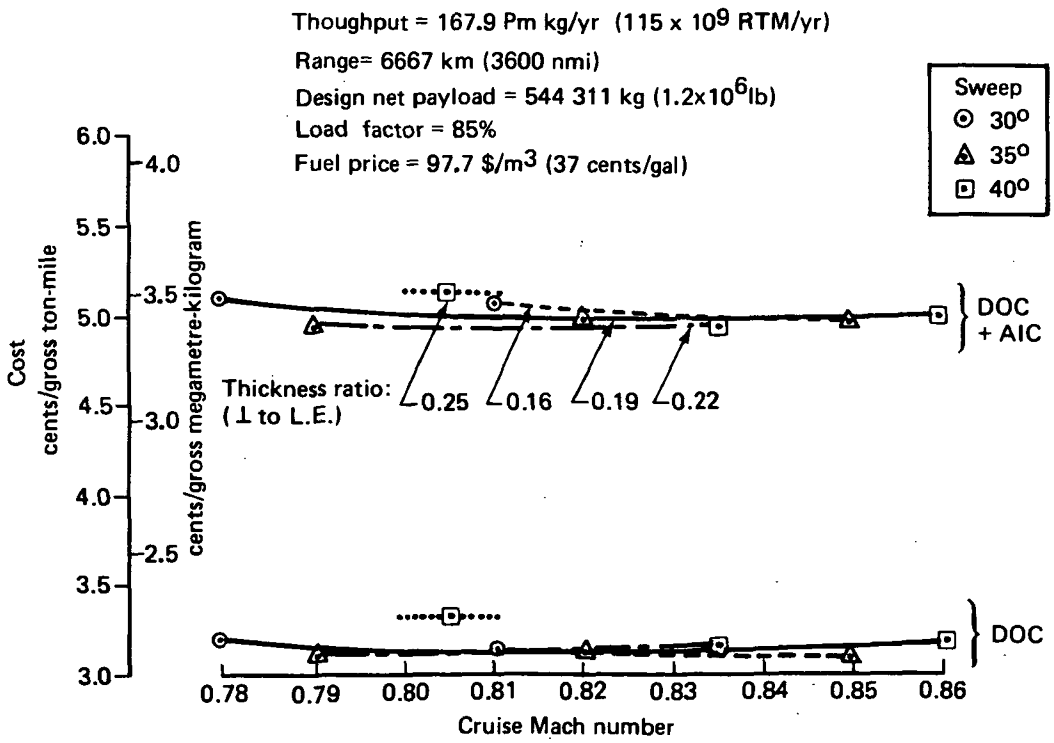


Figure 30 Effect of Airplane Configuration on Economics

included in what would normally be wasted space. Since these airplanes are not constrained by MZFW, this additional space (see Figure 31) could be utilized to either increase the payload flexibility (carry a mix of 8 x 8 and LD-7 containers) or increase the maximum payload and hence decrease the overall operating costs. The combination of low thickness ratio and moderate sweep gives very good fuel use efficiency, and high cruise Mach number (M = 0.85) yields high mission flexibility permitting longer range city pairs to be served with a single crew, and resulting in higher utilization and ultimately lower costs.

Table 4 summarizes the selection rationale for the selected civil configuration. Figure 31 shows the wing cross-section and identifies the areas used for fuel and cargo.

Table 4 Selection Rationale—Civil Configurations

Economic criteria: Minimum DOC + A/C at constant throughput of 167.9 Pm kg/yr (115×10^9 RTM/yr) • DOC: Revised 1976 ATA formula • A/C: Airplane investment cost based on CAB guidelines of 12% ROI		
	Best choice	Reason
Additional considerations		
• Fuel use efficiency	t/c = 0.16 sweep = 30°	High cruise efficiency
• Performance growth potential		
• Increase in gross weight	t/c = 0.16 sweep = 35°	Shortest TOFL
• Cargo flexibility	t/c = 0.16	Space for additional LD-7 bay
• Mission flexibility		
• Low cost at long range	t/c = 0.16 sweep = 30° or 35°	High cruise efficiency
• Low block time	t/c = 0.16 sweep = 35° t/c = 0.19 sweep = 40°	M = 0.85 M = 0.86
• Technical margins		
• Takeoff lift coefficient	t/c = 0.16 sweep 30° or 35°	Low wing loading
• Drag confidence level	t/c = 0.16 sweep = 30° or 35°	Less extrapolation from data base
Selected configuration: net payload = 544 311 kg (1 200 000 lb) t/c = 0.16 Sweep = 35°		

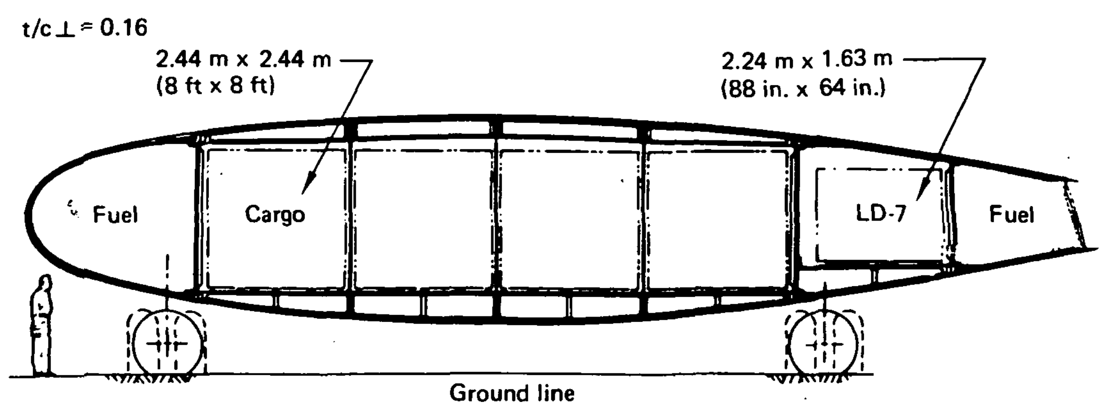


Figure 31 Selected Civil Wing Cross Section—Model 759-195

5.2.3 MILITARY CONFIGURATION TRADES

The military configuration parametric study made maximum use of the civil configuration parametric results. The configurations considered will be limited to those designed to a 272 155-kilogram (600 000-pound) payload capability. The configuration concept used the same wing geometry as the civil configurations and provides the required outsized cargo capability in an outsize cargo pod. In addition, the military configurations have appropriately strengthened floors, ramps, and pressurization. The impact of these modifications on the parametric data was investigated to the extent necessary for selection of the Final Military configuration. Also, the parametric data were adjusted for range and field length effects. A listing of these mission requirements follows:

- Net payload 272 155 kg (600 000 lb)
    - 2.44 x 2.44 x 6.10 m/2.44 x 2.44 x 12.19 m (8 x 8 x 20 ft/8 x 8 x 40 ft) containers
    - 463L pallets
    - Military equipment: roll-on and roll-off
    - Two 4.11 m (13.5 ft) h x 5.18 (17 ft) w x 12.19 (40 ft) l cargo bays
- Design range 10 186 km (5500 nmi)
- Design critical field length 3048 m (10 000 ft)
- Cargo compartments pressurized

Additional considerations

- Capability to air-launch:
    - ALLRC missiles
    - M-X missiles

The baseline for the military Phase II (shown in Figure 32) is a three-bay configuration. During the study it was discovered that higher aspect ratios than could be provided by the three-bay configurations were required to meet the military range and field length requirements. The study was expanded to include four two-bay configurations (see Figure 33).

Range, km(nmi)	10 186	(5500)
Net payload, kg (lb)	272 155	(600 000)
MTOW, kg(lb)	1 204 288	(2 655 000)
OEW, kg(lb)	418 757	(923 200)
Wing area, $m^2(ft^2)$	1749	(18 826)
Basic aspect ratio	3.45	
Effective aspect ratio	6.06	
Basic wing span, m(ft)	78	(255)
Total wing span, m(ft)	92	(301)
Sweep, degrees	35	
t/c⊥	0.19	
Cruise Mach number	0.82	
Engines BPR	9.5	
TSLS, N(lb)	422 581	(95 000)

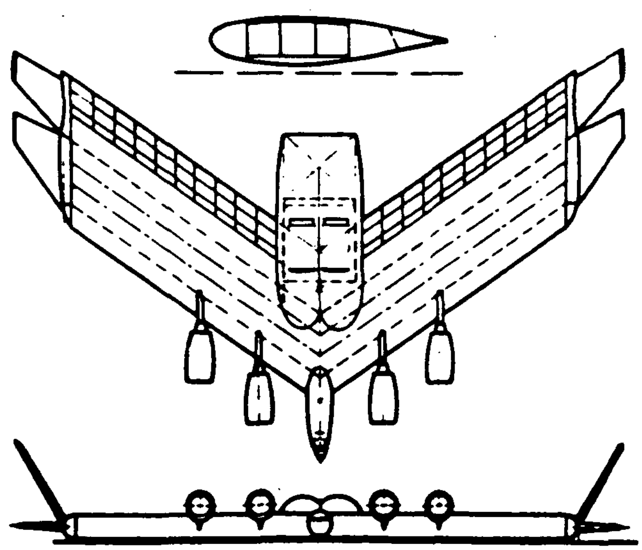


Figure 32 Baseline Military Configuration—Model 759-192M

Payload = 272 155 kg (600 000 lb)	t/c⊥	Sweep = 30°	Sweep = 35°	Sweep = 40°
c̄ = 18.44 m (726 in.)	0.19	M = 0.78 AR = 3.824 -205M	M = 0.82 AR = 3.451 -192M Baseline	
c̄ = 16.64 m (655 in.)	0.22		M = 0.79 AR = 3.825 -206M	M = 0.835 AR = 3.378 -207M
c̄ = 19.81 m (780 in.)	0.16		M = 0.85 AR = 4.704 -210M	
c̄ = 16.64 m (655 in.)	0.19		M = 0.82 AR = 5.517 -209M	
c̄ = 14.91 m (587 in.)	0.22		M = 0.79 AR = 6.250 -208M Selected	
c̄ = 13.46 m (530 in.)	0.25			M = 0.805 AR = 6.095 -204M

Figure 33 Military Configurations—Phase II

Figure 34 shows the effect of the military requirements on the baseline parametric airplane, 759-192, which would meet the military requirement at 6667 kilometres (3600 nautical miles) but would not meet them satisfactorily at 10 186 kilometres (5500 nautical miles). In order to meet the 3048-metre (10 000-foot) field length requirement, the two-bay cross section was developed, resulting in a higher aspect ratio and thus reduced drag. The curves show that the -208M meets the 10 186 kilometre (5500 nautical mile) range as well as the 3048 (metre (10 000 foot) field length. Aerodynamic efficiency is compared in Figure 35 and structural efficiency in Figure 36. The military aircraft exhibited the same trends as the civil aircraft but pays the extra penalty in structural weight in order to carry outsized cargo such as bridge loaders and M-60 tanks. Figure 37, fuel use efficiency, reflects the trends found in the civil aircraft, the reduced level being a reflection of the increased range and reduced payload. As expected, the aircraft with a 25-percent thickness ratio and 40-degree sweep also suffers from very low cruise lift-to-drag ratios. A comparison of fuel use efficiency between civil and military configurations is contained in Figure 38.

Life cycle costs for the four two-bay military configurations are compared in Table 5. Model 759-208M was selected from these configurations on the basis of the following selection rationale:

- Formal criterion
    - Minimum 20-year life cycle costs

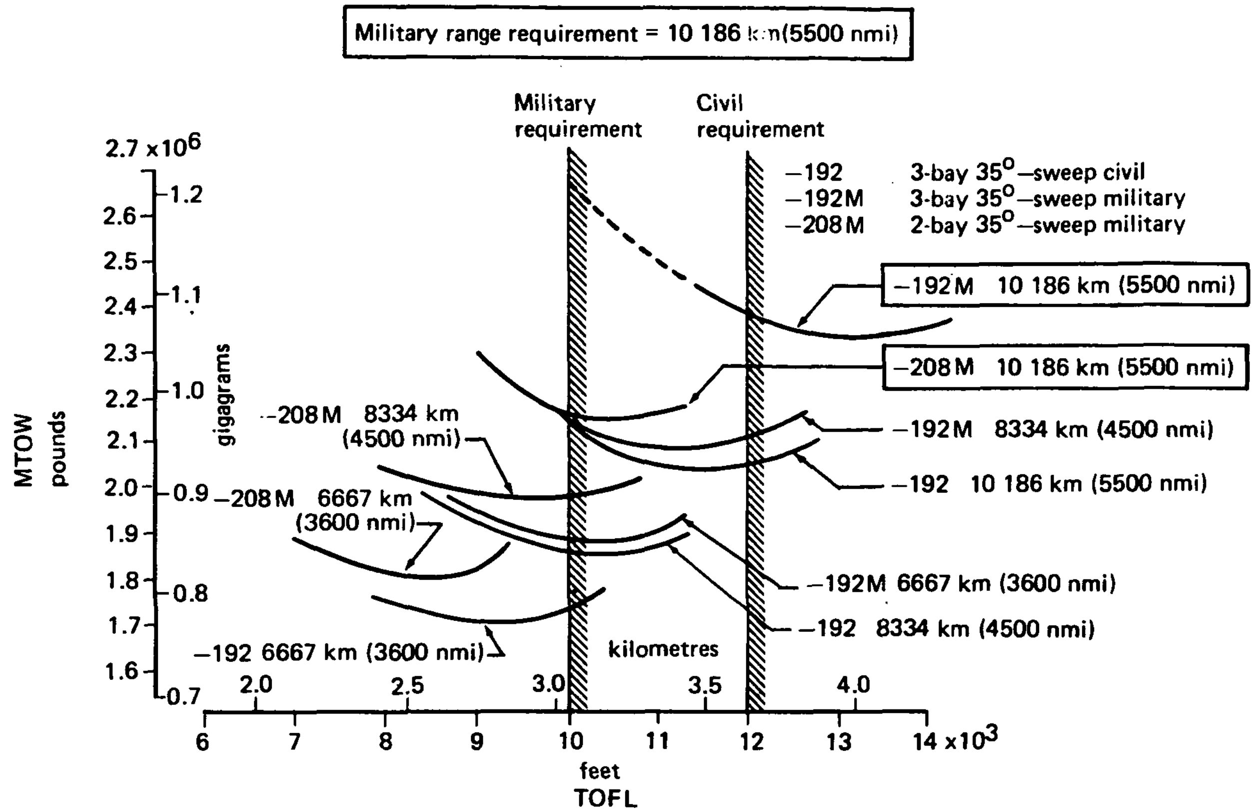


Figure 34 Effect of Military Requirements

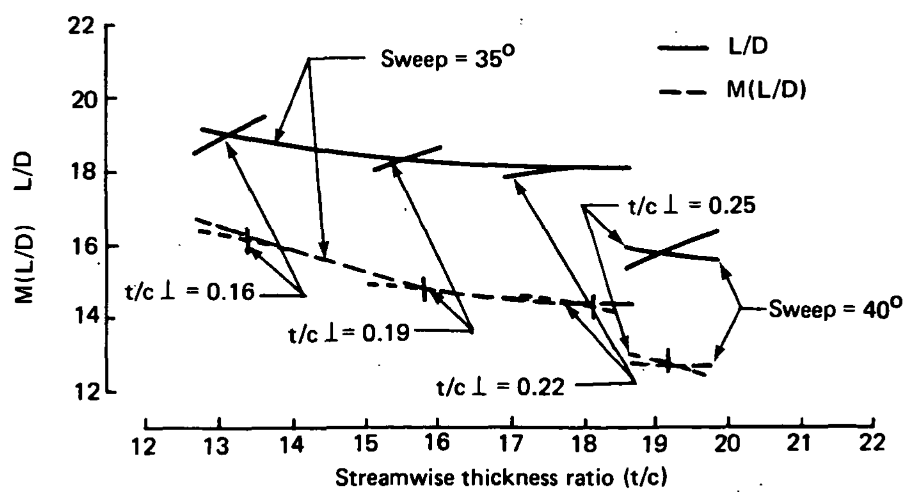


Figure 35 Aerodynamic Efficiency—Phase II (Military—Two Bay)

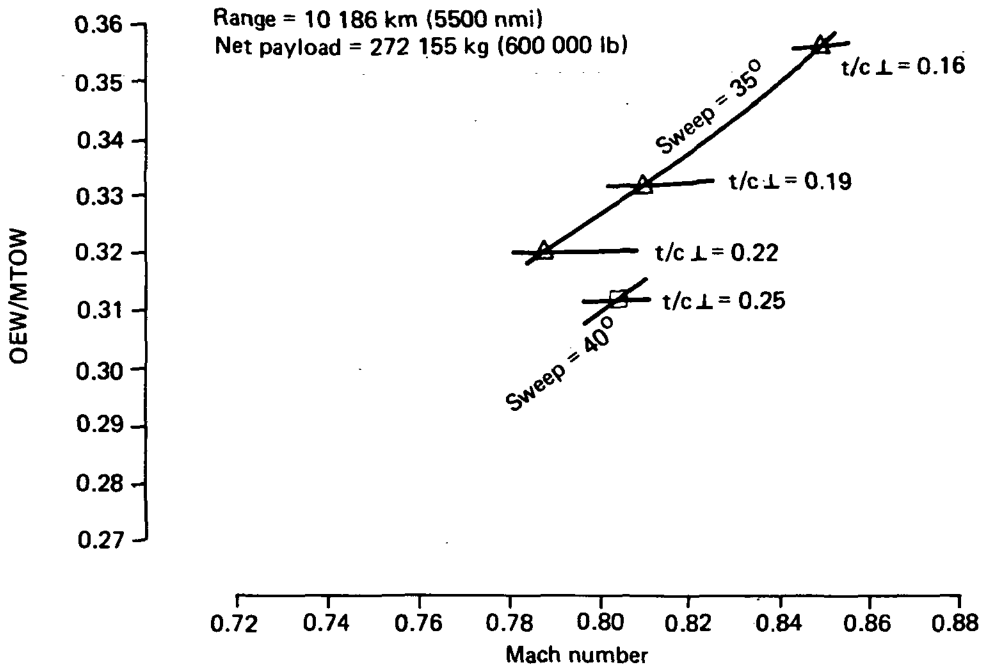


Figure 36 Structural Efficiency—Phase II (Military)

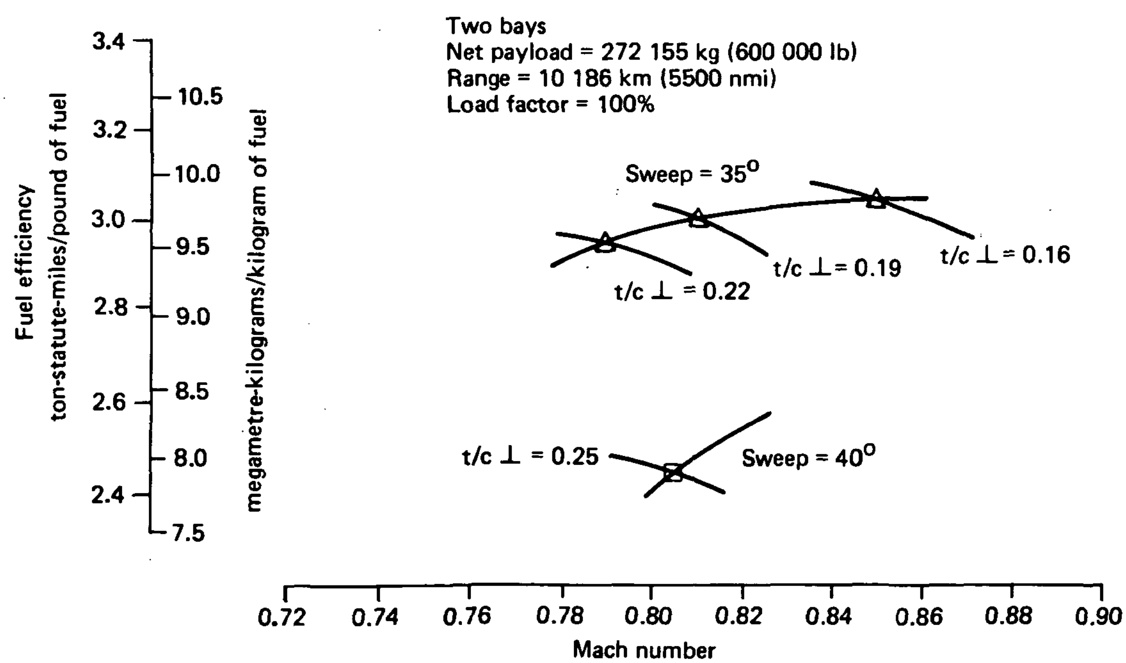


Figure 37 Fuel Efficiency Comparison—Phase II (Military)

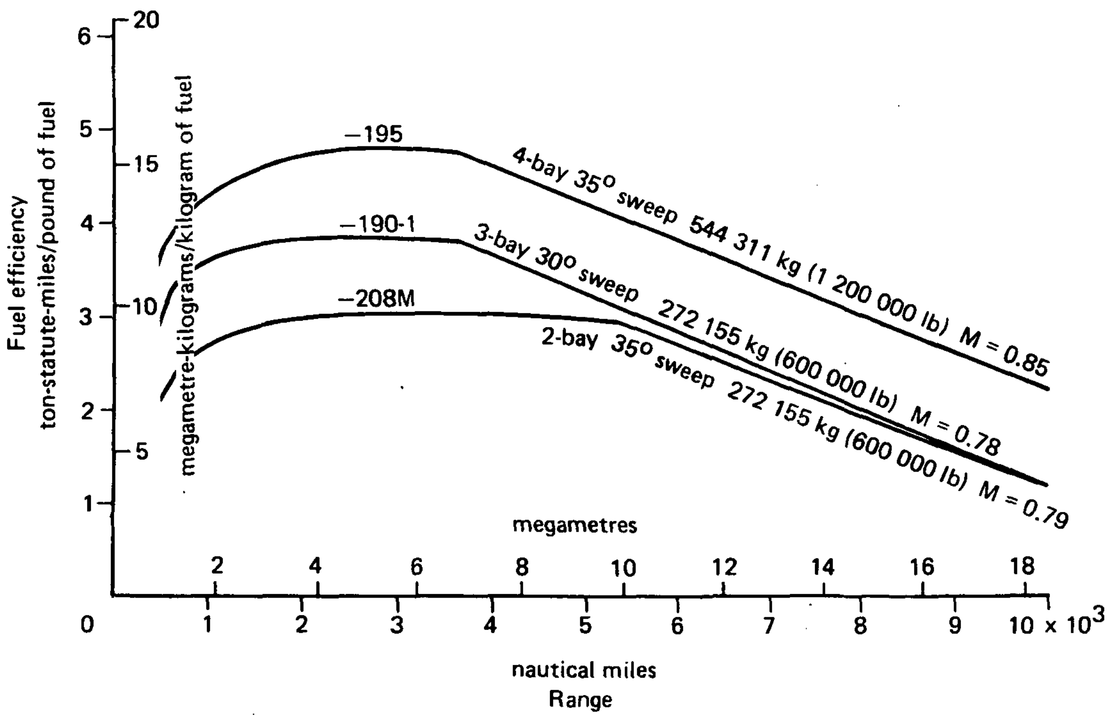


Figure 38 Fuel Use Efficiency—Comparison of Military and Civil Configurations

Table 5 20-Year Life-cycle Cost—Parametric Military Aircraft (Dollars in Millions)

	759-208M*	759-204M	759-209M	759-210M
Development and production	9986	13 180	10 241	11 835
Support equipment investment	1390	1860	1425	1537
Operations and support				
Fuel	7520	9303	7615	7808
Other	7099	7584	7184	7416
Total life cycle cost	25 995	31 927	26 465	28 596
Ground rules				
1976 dollars				
125 UE				
7 squadrons of 16 UE				
13 command support UE				
1000 flight hours/year				
10 186 km (5500 nmi) flights				
272 155 kg (6000 000 lb) net payload/flight				
Fuel price: 97.7 \$/m^3 (37 cents/gallon)				
Validation phase not costed for parametric aircraft				

*Basis for selected military configuration

- Additional considerations (intangibles)
  - Fuel use efficiency
  - Performance potential
  - Design flexibility
  - Technical margins
  - Combined civil/military program options

The 759-208M has a wing sweep of 35 degrees, a perpendicular thickness ratio of 0.22, and it cruises at Mach 0.79. The selected airplane payload vs. range curve is shown in Figure 39. This configuration evolved into the final military configuration, Model 759-213M, through changes to the crew compartment, fuel tank location, wing size, and airfoil shape (see Section 5.3.1.2 for details).

5.2.3.1 Military Tanker

The final military configuration, Model 759-213M, provides internal volume available for fuel in the wing leading edge and aft of the container compartment, between the aft spar and the auxiliary spar (see Figure 40). This volume exceeds the requirements for mission and payload fuel. This tanker version is capable of carrying 315 882 kilograms (696 400 pounds) of fuel as payload. The additional tanks required an increase in the OEW by 1179 kilograms (2600 pounds). As a permanent installation, no conversion time from cargo to tanker would be required. Arrangement of the tanks is shown in Figure 40.

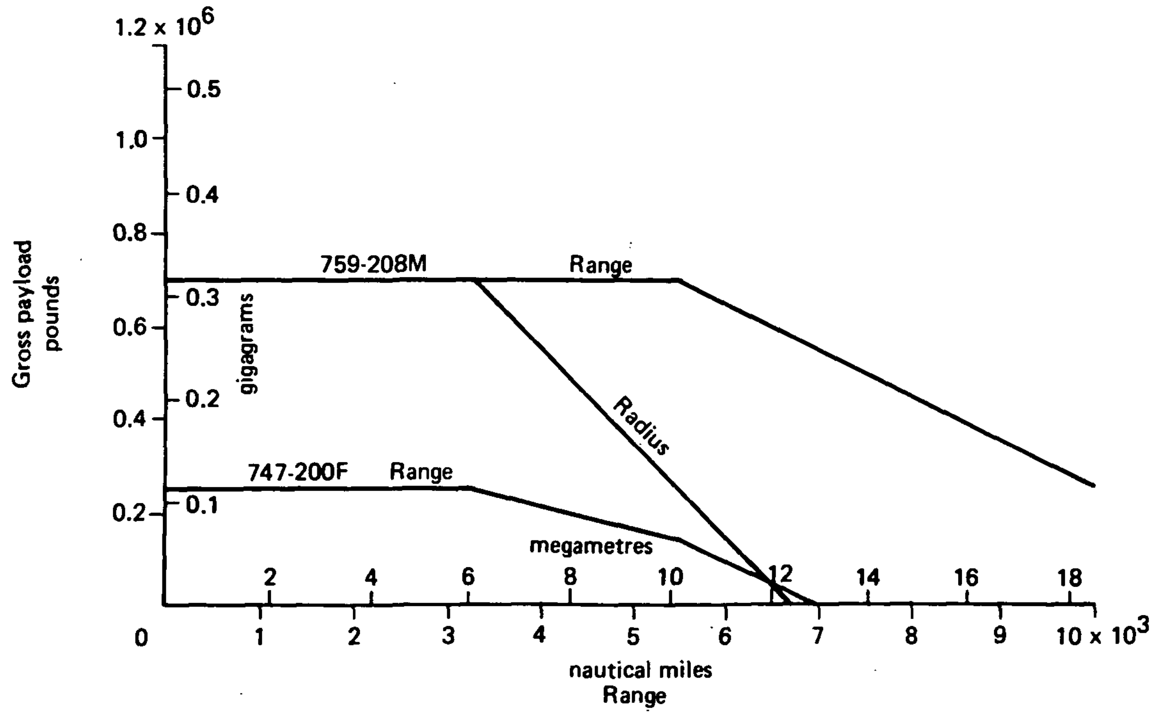


Figure 39 Selected Military Airplane Payload/Range (Payload/Radius)

MTOW = 887 921 kg (1 957 530 lb)
Maximum fuel = 592 144 kg (1 305 453 lb)

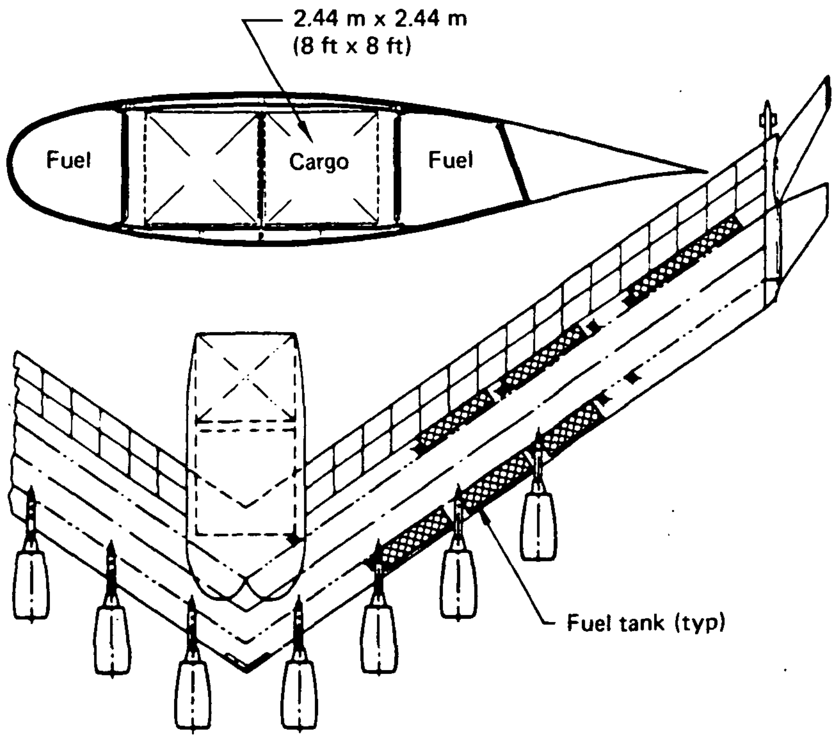


Figure 40 Military Tanker

5.2.3.2 Air Launch Long Range Cruise Missile (ALLRCM)

A stowage and handling concept that can be readily installed into the military DLF utilizes a modified SRAM-type rotary launcher. The launchers are mounted in tracks on special pallets that are moved into the airplane with its cargo-handling system. The racks are mounted in the first bay as shown in Figure 41. The racks are moved to the launch position at the wing tips where the missiles are ejected through a small launch door. After the complement of eight missiles is launched, the rack is transferred to the second bay for storage and another rotary rack is moved to the launch platform and the launch sequence is repeated.

The military DLF will accommodate 26 rotary racks and 208 AGM-86A, Class IIA missiles as shown in Figure 41. The gross payload is 590 000 pounds allowing for some increase in range or loiter time.

5.2.3.3 Air Launched M-X Missiles

The missiles considered are based on Boeing M-X studies and are shown in Figure 42. The baseline mission chosen for this study was the 81 647-kilogram (180 000-pound), 2.29-metre (90-inch) missile.

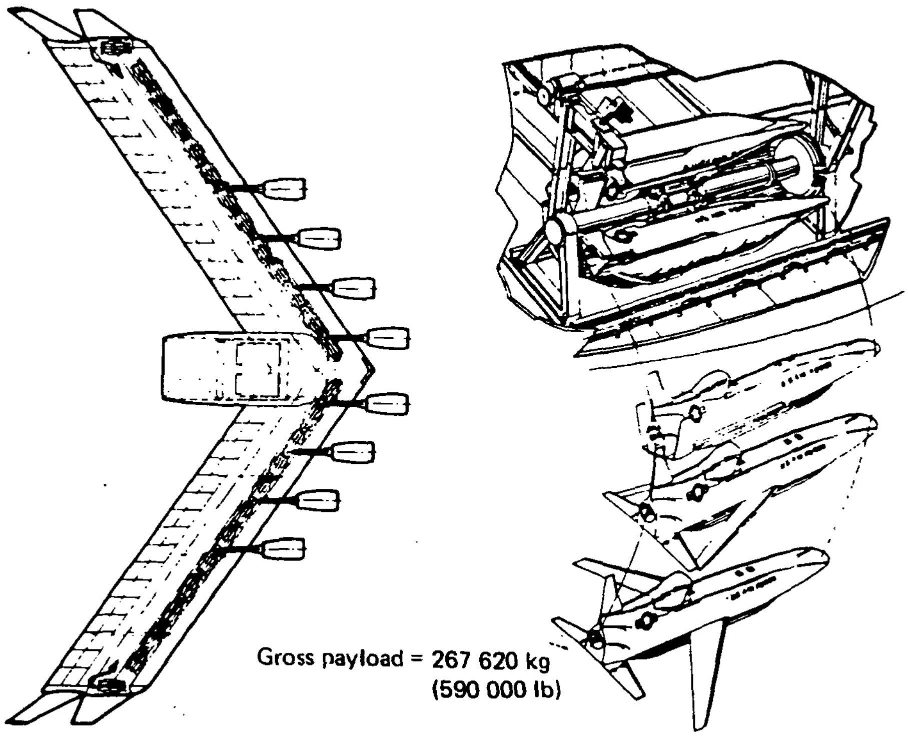


Figure 41 Military Payload—208 ALLRCM

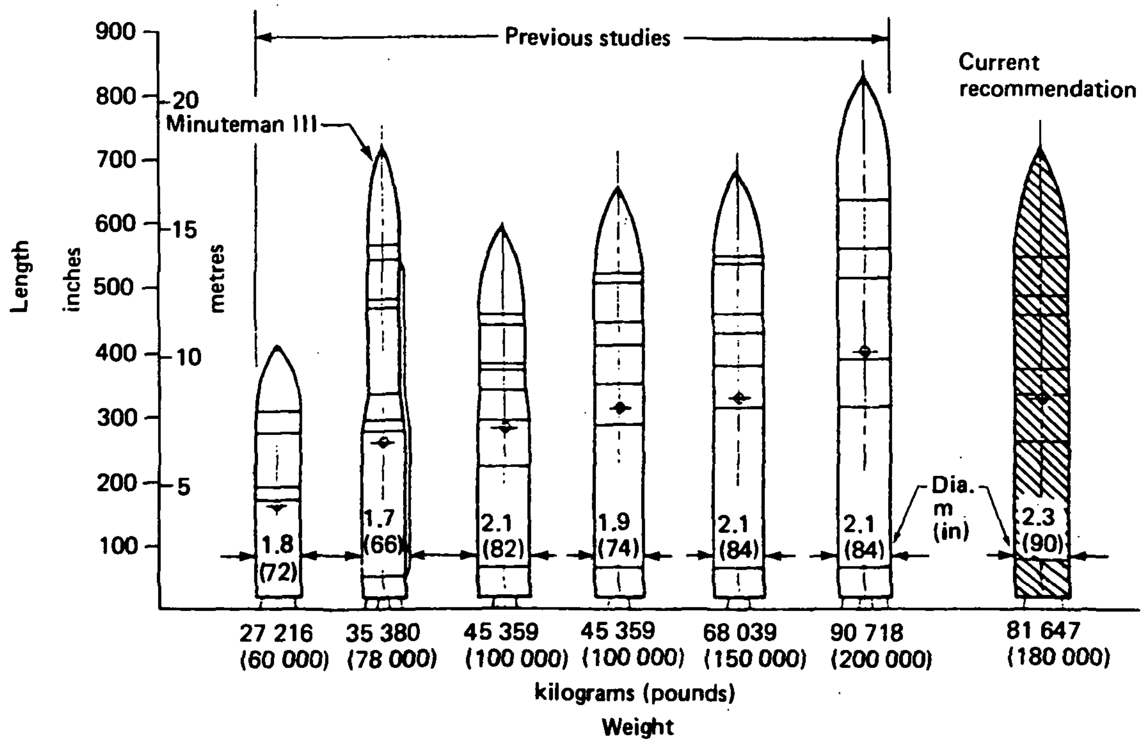


Figure 42 Candidate M-X Missiles—Air Mobile

Two launch methods were considered:
- Aft egress with parachute extraction
- Wing tip launched

Concept I Parachute Launch–As part of the M-X demonstration program, a Minuteman missile was launched by parachute launch initiation. After the missile was ejected and stabilized, first stage launch was successfully initiated.

