AVION

A DETAILED REPORT ON THE PRELIMINARY DESIGN OF A 79-PASSENGER, HIGH-EFFICIENCY, COMMERCIAL TRANSPORT AIRCRAFT

A design project by students in the Department of Aerospace Engineering at Auburn University under the sponsorship of the NASA/USRA University Advanced Design Program.

Auburn University
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Submitted to: Dr. J.O. Nichols

Date Submitted: 23 February 1990

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"Progress in aviation, from its very inception, has probably been best characterized as the product of research, or the application of the 'scientific method.' This method is the logic, or the examination and reasoning process, by which a particular problem or objective is approached. Stepwise, the process involves the collection of available pertinent knowledge, formulation of new hypotheses or theories, critical investigation and experimentation, and, finally, formulation of acceptable conclusions leading to new or revised laws. With sound engineering judgment, this approach translates into careful, systematic study, isolation of variables to evaluate their individual effects, and close attention to details. This is the fundamental research philosophy, or method of inquiry, that is threaded through the story of aviation."

- James F. Connors

(1:2-3)

We would like to thank Dr. James O. Nichols for his wisdom, guidance, and contributions to the preliminary design process included in this report.

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The splinters were worth it!!!

We would like to thank Dr. Jan Roskam (University of Kansas) for supplying us with his *Airplane Design* series which was used as the basis for the methodology included in this report.

Abstract

The Avion is the result of an investigation into the preliminary design for a high-efficiency commercial transport aircraft. The Avion is designed to carry 79 passengers and a crew of five through a range of 1,500 nm at 455 kts (M=0.78 at 32,000 ft). It has a gross take-off weight of 77,000 lb and an empty weight of 42,400 lb. Currently there are no American-built aircraft designed to fit the 60-90 passenger, short/medium range marketplace. The Avion gathers the premier engineering achievements of flight technology and integrates them into an aircraft which will challenge the current standards of flight efficiency, reliability, and performance. The Avion will increase flight efficiency through reduction of structural weight and the improvement of aerodynamic characteristics and propulsion systems. Its design departs from conventional aircraft design tradition with the incorporation of a three-lifting-surface (or tri-wing) configuration. Further aerodynamic improvements are obtained through modest main wing forward sweeping, variable incidence canards, aerodynamic coupling between the canard and main wing, leading edge extensions, winglets, an aerodynamic tailcone, and a T-tail empennage. The Avion is propelled by propfans, which are one of the most promising developments for raising propulsive efficiencies at high subsonic Mach numbers. Special attention is placed on overall configuration, fuselage layout, performance estimations, component weight estimations, and planform design. Leading U.S. technology promises highly efficient flight for the 21st century; the Avion will fulfill this promise to passenger transport aviation.

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Symbol	Description	Dimension
A	Aspect Ratio	---
b	Wing Span	ft
C_D	Drag Coefficient	---
C_L	Lift Coefficient	---
c	Chord Length	ft
c_j	Specific Fuel Consumption	lb/lb/hr
CGR	Climb Gradient	radians
D	Drag	lb
E	Endurance	hrs
e	Oswald's Efficiency Factor	---
e	Inertia Calculation Variable	ft
f	Equivalent Parasite Area	ft^2
g	Gravity	ft/sec^2
GW	Gross Weight	lb
$I_{??}$	Moment of Inertia About ?? Axis	$slug/ft^3$
i	Angle of Incidence	° (degrees)
K	Correction Factor	---
L	Lift	lb
L	Overall Fuselage Length	ft
L/D	Lift-to-Drag Ratio	---
M	Mach Number	---
M_{ff}	Mission Fuel Fraction	---
q	Dynamic Pressure	lb/ft^2
R	Range	nm
R_N	Reynolds Number	---

Symbol	Description	Dimension
$R_?$	Radius of Gyration About ? Axis	ft
$\underline{R}_?$	Non-Dimensionsal Radius of Gyration About ? Axis	ft
S	Reference Area	ft^2
s	Distance	ft
T	Thrust	lb
t	Wing Thickness	ft
TOP_{25}	FAR 25 Take-Off Parameter	lb/ft^2
T/W	Thrust-to-Weight Ratio	---
V	Velocity	kts
V_h	Horizontal Tail Volume	---
V_v	Vertical Tail Volume	---
W	Weight	lb
$W_{\#}$	Mission Phase End Weight	lb
W/S	Wing Loading	lb/ft^2
x	Distance from C.G.	ft
α	Angle of Attack	° (degrees)
λ	Taper Ratio	---
Λ	Sweep Angle	° (degrees)
Γ	Dihedral Angle	° (degrees)
σ	Air Density Ratio	---