Figure 43 illustrates this concept as applied to the military DLF configuration carrying two M-X missiles in the pod. This concept adapts well to a highly common logistics configuration with an aft loading door.

Concept II Wing Tip Launch–Wing tip carriage and launch of M-X missiles is a more effective method of launching missiles because of the initial velocity that is imparted to the missile.

The regular wing tip is replaced by a special tip, designed to launch the missile (Figure 44).

By off-loading approximately 45 359 kilograms (100 000 pounds) of fuel, it is possible to carry four M-X missiles.

5.2.4 SENSITIVITY STUDIES

5.2.4.1 Payload Container Size

Container Height Study–The impact of unconstrained container height was evaluated to determine the optimum cargo bay height using the selected parametric civil configuration, Model 759-195. Container heights of 1.83, 2.44, and 3.05 metres (6, 8, and 10 feet) were used. The t/c ⊥ of 0.16 and the net payload of 544 311 kilograms (1.2 million pounds) were held constant. Figure 45 compares the characteristics for each configuration. Figure 46 shows the weight trends as container height is varied.

The wing area and OEW increased as the container height decreased. The L/D increased as the container height decreased, cancelling out the OEW increase with a reduction in block fuel, resulting in little change in MTOW. The economics slightly favor the 8-foot high container as shown in Figure 47.

The wind loading is only 3160 pascals (66 pounds per square foot) for the 1.83-metre (6-foot) high container airplane; therefore, the effect of increasing t/c ⊥ to 0.19, thus increasing the wind loading, was investigated and the results are compared to the 16 percent wing in Figures 45 through 47. The economics continue to slightly favor the 2.44-metre (8-foot) high container airplane with the higher t/c (Figure 47).

Two Container Study–The impact of using two container sizes, the 2.44 x 2.44 metre (8 x 8 foot) and one of smaller cross section, was investigated in order to effectively utilize the available space for cargo. The selected civil configuration, Model 759-195, was used for the study. It has an excess of volume above that required for payload and fuel due to the relatively thin wing thickness ratio of 16 percent. The area aft of the rear spar is large enough to accommodate 46 LD-7 type containers [dimensions 2.24 x 1.63 x 3.14 metres (88 x 64 x 125 inches)] which would increase the net payload by 72 611 kilograms (160 080 pounds), see Figure 31. With a tare weight of 257 kilograms

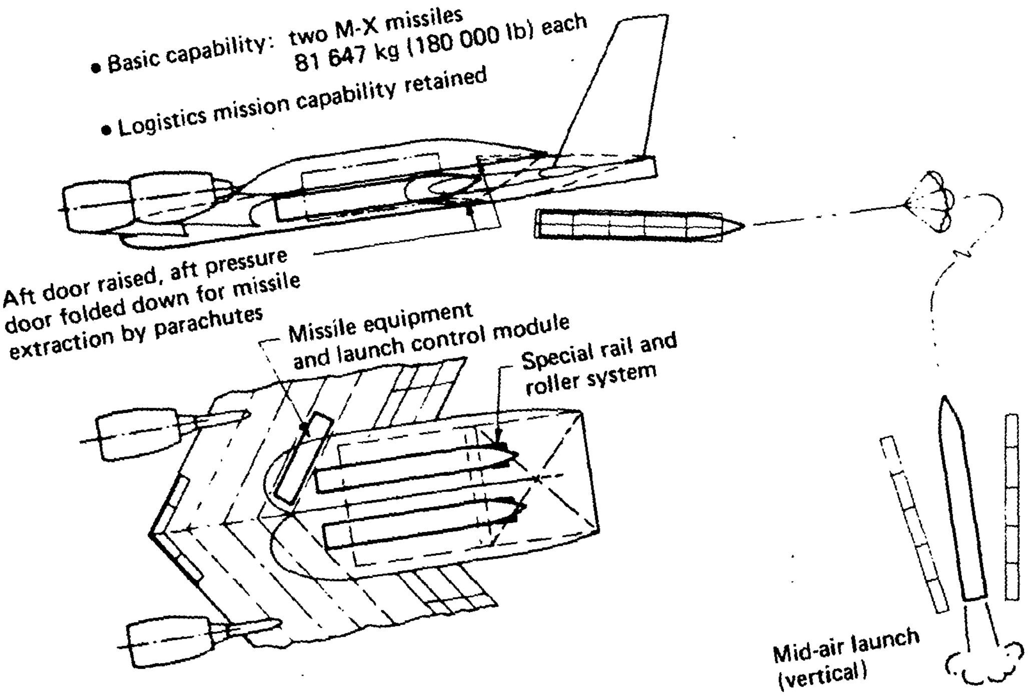


Figure 43 M-X Missile—Parachute Launch

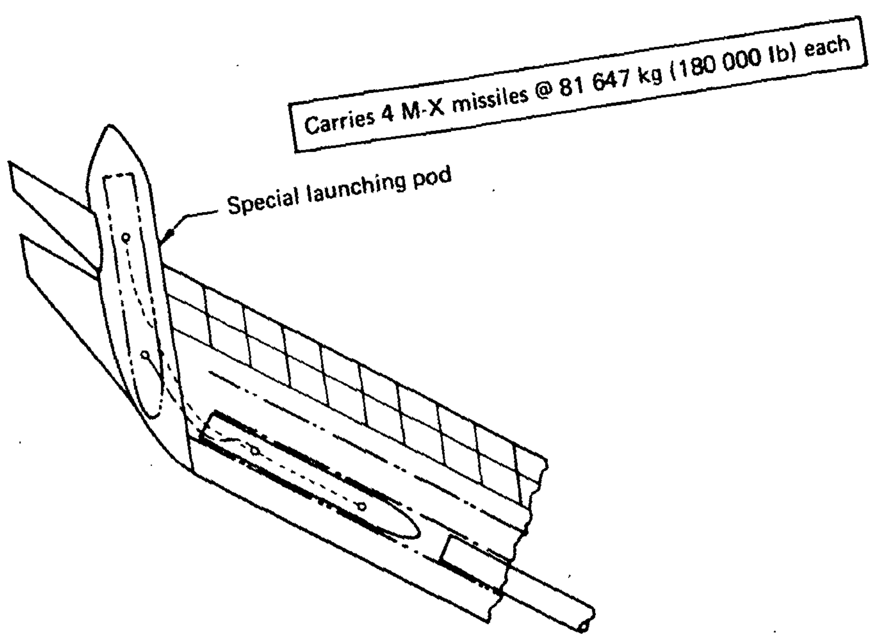


Figure 44 M-X Missile–Wing Tip Launch

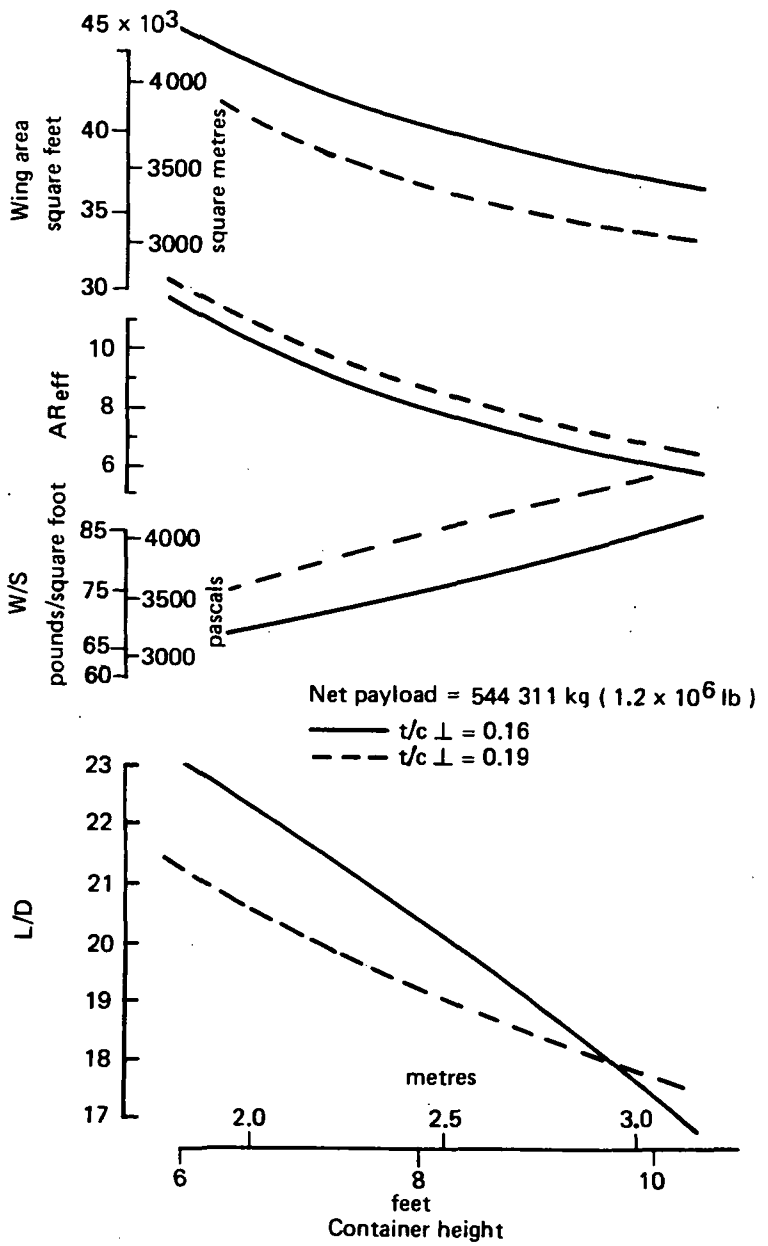


Figure 45 Configuration Characteristics—Container Study

(567 pounds) each the gross payload is increased by 84 442 kilograms (186 162 pounds). This is an increase of 13.3 percent in payload, resulting in a reduction of DOC + AIC of 2.2 percent (Table 6).

5.2.4.2 Range Study

Civil Range—The effect of design range upon economics was investigated using values of 5556, 6667, 8334, and 10 186 kilometres (3000, 3600, 4500, and 5500 nautical miles). Figure 48 shows the economic sensitivity to design range, including the off-design range. These curves show the minimum costs, DOC plus AIC, occur between 6667 and 8334 kilometres (3600 and 4500 nautical miles), and illustrate the outstanding efficiency of these configurations.

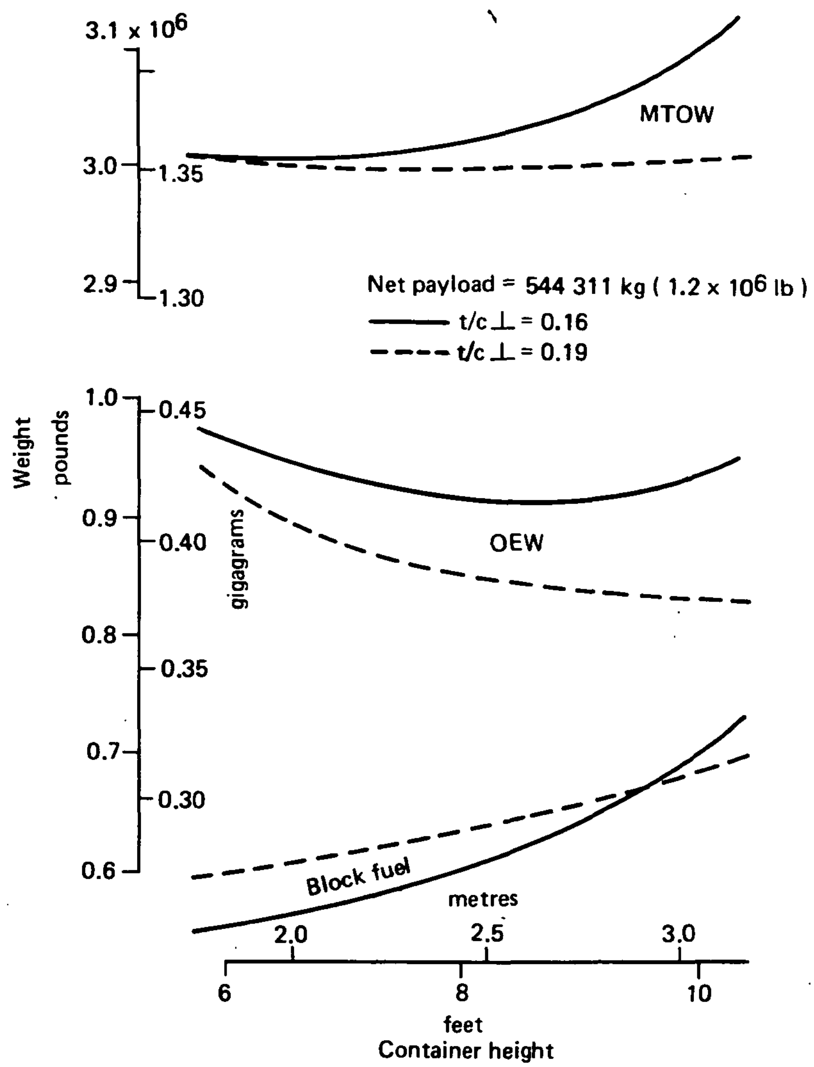


Figure 46 Weight Trends – Container Study

Military Range–The effect of design range upon airplane weight and block fuel using values of 6667, 8334 and 12 038 kilometres (3600, 4500, and 6500 nautical miles) was evaluated. Figure 49 shows the sensitivity of weight and block fuel to design range. Because the OEW increased very little as design range increased, the increase in takeoff gross weight is due mainly to the increase in block fuel. The off-design fuel for the 12 038 kilometre (6500 mile) design range airplane was compared with the design range fuel, as shown in Figure 49. The figure indicates that the design range could be increased to 12 038 kilometres (6500 nautical miles) with a small penalty in fuel burn for the shorter range missions.

5.2.4.3 Terminal Area

The landing gears are distributed and arranged along the wing span on the DLF airplanes to meet the following conditions:

- Each gear has a single wheel load below 26 981 kilogram (59 500 pounds) which is the maximum load of the 15.8 x 6.2-7 metre (52 x 20.5-23) size tire.

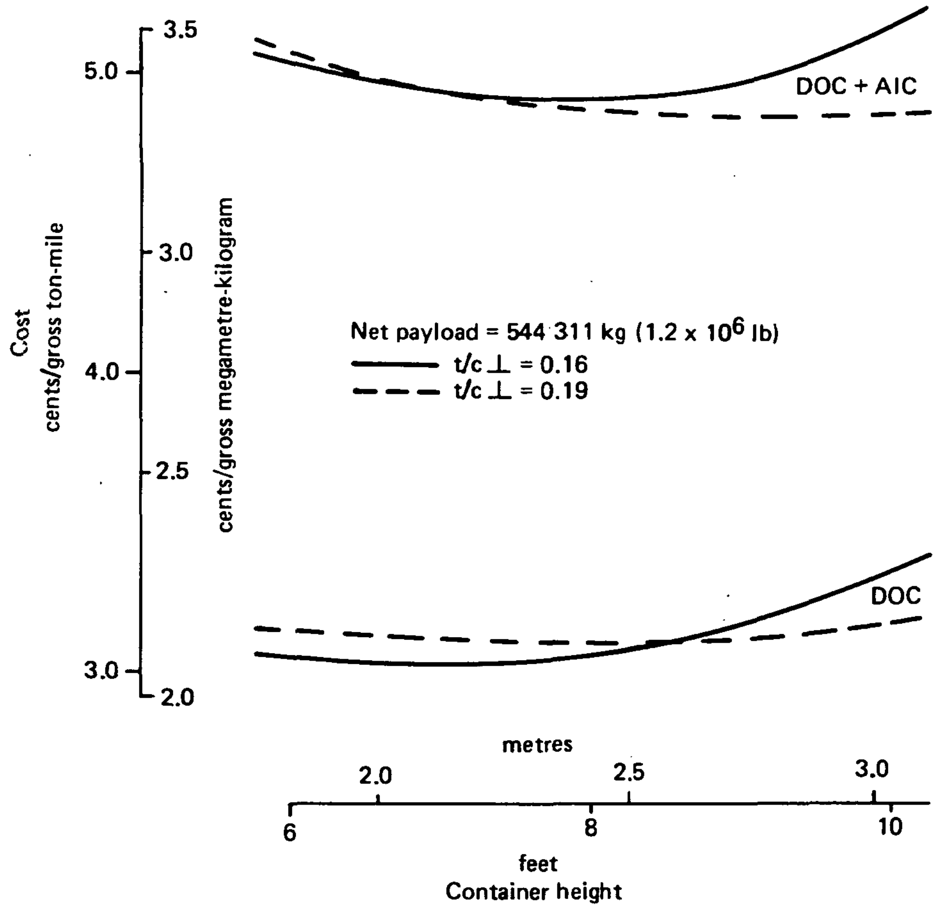


Figure 47 Economics – Container Study

- The airplanes are balanced by arranging the gear patterns fore and aft of the center of gravity.
- The outboard gears are located as far inboard as possible to keep runway width to a minimum without increasing runway thickness, and without increasing wing bending moment for the taxi condition above the maximum wing bending moment for the critical flight condition.

The landing gear arrangement for the final civil configuration, Model 759-211, is shown in Figure 50. Landing gear flotation for this arrangement is shown in Figure 51. Using a maximum runway stress of 2.76 megapascals (400 pounds per square inch), the required concrete runway thickness is 0.38 metre (15 inches). The DLF airplane is not able to use existing runways because the landing gear tread is wider than the runways. Since dedicated airports will be required, landing gear flotation need not be restricted to the current commercial jet airplane design limit.

The landing gear arrangement for the final military configuration, Model 759-213M, and landing gear flotation for this arrangement are shown in Figures 52 and 53 respectively. The required concrete runway thickness is 0.36 metre (14 inches).

The runway width requirement is primarily related to operations conducted under reduced visibility, the degree of control, maneuverability, stability of the airplane during final approach and landing, and additionally, to certain dimensional characteristics of the airplane.

Table 6 Two Container Airplane Comparison (SI Units)

Model	759-195	759-195A
Containers, m	2.44 x 2.44	2.44 x 2.44 + LD7
MTOW, kg	1 377 017	1 511 634
OEW, kg	418 458	438 732
Payload, kg		
Net	544 311	616 886
Gross	633 215	717 583
TSLS, N	391 444	417 243
TOFL, m	3094	3495
Fleet size	103	90
DOC (¢/Mmkg)	2.121	2.047
DOC + AIC (¢/Mmkg)	3.406	3.401

Table 6a Two Container Airplane Comparison (Customary Units)

Model	759-195	759-195A
Containers, ft	8 x 8	8 x 8 + LD7
MTOW, lb	3 035 803	3 332 583
OEW, lb	922 543	967 238
Payload, lb		
Net	1 200 000	1 360 000
Gross	1 396 000	1 582 000
TSLS, lb	88 000	93 800
TOFL, ft	10 151	11 466
Fleet size	103	90
DOC (¢/GTM)	3.097	2.988
DOC + AIC (¢/GTM)	4.973	4.866

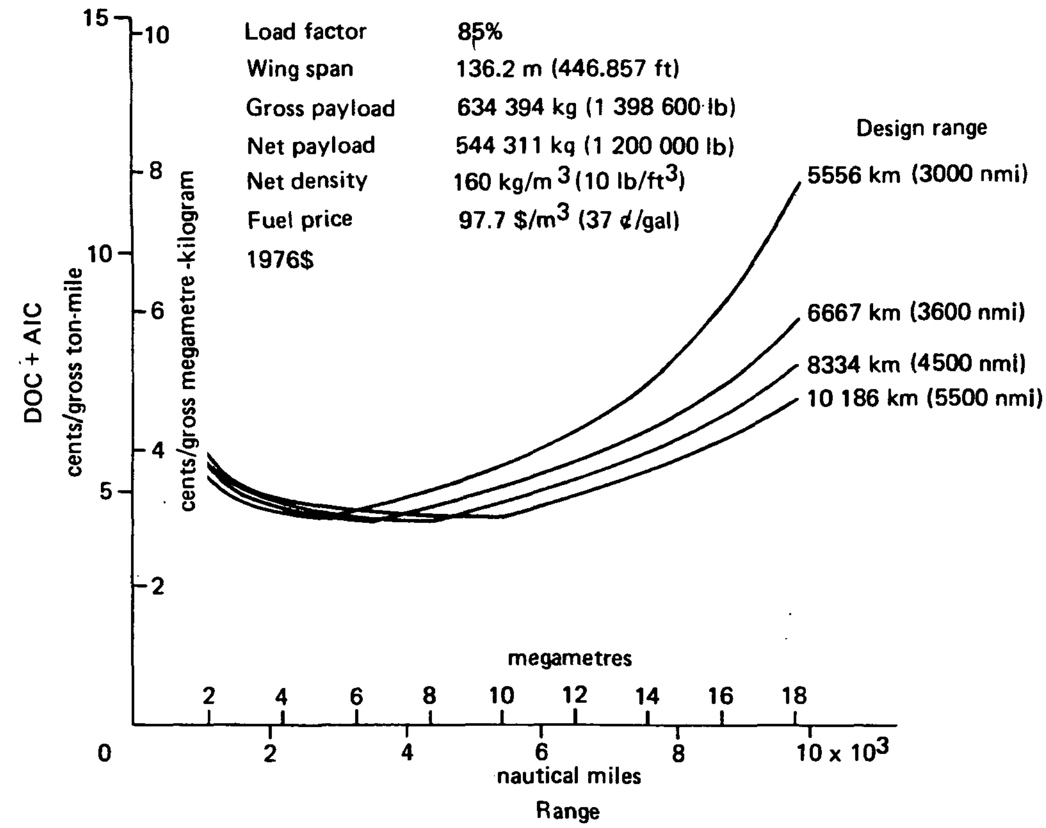


Figure 48 Design Range Sensitivity—Model 759-211

Reference 5 requires a 61-metre (200-foot) runway width to land the 747 which has a 59.7-metre (196-foot) wing span and 12.5-metre (41-foot) landing gear width. If DLF airplane control and maneuverability during approach are comparable to that of the 747, then a runway width allowing 24.4 metres (80 feet) between the outboard gear and the edge of the runway will be acceptable. Using these criteria the required runway width for the final civil configuration would be 149.4 metres (490 feet). The required runway width for the final military configuration would be 137.2 metres (450 feet).

5.3 FINAL CONFIGURATION STUDY

Selection Rationale—Selection rationale consists of comparing the various candidate configurations on the basis of a formal prescribed economic measurement criterion and applying the best judgment to a consideration of intangibles, analysis assumptions, and secondary considerations that could influence the final choice of configuration. The formal measurement of civil economic performance is the same (DOC + AIC) as used in the previously distributed load study (Reference 1). The formal measurement of military performance is the minimum 20-year life cycle cost.

Consideration of intangibles reflects additional experience and insight into the sensitivity of both aerodynamic and market effects that has been gained since the previous study was completed. A low speed wind tunnel test made since then has given the contractor experimental data with which to assess some of the potential technical risks of departing from the baseline configuration. Similarly, additional marketing studies have been conducted that provide an understanding of the relative impact of airplane characteristics on the ultimate market.

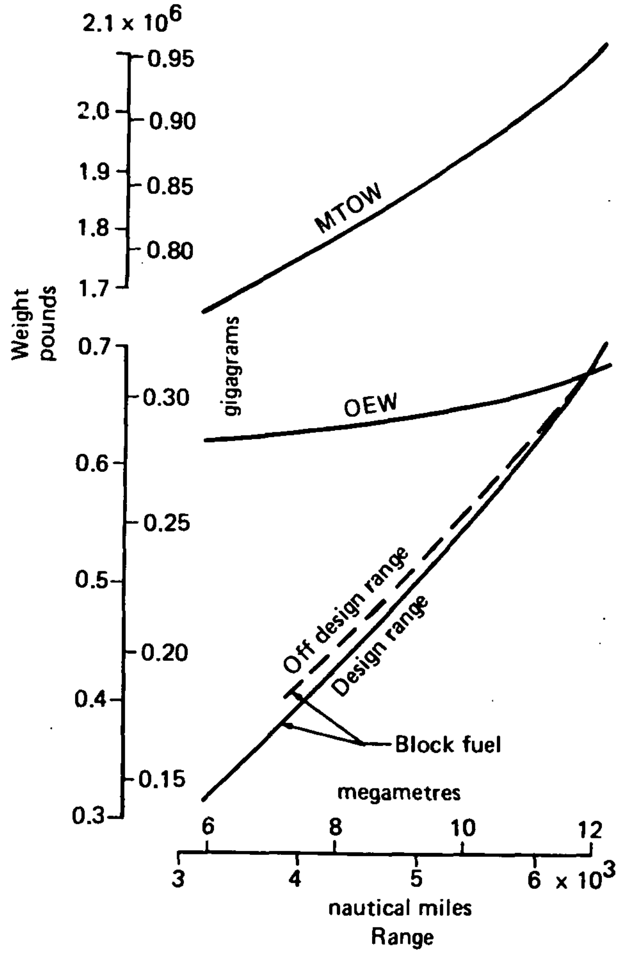


Figure 49 Military Design Range Sensitivity—Model 759-213M

Civil Market—Boeing has concluded that the annual 167.9 revenue petametre-kilogram (115 billion revenue ton-statute mile) total fleet productivity by 1995 specified in this study is a reasonable level at which to compare distributed load airplane systems. It is approximately midway between upper and lower levels of market size predicted by different assumptions. It results in reasonable fleet sizes for efficient production and the projected total transportation costs should include a reasonable margin for profit.

Consideration of Intangibles—The purpose of considering intangibles is to reduce the risk of choosing a configuration on the basis of this limited study and later find that more sophisticated and detailed analysis would have produced a different configuration. The objective is to consider and avoid technical and program risks that can be anticipated. The following list shows some of these intangibles and considerations:

- Sweepback—Thickness Ratio Trades—The critical low speed problem for these distributed load airplane designs is the takeoff lift coefficient. Since these airplanes do not rotate in pitch at takeoff, it is the lift coefficient on the ground at ground roll attitude which must be used. In the parametric study, the same design flap system was assumed to be used on all of the study airplanes. This single slotted flap, combined with a flexible deflected upper surface cove ahead of it, was tested on the baseline parametric design airplane in the Boeing 1.5 x 2.4-metre (5 x 8-foot) wind tunnel. The tests were successful and demonstrated trimmed lift coefficients

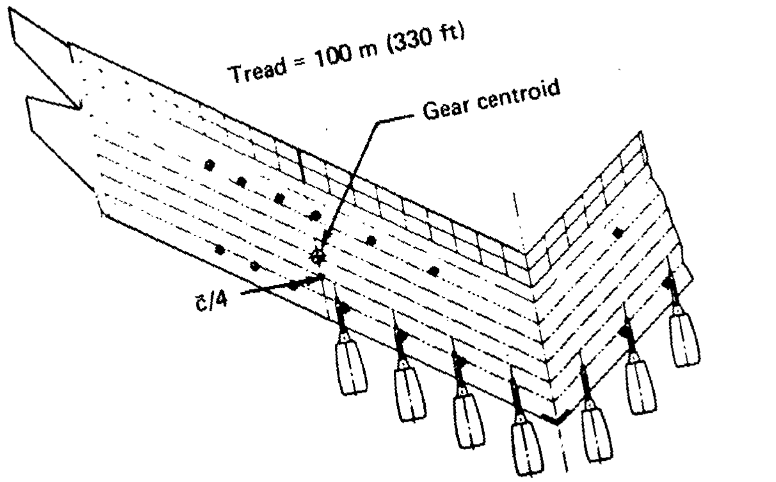


Figure 50 Landing Gear Arrangement—Model 759-211

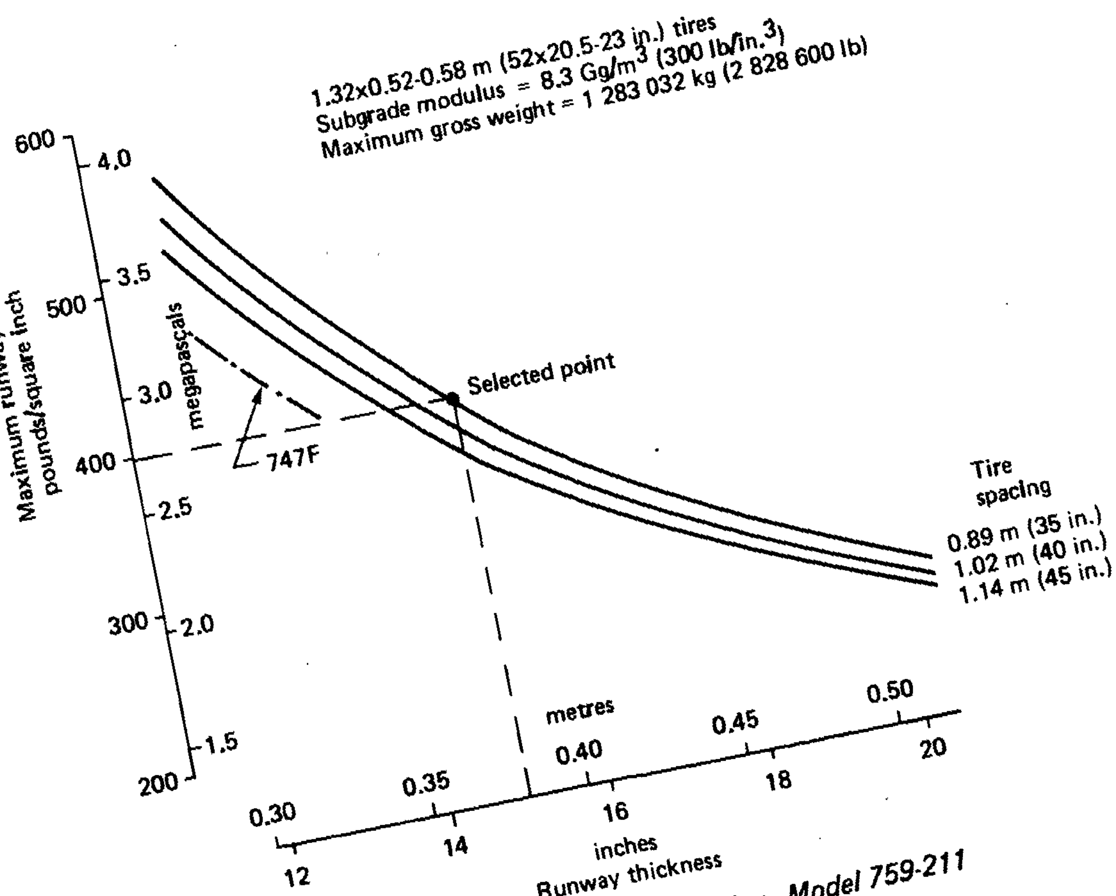


Figure 51 Rigid Pavement Flotation—Model 759-211

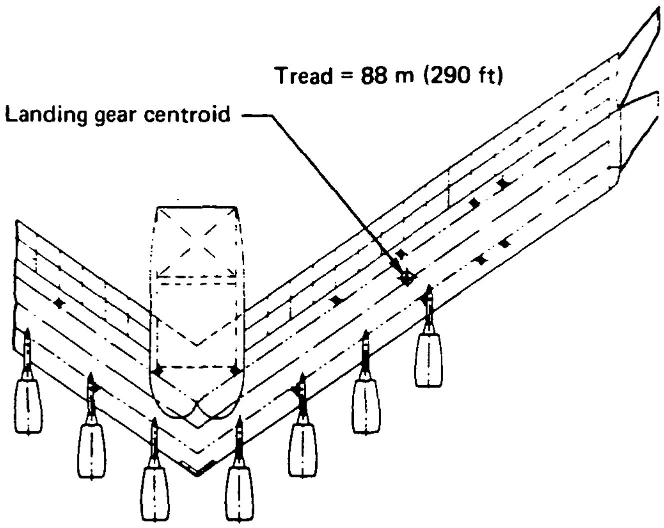


Figure 52 Landing Gear Arrangement— Model 759-213M

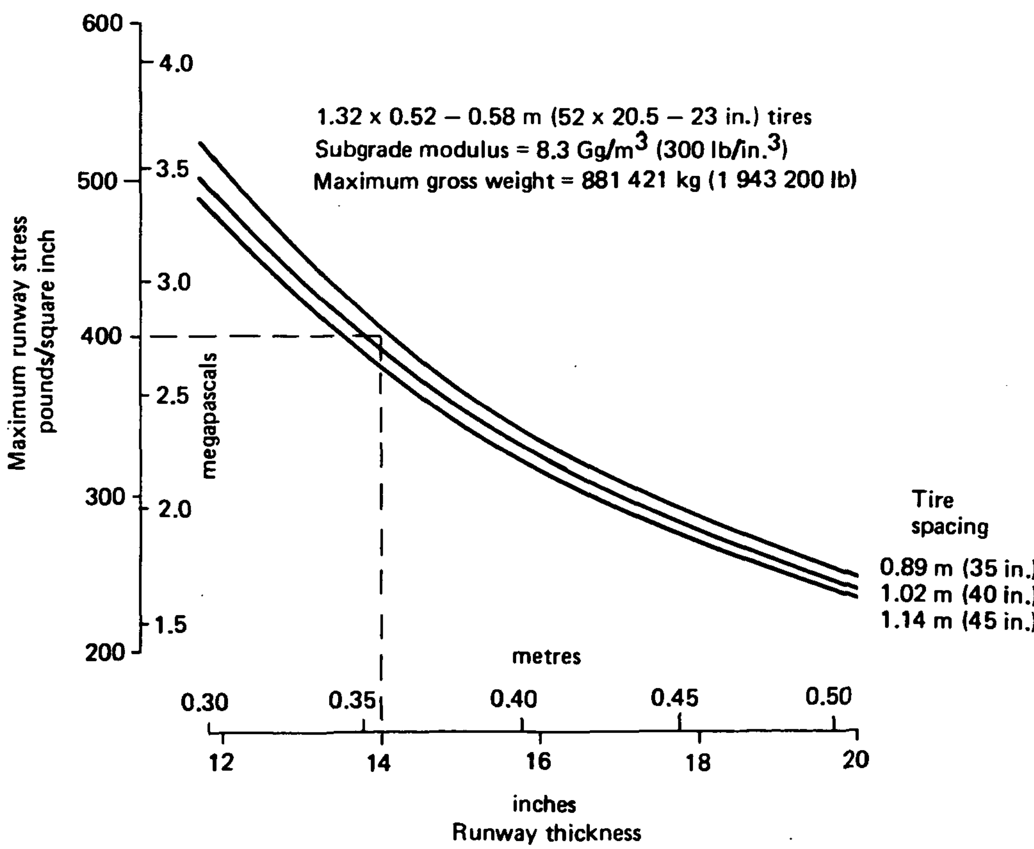


Figure 53 Rigid Pavement Flotation—Model 759-213M

higher than required for the 30-degree sweepback configuration. The effect of sweepback decreases this lift coefficient. There is sufficient lift to accommodate approximately 35 degrees of sweep with this design. Higher sweepback will require a larger, heavier trailing edge flap design. Hence, 40-degree sweptback designs would not be chosen unless they show sufficiently better economics to warrant a more detailed analysis of high lift system alternatives. Similarly, with thickness ratio choices, there are minimum and maximum thickness ratios beyond which the takeoff lift coefficient becomes difficult to obtain. The 16 to 25 percent thickness ratios used in the parametric study cover the range wherein the required lift can probably be attained without leading edge devices.

- Minimum Runway Width–Marketing studies based on the hub and spoke network concept, in which only a small number of hub cities are connected by DLF service, show that runway costs do not have a significant effect on the system economics. Therefore, runway width is not a factor affecting configuration choice.

- Military/Civil Combined Programs–Certain configuration designs involve less compromise in converting from Military to Civil application or vice versa. This is considered a favorable intangible.

- Speed or Range Potential–The higher Mach number designs can be considered for daily round trip non-stop schedules from the West Coast to Europe or Japan. This is a definite service and scheduling advantage that will result in higher utilization and ultimately lower costs than assumed by this preliminary study. Similarly, outstanding range performance can open up new civil markets (Australia to U.S., East Coast to Japan, South America to U.S., etc.) and military missions (fly out to destination and return unrefueled) that cannot be accommodated by present airfreight designs.

- Cross Section Flexibility–The four-bay wing cross section designed for 2.44-metre (8-foot) high containers can handle 2.74 and 3.05-metre (9- and 10-foot) high cargo in the center two bays with less compromise than the three-bay. Four-bay cross sections that are over 21.5 percent thick can handle the M-60 tank internally.

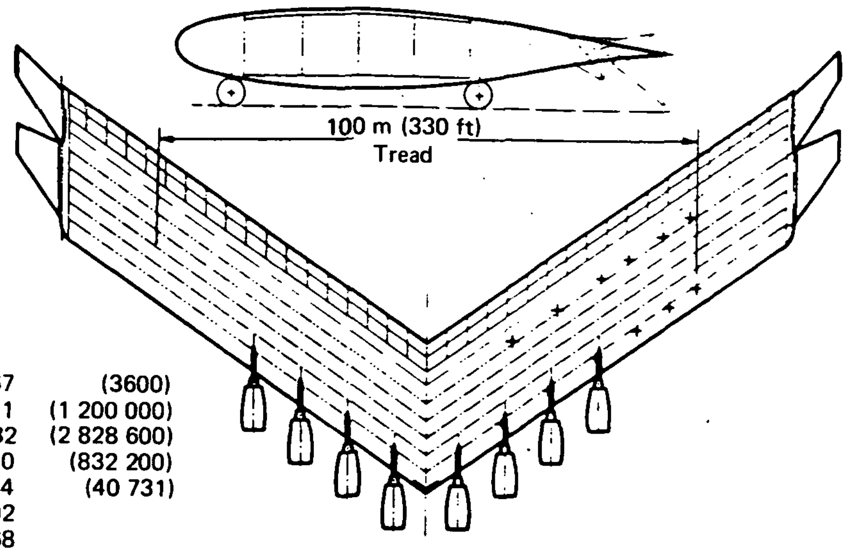
5.3.1 FINAL CONFIGURATION DEFINITION

5.3.1.1 Civil Configuration

The final civil configuration (see Figure 54) is the same as the parametric selected configuration, 759-195, except for the following:

- Crew compartment installed in leading edge of the wing, eliminating the body
- L.E. fuel tanks removed outboard of the outboard engines, eliminating bird strike penalty and the requirement for 9-g cargo restraint in these areas
- Wing sized to the nearest number of whole containers to match the payload
- Wing airfoil cambered to reduce drag due to lift
- Eliminate alternate field fuel reserve requirements with a Class IIIC landing system
- Thrust reversers removed

The configuration characteristics for the final civil, Model 759-211, are contained in Table 7.



Range, km(nmi)	6667	(3600)
Net payload, kg (lb)	544 311	(1 200 000)
MTOW, kg(lb)	1 283 032	(2 828 600)
OEW, kg(lb)	377 480	(832 200)
Wing area, m^2(ft^2)	3784	(40 731)
Basic aspect ratio	4.902	
Effective aspect ratio	7.868	
Basic wing span, m(ft)	136	(447)
Total wing span, m(ft)	153	(503)
Sweep, degrees	35	
t/c⊥	0.16	
Cruise Mach number	0.85	
Engines BPR	9.5	
TSLS, N(lb)	375 875	(84 500)

Figure 54 Final Civil Configuration—Model 759-211

The systems description follows:

- Pneumatic system (Figures 55 and 56)—Engine bleed air is used for flight deck air-conditioning and pressurization, cargo compartment heating, and engine inlet thermal anti-icing. Engine bleed air system is similar to the Model 747.
  - Engine starting—Pneumatic starters are driven by either APU air or engine bleed air during cross starting.
  - Engine inlet TAI—The pneumatic thermal anti-icing system is the same as the Model 747 but scaled for engine size.
  - Air-conditioning—Two small simple boostrap air cycle units similar in size and arrangement to Gulfstream II/business jets air-conditioning packs are used for the flight deck air-conditioning and pressurization.

    Four air cycle air-conditioning packs similar to the Model 747 but scaled for increase in pack airflow are used in cargo compartment air-conditioning. Four cargo compartment air recirculation fans are installed in the return air ducts to mix the compartment air with a fresh supply of air.

- Hydraulic system—A 27.6 megapascal (4000 psi) hydraulic system supplies a conventional flap drive system. Total hydraulic power requirement for driving the trailing edge flaps is approximately 5.07 megawatts (6800 horsepower). This hydraulic power is sufficient to actuate 50 percent of the trailing-edge flaps at the maximum rate.

Table 7 Configuration Characteristics Final Civil–Model 759-211 (SI Units)

	Wing	Horizontal (each)	Vertical (each)
Area, m^2	3784.090	82.710	190.723
Span, m	136.202	8.505	23.935
Aspect ratio	4.902	0.874	3.003
Effective aspect ratio	7.868		
$\bar{c}/4$ sweep, deg	35	50	30
$\bar{c}/4$ station, m	56.304	83.908	98.788
MAC (Norm. to LE), m	22.758		
MAC (S.W.), m	27.782	10.319	8.218
t/c (Norm. to LE theoretical chord), %	16		
t/c (S.W. theoretical chord), %	13.016	12	12
$\bar{y}$, m	34.050	3.645	10.742
Root chord, m	27.782	13.891	10.415
Tip chord, m	27.782	5.556	5.520
Taper Ratio	1	0.4	0.53
Tail arm, m		27.603	3.540
Volume coefficient (total)		0.04271	0.03144

Power plants	Number	Type	BPR	TSLS (N)
	8	TF1990	9.5	375 875

Landing gear	No. gear	No. tires	Tire size, m
	24	48	1.3208 x 0.5207-0.5842

Gross containerized volume = 3793 m^3
Number of 2.44 x 2.44 x 6.10 m containers = 104
Number of bays = 4
Vertical cant. = 20.81 deg
Horizontal span/wing span ratio = 0.12490
Front spar location, % chord = 13.726
Rear spar location, % chord = 59.81
Front spar to rear spar length = 10.490 m

Hydraulic pumps are powered by the main engines and 1.49 megawatt (2000 horsepower) APUs.

- Electrical system–Four 60-KVA IDGs are installed on the main engines to provide airplane electrical power. In addition, two 90-KVA generators are installed on each APU to supply electrical power for cargo loading.

- Avionics–Adequate avionics are provided for category III B/C landing capability.

5.3.1.2 Military Configuration

The final military configuration (see Figure 57) is the same as the parametric selected configuration, Model 759-208M, except as follows:

- Crew compartment installed in leading edge of the wing, eliminating the body
- L.E. fuel tanks removed outboard of the outboard engines, eliminating bird strike penalty and the requirement for 9-g cargo restraint in these areas.

Table 7a Configuration Characteristics Final Civil—Model 759-211 (Customary Units)

	Wing	Horizontal (each)	Vertical (each)
Area, ft^2	40 731.607	890.291	2052.934
Span, ft	446.857	27.906	78.529
Aspect ratio	4.902	0.874	3.003
Effective aspect ratio	7.868		
$\bar{c}/4$ sweep, deg	35.000	50.000	30.000
$\bar{c}/4$ station, in.	2216.710	3303.487	3889.327
MAC (Norm. to LE), in.	896.000		
MAC (S.W.), in.	1093.814	406.273	323.573
t/c (Norm. to LE theoretical chord), %	16.000		
t/c (S.W. theoretical chord), %	13.106	12.000	12.000
$\bar{y}$, in.	1340.573	143.517	422.930
Root chord, in.	1093.814	546.907	410.073
Tip chord, in.	1093.814	218.762	217.338
Taper ratio	1.000	0.400	0.530
Tail arm, ft		90.564	139.384
Volume coefficient (total)		0.04271	0.03144

Power plants	Number	Type	BPR	TSLS (lb)
	8	TF1990	9.5	84 500

Landing gear	No. gear	No. tires	Tire size, in.
	24	48	52x20.5-23

Gross containerized volume = 133 952 ft^3
Number of 8 x 8 x 20 ft containers = 104
Number of bays = 4
Vertical cant. = 20.81 deg
Horizontal span/wing span ratio = 0.12490
Front spar location, % chord = 13.726
Rear spar location, % chord = 59.81
Front spar to rear spar length = 413 in.

- Wing sized to the nearest number of whole containers to match the payload
- Wing airfoil cambered to reduce drag due to lift

This configuration was designed to meet the following military mission requirements:

- Net payload 272 155 kg (600 000 lb)
  - 2.44 x 2.44 x 6.1/2.44 x 2.44 x 12.19 m (8 x 8 x 20/8 x 8 x 40 ft) containers
  - 463L pallets
  - Military equipment—roll on and roll off
  - Two 4.11 m (13.5 ft) h x 5.18 m (17 ft) w x 12.19 m (40 ft) l cargo bays
- Design range 10 186 km (5500 nmi)
- Design critical field length 3048 m (10 000 ft)
- Cargo compartments pressurized

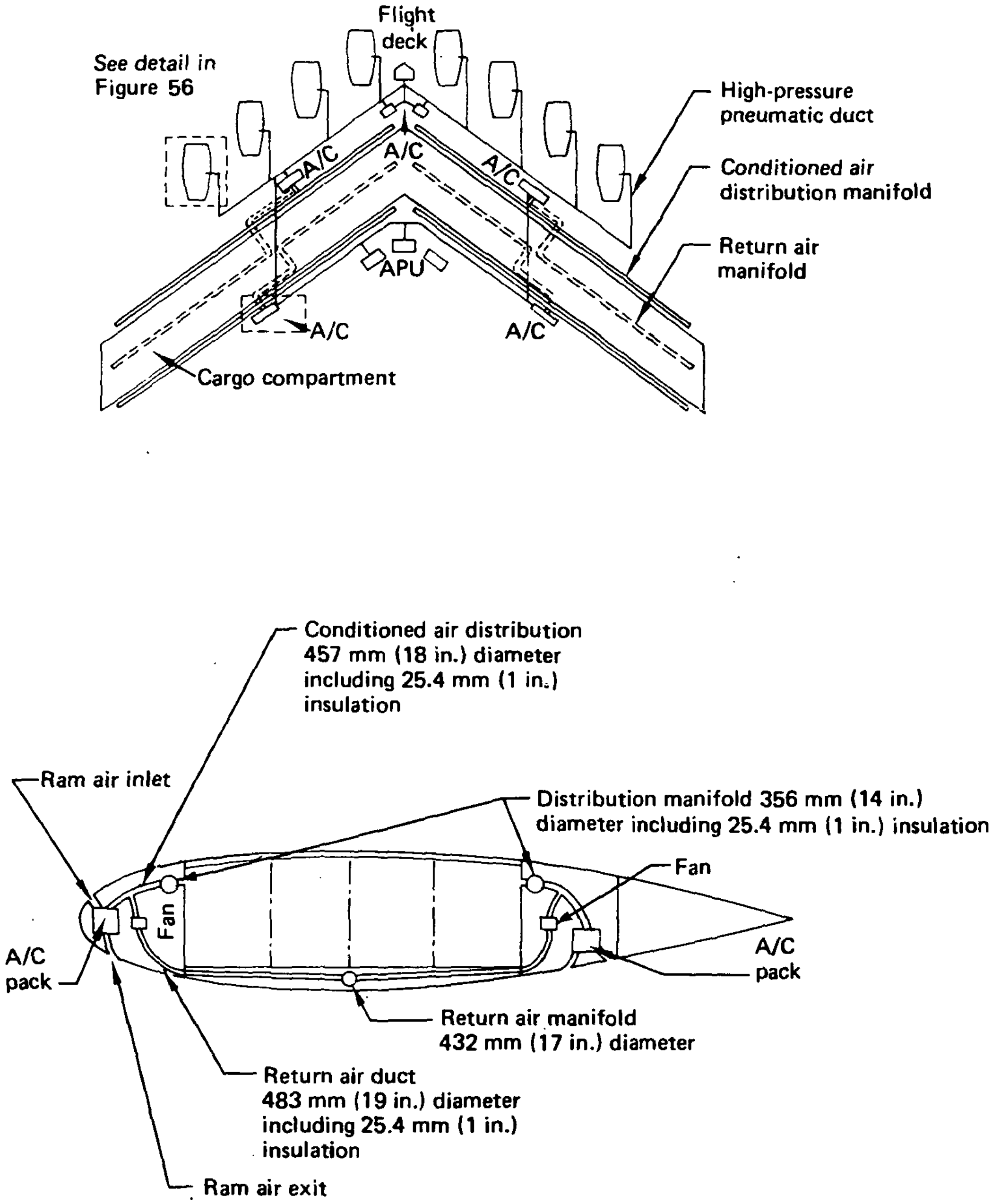
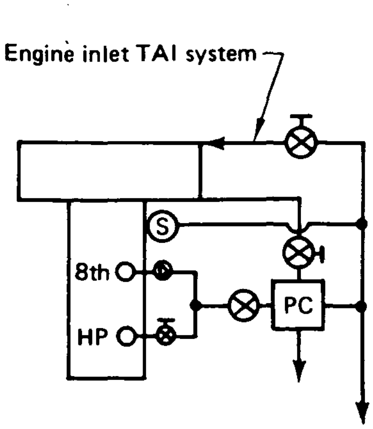
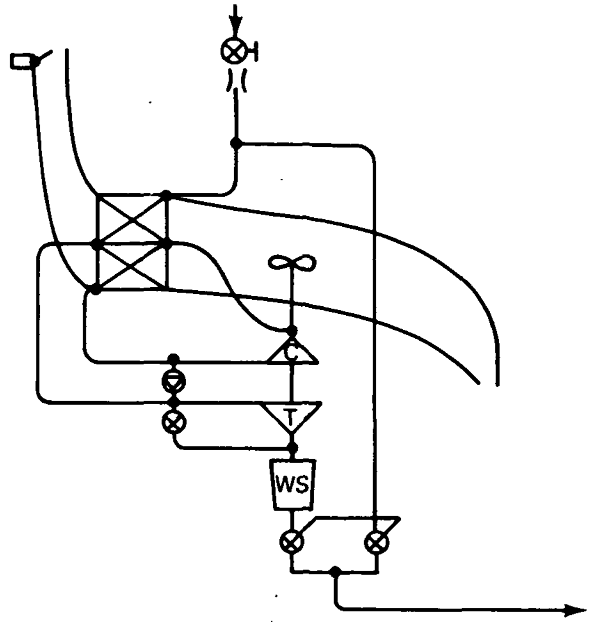


Figure 55 Environmental Control Systems—Model 759-211



Engine bleed system



Air conditioning pack

Note:

WS – water separator
C – compressor
T – turbine
PC – pre-cooler
S – starter
HP – high pressure

Figure 56 Environmental Control System Details

Range, km(nmi)	10 186	(5500)
Net payload, kg (lb)	272 155	(600 000)
MTOW, kg(lb)	887 907	(1 957 500)
OEW, kg(lb)	295 742	(652 000)
Wing area, $m^2(ft^2)$	2114	(22 753)
Basic aspect ratio	6.380	
Effective aspect ratio	9.785	
Basic wing span, m(ft)	116	(381)
Total wing span, m(ft)	127	(417)
Sweep, degrees	35	
t/c⊥	0.22	
Cruise Mach number	0.79	
Engines BPR	9.5	
TSLS, N(lb)	245 542	(55 200)

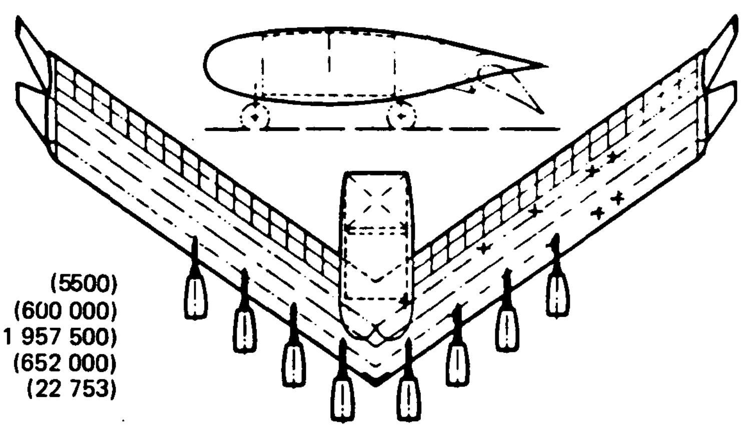


Figure 57 Final Military Configuration—Model 759-213M

Additional considerations

- Capability to air launch:
    - ALLRC missiles
    - M-X missiles

The configuration characteristics for the final military, Model 759-213M, are contained in Table 8. The systems description for this configuration is the same as for the civil configuration (see Section 5.3.1.1).

Figure 58 shows the cross sections of the pod and wing cargo compartments. Figure 59 shows some of the military payloads that can be carried in the wing. The -212M (Figure 60) is the same as -213M except an air cushion landing gear is included to show the relative performance and costs for that option.

Table 8 Configuration Characteristics, Final Military—Model 759-213M (SI Units)

	Wing	Horizontal (each)	Vertical (each)
Area, m^2	2113.833	34.809	103.591
Span, m	116.135	5.464	19.844
Aspect ratio	6.38	.857	3.801
Effective aspect ratio	9.785		
$\bar{c}/4$ sweep, deg	35	50	26
$\bar{c}/4$ station, m	50.396	71.822	81.906
MAC (Norm. to LE), m	14.910		
MAC (S.W.), m	18.201	6.761	5.384
t/c (Norm. to LE theoretical chord),%	22		
t/c (S.W. theoretical chord), %	18.201	12	12
$\bar{y}$, m	29.034	2.342	8.906
Root chord, m	18.201	9.101	6.824
Tip chord, m	18.201	3.640	3.617
Taper ratio	1	0.4	0.53
Tail arm, m		21.425	31.510
Volume coefficient (total)		0.03780	0.02659

Power plants	Number	Type	BPR	TSLS (N)
	8	TF1990	9.5	245 542

Landing gear	No. gear	No. tires	Tire size, m
	18	36	1.3208 x 0.5207-0.5842

Gross containerized volume = 1605 m^3
Number of 2.44 x 2.44 x 6.10 m containers = 44
Number of bays = 2
Vertical cant. = 15.98 deg
Horizontal span/wing span ratio = 0.09410
Front spar location, % chord = 17.391
Rear spar location, % chord = 52.48
Front spar to rear spar length = 5.232 m

Air Cushion Landing System (ACLS) For Mode 759-212M—The ACLS definition follows:

- Air cushion—ACLS used large volumes of low pressure air to support an airplane on the ground. An inflatable trunk serves as a skirt to trap a large volume of low pressure air underneath the airplane. Cushion air is discharged through a series of holes placed at the bottom of the trunk.

  All the ACLSs developed in the past used the same doughnut-shaped trunk. However, it would be impractical to design a doughnut-shaped trunk for the DLF.

  The air cushion design for the military DLF consists of inflatable trunks at the leading edge and trailing edge and two sides of the wing to enclose cushion air as shown in Figure 61. The air cushion covers approximately 75 percent of the cargo floor area; therefore, the cargo floor structures are evenly supported on the ground.

Table 8a Configuration Characteristics, Final Military—Model 759-213M (Customary Units)

	Wing	Horizontal (each)	Vertical (each)
Area, ft^2	22 753.108	374.686	1115.042
Span, ft	381.020	17.927	65.105
Aspect ratio	6.380	0.857	3.801
Effective aspect ratio	9.785		
$\bar{c}/4$ sweep, deg	35.000	50.000	26.000
$\bar{c}/4$ station, in.	1984.106	2827.628	3224.656
MAC (Norm. to LE), in.	587.000		
MAC (S.W.), in.	716.594	266.163	211.983
t/c (Norm. to LE theoretical chord),%	22.000		
t/c (S.W. theoretical chord), %	18.021	12.000	12.000
$\bar{y}$, in,	1143.061	92.196	350.635
Root chord, in.	716.594	358.297	268.652
Tip chord, in.	716.594	143.318	142.385
Taper ratio	1.000	0.400	0.530
Tail arm, ft		70.293	103.379
Volume coefficient (total)		0.03780	0.02659

Power plants	Number	Type	BPR	TSLS (lb)
	8	TF1990	9.5	55 200

Landing gear	No. gear	No. tires	Tire size, in.
	18	36	52x20.5-23

Gross containerized volume = 56 672 ft^3
Number of 8 x 8 x 20 ft containers = 44
Number of bays = 2
Vertical cant = 15.98 deg
Horizontal span/wing span ratio = 0.09410
Front spar location, % chord = 17.391
Rear spar location, % chord = 52.48
Front spar to rear spar length = 206 in.

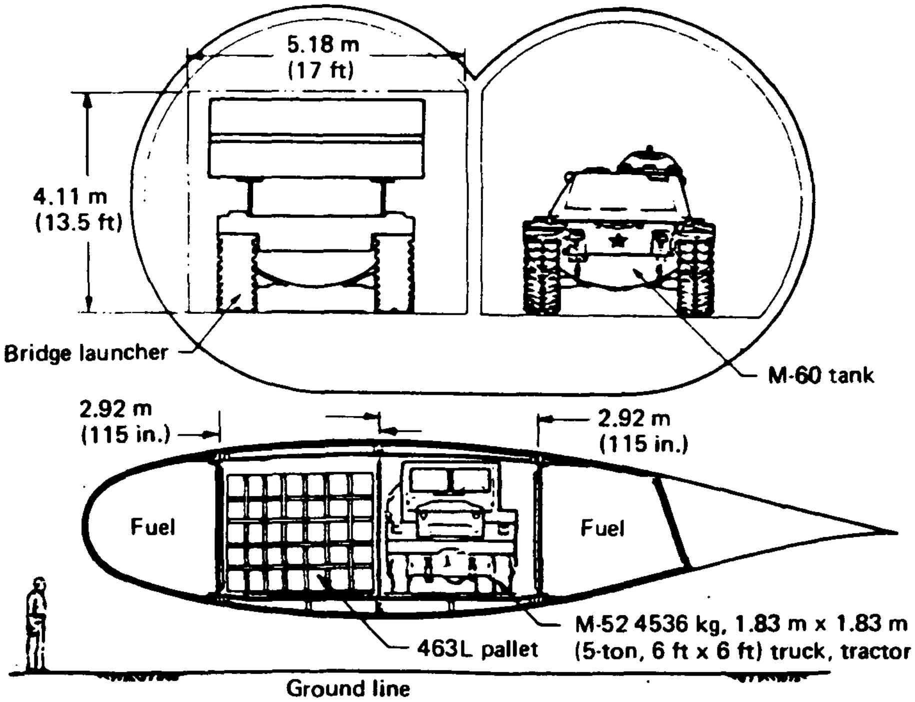


Figure 58 Military Payload Arrangement

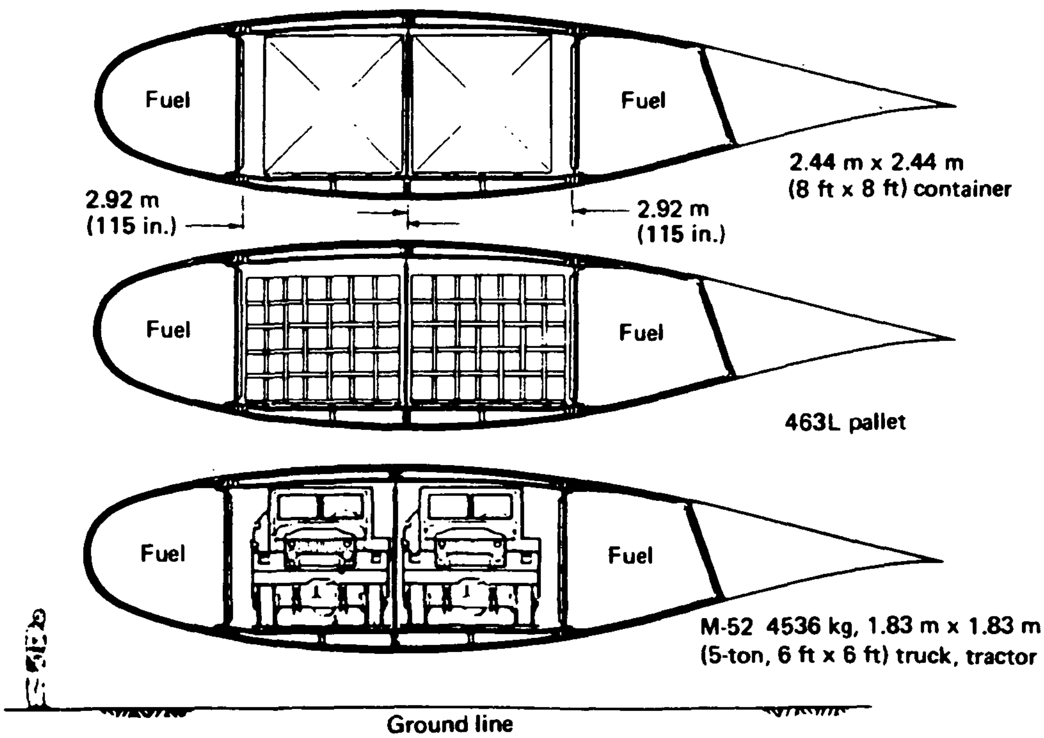
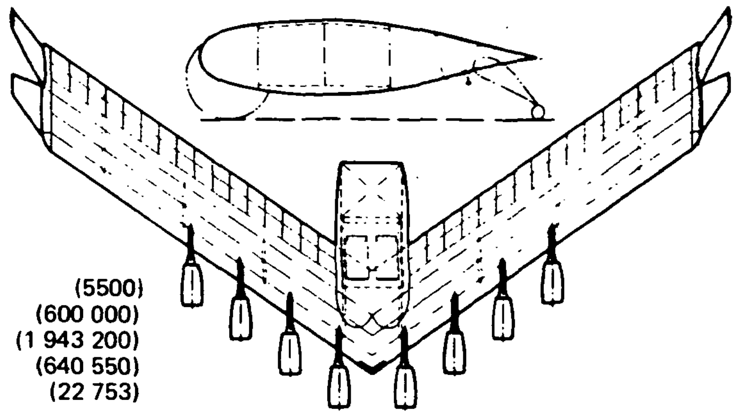
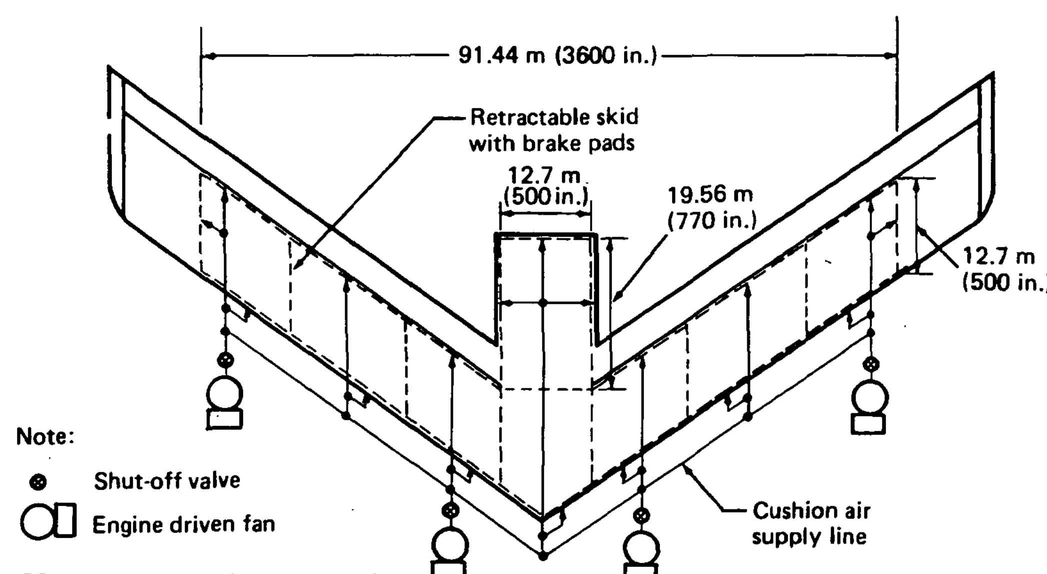


Figure 59 Military Payload—t/c = 0.22



Range, km(nmi)	10 186	(5500)
Net payload, kg (lb)	272 155	(600 000)
MTOW, kg(lb)	881 421	(1 943 200)
OEW, kg(lb)	290 549	(640 550)
Wing area, $m^2(ft^2)$	2114	(22 753)
Basic aspect ratio	6.380	
Effective aspect ratio	9.785	
Basic wing span, m(ft)	116	(381)
Total wing span, m(ft)	127	(417)
Sweep, degrees	35	
t/c⊥	0.22	
Cruise Mach number	0.79	
Engines BPR	9.5	
TSLS, N(lb)	241 983	(54 400)

Figure 60 Final Military Air Cushion Option—Model 759-212M



MTOW = 881 421 kg (1 943 200 lb)
Cushion area = 1384 m^2 (2 145 000 $in.^2$)
Cushion pressure = 6205 Pa (0.9 psig)
Trunk pressure = 12 411 Pa (1.8 psig)

Figure 61 Air Cushion Landing System—759-212M

The trunks are made of one-way stretch material and inflated with 12.4 kilopascals (1.8 psig) air pressure, which is approximately twice the cushion air pressure. Use of stretch material eliminates the need for mechanical drums to restow the trunks in flight.

Several stationary skids are placed under the wing longitudinally and divide the air cushion into eight sections. These skids are used for parking support and also to hold the inflatable boot for the brake pads. The stationary skids may add some aerodynamic drag in flight, but eliminate the need for retracting mechanism. The leading edge and trailing edge trunks are segmented into seven sections. Therefore, rupturing any section would not cause complete collapse of the entire air cushion.

- Air supply—The cushion airflow requirement is approximately 454 kilograms (1000 pounds) per second. The cushion air is supplied by low pressure fans powered by four gas turbine units as shown in Figure 62. Four 2.98-megawatt (4000-horsepower) gas turbine units will be required to generate the cushion air.

  The amount of cushion air required for landing requires substantially less than takeoff condition; therefore, the airplane can make normal landings with one gas turbine unit having failed.

- Ground clearance—The amount of air supply selected here provides one-half inch of air gap beneath the trunks. Whether this amount of gap is sufficient for this size of aircraft is not known at this time. Since the amount of air supply required is proportional to the height of the air gap, increase of the height would proportionally increase the power requirement of the air source.

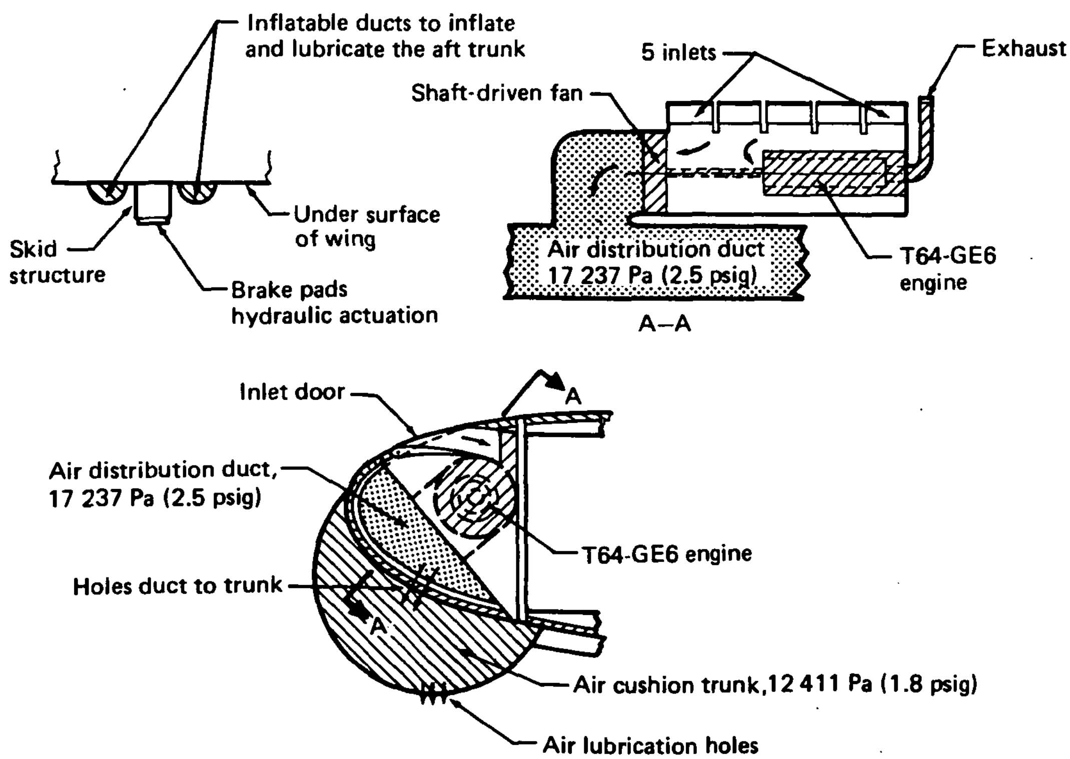


Figure 62 Air Cushion Landing System— Details (759-212M)

• Ground directional control–At low airplane speed, airplane flight control surfaces are not effective in providing directional control. Positive directional control or steering capabilities are needed to properly control the airplane in side-wind conditions. Partially increasing engine thrust or reversing thrust on one side of the airplane creates sufficient differential thrust which would provide yaw control in cross-wind conditions. Various compartments of the compartmentalized air cushion can be partially deflated to produce a high differential braking force which would also produce suitable yaw moments during cross-wind taxiing.

  The use of auxiliary wheels with a steering capability to support approximately 10 percent of the airplane weight would provide adequate ground directional control on improved runways.

5.3.1.3 Convertible Configuration

Figure 63 illustrates the relationship between the various military and civil/military configurations. Table 9 indicates the differences used to define each configuration. The final military configuration 759-213M (Figure 57) is an uncompromised airplane designed to meet the military mission. It is assumed that 125 of these airplanes would be built, with the nonrecurring cost distributed among the 125 airplanes in the fleet.