Subscripts

A	Approach
alt	Flight to Alternate
av	Mean or Average
cr	Cruise
c/4	Quarter Chord
E	Empty
e	Elevator
f	Fuel
feq	Fixed Equipment
FL	Field Length
h	Horizontal Tail Surface
L	Landing
LOF	Lift Off Field
ltr	Loiter
max	Maximum
min	Minimum
ME	Manufacturer's Empty
o	Minimum
OE	Operating Empty
PL	Payload
pwr	Powerplant
r	Root
r	Rudder
reqd	Required
res	Reserve
s	Stall
struct	Structure
t	Tip
tent	Tentative Guesstimation
tfo	Trapped Fuel & Oil
TO	Take-Off
us	Unswept
v	Vertical Tail Surface
W	Wing
wet	Wetted
2	Landing Gear Retraction

1.0 Introduction

The Avion is a necessary addition to the American aircraft industry, which has been stagnant in the development of bold and entirely new aircraft designs. The Avion gathers the premier engineering achievements of flight technology and integrates them into an aircraft which will challenge the current standards of flight efficiency, reliability, and performance. Leading U.S. technology promises highly efficient flight for the 21st century. The Avion will fulfill this promise to passenger transport aviation, not only in the U.S., but also in the world abroad.

1.1 Problem Statement

The evolution of U.S. commercial passenger transport aircraft has maintained a trend of increasing size, range, and efficiency over predecessors. Since the discontinuation of the 727, 737-100, DC-8, and DC-9 series of aircraft, Boeing and McDonnell Douglas have focussed on aircraft carrying between 130 and 500+ passengers through medium to long ranges.

Recent changes in U.S. business travel practices have opened a commercial aircraft marketplace in which there is no production U.S. passenger transport to compete. One phase of this market includes airline shuttle services, in which many daily flights carry relatively few passengers through short distances between major metropolitan areas. Another phase of this market encompasses flights from smaller cities to hub airports. Airlines currently renewing their fleets are purchasing foreign aircraft since they have been left with no U.S. alternatives to adequately fit these routes.

This report undertakes the preliminary design of a 79 passenger, short/medium range aircraft to compete and gain control of this market for the U.S. in routes both here and abroad. To accomplish this, the design focusses

heavily upon higher efficiency without sacrificing performance or reliability. The Avion design approach integrates already-proven technology with new technology. The features to be incorporated are as follows: a tri-wing configuration, propfan powerplants, forward-swept wings, winglets, aerodynamic coupling, strakes, T-tail empennage, and an aerodynamic tailcone. The Avion preliminary design can be found in Figure 1.1.

1.2 Design Approach

To achieve higher efficiency, it was initially recognized that the Avion would need to incorporate fundamental design differences from conventional aircraft. Since efficiency was the governing factor in the design, it became evident that the Avion would indeed evolve into a unique aircraft.

Past improvements in aircraft efficiency have come from efforts to increase size and speed. Neither of these techniques has been particularly successful. For example, large aircraft, such as the 747, often fly with a significant number of empty seats. These situations result in lower efficiency since passenger traffic per seat mile is not maximized. Also, faster aircraft such as the Concorde SST are inefficient in cost per passenger mile. Therefore, it should be noted that efficiency does not necessarily increase with Mach number.

Despite the above reasons for changing the trends of future passenger aircraft, the industry continues in a state of stagnation with respect to bold, new designs. As an example, two recently developed production airliners, the Boeing 757 and 767 series, did little more than upgrade their older counterparts. The design of these aircraft was perhaps the epitome of conservatism. Analogous to Newton's First Law, the aircraft industry continues in its reluctance to change. Entirely redesigned aircraft have been rebuffed, possibly because of the considerable amount of risk and money involved. Even new technology, as