The civil/military versions are composed of the 125 organic military airplanes, identified as 759-213MX, which will be the same airplane as the 759-213M except for the difference required to maintain commonality between the -213C and -213MX. The -213C (Figure 64) is a commercial airplane designed for 6667 kilometres (3600 nautical miles) with provisions incorporated for conversion to the -213MX configuration by means of kits that could be installed within a two- to three-day time period. Provisions would be made in the airframe for quickly attached engine pod and landing gear so that for the military configuration, where long range is required, the thrust and the landing gear systems would be increased along with the gross weight to achieve a 10 186 kilometre (5500 nautical mile) range. The civil version would be delivered with all the provisions for the extra landing gear and engines but would be operated in a more efficient configuration for commercial service. The penalties created by increasing the fuel capacity for the longer ranges is very small,

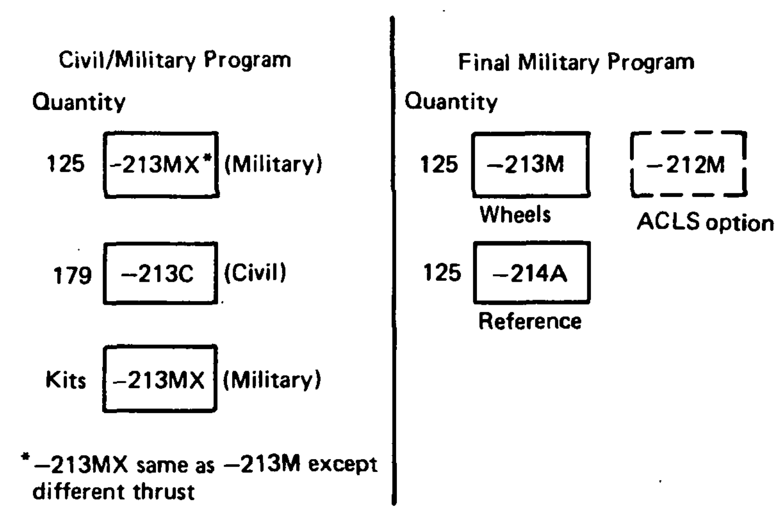


Figure 63 Program Comparison

Table 9 Configuration Comparisons

Civil and Military Programs

Final civil (-211)
- Quantity based on market RTM (84 A/P)
- Net payload: 544 311 kg (1 200 000 lb)
- Wheels
- Non-pressurized

Final military (-213M)
- 125 A/P
- Net payload: 272 155 kg (600 000 lb)
- Wheels
- Pressurized
- IFR
- Drive on/drive off
- Permanent floor (bolt-in)
- Loose equipment
- 463L pallet provision

Alternate final military (-212M)
(Same as -213M except ACLS instead of wheels)

Reference military (-214A)
- 125 A/P
- Net payload: 272 155 kg (600 000 lb)
- Wheels
- Pressurized
- IFR
- Drive on/drive off
- Structurally integrated floor
- Loose equipment
- 463L pallet provision

Civil/military Program

- 125 military + civil quantity based on market RTM (179 A/P)

Dedicated military (-213MX*)
- 125 A/P
- Pressurized
- Wheels
- IFR
- Drive on/drive off
- Permanent floor
- Loose equipment
- 463L pallet provision

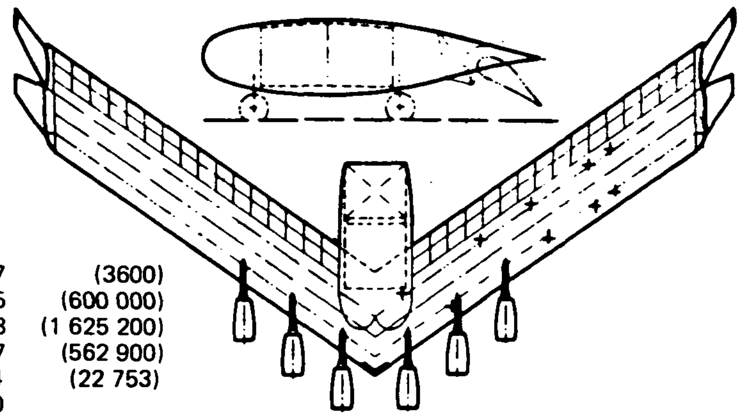
Civil in civil use (-213C)
- 179 A/P
- Pressurized
- Wheels
- IFR
- Provision for drive on/drive off
- Provision for permanent floor

Add:
- Engines
- Landing gear
- Floor

Civil in military use (-213MX*)
- 179 A/P
- Pressurized
- Wheels
- IFR
- Drive on/drive off
- Permanent floor
- Loose equipment
- 463L pallet provision

* Same as -213M except engine thrust is different.

because integral fuel tanks are used with a bonded structure that is sealed in the bonding process. The quick engine and landing gear installations were developed for a previous study, and are applicable to this present contract. The nonrecurring costs for the civil/military airplanes will be distributed through both the military and commercial fleet with the exception that the military features carried by the -213C will be included in the nonrecurring costs for -213MX.



Range, km(nmi)	6667	(3600)
Net payload, kg (lb)	272 155	(600 000)
MTOW, kg(lb)	737 178	(1 625 200)
OEW, kg(lb)	255 327	(562 900)
Wing area, $m^2(ft^2)$	2114	(22 753)
Basic aspect ratio	6.380	
Effective aspect ratio	9.785	
Basic wing span, m(ft)	116	(381)
Total wing span, m(ft)	127	(417)
Sweep, degrees	35	
t/c⊥	0.22	
Cruise Mach number	0.79	
Engines BPR	9.5	
TSLS, N(lb)	289 134	(65 000)

Figure 64 Civil/Military Configuration—Model 759-213C (Civil Version)

The pod for the outsized cargo in the military version was not removed for the civil version because the remaining cargo volume in the wing was insufficient to contain the required design payload of 272 155 kilograms (600 000 pounds). A considerable performance benefit could result by removing the pod and operating the airplane at a lighter gross weight.

5.3.2 TECHNOLOGY APPLICATION AND ANALYSIS

5.3.2.1 Aerodynamic Analysis and Design

The aerodynamic analysis conducted on the study configurations is based on the methods discussed in Appendix A and the technology definition as presented in Section 5.2.1.1. However, for the final civil configuration, the effects of pitching moments due to wing camber and engine thrust, and overwing blowing effects due to the engines, were considered. Specific areas of interest not covered therein are the effects of tip fins and overwing engine location.

Effect of Tip Fins—An important feature of the airplanes analyzed in this study is the use of tip fins. A combination of a horizontal and vertical fin is used to shift the center of lift aft so that aerodynamic balance is achieved with the wing span fully loaded with cargo. This method of stabilization offers several advantages.

The fins increase the theoretically effective aspect ratio with a corresponding significant reduction in induced drag. The induced drag benefits increase directly with fin size; however, for conventional airplanes, these benefits are traded off against wing weight increases which usually result in smaller fins relative to the wing, than in the present case. In this study, wherein the full-span loading inherently results in very low net bending moments and wing weights, the fins are sized primarily for stability and control purposes and induced drag reductions are an extra benefit. The adverse effects of the relatively large fins on bending loads in the wing are alleviated by controlling the vertical fin incidence and adjusting the outboard trailing-edge devices. Analyses have shown that the spanwise lift distribution can be appropriately modified in this way to keep the 1-g bending moments within desired limits while reserving most of the available induced drag benefits.

Using tip fins as stabilizers instead of a conventional stabilizer, mounted on tail booms for example, has advantages in addition to avoiding the need for booms and reducing induced drag. Wing tip stabilizers operate in the strong upwash around the tips and carry a positive load. This results in a higher C_{L_α}, which in turn means a smaller required angle of attack and a reduction in wing profile drag. Proper tailoring of the fin geometry can reduce fin profile drag and wing-fin interference drag. Proper tailoring of the fin geometry can reduce fin profile drag and wing-fin interference drag (see Reference 2 for example) so that the overall effect of using fins rather than a conventional tail is to substantially reduce airplane trim drag.

In addition to contributing to longitudinal stability and reducing induced drag, the all-movable vertical fins provide the required lateral directional control and directional stability. Their multi-purpose role is an indication that they are efficiently integrated airplane components. It is assumed that advanced technology will allow the development of a control system to insure the success of such highly integrated components.

Effect of Overwing Engine Location—A careful examination was made of the results presented in Reference 4 which showed an induced drag benefit associated with engines being located above the wing. The data presented in the reference was appropriately modified so as to apply to the plan-form/engine characteristics of the airplanes analyzed in this study. The drag benefit at cruise was found to be negligibly small; however, a low speed drag reduction of five to six percent was indicated. Since the airplanes in this study were cruise thrust limited, these benefits did not affect engine sizing requirements. However, a low speed performance benefit can be expected. The quoted take-off field lengths for the final configurations include this effect.

The drag increments associated with the overwing engine placement were evaluated for the selected configurations and added directly to the drag results from the optimizer (see Appendix A). Lift and pitching moment increments, although favorable, were negligibly small.

5.3.2.2 Propulsion System

The application of propulsion technology to the final configurations is identical to that considered in parametric study and is described in Section 5.2.1.2.

5.3.2.3 Structural Design

The general structural concept was based on the structural definition contained in Reference 1. The use of GR/EP (versus aluminum honeycomb used in the referenced study) would necessitate some changes in the details but not in the major load paths which are shown in Reference 1. Wing structural material and construction techniques were selected consistent with a 1995 airplane certification date. Because the configuration concept leads to low structural loads, bonded honeycomb construction for skins, ribs, spars, and intercostals is expected to be weight and cost effective. The face sheets for the honeycomb were designed of graphite-epoxy in a 0/±45/90 layup. The caps on the ribs, spars, and intercostals were designed using graphite-epoxy pultrusions. The graphite-epoxy material contains 60 percent fiber and has a density of 1522 kilograms per cubic metre (0.055 pound per cubic inch). Allowables for the honeycomb face sheets are:

- Tension ultimate 441 MPa (64 000 lb/in^2)
- Compression ultimate 441 MPa (64 000 lb/in^2)
- Shear ultimate 193 MPa (28 000 lb/in^2)

The stiffeners are strain limited by the face sheets of the honeycomb surfaces.

Minimum gage requirements for graphite-epoxy honeycomb primary structure were established based on considerations of manufacturing, maintenance, hail, and lightning. Minimum face sheets of 0.533 millimetre (0.21 inch) are a handling requirement for honeycomb panels. In addition, the inner face sheet gage cannot be less than 25 percent of the outer face sheet. In order to allow walking on the upper surface, a minimum gage of 1.067 millimetres (0.042 inch) is required for the outer face sheet. In areas exposed to damage from tires, two layers of fiberglass, 0.508 millimetre (0.02 inch), are required over the outer face sheet.

Considerations of hail damage to the leading edge result in a minimum outer face sheet of 1.067 millimetres (0.042 inch). Lighning protection of bonded composite structure will be provided by a weight allowance of 0.49 kilogram per square metre (0.1 pound per square foot) in the primary strike zone and 0.24 kilogram per square metre (0.05 pound per square foot) in the secondary strike zone. Bird hazard is aggravated on the final configurations because the wing leading edge fuel tanks are in the vicinity of engine exhaust. The aluminum honeycomb panels in the leading edge were designed so that the fuel tank would not be penetrated by the impact of a four-pound bird at the maximum cruise speed. The minimum-gage requirements are shown in Figure 65.

The upper and lower ribs were considered to be one structural unit tied together at the spars by vertical stiffening members and by three tension rods between the bays. The joints between the ribs and the vertical stiffeners at the spars were considered rigid.

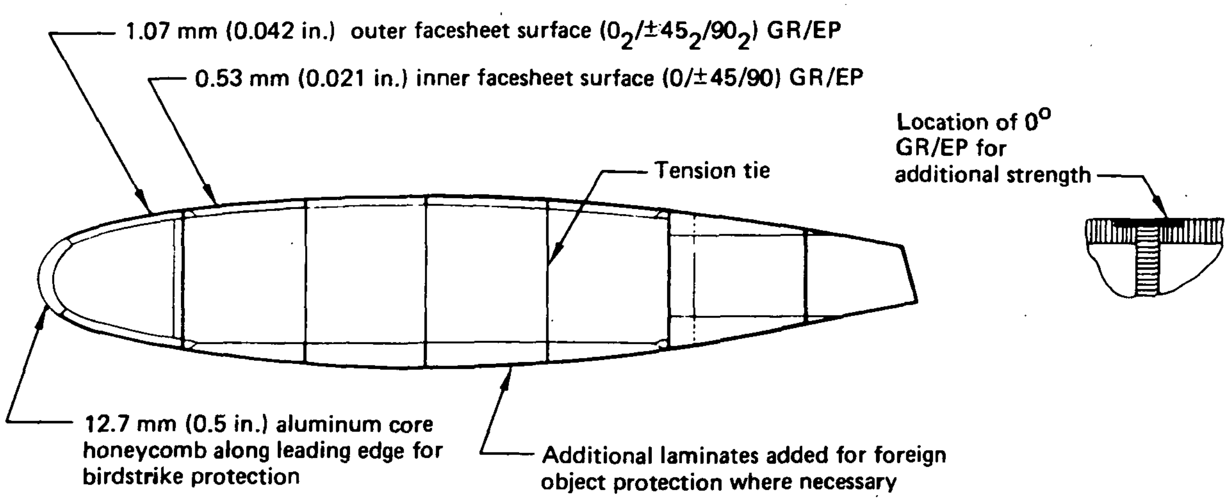


Figure 65 Typical Wingbox Structure

5.3.2.4 Loads and Structural Analysis

Preliminary loads and structural analyses were performed to assess the airplane performance sensitivity to structural weight. Structural weight data for this study, including the final configuration evaluation, were developed parametrically from a previous study of a straight-wing distributed load freighter (Reference 1). Because the inertial and aerodynamic forces are approximately balanced along the span, once-per-flight loads may constitute a large proportion of the ultimate design loads and lead to early fatigue damage. To avoid this situation, ultimate design loads were required to be no smaller than three times the 1-g loads for any flight condition and at least two and one-half times the 1-g loads for any ground taxi condition. In addition to these minimum requirements, the usual flight and ground design conditions were considered. The 2.5-g flight maneuver loads and gust loads were assumed to be held within the above limits by means of a load alleviation system. Since the straight wing airplane had a boom-supported tail, three assumptions were necessary in order to apply this parametric analysis to the swept wing airplane:

- The wing box unit weight is not affected by the use of wing control surfaces rather than a boom-supported tail for stability, trim, and control.
- Wing strength and stiffness requirements for a distributed load freighter are independent of airplane size.
- Adjustment factors can be applied to account for structural material differences and section geometric differences.

These assumptions could result in a wing structural weight estimate for the 759-211 configuration which is too low because it has a higher aspect ratio and a greater structural span than the reference configuration.

The structural design speed altitude envelope appropriate to the swept-wing distributed load airplane is shown in Figure 66 and is based on the standard FAR format. The reference speeds shown (i.e., V_A, V_B, V_C and V_D) are significant primarily in relation to gust load conditions. Since gust loads are substantially reduced by active controls, the impact of speeds higher than those associated with the straight-wing airplane is expected to be minimal.

The effect of payload distribution can also be significant in relation to the other critical design conditions since the combined shear and bending moment are needed to determine structural component sizing. Figure 67 shows the results of analysis conducted to define a representative envelope of payload distributions. Three distributions are shown, one of which is asymmetric spanwise. This imbalance is rectified by a small lateral control input with a minimal effect on overall bending moment. Local anomalies in payload shear such as those shown are expected to be accommodated by the structural capability needed to support the overall bending moment distribution and stiffness requirements.

An evaluation of the configuration characteristics of the swept-wing airplane showed that a more accurate wing structural weight estimate would have to be based on design loads that included the effects of design requirements, control system function and logic, and aeroelasticity. This would require that the control system synthesis be performed in conjunction with the aeroelastic loads analysis and structural sizing. Since an analysis of this magnitude and type is totally different from conventional aircraft, and logically beyond the scope of the study, it was decided to determine the airplane performance sensitivity to variations from the parametric structural weight estimate. The significant configuration characteristics that led to this decision and the structural

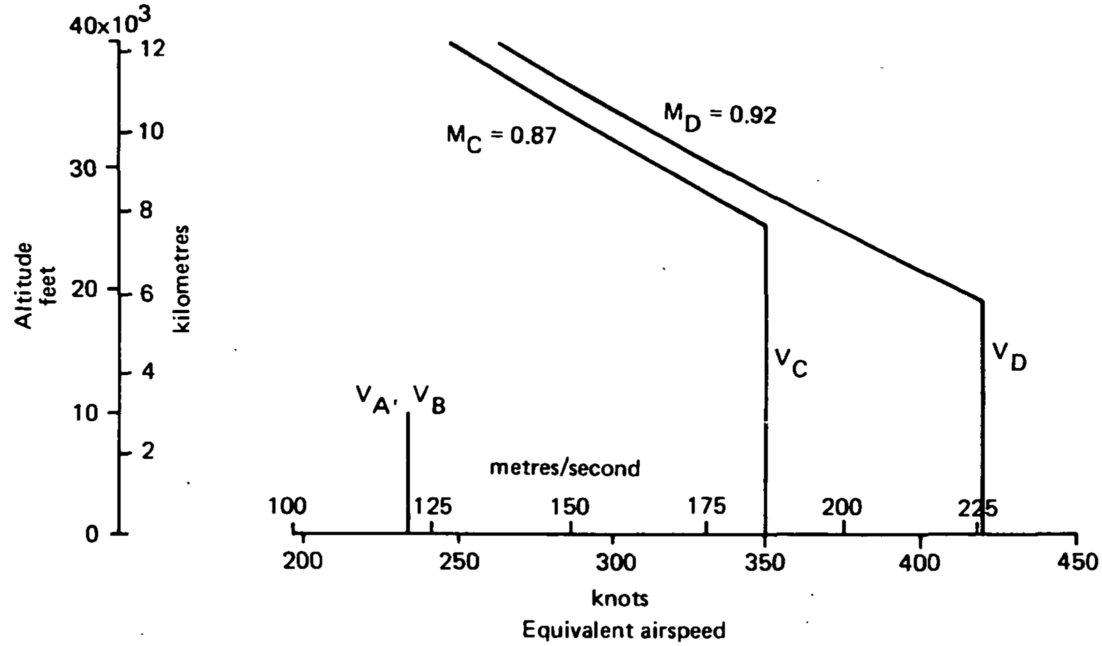


Figure 66 Structural Design Speeds

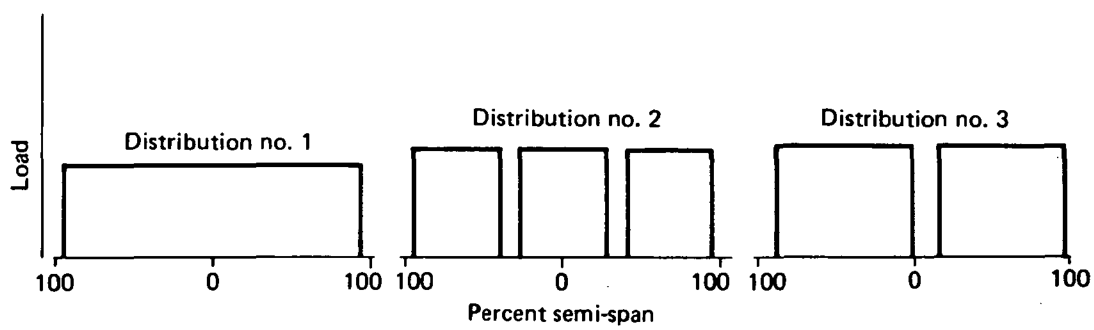


Figure 67 Design Payload Variation

weight sensitivity evaluation are discussed below. However, limited work to evaluate the effects of aeroelasticity is presently being conducted.

Span-loaded airplanes are designed, insofar as possible, to balance the inertial forces with the external forces. During the 1-g flight, this balance is limited by the requirement that the wing control surfaces provide pitch trim while maintaining a low-drag lift distribution. This means that the trim logic must be established before 1-g loads can be determined. For design maneuvers, the wing control surfaces should provide the maneuvering forces in such a way as to minimize the structural loads. This control logic must be established concurrent with the maneuver loads. In this case, there is no requirement for a low-drag lift distribution. In addition to the trim and control functions, the control surfaces must also provide pitch stability and gust load alleviation.

The control logic for gust load alleviation may be somewhat incompatible with the logic required to maintain pitch trim and stability. For example, to reduce loads during an up-gust, upward deflection of the trailing edge control surfaces could be applied. Under these same circumstances, downward deflection of some of the control surfaces would be required to avoid pitch-up and maintain stability.

The effectiveness of the control surfaces in performing these diverse functions is further complicated by aeroelasticity. Relative to a more conventional configuration, the strength requirements of the structure are reduced due to the distributed load concept. This leads to a structure which has very low natural frequencies with significant consequences on the design loads and strength required.

An analysis was conducted to show the weight sensitivity of providing for bending moment and structural stiffness. Figure 68 shows the relationship of increment in ultimate wing bending moment

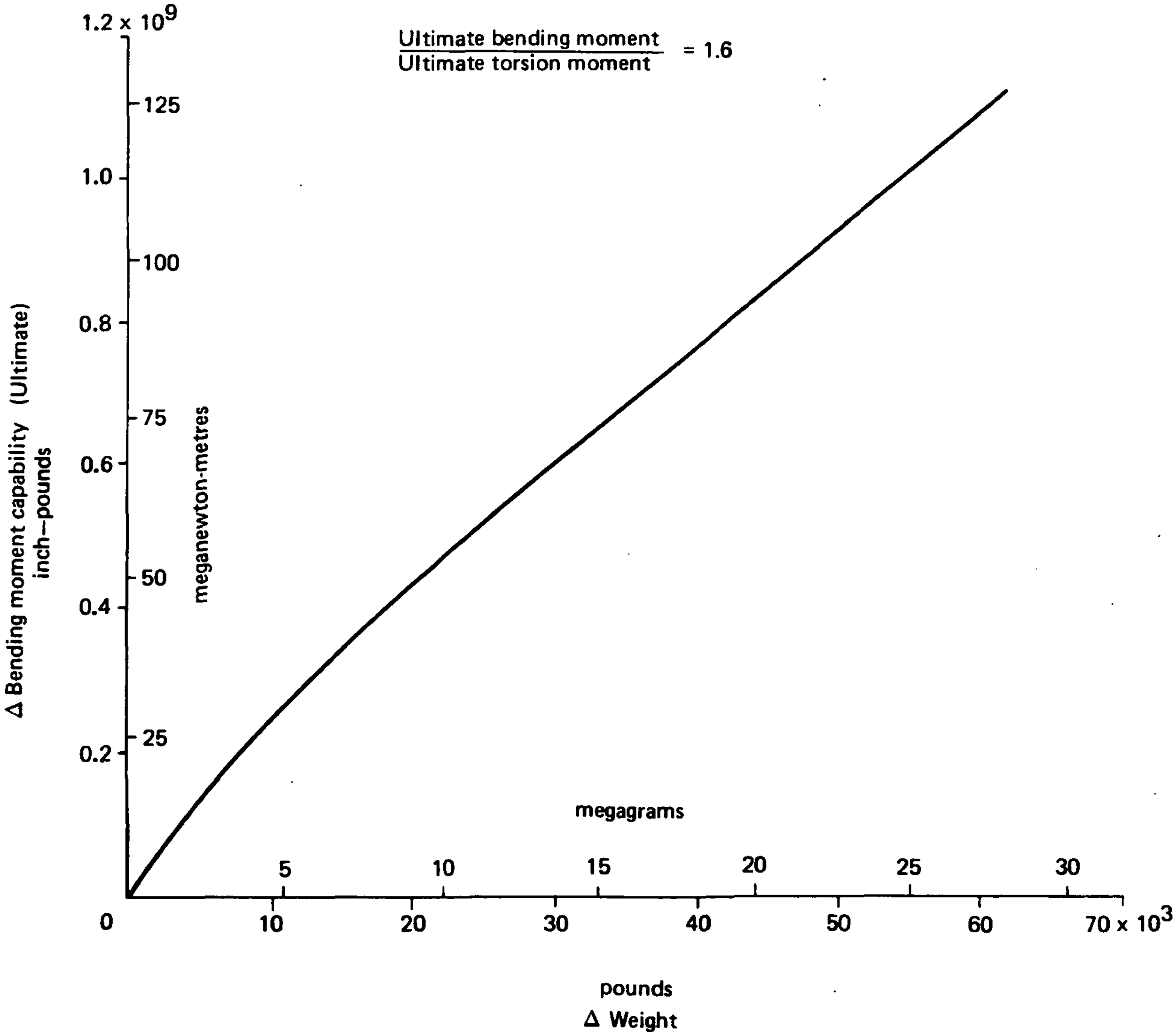


Figure 68 Section Bending Capability

to the incremental weight of stiffeners and spar caps which must be provided to increase the bending capability. The requirement to provide adequate torsion capability was considered by maintaining ultimate torsion moment. Not included in these data are weight increments for nonoptimum components, ribs, and a centerline splice which also may be affected.

As an example of the potential impact of critical design conditions, consider the effect of a requirement for an incremental bending capability of 113 meganewton-metres (10^9 inch-pounds), which is somewhat greater than that identified in the preliminary analyses of potential critical conditions such as ground loads and gust. This would require an additional 24 948 kilograms (55 000 pounds) of bending material. The effect of this material would be to increase the OEW/MTOW ratio by 4.4 percent (from 0.294 to 0.307) and reduce the PL/MTOW ratio by 2.6 percent (from 0.495 to 0.482), while maintaining constant airplane geometry and engine thrust. The results of this analysis indicate that the efficiency of the airplane should not be seriously compromised by a possible requirement for additional strength.

5.3.2.5 Weight and Balance

Weight data for airplanes in the final configuration study phase are presented in Table 10. Functional group weight statements in Tables 11 and 12 conform to MIL-STD 1374 Part I definition. A

Table 10 Weight Data for Final Configurations (SI Units)

Model	OEW*, kg	Cycled (Sized) MTOW, kg	TSLS N/engine	No. of Engines	OEW* per 20-foot Container, kg
759-211	377 480	1 283 031	375 875	8	3630
759-213C	255 327	737 178	289 134	6	4910
759-182A	175 087	469 618	232 642	4	5472
759-212M	290 526	881 421	241 983	8	
759-213M	295 742	887 907	245 542	8	
759-213MX	307 672	900 245	289 134	8	
759-214A	373 960	977 401	317 158	6	

Note: All values are cycled (sized) * 1990 technology

Table 10a Weight Data for Final Configurations (Customary Units)

Model	OEW*, lb	Cycled (Sized) MTOW, lb	TSLS lb/engine	No. of Engines	OEW* per 20-foot Container, lb
759-211	832 200	2 828 600	84 500	8	8002
759-213C	562 900	1 625 200	65 000	6	10 825
759-182A	386 000	1 035 330	52 300	4	12 063
759-212M	640 500	1 943 200	54 400	8	
759-213M	652 000	1 957 500	55 200	8	
759-213MX	678 300	1 984 700	65 000	8	
759-214A	824 440	2 154 800	71 300	6	

Note: All values are cycled (sized) * 1990 technology

Table 11 Group Weight Statements—Civil Airplanes (SI Units)

Functional Group (Weights in kg)	759-211	759-213C	759-182A
Wing	150 807	97 065	49 442
Horizontal tail	4604	1937	2989
Vertical tail	14 154	7687	2073
Body	– – –	27 941	44 157
Main landing gear	56 582	32 510	21 999
Nose landing gear (steering)	– – –	– – –	3801
Nacelle and strut	27 636	16 154	8759
Total structure	253 783	183 295	133 220
Engine	48 531	26 918	13 975
Engine accessories	931	614	367
Engine controls	183	122	73
Starting system	181	136	91
Fuel system	2587	2938	1415
Thrust reverser	0	0	0
Burst protection	10 744	4584	1914
Total propulsion system	63 158	35 313	17 835
Instruments	424	424	422
Surface controls	6902	4663	4454
Hydraulics	3838	2420	1324
Pneumatics	996	1595	871
Electrical	1657	997	1279
Electronics	2522	1947	1406
Flight provisions	468	443	608
Cargo handling	21 317	11 133	6677
Emergency equipment	582	525	240
Air conditioning	2607	1026	957
Anti-icing	1034	780	236
Auxiliary power unit	2138	1229	36
Insulation-cargo compartment	12 845	7149	3125
Total fixed equipment	57 331	34 332	21 636
Exterior paint	596	332	331
Options	1069	614	680
Manufacturers empty weight	375 936	253 887	173 703
Standard and optional items	1544	1440	1383
Operational empty weight	377 480	255 327	175 087
Gross payload	634 394	317 061	194 773
Engines (quantity/designation)	8/TF 1990	6/TF 1990	4/TF 1990
Engine thrust (SLS), N	375 875	289 134	232 642
Cargo containers - quantity (type)	104(SAE)	52(SAE)	32(SAE)
size, m	←	2.44 x 2.44 x 6.10	→
Zero fuel weight	1 011 874	572 388	369 859
Maximum design takeoff weight	1 283 031	737 178	469 618

Table 11a Group Weight Statements—Civil Airplanes (Customary Units)

Functional Group (Weights in lb)	759-211	759-213C	759-182A
Wing	332 473	213 993	109 000
Horizontal tail	10 149	4271	6590
Vertical tail	31 205	16 948	4570
Body	——	61 600	97 350
Main landing gear	124 743	71 672	48 500
Nose landing gear (steering)	——	——	8380
Nacelle and strut	60 926	35 613	19 310
Total structure	559 496	404 097	293 700
Engine	106 993	59 345	30 810
Engine accessories	2053	1354	810
Engine controls	404	268	160
Starting system	400	300	200
Fuel system	5703	6478	3120
Thrust reverser	0	0	0
Burst protection	23 686	10 106	4220
Total propulsion system	139 239	77 851	39 320
Instruments	935	935	930
Surface controls	15 217	10 281	9820
Hydraulics	8462	5335	2920
Pneumatics	2196	3517	1920
Electrical	3653	2198	2820
Electronics	5561	4293	3100
Flight provisions	1031	977	1340
Cargo handling	46 996	24 545	14 720
Emergency equipment	1282	1157	530
Air conditioning	5747	2262	2110
Anti-icing	2280	1720	520
Auxiliary power unit	4714	2709	80
Insulation-cargo compartment	28 319	15 761	6890
Total fixed equipment	126 393	75 690	47 700
Exterior paint	1313	733	730
Options	2356	1354	1500
Manufacturers empty weight	828 797	559 725	382 950
Standard and optional items	3403	3175	3050
Operational empty weight	832 200	562 900	386 000
Gross payload	1 398 600	699 000	429 400
Engines (quantity/designation)	8/TF 1990	6/TF 1990	4/TF 1990
Engine thrust (SLS), lb	84 500	65 000	52 300
Cargo containers - quantity (type)	104(SAE)	52(SAE)	32(SAE)
size, (ft)	←	8 x 8 x 20	→
Zero fuel weight	2 230 800	1 261 900	815 400
Maximum design takeoff weight	2 828 600	1 625 200	1 035 330

Table 12 Group Weight Statements—Military Airplanes (SI Units)

Functional group (Weights in kg)	759-212M	759-213MX	759-213M	759-214A
Wing	125 134	125 134	125 134	112 740
Horizontal tail	1937	1937	1937	9525
Vertical tail	7687	7687	7687	6264
Body	27 941	27 941	27 941	100 775
Main landing gear	31 732	39 700	39 157	38 474
Nose landing gear (steering)	3175	---	---	7548
Nacelle and strut	18 187	21 539	18 441	17 640
Total structure	215 795	223 939	220 299	292 966
Engine	29 246	35 891	29 742	29 942
Engine accessories	750	819	756	644
Engine controls	149	162	150	127
Starting system	181	181	181	136
Fuel system	2847	3077	2945	3157
Thrust reverser	0	0	0	0
Burst protection	4168	6112	4301	5593
Total propulsion system	37 342	46 242	38 074	39 599
Instruments	424	424	424	454
Surface controls	4663	4663	4663	8160
Hydraulics	2807	2854	2822	3334
Pneumatics	995	997	997	1497
Electrical	1614	1616	1614	2132
Electronics	2109	2128	2115	1315
Flight provisions	450	451	451	318
Cargo handling	6123	6123	6123	6314
Emergency equipment	542	543	542	953
Air conditioning	1131	1026	1026	2667
Anti-icing	865	870	858	363
Auxiliary power unit	1469	1500	1480	2858
Insulation-cargo compartment	7149	7149	7149	3983
Total fixed equipment	30 341	30 347	30 264	34 346
Exterior paint	332	332	332	227
Options	734	750	740	---
Manufacturers empty weight	284 545	301 610	289 709	367 138
Standard and optional items	5981	6061	6033	6804
Operational empty weight	290 526	307 672	295 742	373 942
Gross payload	317 061	317 061	317 061	317 061
Engines (quantity/designation)	8/TF 1990	8/TF 1990	8/TF 1990	6/TF 1990
Engine thrust (SLS), N	241 983	289 134	245 542	317 158
Cargo containers-quantity (type)	52(SAE)	52(SAE)	52(SAE)	52(SAE)
size, m	2.44 x 2.44 x 6.10			
Zero fuel weight	607 587	624 733	612 803	691 003
Maximum design takeoff weight	881 421	900 243	887 907	977 401

Table 12a Group Weight Statements—Military Airplanes (Customary Units)

Functional group (Weights in lb)	759-212M	759-213MX	759-213M	759-214A
Wing	275 874	275 874	275 874	248 550
Horizontal tail	4271	4271	4271	21 000
Vertical tail	16 948	16 948	16 948	13 810
Body	61 600	61 600	61 600	222 170
Main landing gear	69 957	87 523	86 327	84 820
Nose landing gear (steering)	7000	— — —	— — —	16,640
Nacelle and strut	40 096	47 485	40 656	38 890
Total structure	475 746	493 701	485 676	645 880
Engine	64 477	79 126	65, 569	66 010
Engine accessories	1654	1805	1666	1420
Engine controls	328	357	330	280
Starting system	400	400	400	300
Fuel system	6277	6784	6492	6960
Thrust reverser	0	0	0	0
Burst protection	9189	13 474	9482	12 330
Total propulsion system	82 325	101 946	83 939	87 300
Instruments	935	935	935	1000
Surface controls	10 281	10 281	10 281	17 990
Hydraulics	6188	6292	6221	7350
Pneumatics	2193	2198	2198	3300
Electrical	3559	3562	3559	4700
Electronics	4650	4692	4662	2900
Flight provisions	993	995	994	700
Cargo handling	13 500	13 500	13 500	13 920
Emergency equipment	1194	1198	1195	2100
Air conditioning	2493	2262	2262	5880
Anti-icing	1906	1919	1891	800
Auxiliary power unit	3238	3308	3262	6300
Insulation-cargo compartment	15 761	15 761	15 761	8780
Total fixed equipment	66 891	66 903	66 721	75 720
Exterior paint	733	733	733	500
Options	1619	1654	1631	— — —
Manufacturers empty weight	627 314	664 937	638 700	809 400
Standard and optional items	13 186	13 363	13 300	15 000
Operational empty weight	640 500	678 300	652 000	824 400
Gross payload	699 000	699 000	699 000	699 000
Engines (quantity/designation)	8/TF 1990	8/TF 1990	8/TF 1990	6/TF 1990
Engine thrust (SLS), lb	54 400	65 000	55 200	71 300
Cargo containers-quantity (type)	52(SAE)	52(SAE)	52(SAE)	52(SAE)
size, ft	8x8x20	8x8x20	8x8x20	8x8x20
Zero fuel weight	1 339 500	1 377 300	1 351 000	1 523 400
Maximum design takeoff weight	1 943 200	1 984 700	1 957 500	2 154 800

typical center of gravity management and loadability diagram for the 759-211 airplane is illustrated in Figure 69. It shows the typical degree of management versatility on swept-wing DLFs. All cg management between Zero Fuel Weight (ZFW) and Maximum Design Takeoff Weight (MTOW) is performed by programming fuel usage. The wide longitudinal cg range variation is available due to unused space in the leading and trailing edges. Fuel tanks can be located to achieve the cg envelope shown in Figure 69.

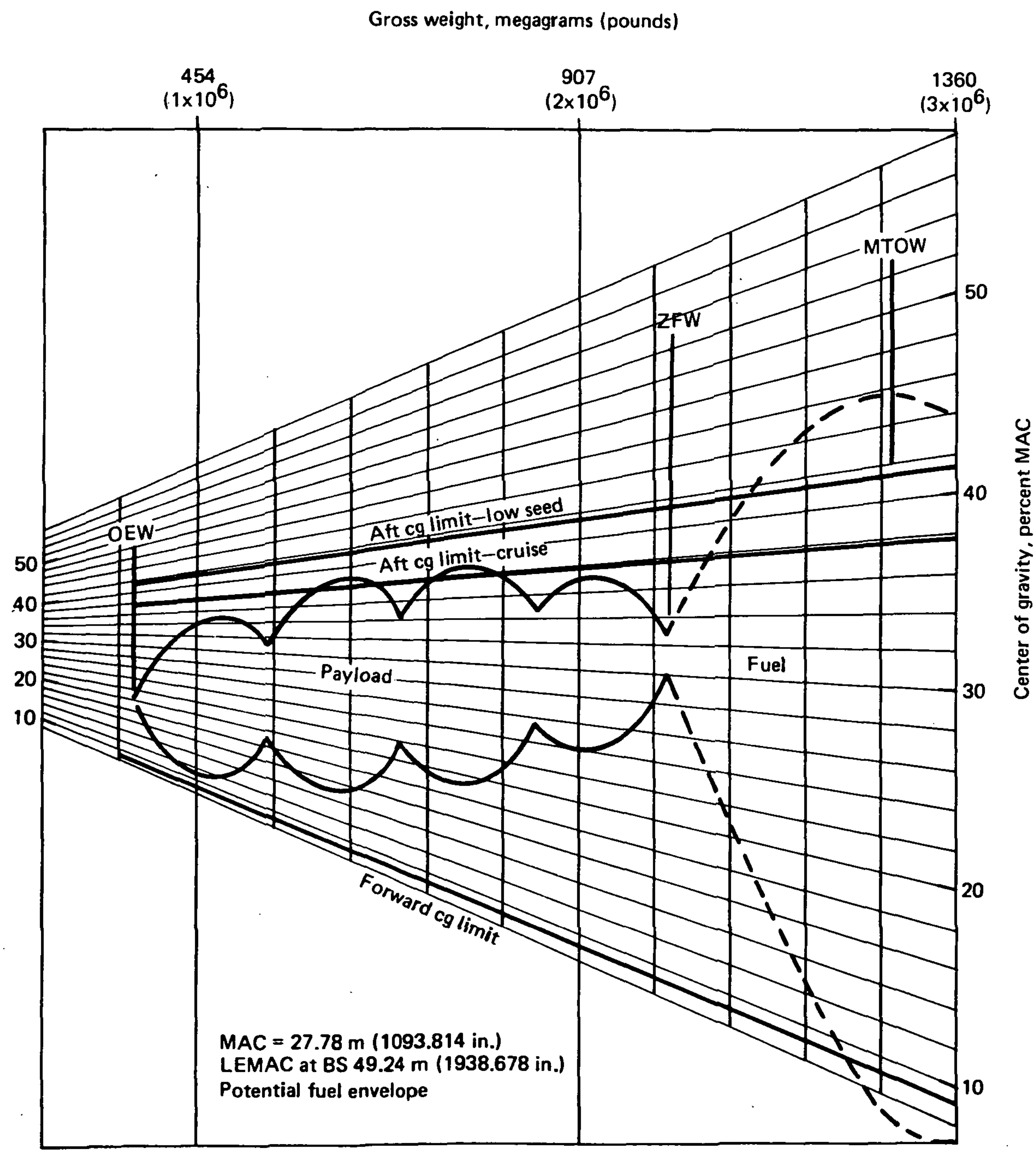


Figure 69 Center of Gravity Management–Model 759-211

Unit weights based upon the 759-183 airplane analysis (Reference 1) were applied to the swept-wing airplanes in this study. In order to maintain consistency and achieve minimum turn-around time, weight equations for civil (commercial) airplanes were programmed on an electronic computer. Five equations were developed for weight estimation of the wing, identifying leading edge, box, fixed trailing edge, movable trailing edge, and tip structure. Adjustments were included for thickness ratio, number of cargo bays (rib length), maximum dynamic pressure, and graphite composite material of construction. Allowable stresses and strength capability were developed for this study and are also representative of those presented in Reference 1. However, stiffness requirements have not been defined. No adjustment was made for sweep angle. The vertical tail unit weight is representative of a detailed loads and structural analysis. The body weight was derived by a detailed component analysis (military airplanes). Standard and operational item weights were expressed for crew and crew services, engine oil, and unusable fuel. All other functional group weights were derived from an equation for the entire functional group. Detailed components for military airplanes were identified and analyzed.

A conventional analysis was used for center of gravity management.

The primary objective of the final configuration study phase was to determine the performance and cost potential of an airplane which would result from an adequate research effort to optimize the airplane design definition. Some risk has been assumed because the required analysis and definition is beyond the scope of this study contract. More detailed aeroelastic loads and structural analyses would be required to establish a baseline airplane which definitely meets minimum design requirements and criteria.

Ground rules, airplane definitions, design concepts, structural materials, design requirements, and design criteria assumed as the basis for weight data are listed in Table 13. Advanced technology definitions are expanded in Table 14 which includes corresponding weight increments from current technology. Refinements were made to equivalent definitions used for the parametric study phase.

5.3.2.6 Stability and Control

Specific stability and control studies were not conducted for the final selected configurations. The longitudinal balance, wing-tip tail sizing, and cg limits were determined from data already available from the parametric and other DLF studies. Criteria and ground rules discussed in the parametric study were unchanged for the final configuration studies.

5.4 FINAL CONFIGURATION PERFORMANCE CHARACTERISTICS

5.4.1 AERODYNAMIC PERFORMANCE

The flight profile and associated time, fuel and distance for the civil aircraft are based on 1967 ATA International mission rules, with the exception of the exclusion of 370.4 kilometre (200 nautical-mile) alternate fuel reserves, and military transport mission rules (Reference Mil-C-5011A) for the military aircraft. The Boeing Thumbprint program, as described in Appendix A, was used to compute the performance values given in Tables 16 and 17 (see Section 5.7.1). The two missions, civil and military were calculated for a design range of 6667 and 10 186 kilometres (3600 and 5500 nautical miles), respectively. Aerodynamic technology corresponds to the 1990 time period as described in Section 5.1.1, Technology Definition.

Table 13 Weight Definition Assumptions for Final Study Configurations

Item	Definition
Configuration/interior arrangement	
Airplane geometry	Per general arrangement (see Figure 54)
Fuel capacity	Mission fuel for maximum payload + trade fuel -12 038 km (6500 nmi) range @ constant MTOW
Design concepts/materials	
Structural and systems concepts for performance baseline	1990 technology (1995 certification)
a. Wing, horizontal tail, vertical tail	Graphite composite skins* and graphite composite honeycomb core. Graphite composite (pultrusion) chords, tubes and fittings where feasible. Multi-spar slap horizontal and vertical tails.
b. Body	None on commercial airplane; graphite on military airplane
c. Landing gear	Conventional steel, two wheel truck geometrically similar to 747 nose gear, modified for DLF loads and brakes added. No powered wheels.
d. Brakes	Conventional disc, carbon
e. Nacelle and strut	Graphite composite honeycomb sandwich
f. Hydraulic actuator	Conventional, steel
g. Thrust reversers	None (plug installed in nacelle)
h. Wing leading edge	Aluminum honeycomb sandwich for bird strike protection
i. Engine burst protection	Kevlar membrane on all engines (for interchangeability considerations)
j. Fuel system	Scavenger pumps, integral tanks (no sealant required)
k. Hydraulic system	Conventional, 27.6 MPa (4000 psi)
l. Anti-icing	Conventional, engine inlet only
m. APU system	Rubberized L-1011 system with PT6 engine, 746 kW (1000 HP)
n. Flight controls system	Same as 747, except full time flight critical stability augmentation system

* Graphite fibers and S glass in epoxy matrix.

C-2

Table 13 Weight Definition Assumptions for Final Study Configurations (Continued)

Item	Definition
o. Signal wires	Conventional (not fiber optics)
p. Cargo compartment floor—	
Commercial airplane	None (container handling system only)
Military airplane	Removable (internal roller trays)
q. Cargo lane width	
Commercial airplane	2.62 m (103 in.) (tension ties used in wing)
Military airplane	2.92 m (115 in.)
Criteria/requirements	
Commercial airplane compliance	FAR
Military airplane compliance	FAR and MIL specifications
Design criteria	Assumed same criteria as 757-183** configuration
Static loads	Not available. Assumed equivalent to 759-183.
Dynamic/aeroelastic loads	Not available. Assumed not critical
Minimum gage requirements	Same as 759-183
Bending/torsional stiffness (EI/JG) requirements	Not defined (assumed not critical)
Kinetic energy in fan stage for engine burst	4.4 MNm (39×10^6 in-lb) for TSLS = 266 893 N (60 000 lb)
Maximum dynamic pressure (q)	Proportional to V_{mo} of 759-183
Cargo compartment pressure	
Differential - commercial airplane	Zero
- military airplane	31 kPa (4.5 psi) -5486 m (18 000 ft) equivalent cabin altitude
Load alleviation capability	Not defined
Takeoff and landing field length	Commercial: $<$3658 m (12 000 ft); Military: $<$3048 m (10 000 ft)
Data source	
Weight data baseline	759-183 (1985 technology)
Balance criteria	
Cargo centroid variation	±5% Container length and ± 10% container width
OEW variation	± 1% MAC
Center of gravity management	Fuel transfer during cruise to achieve cg for maximum aerodynamic efficiency
Crew comfort level and safety	
Number of flight crew	Two
Services	Food warming and beverage provisions included

** Selected straight-wing configuration from Reference 1.

Table 14 Advanced (1990) Technology Definition and Weight Improvement for Final Study Airplanes

Component	Final (selected) Commercial and Military Airplanes		Reference Commercial and Military Airplanes	
	Definition	1	Definition	1
Wing upper and lower surfaces, ribs, spars	Same as parametric study airplanes	-15	Same as parametric study airplanes, except bolted joints where shear loads exceed the capability of bond	-20
Wing leading edge	Same as parametric study airplanes	0	Graphite composite honeycomb sandwich	-10
Wing trailing edge	Same as parametric study airplanes	-20	Same as parametric study airplanes	-15
Control surfaces	Same as parametric study airplanes	-25	Same as parametric study airplanes	-25
Vertical tail box	Same as parametric study airplanes	-25	Same as parametric study airplanes	-25
Horizontal tail box (DLF wing tip fin)	Same as parametric study airplanes	-20	Same as parametric study airplanes	-25
Landing gears	Conventional steel with selective use of titanium	- 2	Same as selected airplanes	- 2
Brake assembly	Same as parametric study airplanes	-40	Same as parametric study airplanes	-40
Body	None on commercial airplane; graphite-epoxy honeycomb sandwich on military airplane	NA 2 -15 3	Graphite composite honeycomb sandwich	-15
Nacelle	Graphite-epoxy honeycomb sandwich, improved design	- 2	Same as selected airplanes	- 2
Strut	Graphite-epoxy honeycomb sandwich; titanium fittings	-12	Same as selected airplanes	-12
Lift augmentation	None	----	None	----
Maneuver load control	Yes	4	Yes	4

1 Weight increment from current technology - percent of component

2 Commercial airplane

3 Military airplane

4 Included in baseline loads

5.4.1.1 Civil Configurations

Payload, fuel efficiency, and cruise drag polars are presented for the final civil configurations in Figures 70 and 71.

5.4.1.2 Military Configuration

Payload, fuel efficiency, and cruise drag polars are presented for the final military configurations in Figures 72 and 73.

Figure 74 illustrates radius-range and loiter-time characteristics for the 759-213M. Radius-range data were developed under the assumption that all cargo was off-loaded at the destination and that the aircraft returned empty. The mission itself followed Mil-C-5011A rules. Loiter times were calculated for a maximum L/D condition which resulted in a velocity of M = 0.65 and an altitude of 7010 metres (23 000 feet).

5.4.2 TAKEOFF, LANDING, AND LOW SPEED PREDICTIONS

In order to obtain a more comprehensive data base, a wind tunnel test was conducted to determine the low speed aerodynamic characteristics of the swept, flying wing distributed load freighter configuration. The test was conducted concurrently on a company-funded basis. The purpose of the test was to evaluate in-ground and out-of-ground effects of a single slotted, full-span flap system of 12 spanwise flap segments with horizontal, and near vertical movable fins attached to the wing tips. The flaps were evaluated for high lift capability, effects of spanwise trailing edge camber variation through differential spanwise flap segment deflections, and control in pitch and roll. The tipfins were evaluated for pitch, roll, and yaw control, and for variation of span loading to reduce drag due to lift.

The results of the test indicate:

- Takeoff can be achieved without rotation when full span flaps are deployed to 40 degrees.
- The vertical tipfins can trim the airplane in-ground and out-of-ground with flaps down 40 degrees, the cg at 40 percent MAC and all engines at maximum thrust.
- Approach speed ($V_{APP} = 1.3V_S$) is comparable to that of long range commercial aircraft when the flaps are down 40 degrees and the cg is at 35 percent MAC.
- Pitch trim for takeoff and landing flares can be accomplished by deflection of the vertical tipfins.
- Wing tip fins are effective for reducing drag due to lift in cruise. Wind tunnel data produced excellent correlation with the theoretical calculations of the method described in Section 2.7.7 of Appendix A.

The wind tunnel model lower wing surface was modified to simulate an air cushion landing system. One air cushion was located under the wing leading edge and a second one was located at the trail-

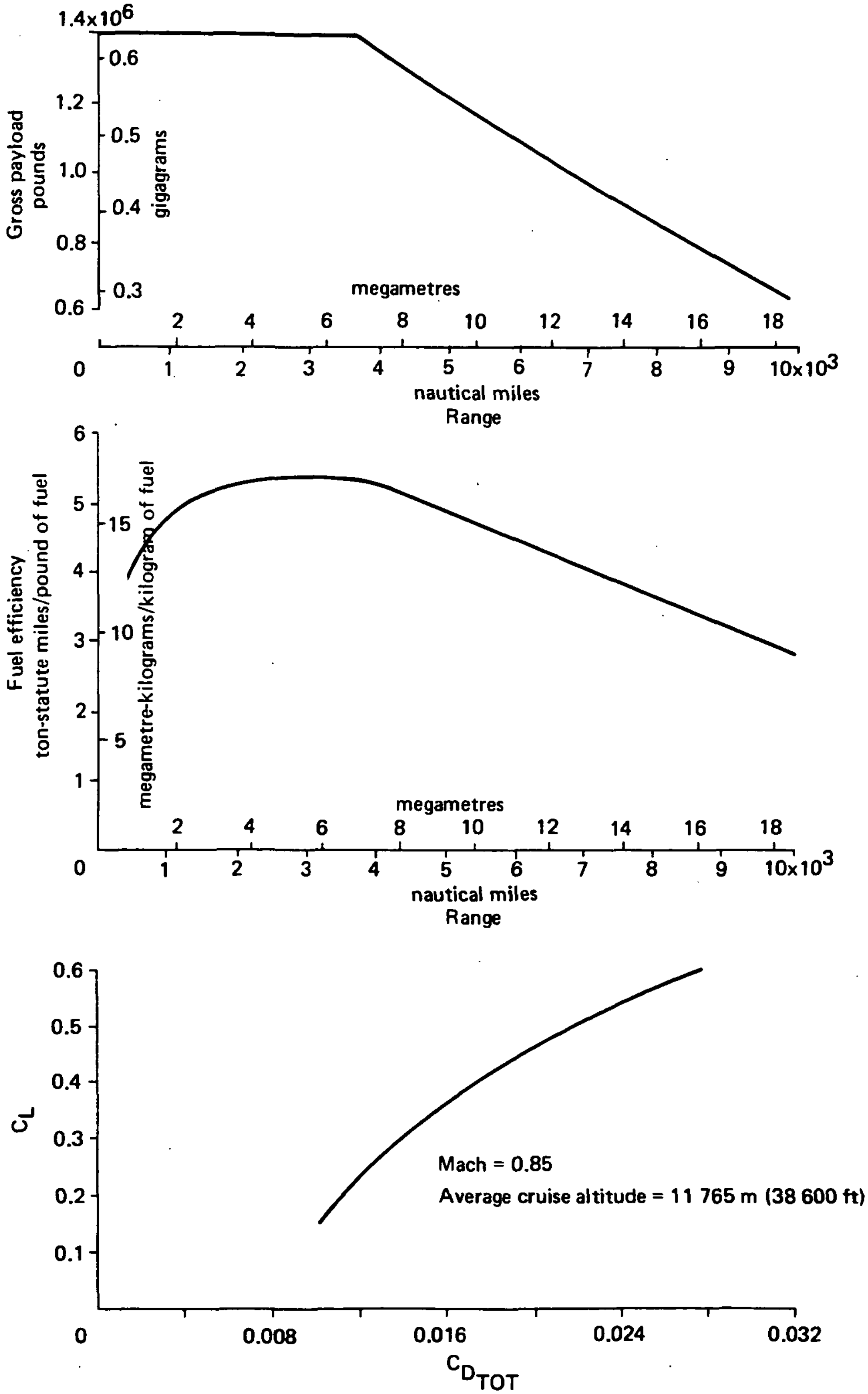


Figure 70 759-211 Performance Characteristics

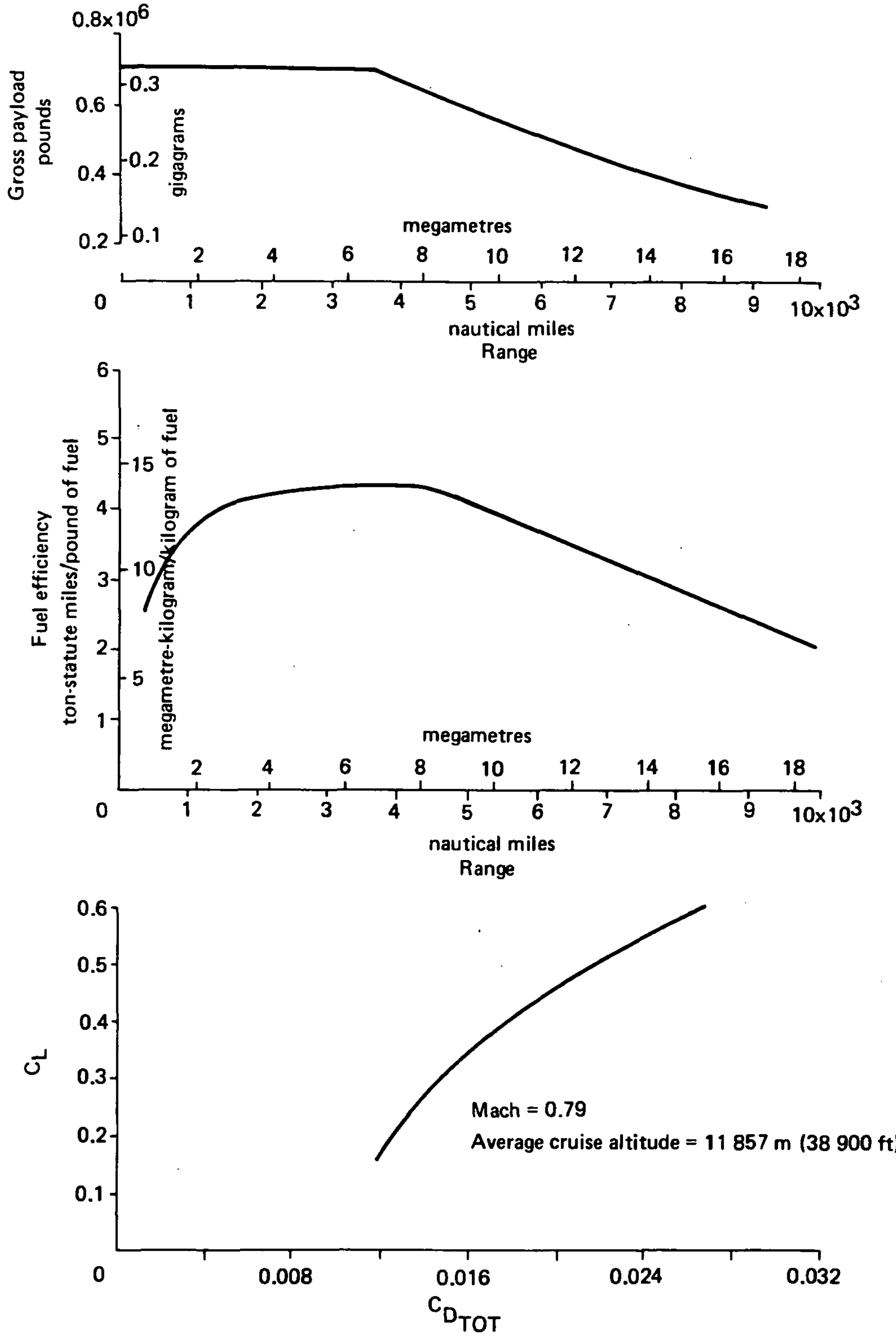


Figure 71 759-213C Performance Characteristics

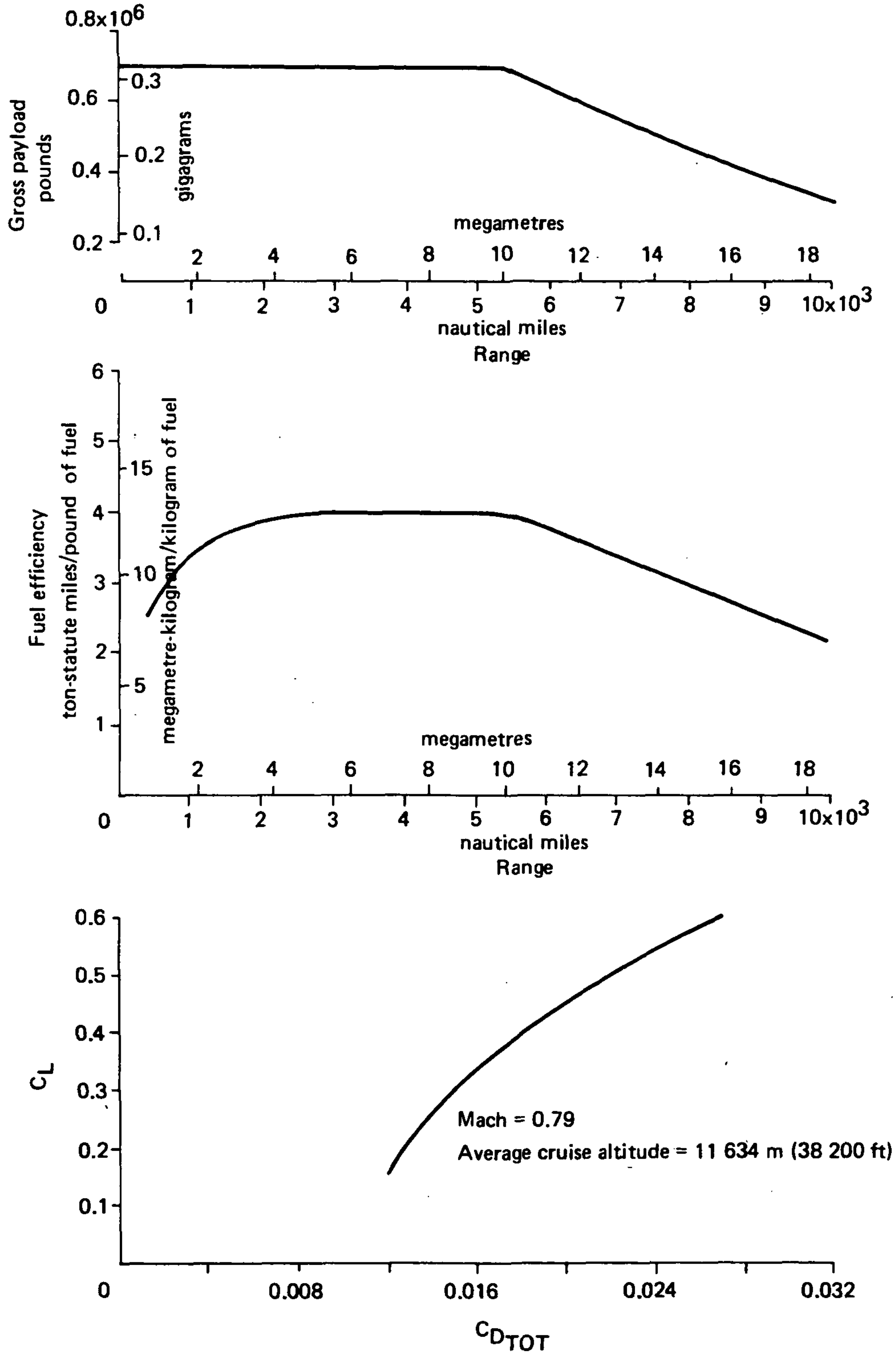


Figure 72 759-213M Performance Characteristics

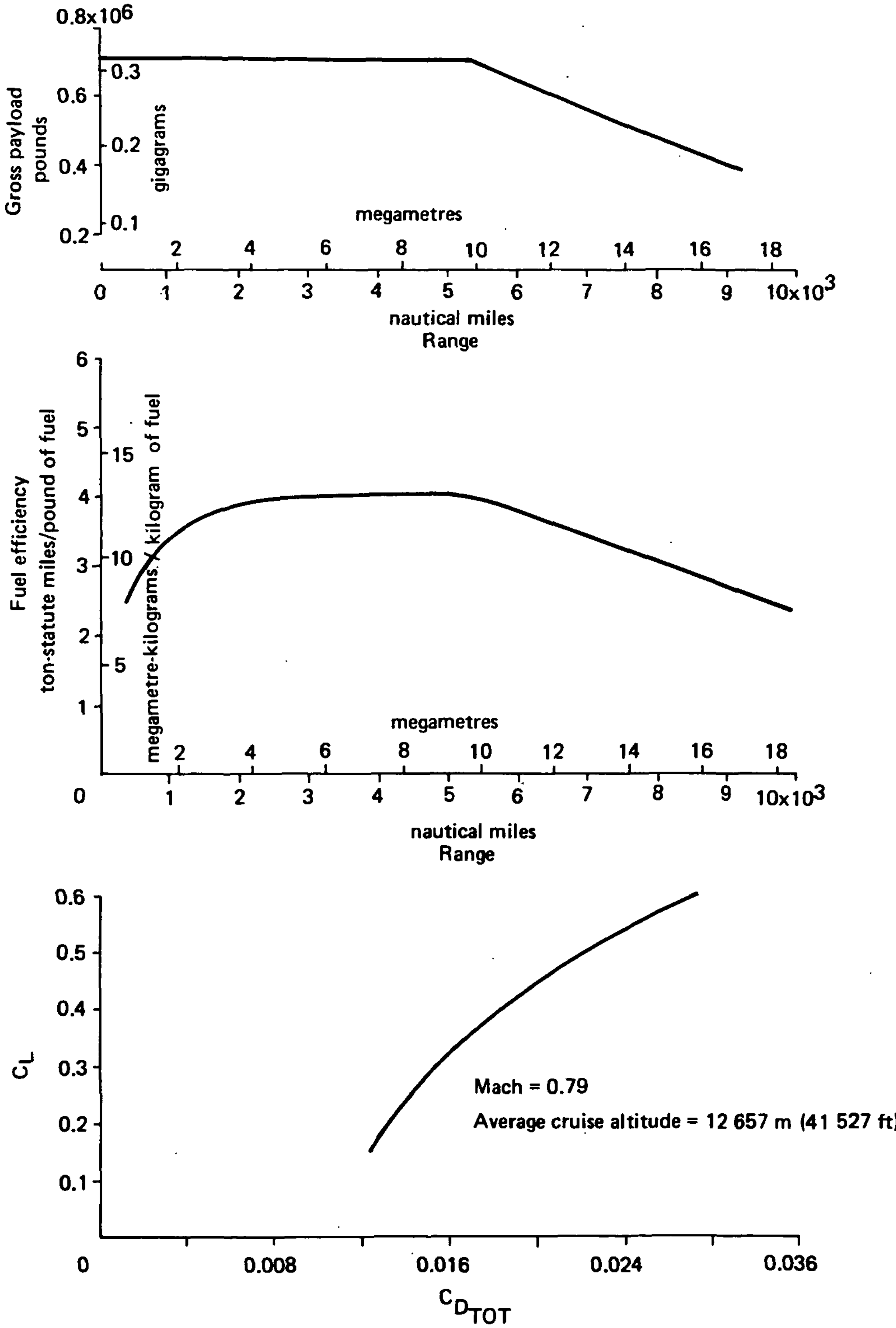


Figure 73 759-213MX Performance Characteristics

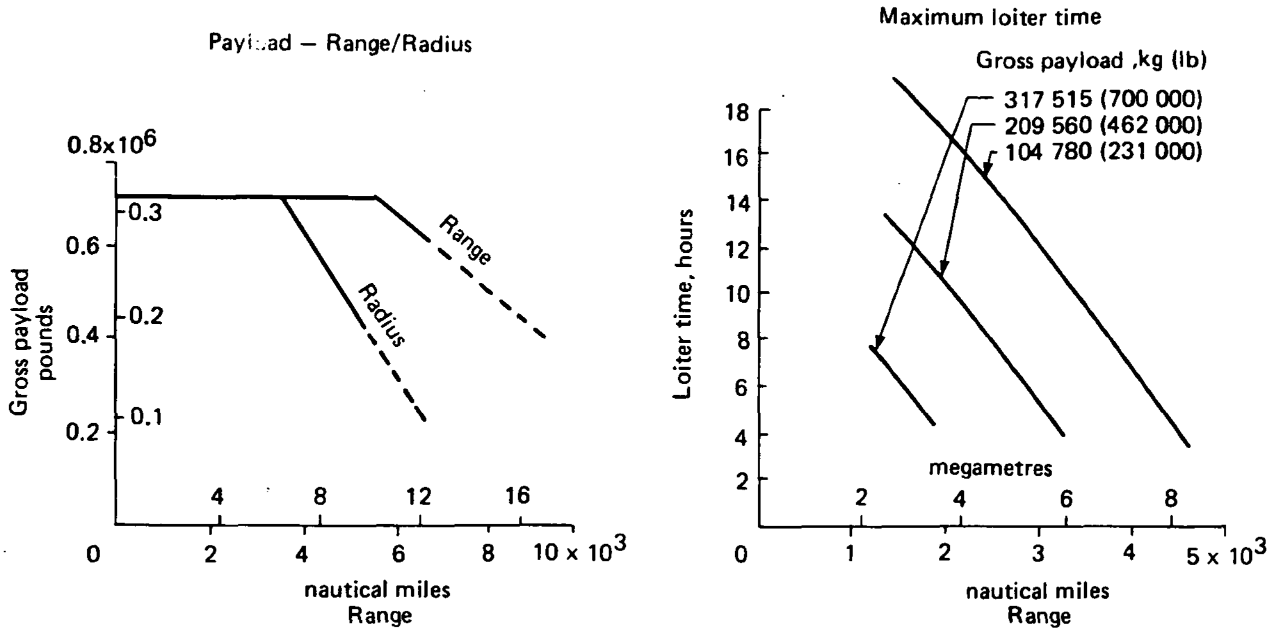


Figure 74 Military Configuration Characteristics–Model 759-213M

ing edge of the flap. Each extended to 75 percent of wing span with a chordwise fence placed at the ends to contain the high pressure air underneath the wing. The flap was lowered to 52 degrees to just clear the runway surface. Lift drag and nose-down pitch was reduced on the ground. Drag was increased out-of-ground while lift and nose-down pitch were further reduced.

5.4.3 NOISE ANALYSIS

Nominal community noise values have been predicted for the DLF civil configuration, Model 759-211. These values represent engine and airframe noise that would be measured at the present standard community noise stations:

- Takeoff 6.482 kilometres (3.5 nautical miles) from brake release
- Sideline 0.463 kilometre (0.25 nautical mile)
- Approach 1.852 kilometres (1 nautical mile) from threshold [113-metre (370-foot) altitude for 3-degree glide slope]

The engine noise predictions represent eight engines having 1990 technology, and 9.5 BPR geared fan, 3/4 length duct (fully lined) and a maximum takeoff thrust of 375 875 newtons, 84 500 pounds each.

The predicted values comply with the present FAR-36 rule and are 1.0 dB higher than the proposed FAR-36 NPRM 75-37-C takeoff limit after trades, see Table 15. For a high probability of certification, predicted values several decibels below the limit are required. However, the engineering and operational options for noise reduction on this airplane have not been exhausted. The dominant noise source on takeoff is fan noise which suggests using a full, rather than a three-quarter length, duct. Alternatively, a steeper takeoff flight profile would probably result in lower values at the 6.482-kilometre (3.5-nautical mile) measuring point.

Table 15 Nominal Community Noise—Model 759-211

	Noise level (EPNdB)		
	Predicted Total/Airframe	FAR-36 (Current)	FAR-36 NPRM 75-37-C
Takeoff[1] (No cutback)	109.0/79.5	108	106
Sideline[2]	101.5/72.1	108	103
Approach[3]	104.5/95.0	108	105

(1) 6.482 km (3.5 nmi) from brake release
243.84 m (800 ft) altitude
2.8° climb angle

(2) 0.463 km (0.25 nmi)
262.128 m (860 ft) altitude

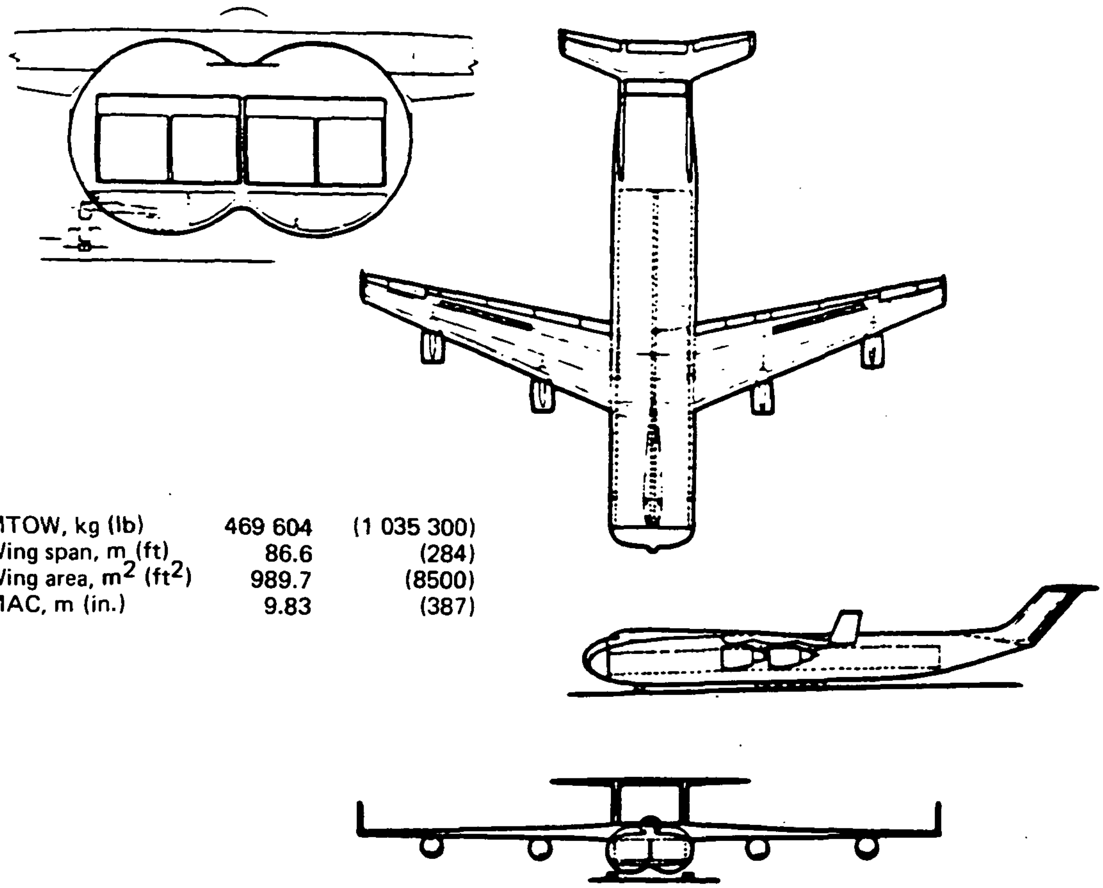
(3) 1.852 km (1 nmi) from threshold
112.776 m (370 ft) altitude
30% thrust

5.5 REFERENCE CONFIGURATIONS

5.5.1 REFERENCE CIVIL CONFIGURATION

The reference civil configuration (Model 759-182A) was chosen from the dedicated air freighter (DAF) studies performed in the Boeing Preliminary Design group. This is the same reference configuration used for the previous DLF study (Reference 1). Figure 75 provides a three-view drawing of the fuselage-loaded airplane, which is an outgrowth of those studies. Developed as an intercontinental air freighter with a wide (double-lobe) fuselage, it offers several advantages to the operator. All cargo is carried on one deck level, with loading accomplished through a nose door with a sill height of 2.15 metres (84 inches) above ground using a kneeling landing gear. The cargo compartment was sized for 2.44 x 3.05-metre (8 x 10-foot) containers and military cargo, but for this study the cargo volume is equivalent to thirty-two 2.44 x 2.44 x 6.10-metre (8 x 8 x 20-foot) containers. The double-lobe shaped fuselage is adaptable to pressurization if this becomes a requirement.