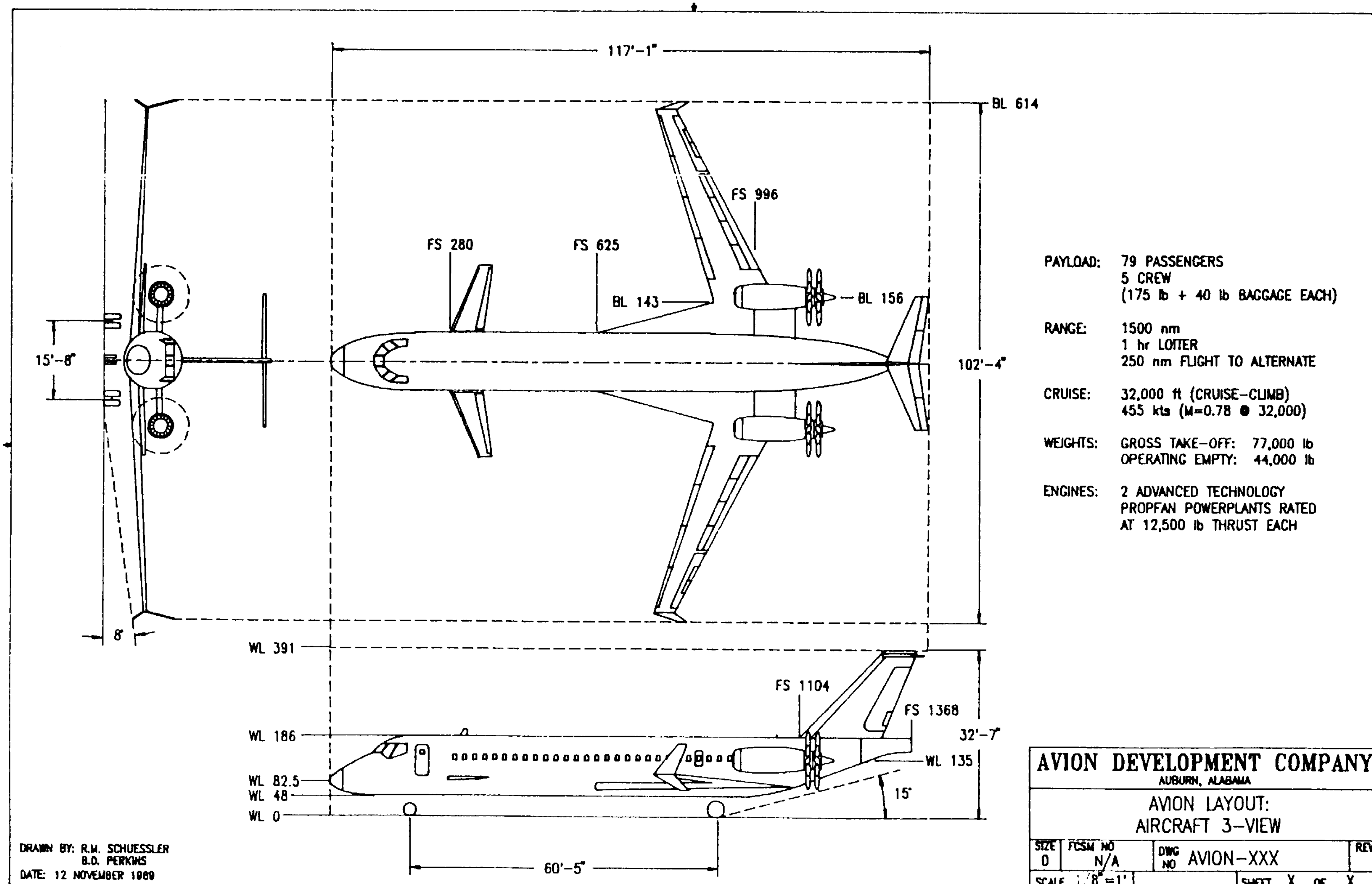


Figure 1.1 Avion Configuration 3-View

demonstrated on the propfan-propelled MD-91X, has been restrained by the industry's inertia. But, eventually the day will come when this conservative policy is no longer profitable. Fuel costs will continue to rise and foreign competition will only improve, leaving the traditional aircraft designs inadequate at their efficiency levels.

The Avion preliminary design started effectively from scratch, with no preconceived ideas or limitations. The targeted Avion achievement goal was the development of an aircraft which pushed the limits of flight efficiency, reliability, and performance. In this attempt, each aircraft component was looked at individually in order to seek and find its practicality of usage in the final design. If drawbacks and disadvantages were encountered, methods of circumventing or remedying the problems were considered. Individual component effects on the others were carefully examined. Constant emphasis was placed on all the components acting collectively in the final configuration.

Weight, simplicity, accessiblity, maintainability, and cost are critical in the design of any aircraft. In view of these items, it is recognized that above certain cost levels, no aircraft will be sold. There is a need, therefore, to minimize the expense of new research and development. However, since the conceptual design has focussed upon a higher efficiency configuration, it is reasonable to expect that the Avion will be marketed effectively with a price tag higher than other aircraft in its category. The applications of newly developed technologies were considered for use in the design because the expenses of their integration were offset by the fuel savings of increased efficiency.