Flight control system requirements differ from those of the DLF selected configuration, principally because of the more conventional geometric configuration of the reference airplane. Low speed control and takeoff rotation requirements establish the minimum horizontal tail size of the reference configuration. The minimum tail size, as established by control requirements, satisfies the unaugmented longtiudinal stability criterion of time-to-double-amplitude of six seconds, permitting use of a handling qualities SAS to meet handling qualities criteria. There is therefore little advantage in decreasing horizontal tail size to meet the relaxed stability criterion of time-to-double-amplitude of two seconds with the consequent necessity of hard SAS implementation. Analyses of the lateral directional stability characteristics of the reference configuration demonstrate satisfactory Dutch roll frequency and damping, and spiral stability so that no requirement exists for a lateral directional stability augmentation system.



MTOW, kg (lb)	469 604	(1 035 300)
Wing span, m (ft)	86.6	(284)
Wing area, m^2 (ft^2)	989.7	(8500)
MAC, m (in.)	9.83	(387)

Figure 75 Reference Civil Configuration—Model 759-182A

Table 11 contains the weight statement for the 759-182A. The advanced technology items utilized and the associated weight impact are listed in Table 14. The reference configuration loadability diagram is shown in Figure 76. Tolerances and allowances used were similar to those employed in the selected configuration.

The same criteria and rationale used to develop the 1990 technology levels for the DLF final configuration were applied to obtain the reference configuration levels. The performance results for the two configurations are compared in Table 16. The 1990 reference configuration payload/range curve and drag polar are presented in Figure 77. The procedures used for drag polar computation are outlined in Appendix A.

5.5.2 REFERENCE MILITARY CONFIGURATION

The reference configuration for the military (Model 759-214A), as the civil reference configuration, is based upon the dedicated air freighter (DAF) studies performed in the Boeing Preliminary Design group. Figure 78 presents a three-view drawing of this wide-fuselage (triple-lobe) freighter. The cargo compartment was sized to carry the 272 155 kilogram (600 000 pound) net payload in 2.44 x 2.44-metre (8 x 8-foot) containers in six bays. The center bays are sized to accommodate the 4.1-metre (13.5-foot) high by 5.2-metre (17-foot) wide outsized cargo. Loading is accomplished through a nose door and by use of a loading ramp aft.

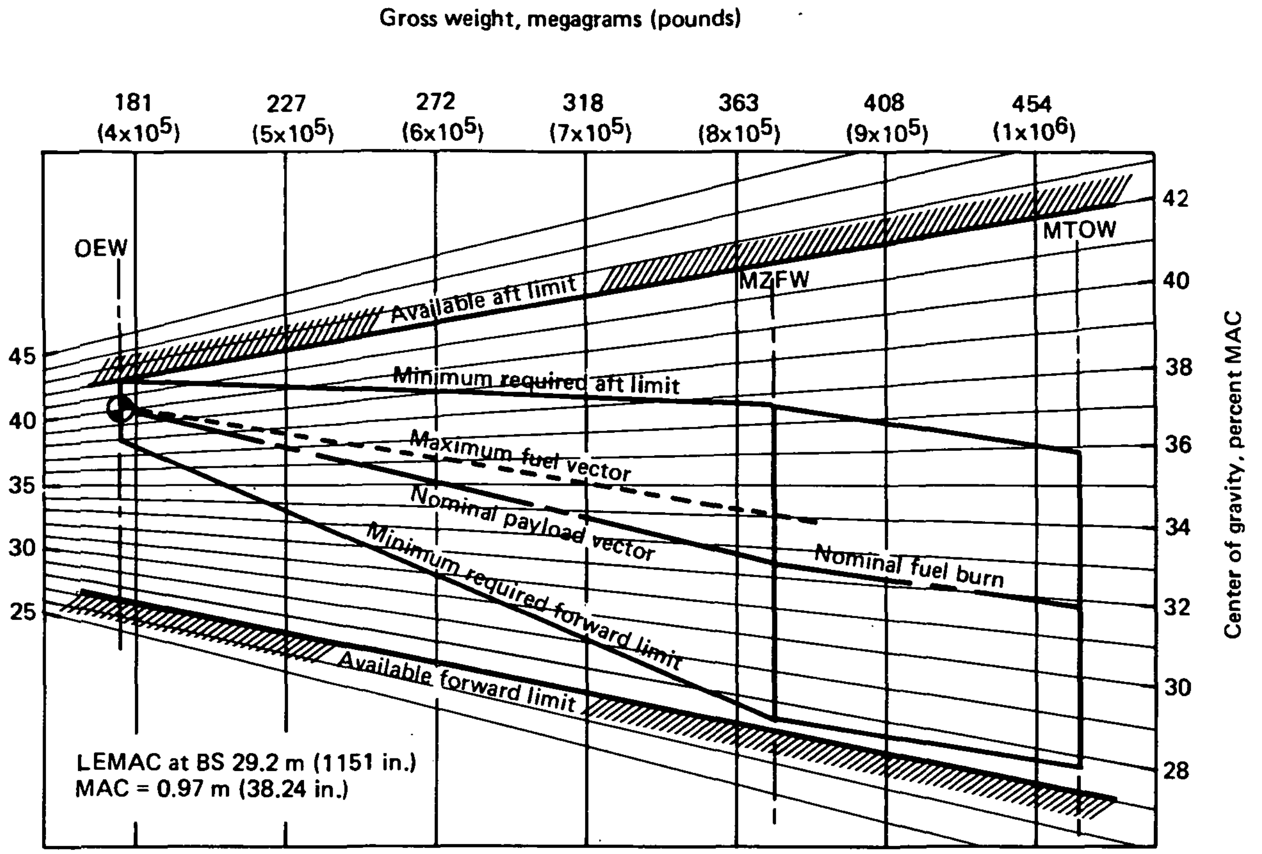


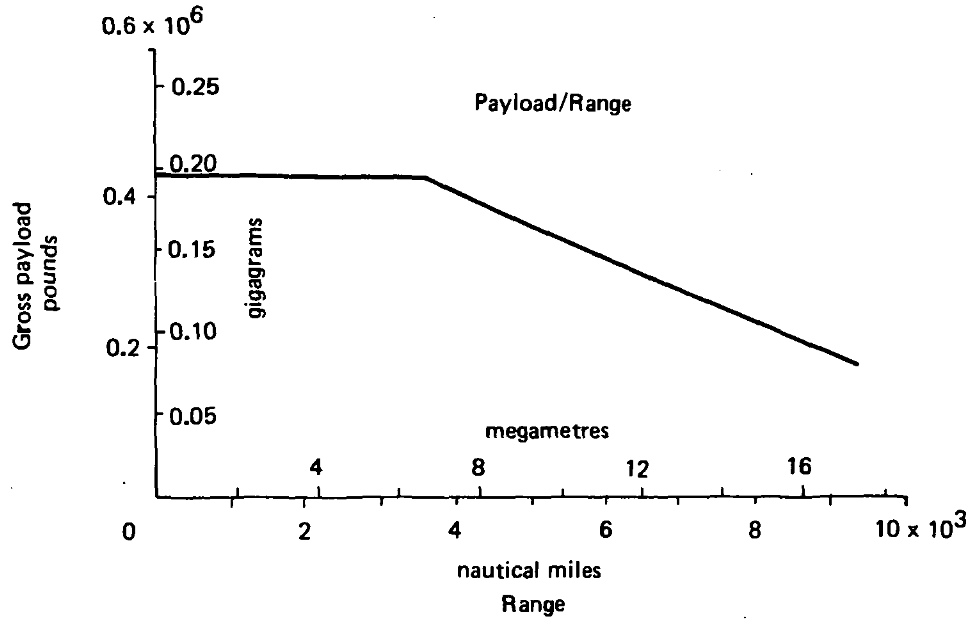
Figure 76 Reference Civil Configuration Loading Diagram—Model 759-182A

Table 12 contains the weight statement for the 759-214A. The advanced technology items utilized and associated weight impact are listed in Table 14. The performance results for the reference and final military configurations are compared in Table 17 (see Section 5.7.1). The payload/range curve and drag polar for the 759-214A are contained in Figure 79.

5.6 ECONOMIC STUDIES

5.6.1 CIVIL CONFIGURATION ECONOMICS

The economics for the parametric study and the final configuration study were conducted by utilizing the Costing and Pricing Methodology contained in Appendix B. In summary, manufacturing cost estimates of all configurations were derived from in-house methods; a return on the manufacturer's investment is added to the cost estimates to arrive at airplane prices; and direct operating costs (DOC) were computed according to the 1967 ATA formulae updated by Boeing to reflect January 1976 experience. In addition, airplane investment costs (AIC) were computed. The AIC is viewed as the cost to the operator for attracting capital to purchase the airplane plus the cost of taxes incurred during its operation. The AIC computed in this study allows the operator a return on his investment equal to the CAB guideline of 12 percent per year and is amortized over the life of the airplane in the same manner as airplane depreciation.



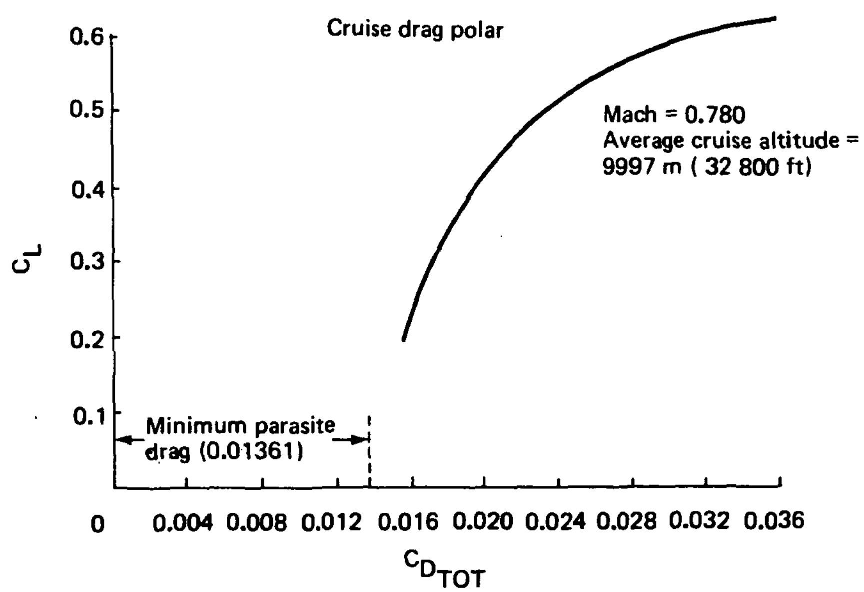
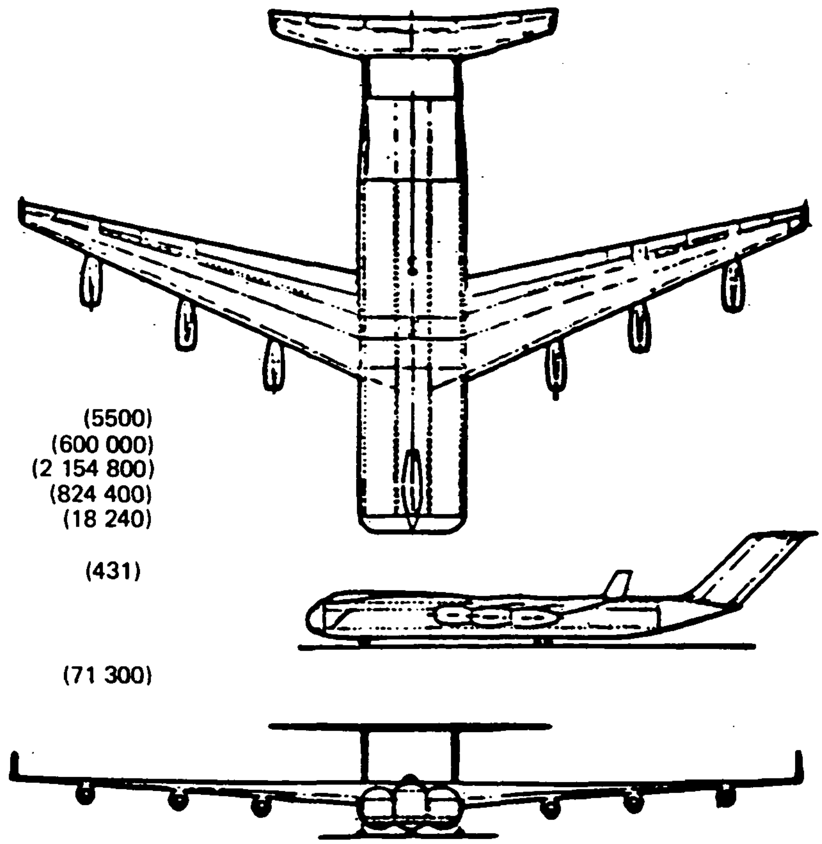


Figure 77 Reference Civil Configuration Characteristics—Model 759-182A



Range, km(nmi)	10 186	(5500)
Net payload, kg (lb)	272 155	(600 000)
MTOW, kg(lb)	977 401	(2 154 800)
OEW, kg(lb)	373 942	(824 400)
Wing area, $m^2(ft^2)$	1695	(18 240)
Wing aspect ratio	10.5	
Wing span, m(ft)	131	(431)
Sweep, degrees	20.0	
Cruise Mach number	0.74	
Engines BPR	9.5	
TSLS, N(lb)	317 158	(71 300)

Figure 78 Reference Military Configuration—Model 759-214A

In arriving at the airplane cost it is first necessary to define the production quantity. Airplane quantities are predicated on a fleet annual throughput of 167.9 revenue petametre-kilograms (115 billion RTM), a design range of 6667 kilometres (3600 nautical miles) and a load factor of 85 percent.

Airplane utilization was based on 5683 hours per year available for block time and turnaround time (assumed to be 0.5 hour). The quantities and the cost-based prices for the final civil, reference, and convertible configuration follow:

Model	Quantity	Airplane Price
759-211	84	190 million dollars
759-182A	291	70 million dollars
759-213C	179	75 million dollars

The price of the convertible configuration (759-213C) is based on the Costing and Pricing Methodology assumption that the government would be charged for common developmental costs. If the civil operator were required to reimburse the government for a pro rata share of the common developmental costs, the 759-213C price would increase to 90 million dollars.

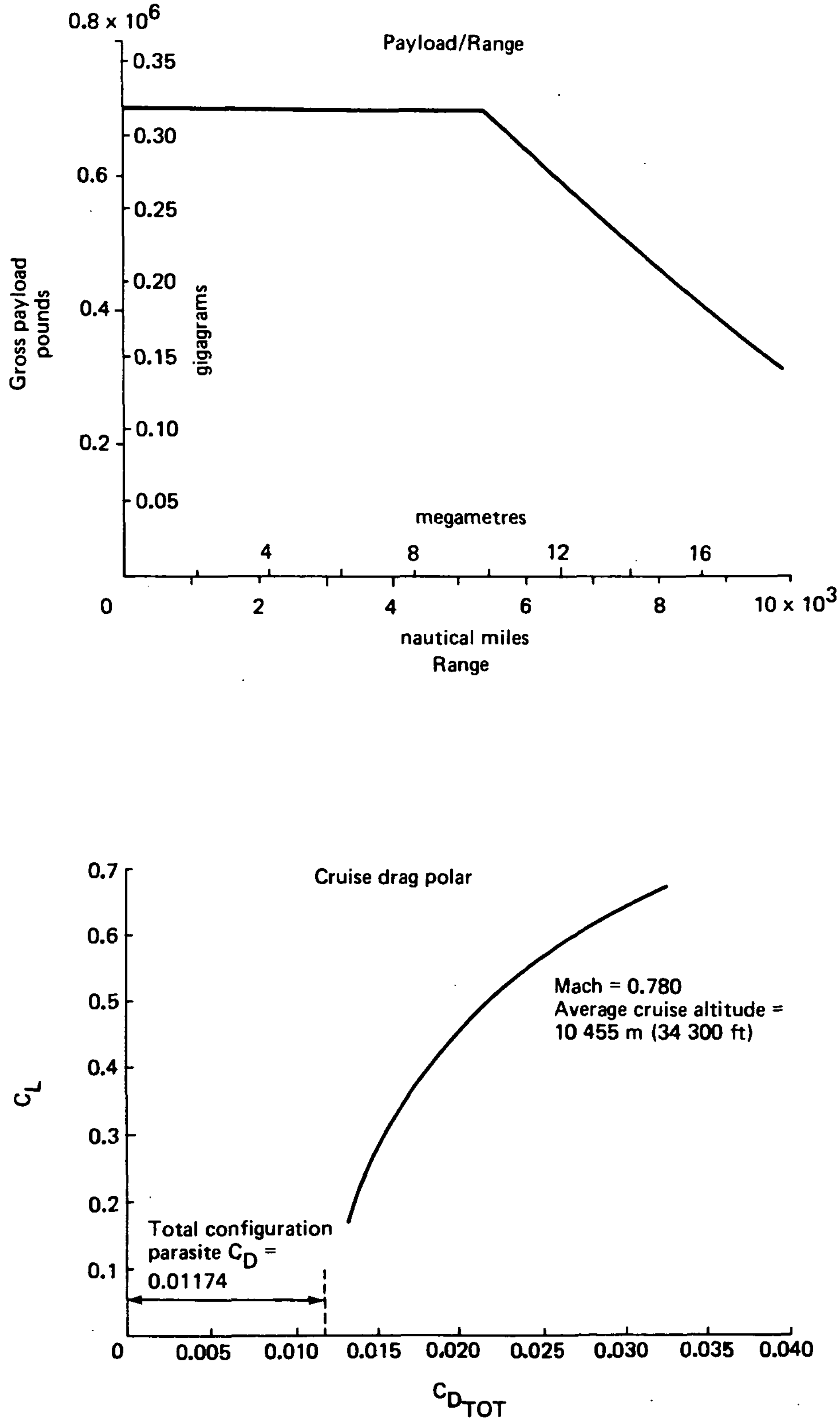


Figure 79 Reference Military Configuration Characteristics—Model 759-214A

The airplane prices presented above are based on manufacturing costs plus a return on investment, and the economics presented herein are predicated on these cost-based prices. However, the actual prices of commercial airplanes are dictated by market forces including such factors as airline ROI requirements, operating cost, break-even load factor, competitive airplane pricing, and others. The airline evaluating the purchase of a new airplane will examine all the alternate ways of meeting its needs against these factors, a procedure common to other businesses. Similarly, when evaluating a possible new airplane program, the manufacturer will compare the market dictated price against the price required for a reasonable rate of return (RROR), and, as in other businesses, reject the program if it will not provide a reasonable return.

A projection of these market factors for the 1990s is obviously speculative and judgmental, but signs of market price limiting can be seen despite the uncertainty. In our judgment, the market forces would be expected to limit the price of the 759-211 to 150 million dollars or less. The 70 million dollar RROR price of the 759-182A appears consistent with market pricing. In the case of the 759-213C, the economic windfall from federal government participation in development and financing is reflected in a favorable RROR price (75 million dollars). At this price the airplane should be very attractive on the market, and it may be competitive at a price as high as 100 million dollars.

The market-based prices quoted above assume that the operator will require a higher return on investment for the larger airplanes than for the smaller ones.

Using the cost-based prices and the DOC formulae shown in Appendix B, the civil configuration economics were determined and are displayed in comparative form in Section 5.7, Comparisons. A study of the sensitivity of airplane DOC and DOC + AIC to fuel price is summarized in Figures 80 and 81. While the civil final DLF is appreciably more economic than the reference airplane (759-182A), the information in Figure 81 indicates that the DLF is slightly more sensitive to fuel price increase than is the conventional reference airplane.

5.6.2 MILITARY CONFIGURATION LIFE CYCLE COSTS

This section presents the approach used in estimating the life cycle costs for Models 759-213M, -212M, -213M, -213MX, and -214A. All aircraft costs are for a production quantity of 125. In addition, Model -213M costs were determined for production quantities of 100, 200, and 300.

The life cycle costs, calculated in 1976 dollars, are developed on an annual peacetime flying time of 1000 hours and 20 years of operation. These costs are compared in Section 5.7, Comparisons. Validation, development, and production estimates were made utilizing a detailed Boeing cost model (Appendix B). Operations and support costs were estimated using the Air Force CACE model from AFR 173-10. Included in the costing methodology are the costs of developing, producing, and operating each of the four designs. It is assumed that two validation airplanes will be procured for each design, and that the four developmental airplanes will be refurbished as production articles. The purchase of 125 aircraft is assumed for each design of which 112 are UE and 13 are Command Support. It is also assumed that attrition would come out of the Command Support complement.

Single source production is postulated due to the probable (small) size of the program. Peak rate production is based on 18 aircraft per year. Consistent with current experience, support investment costs are assumed to be 10 percent of production cost for initial spares and 5 percent for AGE and other costs.

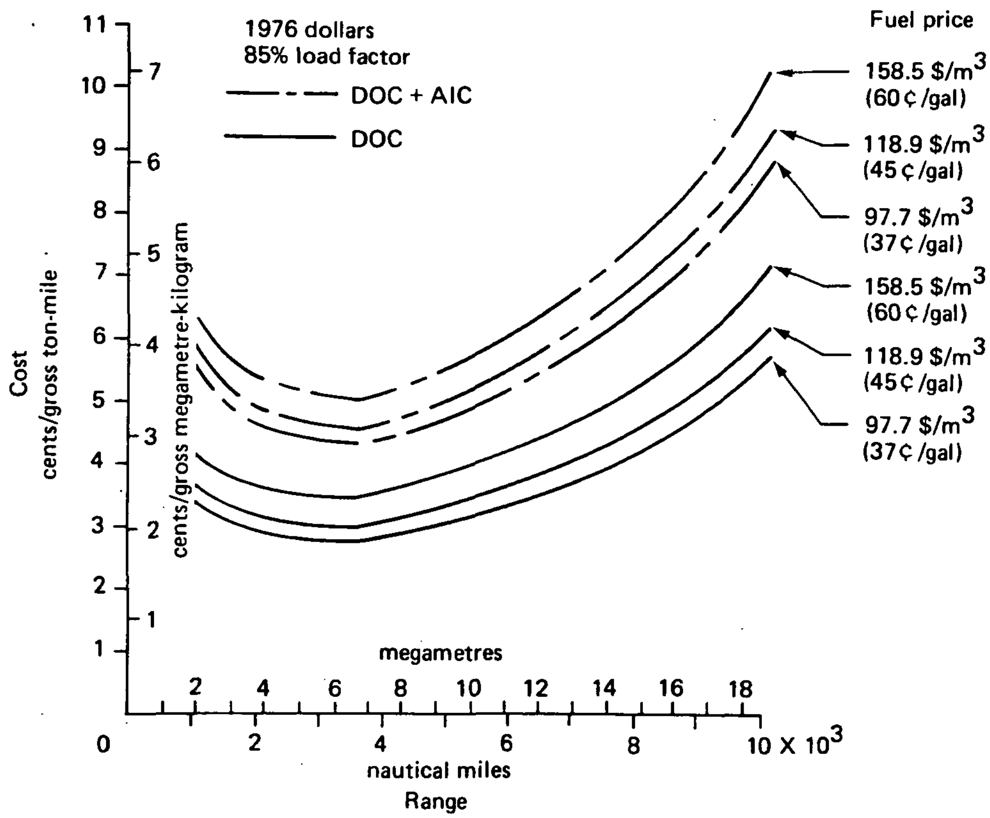


Figure 80 Economic Sensitivity to Fuel Price—Model 759-211

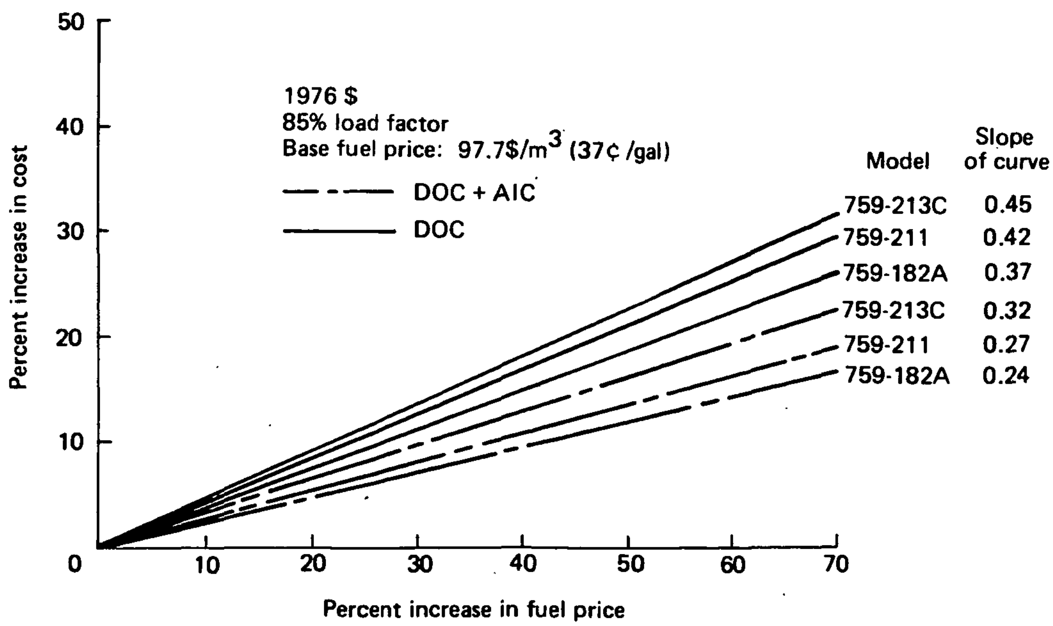


Figure 81 Comparison of Economic Sensitivity to Fuel Price

Operations and support cost for Models -213M, -212M, -213MX, and -214A were based upon 250 four-hour flights per year at full payload for POL consumption. The -213M fuel consumption also was calculated for 250 four-hour flights per year at a 35 percent payload, and 80 12.5-hour per year at full payload.

5.7 COMPARISONS

5.7.1 TECHNICAL COMPARISONS

5.7.1.1 Civil Configurations

The flight profile, mission rules, and procedures used to establish reference configuration performance are identical to those used for the DLF performance. The gross payload, corresponding to a net payload density of 160 kilograms per cubic metre (10 pounds per cubic foot), is 194 591 kilograms (429 000 pounds) for the reference aircraft and 635 029 kilograms (1.4 million pounds) for the DLF. The takeoff gross weight is 469 468 kilograms (1.035 million pounds) for the reference airplane and 1.284 gigagrams (2.83 million pounds) for the DLF. Though considerably smaller than the DLF final configuration, the results are normalized by assuming a constant throughput. Table 16 presents the technical comparison of the selected and reference configurations.

The selected airplane exhibits slightly superior aerodynamic performance relative to the reference configuration. It cruises at higher Mach number (M = 0.85 vs. M = 0.78) at higher aerodynamic efficiency (L/D = 21.66 vs. 21.58) and almost equal airplane cruise efficiency [RF = 34 706 vs. 34 984 kilometres (18 740 vs. 18 890 nautical miles)]. The cruise altitude is higher [11 143 vs. 10 269 metres (36 560 vs. 33 690 feet)]. The airplanes have dissimilar wing spans; the DLF having a 57 percent longer span, and almost five times the wing area.

The structural efficiency of the distributed load selected configuration is considerably better than for the reference conventional airplane (OEW/MTOW = 0.2942 vs. 0.3728). This saving in structural weight fraction is not offset by the increase in fuel weight fraction for the DLF to yield a poorer payload-to-gross-weight fraction as was found in the previous straight wing DLF contract (Reference 1), (DLF PL/MTOW = 0.4944 vs. 0.4147 for the reference configuration). This is a result of the substantially better Mach number and L/D of the swept-wing over the straight-wing DLF.

5.7.1.2 Military Configurations

The flight profile, mission rules, and procedures used to establish the military reference configuration performance are identical to those used for the military DLF performance. The gross payload, corresponding to a net payload density of 160 kilograms per cubic metre (10 pounds per cubic foot), is 317 061 kilograms (699 000 pounds) for both the reference aircraft and the DLF. The takeoff gross weight of the reference is 979 760 kilograms (2.16 million pounds), the DLF has a slightly lower value of 889 041 kilograms (1.96 million pounds). The results can be compared in the same manner as the civil versions. Table 17 presents the technical comparison of the two configurations.

The reference airplane exhibits slightly superior aerodynamic performance relative to the selected configuration. A brief overview reveals that the military reference aircraft cruises at a slightly lower

Table 16 Comparison of Civil Configurations (SI Units)

Design range = 6667 km
Net payload density = 160 kg/m^3
ATA International rules Standard day

	759-211 Final Civil	759-182A Reference Civil	759-213C Civil Convertible of Final Military
MTOW, kg	1 283 031	469 618	737 088
Thrust, N	375 875	232 642	289 134
OEW, kg	377 480	175 087	255 327
OEW/MTOW	0.2942	0.3728	0.3464
Gross payload, kg	634 394	194 773	317 061
Gross payload/MTOW	0.4944	0.4147	0.4301
TOFL, m	2749	3648	2622
Cruise: Mach	0.85	0.78	0.79
ICAC, m	11 143	10 269	11 244
L/D	21.66	21.58	21.61
RF, km	18 740	18 890	18 010
Block fuel, kg	245 561	89 730	150 135
Block time, hr	8.16	8.73	8.72
Block fuel/payload	0.387	0.461	0.474
(Payload/MTOW) Mach	0.4202	0.3235	0.3398
Fuel efficiency, Mmkg/kg of fuel	17.220	14.452	14.066
Landing weight, kg	1 039 384	380 473	590 024
Landing field length, m	1951	1823	1975
V_{APP}, m/s	69	66	70
Cruise SFC, kg/hr/N	0.0575	0.0529	0.0554
AR_{eff}	9.79	10.5	9.79
Wing span, m	136.2	86.6	116.1
S_W, m^2	3784	780	2114
W/S, Pa	3323	5832	3419
Sweep angle, degrees	35	20	35
Streamwise thickness ratio	0.13	0.14	0.18
Price ($millions)	190	70	75
$/kg of OEW	503	399	293
$/kg of gross payload	300	359	236

Mach number and altitude [M = 0.78 vs. 0.79, and 10 445 vs. 11 046 metres (34 270 vs. 36 240 feet)], but displays a considerably higher aerodynamic and cruise efficiency [L/D = 23.17 vs. 21.97, and RF = 34 731 vs. 32 336 kilometres (18 753 vs. 17 460 nautical miles)]. The airplanes have almost equal total wing spans but a considerable difference in wing area.

The structural efficiency of the distributed load military freighter is better than the reference, conventional design, (OEW/MTOW = 0.333 vs. 0.383). This lower structural weight fraction is somewhat offset by the increased fuel weight fraction for the DLF to yield gross weight fractions of PL/MTOW = 0.357 vs. 0.324.

Table 16a Comparison of Civil Configurations (Customary Units)

Design range = 3600 nmi
Net payload density = 10 lb/ft^3
ATA International rules Standard day

	759-211 Final Civil	759-182A Reference Civil	759-213C Civil Convertible of Final Military
MTOW, lb	2 828 600	1 035 330	1 625 000
Thrust, lb	(8) 84 500	(4) 52 300	(6) 65 000
OEW, lb	832 200	386 000	562 900
OEW/MTOW	0.2942	0.3728	0.3464
Gross payload, lb	1 398 600	429 400	699 000
Gross payload/MTOW	0.4944	0.4147	0.4301
TOFL, ft	9020	11 970	8601
Cruise: Mach	0.85	0.78	0.79
ICAC, ft	36 560	33 690	36 890
L/D	21.66	21.58	21.61
RF, nmi	18 740	18 890	18 010
Block fuel, lb	541 370	197 820	330 990
Block time, hr	8.16	8.73	8.72
Block fuel/payload	0.387	0.461	0.474
(Payload/MTOW) Mach	0.4202	0.3235	0.3398
Fuel efficiency, ton-mi/lb of fuel	5.35	4.49	4.37
Landing weight, lb	2 291 450	838 800	1 300 780
Landing field length, ft	6400	5980	6480
V_{APP}, knots	134	128	135.2
Cruise SFC, lb/hr/lb	0.5634	0.5184	0.5437
AR_{eff}	9.79	10.5	9.79
Wing span, ft	446.9	284	381
S_W, ft^2	40 732	8500	22 753
W/S, lb/ft^2	69.4	121.8	71.4
Sweep angle, degrees	35	20	35
Streamwise thickness ratio	0.13	0.14	0.18
Price ($millions)	190	70	75
$/lb of OEW	228	181	133
$/lb of gross payload	136	163	107

Another aircraft which must be considered in a comparison of final military configurations is the air cushion landing system equipped version (759-212M) of the selected military DLF. Due to the significantly lighter landing system (approximately 30%), the -212M has a lower MTOW than the -213M, but due to higher drag its performance is somewhat inferior to the -213M (see Table 18).

5.7.1.3 Convertible Configurations

Civil Version—The convertible civil/military aircraft, when flown in its civil configuration with six engines rather than eight (759-213C), can be compared to the final and reference civil aircraft. This

Table 17 Comparison of Military Configurations (SI Units)

Design range = 10 186 km
Net payload density = 160 kg/m^3
Military rules Standard day

	759-213M Final Military	759-214A Reference Military	759-213MX Final Military W/Civil TSLS Engine
MTOW, kg	887 907	977 401	900 245
Thrust, N	(8) 245 542	(6) 317 158	(8) 289 134
OEW, kg	295 742	373 942	307 672
OEW/MTOW	0.3331	0.3826	0.3418
Gross payload, kg	317 061	317 061	317 061
Gross payload/MTOW	0.3571	0.3244	0.3522
TOFL, m	3048	3048	2655
Cruise: Mach	0.79	0.78	0.79
ICAC, m	11 046	10 445	12 049
L/D	21.97	23.17	22.27
RF, nmi	17 460	18 573	17 900
Block fuel, kg	255 613	264 159	255 146
Block time, hr	12.5	12.52	12.5
Block fuel/payload	0.807	0.832	0.806
(Payload/MTOW) Mach	0.2821	0.253	0.2782
Fuel efficiency, Mmkg/kg of fuel	12.617	12.231	12.617
Landing weight, kg	635 664	714 453	649 041
Landing field length, m	2091	1686	2124
V_{APP}, m/s	72	63	72
Cruise SFC, kg/hr/N	0.0581	0.0577	0.0580
AR_{eff}	9.79	10.5	9.79
Wing span, m	116.1	131.4	116.1
S_W, m^2	2114	1643	2114
W/S, Pa	4118	5832	4175
Sweep angle, degrees	35	20	35
Streamwise thickness ratio	0.18	0.15	0.18
Life cycle cost, $ millions	29 232	36 868	28 943

aircraft, with a gross payload of 317 061 kilograms (699 000 pounds) and net payload density of 160 kilograms per cubic metre (10 pounds per cubic foot), has a takeoff gross weight of 739 376 kilograms (1.63 million pounds). The flight profile, mission rules, and procedures were identical to those used for the two dedicated civil aircraft.

The -213C, being a convertible aircraft, suffers from higher drag and operating empty weight due to the presence of the outsize cargo pod and convertible fittings. The aircraft also has a 2-bay, 22 percent thickness ratio (normal) wing as its basis, thereby somewhat offsetting the higher parasite drag and weight by having a high aspect ratio. The performance of this aircraft compares favorably on

Table 17a Comparison of Military Configurations (Customary Units)

Design range = 5500 nmi
Net payload density = 10 lb/ft^3
Military rules Standard day

	759-213M Final Military	759-214A Reference Military	759-213MX Final Military W/Civil TSLS Engine
MTOW, lb	1 957 500	2 154 800	1 984 700
Thrust, lb	(8) 55 200	(6) 71 300	(8) 65 000
OEW, lb	652 000	824 400	678 300
OEW/MTOW	0.3331	0.3826	0.3418
Gross payload, lb	699 000	699 000	699 000
Gross payload/MTOW	0.3571	0.3244	0.3522
TOFL, ft	10 000	10 000	8710
Cruise: Mach	0.79	0.78	0.79
ICAC, ft	36 240	34 270	39 530
L/D	21.97	23.17	22.27
RF, nmi	17 460	18 573	17 900
Block fuel, lb	563 530	582 370	562 500
Block time, hr	12.5	12.52	12.50
Block fuel/payload	0.807	0.832	0.806
(Payload/MTOW) Mach	0.2821	0.2530	0.2782
Fuel efficiency, ton-mi/lb of fuel	3.92	3.80	3.92
Landing weight, lb	1 401 400	1 575 100	1 430 890
Landing field length, ft	6860	5530	6970
V_{APP}, knots	140.3	121.9	141.8
Cruise SFC, lb/hr/lb	0.5702	0.5656	0.5684
AR_{eff}	9.79	10.5	9.79
Wing span, ft	381	431	381
S_W, ft^2	22 753	17 686	22 753
W/S, lb/ft^2	86.0	121.8	87.2
Sweep angle, degrees	35	20	35
Streamwise thickness ratio	0.18	0.15	0.18
Life cycle cost, $ millions	29 232	36 868	28 943

the basis of cruise Mach number, efficiency, and lift to drag ratio with the reference aircraft. It is not as successful when comparing the cruise efficiencies [RF = 33 355 kilometres (18 010 nautical miles) for the DLF, 34 706 kilometres (18 740 nautical miles) for the reference] but is slightly superior in cruise altitude [11 272 vs. 10 269 metres (36 890 vs. 33 690 feet)] (see Table 16).

The convertible aircraft lies between the two other aircraft when a comparison of structural efficiency is made, and is slightly better than the reference aircraft when comparing the payload to gross weight fraction, (PL/MTOW = 0.4301 vs. 0.4147 for the reference aircraft).

When comparing the convertible aircraft to the final civil configuration, in all cases the larger final civil design is superior.

Table 18 Comparison of Military Configurations (SI Units)

Design range = 10 186 km
Net payload density = 160 kg/m^3
Military rules Standard day

	759-212M Final Military W/ACLS	759-213M Final Military
MTOW, kg	881 421	887 907
Thrust, N	241 983	245 542
OEW, kg	290 526	295 742
OEW/MTOW	0.3296	0.3331
Gross payload, kg	317 061	317 061
Gross payload/MTOW	0.3597	0.3571
TOFL, m	3048	3048
Cruise: Mach	0.79	0.79
ICAC, m	10 961	11 046
L/D	21.88	21.97
RF, km	32 188	32 336
Block fuel, kg	254 574	255 613
Block time, hr	12.5	12.5
Block fuel/payload	0.804	0.807
(Payload/MTOW) Mach	0.2842	0.2821
Fuel efficiency, Mmkg/kg of fuel	12.649	12.617
Landing weight, kg	630 171	635 664
Landing field length, m	2076	2091
V_{APP}, m/s	72	72
Cruise SFC, kg/hr/N	0.0581	0.0581
AR_{eff}	9.79	9.79
Wing span, m	116.1	116.1
S_W, m^2	2114	2114
W/S, Pa	4089	4118
Sweep angle, degrees	35	35
Streamwise thickness ratio	0.18	0.18
Life cycle cost ($millions)	29 704	29 232

Military Version—This airplane was converted from 759-213C which has six 289-kilonewton (65 000-pound) thrust engines. The military version with two additional 289-kilonewton (65 000-pound) thrust engines has a higher gross weight, a higher OEW, but cruises at a higher altitude, resulting in a slightly higher cruise efficiency. This airplane has the lowest life cycle cost of any of the military configurations because the higher production quantity resulting from the civil fleet requirement results in a lower average airplane cost for the military as well as the civil fleet.

The convertible civil/military, when flown in its military configuration with eight engines (759-213MX), can be compared to the reference and final military aircraft (see Table 17). This aircraft,

Table 18a Comparison of Military Configurations (Customary Units)

Design range = 5500 nmi
Net payload density = 10 lb/ft^3
Military rules Standard day

	759-212M Final Military W/ACLS	759-213M Final Military
MTOW, lb	1 943 200	1 957 500
Thrust, lb	(8) 54 400	(8) 55 200
OEW, lb	640 500	652 000
OEW/MTOW	0.3296	0.3331
Gross payload, lb	699 000	699 000
Gross payload/MTOW	0.3597	0.3571
TOFL, ft	10 000	10 000
Cruise: Mach	0.79	0.79
ICAC, ft	35 960	36 240
L/D	21.88	21.97
RF, nmi	17 380	17 460
Block fuel, lb	561 240	563 530
Block time, hr	12.50	12.50
Block fuel/payload	0.804	0.807
(Payload/MTOW) Mach	0.2842	0.2821
Fuel efficiency, ton-mi/lb of fuel	3.93	3.92
Landing weight, lb	1 389 290	1 401 400
Landing field length, ft	6810	6860
V_{APP}, knots	139.7	140.3
Cruise SFC, lb/hr/lb	0.5701	0.5702
AR_{eff}	9.79	9.79
Wing span, ft	381	381
S_W, ft^2	22 753	22 753
W/S, lb/ft^2	85.4	86.0
Sweep angle, degrees	35	35
Streamwise thickness ratio	0.18	0.18
Life cycle cost ($millions)	29 704	29 232

in its military mode, has a gross payload of 317 061 kilograms (699 000 pounds), and a takeoff gross weight of 898 113 kilograms (1.98 million pounds).

The thrust per engine is sized for the commercial aircraft and is greater than the thrust per engine of the -213M. Therefore, the -213MX will have higher total thrust than the -213M. The -213MX convertible aircraft benefits from the increase in thrust available. It is superior in its cruise efficiency and cruise altitude, and the lift to drag ratio increased with the increase in altitude.

The performance benefits of the larger engines are offset by the lower structural and payload efficiencies. The larger engines produce an increase in the operating empty weight thereby increasing the structural weight fraction above the level of the other DLF military aircraft, but remains lower than the value for the reference airplane.

Therefore, it appears that the convertible aircraft is a fairly good compromise in performance when compared to a dedicated military DLF; i.e., 759-213M.

5.7.2 ECONOMIC COMPARISONS

5.7.2.1 Civil Configurations

A comparison of the economics of the final civil, reference civil, and the convertible configurations in commercial use is shown in Figure 82. A comparison of the breakdown of economics for the three civil airplanes is shown in Figure 83.

The economic penalty for operating the convertible configuration in the civil market is estimated to be a 2.5 percent increase in DOC and a 3.3 percent increase in DOC+AIC. Most of this penalty results from the weight of structure required for pressurization. While assumed not required by the civil operator, pressurization is necessary so that convertibility can be effected within three days.

5.7.2.2 Military Configurations

Comparisons of the 20-year life cycle costs for the military DLFs and the reference military configuration are shown in Tables 19 through 21, where validation and development costs are combined into one cost element; i.e., Development. Table 19 shows a comparison of the costs of the various

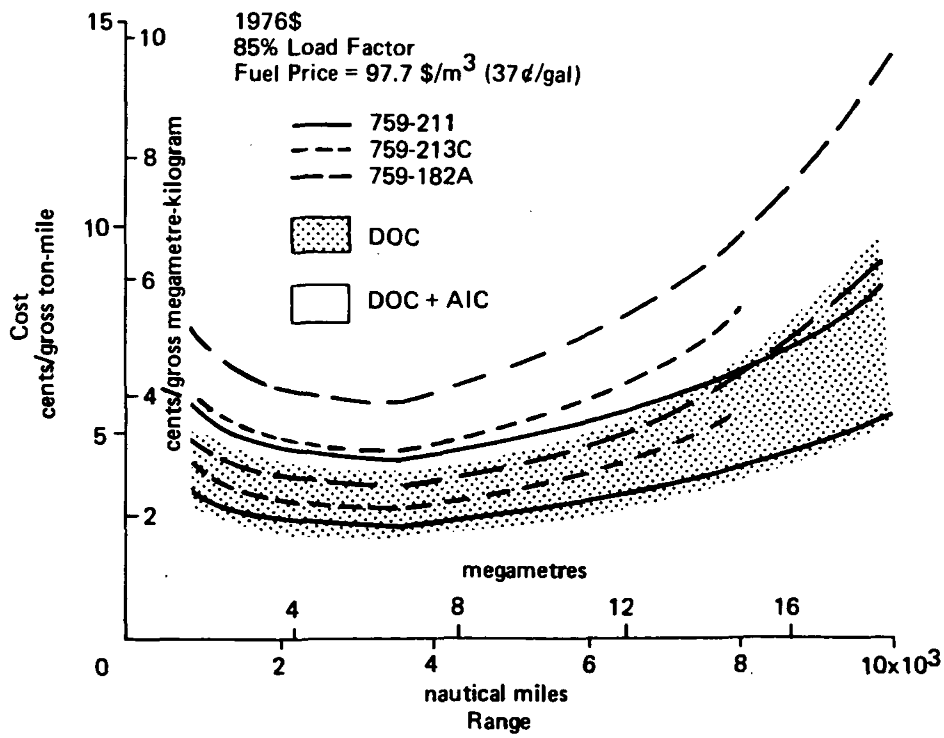


Figure 82 Civil Configuration Economic Comparison

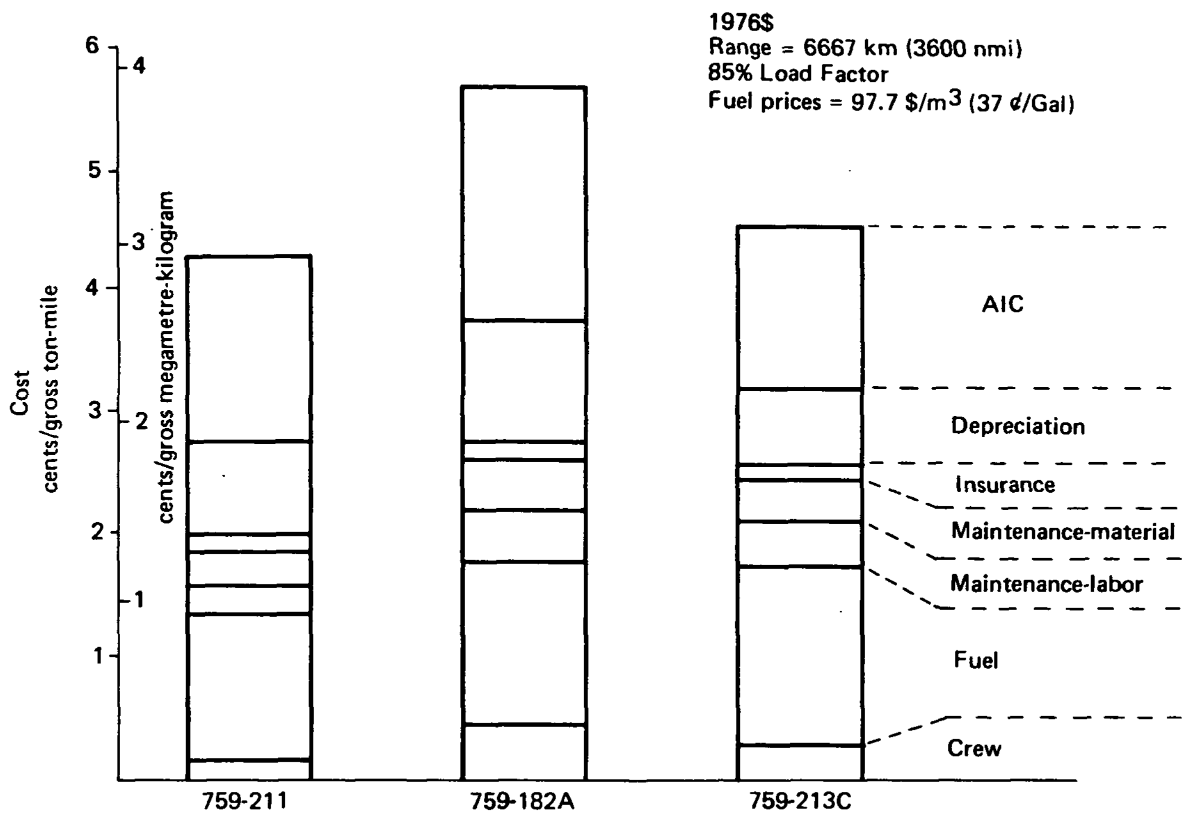


Figure 83 Breakdown of Civil Economics

military configurations. Of particular note is that even though the military is assumed to pay for all common nonrecurring costs of the civil/military program, the 759-213MX experiences the least life cycle cost.

Table 20 presents a comparison of the costs of the final military configuration being operated under different flight-time assumptions, the cost differences merely reflecting differences in fuel used. Table 21 contains a comparison of costs of the 759-213M based on the quantity purchased. For quantities of 100, 200 and 300 airplanes, the Development costs remain constant, and the Operations and Support costs reflect a nearly constant cost per airplane. However, the effect of learning-curve gains is seen in the comparison of Production and Support Investment costs.

5.8 AREAS FOR FURTHER STUDY: POTENTIAL PROBLEM AREAS

This study has established the size and configuration characteristics of swept-wing distributed load airfreighters that provide the most promising economics by making certain simplifying assumptions appropriate to a preliminary design study. The next step in the sequence of events eventually leading to a program definition is to explore those areas that possess high technical leverage, but which have not yet been covered in sufficient technical depth. These problem areas should be investigated in a broad context that will permit the application of solutions to a range of DLF sizes, and will reveal additional considerations impacting the development of the total program.

Table 19 Summary of Life Cycle Cost (Dollars in Millions)

Model Quantity UE Cost Element	-213M 125 112	-212M 125 112	-213MX 125 112	-214A 125 112
Development				
Airframe	$ 2406.2	$ 2562.6	$ 2424.8	$ 4023.0
Engines	571.4	563.6	759.8	1180.5
Avionics	56.0	56.0	56.0	56.0
Flight test A/P				
Airframe	844.7	971.0	850.9	1328.6
Engines	37.1	44.7	41.4	35.9
Avionics	4.5	4.5	4.5	4.5
Total	3919.9	4202.4	4137.4	6628.5
Production				
Airframe	8728.2	8943.2	8029.3	12 686.2
Engines	1545.6	1523.2	1805.4	1497.4
Avionics	280.0	280.0	280.0	280.0
Total	10 553.8	10 746.4	10 114.7	14 463.6
Support investment				
Initial spares	1055.4	1074.6	1011.5	1446.4
AGE, other	527.7	537.3	505.7	723.2
Total	1583.1	1611.9	1517.2	2169.6
Operations and support (1)				
AGE, spares, mods	1097.3	1106.2	1061.9	1412.2
Military pay and allowance	2512.9	2508.9	2516.9	2541.6
Depot maintenance	2466.7	2453.8	2493.4	2624.6
Fuel	5727.7	5705.3	5729.9	5649.3
Pipeline support	313.0	312.6	313.5	315.8
Other	1057.3	1056.2	1058.3	1063.2
Total	13 174.9	13 143.0	13 173.9	13 606.7
Total life cycle cost, millions	$29 231.7	$29 703.7	$28 943.2	$36 868.4

(1) 250 4-hour flights/year @ full payload

Table 20 Summary of Life Cycle Cost—Model 759-213M (Dollars in Millions)

	(1)	(2)	(3)
Quantity	125	125	125
UE	112	112	112
Cost Element			
Development			
Airframe	$ 2406.2	$ 2406.2	$ 2406.2
Engines	571.4	571.4	571.4
Avionics	56.0	56.0	56.0
Flight test airplane			
Airframe	844.7	844.7	844.7
Engines	37.1	37.1	37.1
Avionics	4.5	4.5	4.5
Total	3919.9	3919.9	3919.9
Production			
Airframe	8728.2	8728.2	8728.2
Engines	1545.6	1545.6	1545.6
Avionics	280.0	280.0	280.0
Total	10 553.8	10 553.8	10 553.8
Support investment			
Initial spares	1055.4	1055.4	1055.4
AGE, other	527.7	527.7	527.7
Total	1583.1	1583.1	1583.1
Operations and support			
AGE, spares, mods	1097.3	1097.3	1097.3
Military pay and allowance	2512.9	2512.9	2512.9
Depot maintenance	2466.7	2466.7	2466.7
Fuel	5727.7	4601.0	5837.4
Pipeline support	313.0	313.0	313.0
Other	1057.3	1057.3	1057.3
Total	13 174.9	12 048.2	13 284.6
Total life cycle cost, millions	$29 231.7	$28 105.0	$29 341.4

(1) 250 4-hour flights/year @ full payload
(2) 250 4-hour flights/year @ 35% payload
(3) 80 12.5-hour flights/year @ full payload

Table 21 Summary of Life Cycle Cost—Model 759-213M (Dollars in Millions)

Quantity UE Cost Element	100 90	200 180	300 270
Development			
Airframe	$ 2406.2	$ 2406.2	$ 2406.2
Engines	571.4	571.4	571.4
Avionics	56.0	56.0	56.0
Flight test airplane			
Airframe	844.7	844.7	844.7
Engines	37.1	37.1	37.1
Avionics	4.5	4.5	4.5
Total	3919.9	3919.9	3919.9
Production			
Airframe	7378.7	12 589.0	17 360.1
Engines	1236.5	2473.0	3709.4
Avionics	224.0	448.0	672.0
Total	8839.2	15 510.0	21 741.5
Support investment			
Initial spares	883.9	1551.0	2174.2
AGE, other	442.0	775.5	1087.1
Total	1325.9	2326.5	3261.3
Operations and support (1)			
AGE, spares, mods	913.8	1652.2	2355.1
Military pay and allowance	2019.3	4038.5	6057.8
Depot maintenance	1982.1	3964.3	5946.4
Fuel	4602.6	9205.2	13 807.8
Pipeline support	251.6	503.1	754.7
Other	849.6	1699.2	2548.8
Total	$10 619.0	$21 062.5	$31 470.6
Total life cycle cost, millions	$24 704.0	$42 818.9	$60 393.3

(1) 250 4-hour flights/year @ full payload

The technical limits relative to size have not yet been discovered. Size studies related to aeroelastics, market matching, payload density, and various payload types are recommended. The parametric evaluation of rigid airplanes can indicate the available potential in a given configuration, but will not ensure that this potential is attainable.

The aeroelastic problem may limit size. Size studies covering a wide range of sizes and configurations using aeroelastic methods of evaluation, coupled with proper stability and control augmentation simulated in the aeroelastic model, will reveal the nature of the aeroelastic problem as a function of size. A computer program designed specifically to handle parametric studies at low cost for the purpose of understanding these phenomena is now available.

Market matching studies as a function of a projected market lasting over a projected period of time (e.g., 20 years) would determine if size limitation creates a significant impact upon the market. Studies to date do not indicate that a 544 311-kilogram (1.2 million-pound) payload airplane is too large for the projected market and that adequate frequency of service would be available in a typical hub and spoke system.

Wing loading is heavily influenced by payload density, consequently, such density will also affect the size of the airplane. Some projections of payload density indicate that the average density may increase to about 320 kilograms per cubic metre (20 pounds per cubic foot). If this trend takes place, size studies should include a payload density parameter. Other payloads may be important for special configurations of distributed load freighters designed to haul liquified natural gas at 416 kilograms per cubic metre (26 pounds per cubic foot), or jet propulsion fuel at 801 kilograms per cubic metre (50 pounds per cubic foot). All of these factors will have an impact on the optimum size. If the optimum sized airplane is found to be very large, it will be highly desirable to build a scale demonstrator for the purpose of demonstrating the feasibility of the control system and manufacturing methods. The size studies described above would extend below 454 megagrams (one million pounds) gross weight for the higher payload densities and may indicate that a scale demonstrator could be a useful vehicle in filling the role of a tanker for JP fuel or possibly liquified natural gas.

Thick airfoil technology was found to be very important in the performance of the airplane and size optimization. There is limited high speed high Reynolds number data available on airfoils that may be optimized for distributed load freighters. A systematic study of airfoil shapes, including the possibility of boundary layer suction on the aft portion, or ventilated-base thick airfoils could lead to significant reduction in weight and cost of distributed load freighters. This study concentrated on technology available for beginning of production in 1990. Given a reasonable development program, much of the technology required for these simplified airplanes could be available at least five years earlier.

The installation of engines over the leading edge of the DLF wing is expected to result in a more severe interference problem than for under-the-wing installations. This arrangement is most favored in the design of the DLF, which indicates that details of this installation should be established. It is proposed that test data of past installations of this type be examined with the aim of confirming the present interference levels. A subsequent program to minimize adverse interference by appropriate local contouring and/or acceleration bodies can be accomplished by application of a Boeing Potential Flow Program. A simplified wind-tunnel program should then be performed to confirm the results. A reduction in induced drag due to upper-surface blowing has been suggested in some recent data (Reference 4). The benefits quoted were used in the analysis of the final DLF configurations and were found to be significant only at low speed.

In summary, the technical areas that need further work are: the aeroelastic problem, the airplane handling characteristics, the engine nacelle placement, and the high speed airfoil wing design problem. Although the aeroelastic problem warrants the highest priority, preliminary analyses indicate that the multiple segment undelegated trailing edge control flaps have a powerful effect on the stability of the wing section just forward of each flap segment. The analyses also indicate that the whole trailing-edge can be programmed to produce all the required control responses of the entire wing. The handling characteristics problem can be attacked by using flight simulators. The nacelle placement and high speed airfoil design problems would be initially analyzed using theoretical techniques, followed by wind tunnel cut-and-try variations of the more promising analytical solutions. A small scale flight demonstrator programmed to simulate various sizes of DLF airplanes should be considered, thus, the aeroelastic flight study must include the effect of scale.

6.0 CONCLUDING REMARKS

The swept-wing distributed load freighter airplane concept (Model 759-211) shows promising potential, but the optimum size occurs at about triple the payload weight of the conventional [544 311 kilograms/181 437 kilograms (1.2 million pounds/400 000 pounds)]. The resulting ton-mile costs are one half those of present airfreighters and about 75 percent of the best advanced conventional design incorporating the same technology. This superiority is a result of the DLF configurations' characteristic of continuously improving aerodynamic efficiency with size, while holding or slightly improving the weight fraction. The fuel efficiency is double that of present airfreighters and 19 percent better than the reference advanced conventional design. The airframe noise is relatively low, allowing ample opportunity for noise reduction in the propulsion system.

The present study determined the size and shape of the DLF type for the best economics on the basis of an appropriate set of simplifying assumptions. Further studies of greater technical depth (e.g., aeroelastic effects, handling qualities, and high speed aerodynamics) are required to determine the technical limits on size.

The economic breakthrough for airfreight shown by the DLF concept may be achievable with less risk and cost than alternative approaches. The advent of the digital computer, functioning as a control for large airplanes which have their controls so distributed as to permit a fine tailoring of the air load distribution, has the potential of permitting the controlled operation of very lightweight structures that are relatively inexpensive to build. The complexity of these systems and their development costs may be of lesser magnitude than laminar flow control, which is possibly the only known alternative for achieving very high performance. With regard to the military configuration, it may be noted that the distributed load military configuration 759-213M shows a significantly lower life cycle cost than the reference (conventional) military airplane 759-214A. The life cycle cost of the military version produced by the joint civil/military program (759-213MX) is still lower, even though the pricing groundrules assume that the military would pay all common nonrecurring costs of the program. Although the civil version produced by the civil/military program (759-213C) benefits from military participation, the performance penalty it suffers due to smaller size [272 155-kilogram (600 000-pound) net payload] results in economics slightly inferior to those of the 759-211. A civil/military program based on the larger 759-211, however, would provide a more economic commercial version while reducing the life cycle cost of the military version.

APPENDIX A

AERODYNAMIC PARAMETRIC DATA BASE

1.0 PERFORMANCE METHOD - THUMBPRINT COMPUTER PROGRAM

The airplane performance produced during the course of this study was calculated using the Boeing-developed computer program TEI-004, Computer Application to Airplane Design Selection (Thumbprint Computer Program), as employed in the previous DLF contract (Reference 1). This program is a tool for sizing aircraft that perform given transport missions. It parametrically adjusts base point design input data to generate large numbers of sized variants, analyzes their characteristics, and permits optimum point selection. The program internally calculates variations in takeoff gross weight, thrust, and takeoff field length, thus permitting selection within chosen constraints on these parameters. These tasks are accomplished using aerodynamic, weights, and propulsion preliminary design procedures. A conceptual flow chart of the Thumbprint Computer Program is shown in Figure A-1.

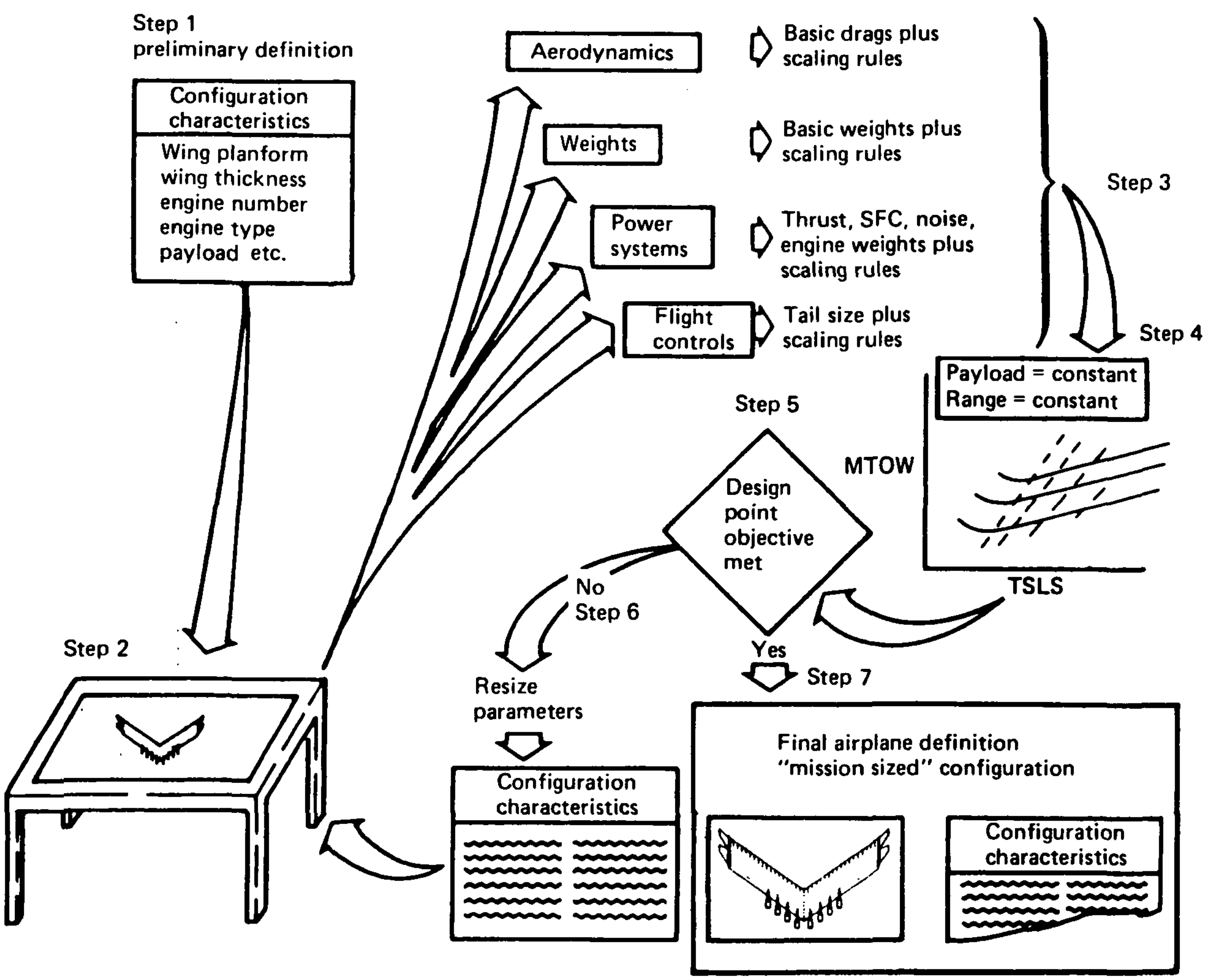


Figure A-1 Aerodynamic Thumbprint Program Flow Chart

Inputs to the program include: (1) an uncycled base point airplane geometry, aerodynamics, weights, and propulsion, and (2) scaling relationships for adjusting the base point values for changes in wing area, engine size, payload and range.

The term uncycled, when used in conjunction with cruise drag polars and minimum parasite drag, refers to an evaluation of an aircraft with nonoptimum engine thrust levels. The aircraft is cycled when Thumbprint has been run and all output, specifically weights, has been checked and approved for the sized aircraft with the engines optimized for takeoff field length and/or cruise flight conditions.

Output of the program as utilized in this study defines the performance weight and aerodynamic characteristics of point design airplanes. Also, off design data for the specific point designs provide the variation of performance for off-loaded conditions.

2.0 AERODYNAMIC PREDICTION BASE

In order to provide needed thick-wing experimental data and to improve confidence in prediction techniques, two exploratory wind tunnel tests were conducted in 1974. These tests provided drag data over a range of Mach numbers and also indicated that high-lift device characteristics were predictable and that ground effects were noncritical.

2.1 AERODYNAMIC DATA BASE

Early studies, plus the results of the above mentioned wing tunnel tests, indicated that three thickness-dependent aerodynamic parameters would be of primary importance in the selection of wing thickness ratio and, hence, the chord, area, aspect ratio, and payload volume for a given span. These three parameters were:

- Subcritical form drag factor
- Drag divergence Mach number
- Degree of drag creep

These three parameters, together with calculable drag items, were used to describe the cruise drag characteristics of payload-in-wing airplanes as illustrated in Figure A-2.

In order to provide aerodynamic inputs for a study in which wing thickness ratio was to be one of the main independent variables, parametric trends of these three variables as a function of thickness ratio were generated, making use of the above wind tunnel results and other pertinent airfoil data.

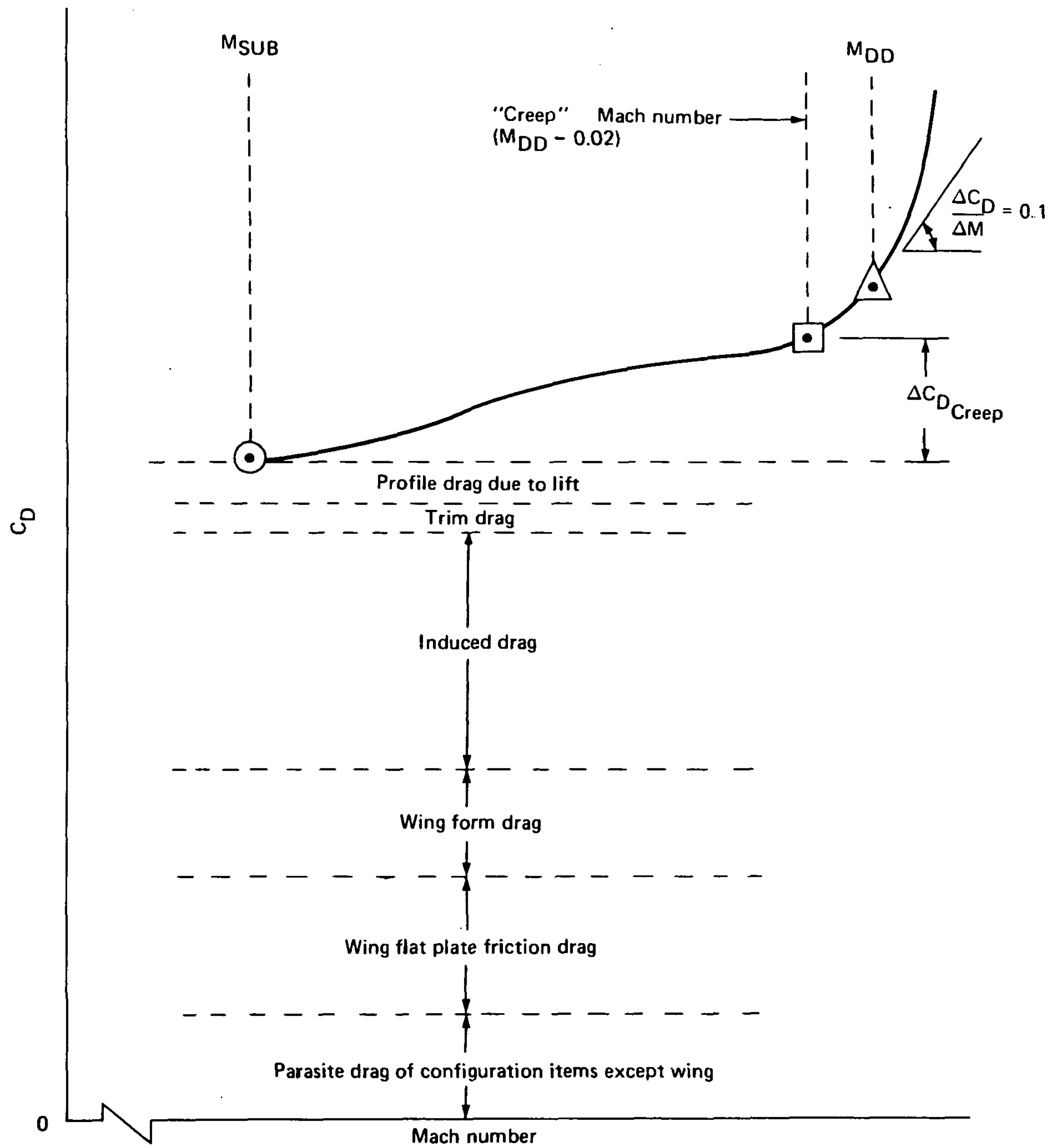


Figure A-2 Cruise Drag Characteristics

2.2 CRUISE DRAG BUILDUP

The established subsonic drag prediction techniques and secondary data obtained from the above mentioned wind tunnel tests, were used to construct cruise drag characteristics in the manner described below.

2.2.1 PARASITE DRAG

The parasite drag for each configuration component was built in the manner shown in Table A-1 for the sample Model 759-211 using four items:

- Flat plate skin friction drag
- Viscous-related form drag
- Pressure and interference drag
- Roughness and excrescence drag

All items in this buildup were computed using internal Boeing methods.

2.2.2 INDUCED DRAG

Induced drag information was obtained by using Vortex Lattice Program A372 (Reference 3). Individual cases were run for each independent control variable to be used in the trimming process. From these cases, a special spanload analysis program was used to calculate induced drag influence coefficients associated with the trim variables. These results were subsequently applied in the trim procedure (see Section 2.2.7).

Table A-1 Parasite Drag Summary—Model 759-211

Item	Flat plate friction drag					Form drag		Pressure and interference drag			Rough and excresence	Total, m^2 (ft^2)
	A_{wet}, m^2 (ft^2)	L_{ref}, m (ft)	$R_e \times 10^{-6}$	$C_{f_{avg}}$	F_{PL}, m^2 (ft^2)	Form factor	Form, m^2 (ft^2)	A_π	C_{D_π}	Pressure and interference		
Wing	7579.747 (81 587.4)	28.35 (93.025)	141.398	0.00194	14.71 (158.335)	(0.2727)	4.012 (43.18)				1.274 (13.717)	19.996 (215.232)
Body (tip caps)	209.386 (2253.8)	27.78 (91.151)	138.549	0.001942	0.407 (4.377)	(≈0.097)	0.039 (.424)				0.035 (0.379)	0.481 (5.18)
Vertical tip fin	776.646 (8359.72)	8.11 (26.613)	40.453	0.00232	1.803 (19.406)	(0.2532)	0.457 (4.915		(0.0024)	(5.535)	0.131 (1.406)	2.904 (31.262)
Horizontal tip fin	336.803 (3625.298)	9.90 (32.478)	49.366	0.00225	0.759 (8.172)	(0.1811)	0.137 (1.480)		(0.0033)	(13.792)	0.057 (0.610)	2.235 (24.054)
Nacelles (B) $TSLS_{Ref}$ = 43 091 kg (95 000 lb)	719.593 (7745.6)	7.59 (24.9)	37.848	0.00232	1.669 (17.970)	(0.39)	0.651 (7.008)	106.38 (1145.1)	(0.03)	(34.354)	0.121 (1.302)	5.633 (60.631)
Nacelle struts {wing (B), other (D)}	340.472 (3664.8)	5.09 (16.7)	25.384	0.00242	0.824 (8.869)						0.057 (0.616)	0.881 (9.485)
Trailing edge (clipped) (base drag ΔC_D = 0.0002												0.757 (8.146)
Total	9962.682 (107 237)				20.172 (217.129)		5.296 (57.007)				(53.681)	32.887 (353.993)

$C_{D_{P_{min}}}$ = (0.00869)

2.2.3 PROFILE DRAG DUE TO LIFT

Profile drag due to lift was computed using internal Boeing methods (see Figure A-3).

2.2.4 COMPRESSIBILITY DRAG

The drag rise curves for varying lift coefficients were constructed by internal Boeing methods, utilizing a modified, "non-peaky" airfoil section, drag rise shape (see Figure A-3).

In the case of the final civil configuration, 759-211, it was noted that the vertical tipfins at 26-degree sweepback were suffering drag rise penalties at a lower Mach number than the wing; therefore, the sweepback was increased to 30 degrees.

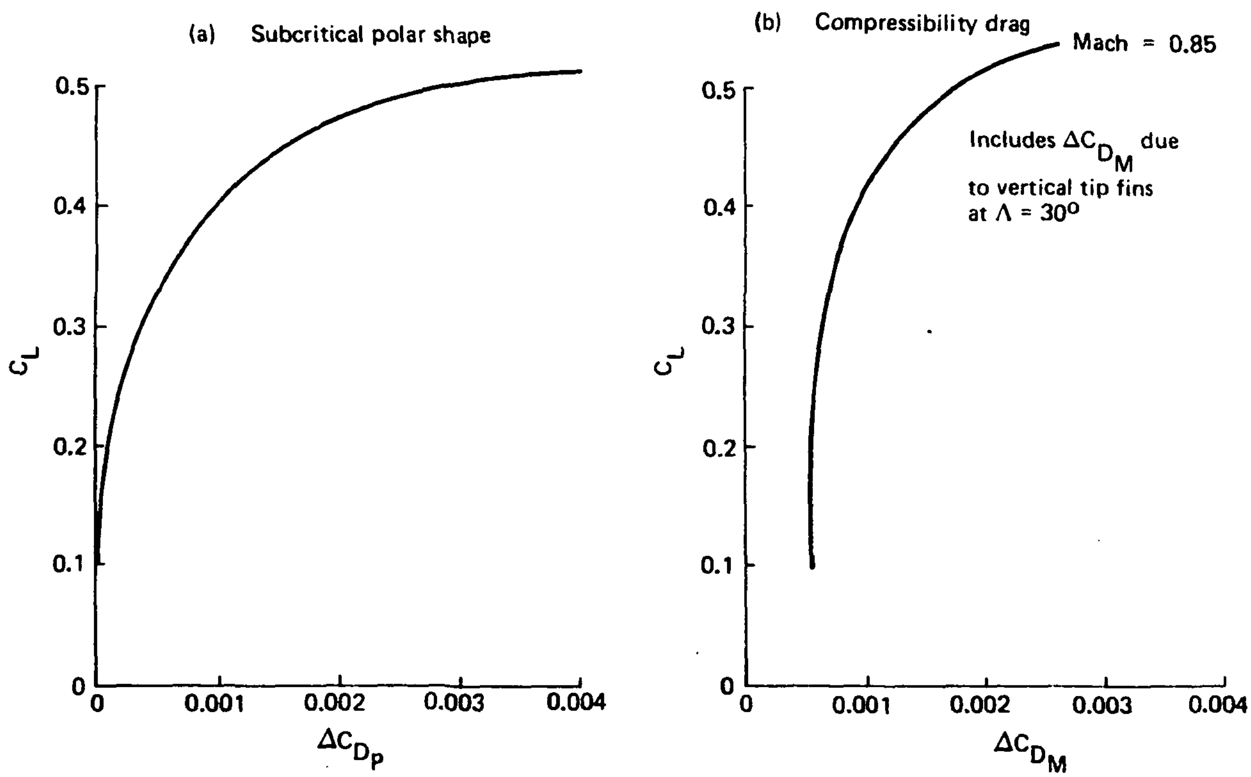


Figure A-3 Cruise Polar Shape and Compressibility Drag—Model 759-211

2.2.5 UNTRIMMED CRUISE POLARS

Untrimmed cruise polars were constructed by summing items 2.2.1 through 2.2.4 above. A typical set of untrimmed polars for the Model 759-211 is indicated in Figure A-4.

2.2.6 FLAP DRAG

Trailing-edge flap drag data were obtained from Reference 4 in which semi-empirical methods were used to relate flap parasite drag to flap area, deflection and tvpe of flap. This information was used in the trim procedure described in the following section.

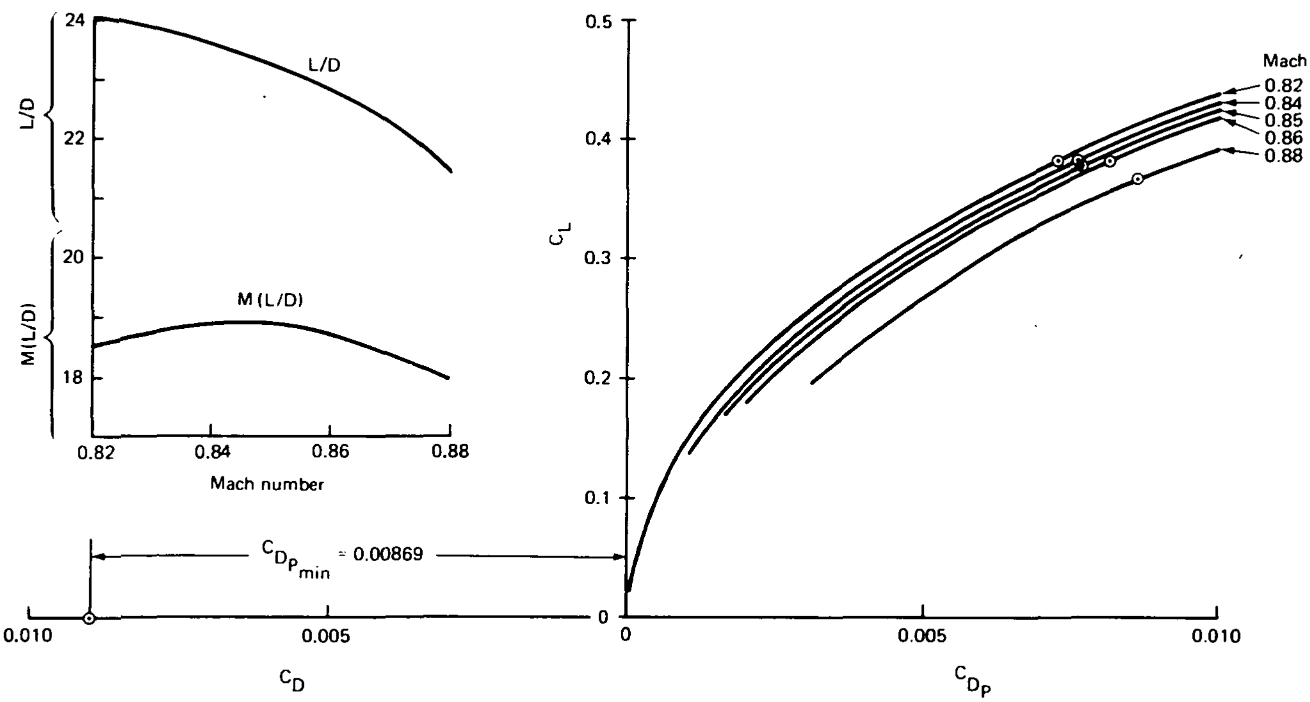
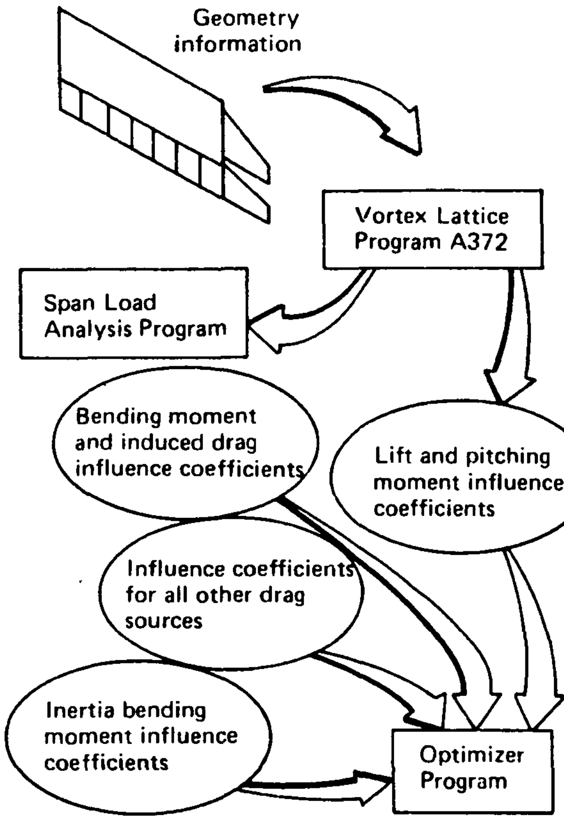


Figure A-4 Untrimmed Cruise Polars—Model 759-211

2.2.7 TRIM PROCEDURE AND DRAG OPTIMIZATION

A special trim procedure was formulated for the airplanes analyzed in this study. Figure A-5 provides an illustration of the approach used. Airplane geometry specifications were placed into Vortex Lattice Program A372 (Reference 3) and cases were executed for each independent variable to be used in the trim procedure. In general, these variables included angle of attack, flap deflections (the wing trailing-edge was divided into a number of flap segments, usually eight), fin deflections (horizontal and vertical), and wing camber. Lift and pitching moment influence coefficients were calculated for each of these variables by employing this program. A special auxiliary program was used to analyze the spanload distribution associated with each variable to generate induced drag and aerodynamic bending moment influence coefficients. Finally, additional drag influence coefficients were generated to approximate the remaining drag sources: flap drag, compressibility drag, and profile drag resulting from lift.

All of this aerodynamic information, along with bending moment influence coefficients associated with the inertia loads (payload, fuel, and OEW), was fed into a special trim/optimization program.



- Vortex Lattice Program A372 is used in conjunction with an auxiliary program to obtain lift, pitching moment, bending moment, and induced drag influence coefficients for each independent aerodynamics loading
- Inertia load information is provided and bending moment influence coefficients are derived
- Aerodynamic and inertia influence coefficients are combined and fed to the optimizer program

Figure A-5 Aerodynamic Approach

This program was used to determine that combination of independent variables (angle of attack, flap deflections, fin deflections, and wing camber) which results in minimum drag subject to the following constraints:

- That trimmed flight occurs at a specified cg location
- That a specified lift coefficient is produced
- That nowhere along the wing span does the 1-g net bending moment (aerodynamic minus inertia) exceed a specified value

Bending moment constraints were not imposed during the parametric study but were used during analysis of the selected configurations. Similarly, wing camber was only used as a variable for the selected configuration analyses.

Drag results from the optimizer program were then used in the Thumbprint Computer Program (see below).

2.2.8 THUMBPRINT INPUTS

As stated previously, cruise drag inputs for the Thumbprint matching and sizing program consist of a parasite drag breakdown such as shown in Table A-1; a curve or subcritical polar shape versus C_L; and curves of compressibility drag versus C_L and Mach number.

Polar shape is defined as all lift-dependent drag items in excess of minimum elliptic induced drag and includes nonelliptic induced drag, profile drag due to lift, and trim drag. Compressibility drag consists of increments to be applied to the subcritical drag polar to yield compressible polars and includes drag creep and trim drag increments. Typical polar shape and compressibility drag inputs are shown in Figure A-3.

The Thumbprint method also accepts parasite drag scalars in order to calculate drag increments created by changes in the sizes of wing, empennage, body, and propulsion system away from the baseline input (uncycled) configuration. The Thumbprint for the final civil configuration (759-211) at its design payload and range is shown in Figure A-6.

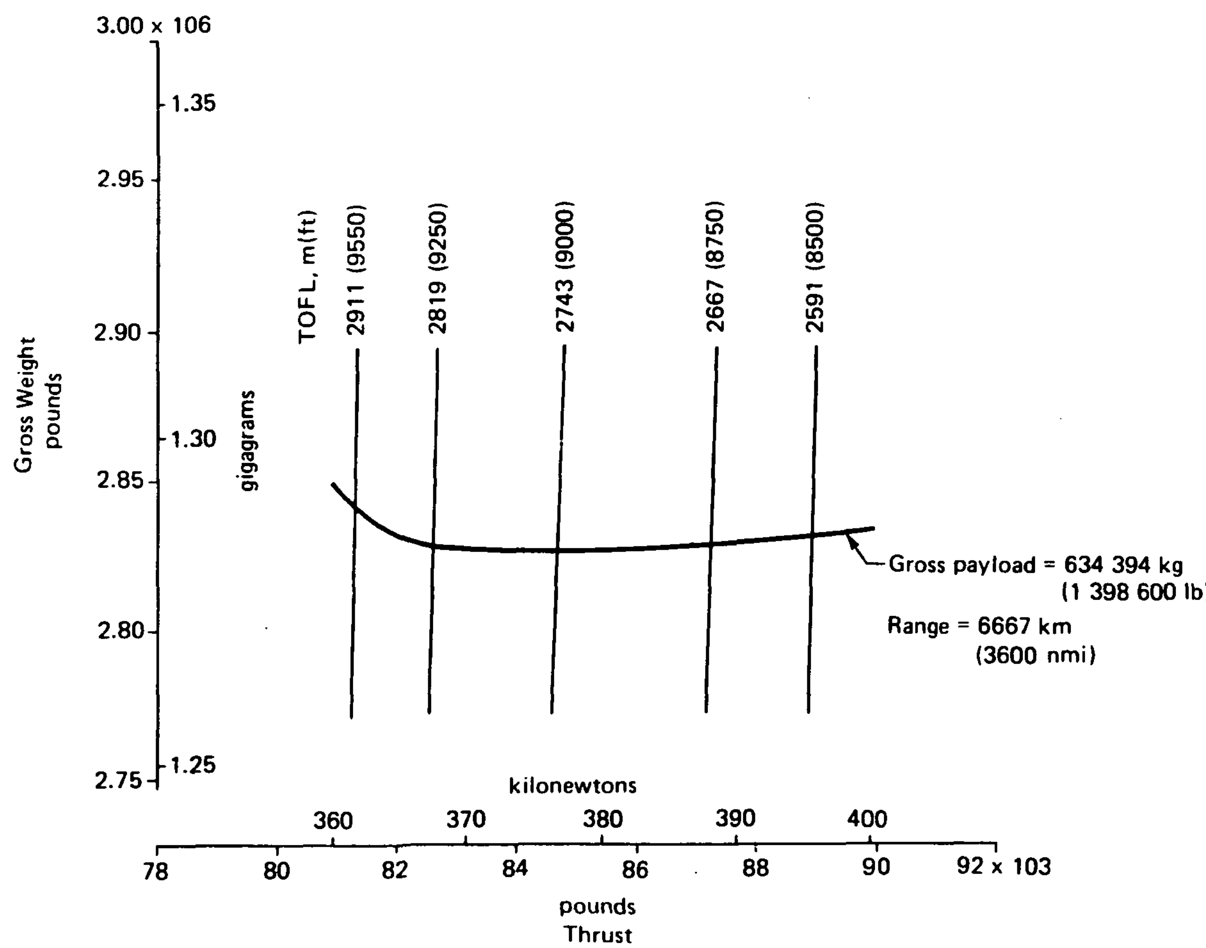


Figure A-6 Thumbprint for Model 759-211

APPENDIX B

SWEPT-WING SPAN DISTRIBUTED LOAD CARGO AIRCRAFT COST AND PRICING METHODOLOGY

1.0 COMMERCIAL COSTING AND PRICING METHODOLOGY

1.1 GENERAL

The objective of the estimating approach to the various DLF configurations is to arrive at consistent cost and prices so that the estimates will reflect design differences. It must be recognized that estimates and prices prepared during a conceptual phase are quite preliminary and are subject to considerable revisions as the program progresses.

It is assumed that required facilities and technology are available prior to program go-ahead. All costs and prices are computed in 1976 dollars. Such prices are calculated to yield a reasonable return on investment to the manufacturer.

1.2 RESPONSIBILITIES AND STUDY FLOW

The Boeing Commercial Airplane Company is organized into functional departments that have specific responsibilities and are repositories of company experience in their particular scope of activities. The Preliminary Design department that is responsible for the DLF study management draws necessary skills from other departments to produce inputs in evaluating the economics of prospective new products. Figure B-1 indicates the responsibilities and flow of information between the responsible groups.

Preliminary Design produces the technical description and drawings of the configurations to be studied. The technical staff analyzes these designs and is responsible for the weight, performance, and stability and control characteristics. This information, along with the configuration definition, is given to Engineering Costs and Schedules for an engineering manhour estimate, and to the Operations (Manufacturing) department for a manufacturing plan, part card, and manhour estimate. The Finance Group in the Business Management department makes an independent estimate, coordinates with departmental input, develops a program schedule, and estimates the final costs and prices. The Requirements and Analysis group in Preliminary Design analyzes these prices and determines the operation costs, investment costs, and indirect costs to assess the market potential. In this manner, the full experience and resources of appropriate authorities in every field are utilized to provide preliminary design answers that represent responsible company output.

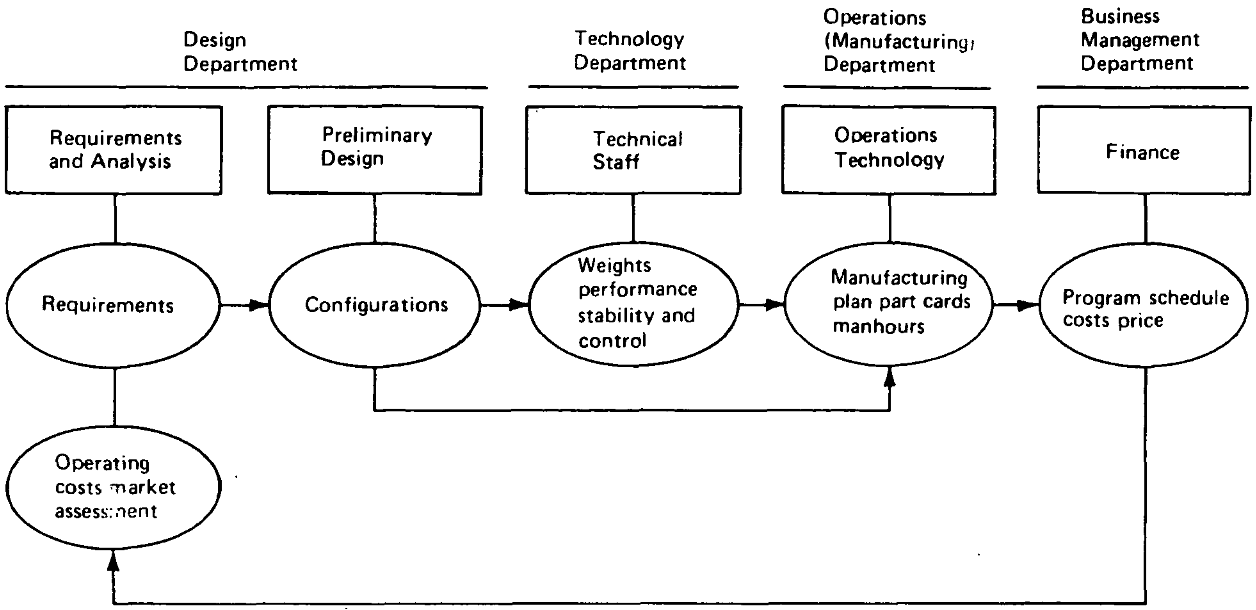


Figure B-1 Pricing and Costing Methodology—Responsibilities and Study Flow

1.3 BASIC REQUIREMENTS AND ASSUMPTIONS

Cost and price data are estimated in 1976 dollars. Market quantities for the civil configuration meet the annual throughput of 167.9 revenue petametre-kilograms (115 billion revenue ton-statute miles). This production run, plus 125 aircraft, is used for the civil version of the military configuration. A production run of 125 aircraft is employed in the military configurations. Load factor is 85 percent of the gross payload (net payload plus container weight). Fuel price is varied at 97.7, 118.9, and 158.5 dollars per cubic metre (37, 45, and 60 cents per gallon). A crew of two is assumed. Direct Operating Cost is calculated using 1967 Air Transport Association (ATA) equations updated with the 1976 Boeing coefficients given on Table B-1.

Analysis techniques used in the development of the airplane prices to be inserted in these DOC equations are described in the following paragraphs.

1.4 COST ESTIMATING METHODOLOGY

The approach used in estimating the costs of distributed load airfreighters is to separate those components of the airplane that are similar to conventional airplanes into one category, and those components that are unique to this concept into another category. The components in the first category are handled by conventional techniques based on correlation with Boeing manufacturing experience. Both the design structure of the main wing box and its trailing-edge surfaces, as well as the manufacturing methods being considered to produce them, fall into the second category.

For these parts, a much greater level of detail on the manufacturing process is undertaken by the Operations and the Engineering Costs and Schedules departments to establish credibility of the estimate. Operations' manhour estimates for the wing box and the trailing-edge surfaces are used as the example to illustrate this activity.

Table B-1 U.S. Intercontinental Direct Operating Cost Formulas (Turbofan; Dedicated Airfreighters)

	Boeing 1976
Crew pay (dollars/block hour) 2-man crew	$22.211 (V_c \times MTOW/10^5)^{0.3} + 44.322$
Fuel, dollars/m^3 (cents/gallon)	97.7, 118.9, 158.5 (37, 45, 60)
Non revenue factor	1.02 on fuel and maintenance
Airframe maintenance—cycle Material, dollars/cycle	$0.89\ (1.235 + 2.261\ Ca/10^6)$
Direct labor, MH/cycle	$0.89\left[\frac{Wa/1000}{0.0419(Wa/1000) + 28.159}\right]$
Airframe maintenance—hourly Material, dollars/FH	$0.89\ (2.508 + 1.736\ Ca/10^6)$
Direct labor, MH/FH	$0.89\left[\frac{Wa/1000}{0.1035(Wa/1000) + 17.919}\right]$
Engine maintenance—cycle Material, dollars/cycle	$1.05\ (16.00\ Ce/10^6 + 19.50)\ Ne$
Direct labor, MH/cycle	$1.05\left[0.0244\ (T/1000) + 0.220\right] Ne$
Engine maintenance—hourly* Material, dollars/FH	$1.05\ (10.256\ Ce/10^6 + 18.115)\ Ne$
Direct labor, MH/FH	$1.05\left[0.0183\ (T/1000) + 0.178\right] Ne$
Burden, MH/direct labor MH	2.0 (MH/direct labor MH)
Maintenance labor rate, dollars/MH	9.00
Investment spares ratio Airframe Engine	 0.06 0.30
Depreciation schedule, years/% residual	15/10 (new airplanes)
Utilization, hours/year	$\frac{t_b\ \text{(AVAILABLE HOURS)}^{**}}{(t_b + 0.5)}$

* For flight hours <2, use: 2-hour cost $-\ 0.73$ [(hourly cost) x (2-flight hours)]

For flight hours >4, use: 4-hour cost + 1.35 [(hourly cost) x 4-flight hours)]

** Parametric study: Available hours = 4649.3 hours per year

Final study: Available hours = 5683 hours per year

Definition of terms and units:

MTOW	Maximum design takeoff gross weight, pounds
Ca	Airframe price, dollars
Ce	Engine price, dollars (excluding reverser)
Ne	Number of engines
T	Sea level static thrust, pounds
V_c	$(715 \times M - 75 \times M^4)$ for $M < 0.9$ $(660 \times M)$ for $M \geqslant 0.9$
Wa	Airframe weight, pounds
M	Mach number
FH	Flight hours
MH	Manhours
CYC	Cycle
t_b	Block time, hours

1.4.1 MANUFACTURING PLAN

The whole cost estimating process starts with a Manufacturing Plan in which the manufacturing methods to be used and the sequence of manufacturing steps for the complete airplane are established. Proposed plant layouts including considerations for handling the very long bonded skin panels approximately 91.44 metres (300 feet) in length, are prepared as part of this activity. A Program Schedule coordinating flow times of the parts production and assembly times of subassemblies and final assemblies is then generated. This is an iterative process requiring reconciled detailed manhour estimates, process flow times, and manloading inputs.

1.4.2 OPERATIONS MANHOUR ESTIMATE EXAMPLE

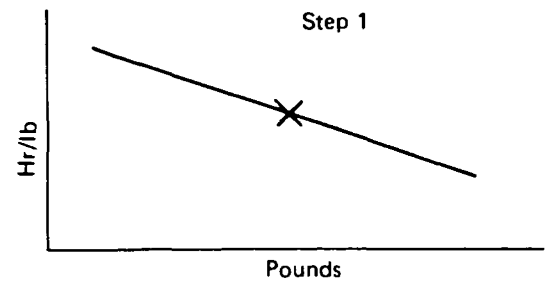
The main wing box plus the fixed leading and trailing edges are entirely built from honeycomb components. Production bonding of these parts and their partial assembly is to be accomplished through the use of a proprietary process in a special facility permitting continuous bonding of parts up to 91.44 metres (300 feet) in length. Considerable depth and detailed analysis are required to produce manhour estimates for these parts to an acceptable level of confidence.

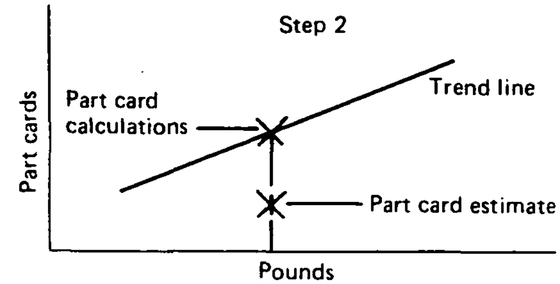
The proposed advanced preliminary design concept can be estimated by relating to company experience with similar projects in the past. The analysis technique used consisted of making a detailed estimate of the flow times and manloading required for each of the steps in the manufacturing process for a particular level of production, e.g., the 100th unit. The contractor has collected and maintained extensive manufacturing experience records such as: comparisons of early preliminary estimates with actual shop performance; learning curve experience; and comparison of shop performance with the ideal performance under controlled conditions. The above preliminary ideal estimate for the 100th unit is subsequently adjusted upward by appropriate historical experience factors and learning curve effects for the particular operation being studied.

The departmental approach described above, in addition to the conventional estimating techniques on the remaining portions of the airplane, is incorporated into a total estimate. These data are compared to the Finance Department estimate that is described as follows:

1.4.2.1 Engineering Labor

The basic estimating approach utilizes hours per pound of design weight for major components of the airplane. Design weight is the weight that Engineering designs rather than the total weight. Examples are the design of landing gear, engine nacelles, and struts. If all are identical, the weight to be considered is the weight of one end item. Adjustment of the base hours is based on the part card deviation from the historical part card versus weight relationship. This particularly affects components of the airframe that have a high degree of commonality within that component.





Formula for a major component of the airplane:

Engineering hours = hours per pounds x pounds x part card estimate/part card calculations

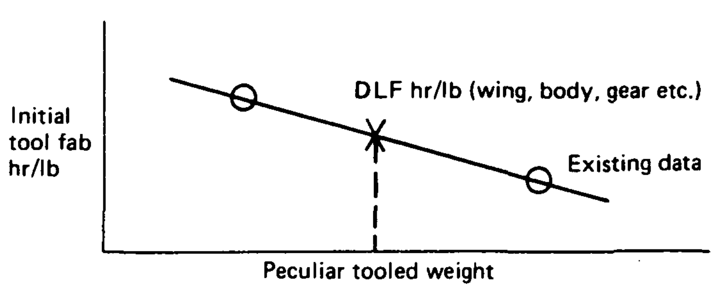
1.4.2.2 Developmental Labor

Developmental labor estimate is composed of tests in support of engineering and the fabrication of mockups. Developmental test labor is estimated by a factor of engineering labor, and developmental mockup is based upon weight as a parameter.

1.4.2.3 Tool Labor

The basic estimating approach utilizes an initial hour-per-pound of peculiar tooled weight extrapolating from existing airplane data. For example, if the nacelles and struts are identical for all locations, the weight of one determines the initial set of tools. Similarly, the wing may have multiple common parts due to nontapered configuration. The initial tooling requirements are based upon only the determined peculiar tooled weight. Adjustments, however, are considered for final assembly or major tools that are not necessarily affected by common parts.

Airplane sectional estimates are made from peculiar weight as follows:

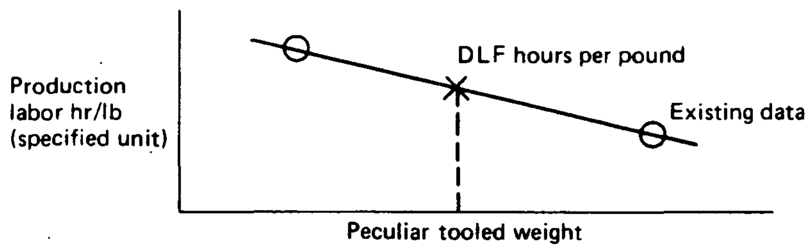


Design and coordination requirements are added as factors of initial fabrication.

Duplication and/or rate tool hours are determined from the production schedule as well as the commonality assessment and are factored from initial tooling. Recurring tooling is estimated as a factor of basic tooling or production labor.

1.4.2.4 Production Labor

As in the case of the tool estimating approach, hours per pound of peculiar weight are used.



For instance, identical nacelles are estimated by unit from historical data and extrapolated for the total program requirements; e.g., six per airplane x 350 airplanes = 2100 units on an improvement curve.

Because of multiple common parts in the wing, the peculiar portion (by weight) is estimated as a unit and extrapolated on an improvement curve to total airplane and program requirements. For example, if the wing is determined to be 40 percent peculiar by weight, each airplane includes 2.5 equivalent units of peculiar construction with cost reductions reflected due to the improvement curve application.

Planning requirements are added as a factor of labor hours. Nonrecurring planning is calculated from part card estimates.

1.4.2.5 Quality Control

Quality control is based on a factor of operations labor.

1.4.2.6 Material

Tool material and developmental material are estimated from historical data as a dollar rate per tool or developmental hour. Production material is calculated as a cost per pound of structure and non-structure weights.

1.4.2.7 Purchased Equipment

Requirements are assessed from existing airplane cost data.

1.4.2.8 Engines

Engines are based on the engine manufacturer's latest available data within The Boeing Company for either existing or study engines.

1.4.2.9 Flight Test

Flight test is estimated as a rate per flight hour.

1.4.3 PARAMETRIC VERSUS POINT DESIGN COSTING

The selected and reference point design configurations are priced and costs determined using the methodology discussed above. The techniques employed for the parametric study differ, however, from the above methods. Because the prime interest is the relative comparison of similar configurations, the parametric study requires less detail. The parametric costing is based on data from previous Boeing studies of distributed-load aircraft.

Recurring costs are estimated based on differences in airframe weight and engine quantities.

1.5 PRICING METHODOLOGY

Commercial pricing incorporates the effect of the program schedule, production rate, quantity of airplanes, program costs, receipts and expenditures. These elements are utilized to establish a price that will yield a reasonable return on the manufacturer's investment.

The relationship between commercial pricing and military pricing of the three programs under consideration is explained in the following information. The commercial program is priced for the delivery of a particular number of airplanes that will produce a fleet productivity of 167.9 Pmkg (115 billion revenue ton-miles) per year delivered in a specified length of time. The pure military is also a straightforward exercise in which 125 military aircraft are delivered in a specified time period. The convertible pricing is based upon cost determination of the common parts for the total number of airplanes [125 Military + 167.9 revenue petametre-kilograms (115 Billion RTM) Civil number], adding a reasonable ROI to the manufacturer for the civil, and cost plus fixed fee for the military. The civil convertible program would be started a short time after the roll-out of the military. The development of the military airplane is charged to the government,and the conversion from military to civil configuration is charged to the civil airplane price.

2.0 MILITARY LIFE CYCLE COST EVALUATION

Airframe costs for the military applications are estimated by using the same cost estimating techniques employed in the civil configurations; however, the pricing is different as noted above. The military operating and support costs are calculated using the CACE (Cost Analysis Cost Estimating) model from the USAF Document AFR173-10, including change 4, dated September 17, 1976. The Boeing version of the CACE model includes maintenance manning calculations as indicated in the following table. This calculation allows the user to vary the estimate of maintenance manhours per flying hour as the air vehicle and operational concepts vary with sortie length, number of sorties, etc. Details of the procedure are illustrated in Table B-2.

Table B-2 Military Operations and Support Costs Using CACE Model (Cost Analysis Cost Estimating Model)

1. **Recurring investment and miscellaneous logistics**
    - **Common support equipment (UE x SE factor)**
    - **Aviation fuel [dollars/gallon = 0.37] (UE x FH x fuel factor)**
    - **Base level maintenance (material only)**
      (UE x FH x BM/FH factor) + (UE x BM/UE factor)
    - **Depot level maintenance**
      (UE x FH x DM/FH factor) + (UE x DM/UE factor)
    - **Class IV modifications (UE x flyaway cost x 0.004494)**
    - **Munitions training**
      (UE x UE related factor) + (UE x CR x crew related factor)
    - **Replenishment spares (UE x FH x replenishment spares factor)**
    - **Vehicular equipment**
      (PPE + BOS/RPM MMY) x (UE factor)
2. **Pay and allowances**
   (officers x pay rate) + (airmen x pay rate) + (civilians x pay rate)
3. **Base operating support**
   (total manpower including PPE + BOS/RPM + MED personnel) x (BOS/RPM factor)
4. **Medical support**
   (PPE + BOS/RPM + MED officers) x (MED factor) + (PPE + BOS/RPM + MED airmen) x (MED factor)
5. **Personnel support**
   (PPE + BOS/RPM + MED officers) x (Permanent change of station factor) + (PPE + BOS/RPM + MED airmen) x (PCS factor)
6. **Pipeline costs**
    - **Acquisition**
      - **Pilots: (UE x CR x pilots/crew) x (turnover rate 0.063) x (AF)**
      - **Non-pilot aircrew-officers: (UE x CR x non-pilots/crew) x (0.059) x (AF)**
      - **Non-pilot officers: (non-rated officers MY) x (0.094) x (AF)**
      - **Airmen: (PPE + BOS/RPM + MED airmen) x (0.134) x (AF)**

Table B-2 Military Operations and Support Costs Using CACE Model (continued)
(Cost Analysis Cost Estimating Model)

6. Pipeline costs (continued)

Training—officers

Pilots: (UE x CR x pilots) x (0.063) x (UPT TNG factor)
Aircrew: (UE x CR x non-pilots) x (0.059) x (TNG factor)
Non-aircrew: (non-aircrew MY) x (0.094) x (TNG factor)

Training—airmen

Maintenance airmen x (0.134) x (TNG factor)
(Total airmen—maintenance airmen) x (0.134) x (TNG factor)

Maintenance manning calculations

MMH/FH x FH/UE/YR x UE x 1.21
Productive MH/month x 12

MMH/FH = maintenance manhours/flying hour
1.21 = maintenance supervision factor x AGE maintenance factor
Maintenance manning distribution factors

0.02 officer
0.98 airmen
-0- civilian

Crew manning calculation

Crew x crew ratio x UE [officers and airmen calculated separately]

Definition of terms:

AF =	acquisition factor
BM =	base maintenance
BOS/RPM =	base operating support and real property management personnel
CR =	crew ratio
DM =	depot maintenance
FH =	flying hours/UE/year
MED =	medical personnel
MH =	manhours
MMH =	maintenance manhours
MMY =	military man years
MY =	man years
PCS =	permanent change of station
PPE =	primary program element (personnel assigned directly to weapon system)
SE =	support equipment
TNG =	training
UE =	unit equipment per squadron
UPT =	undergraduate pilot training
YR =	year

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