1.3 Mission Specification

The mission specification for the Avion was defined based upon the competition aircraft and the design market. (See Figure 1.2 and Table 1.1.)

AVION MISSION SPECIFICATION

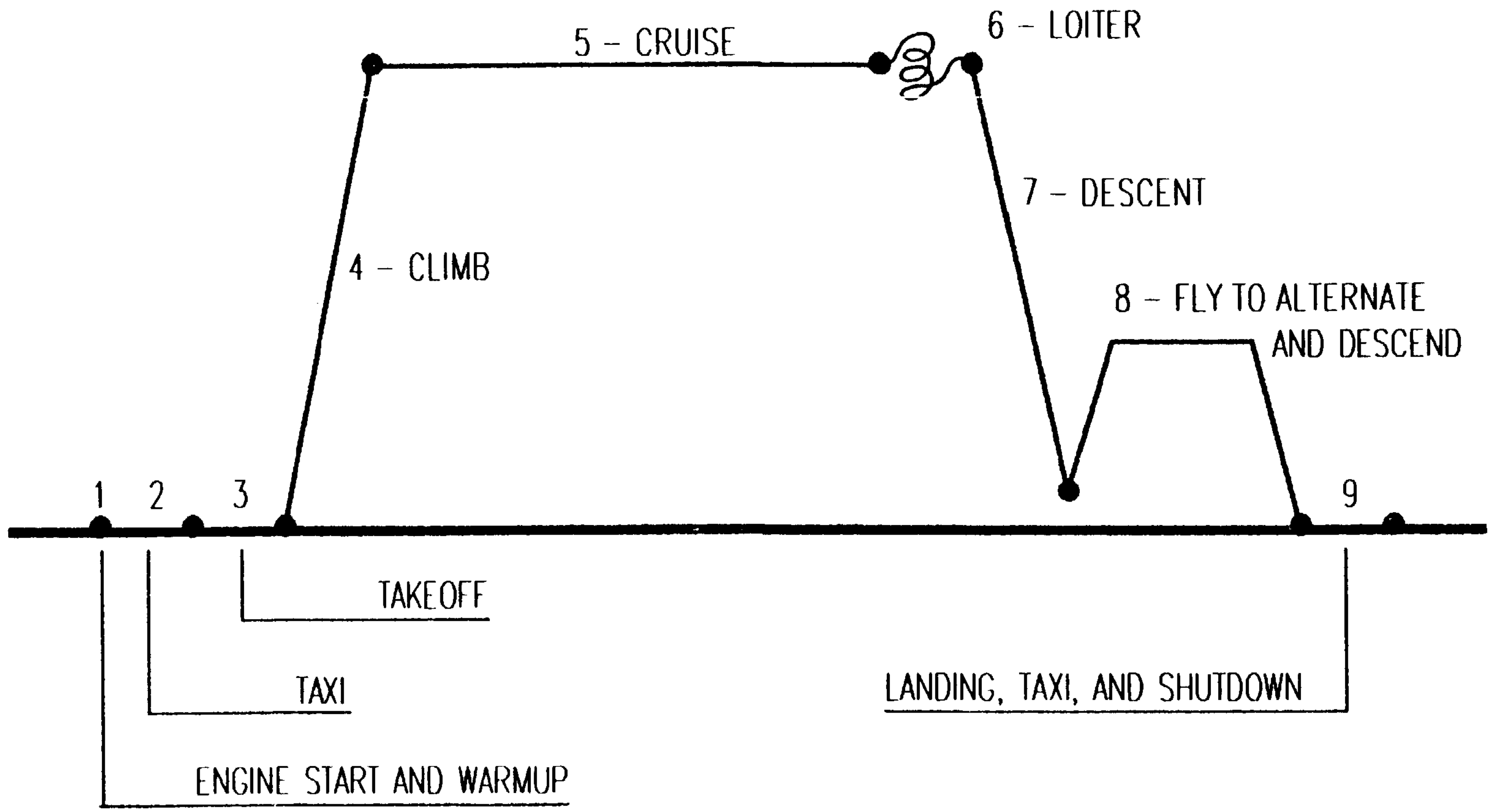


FIGURE 1.2 AVION MISSION SPECIFICATION

Table 1.1 Avion Mission Specification

Payload:	79 passengers at 175 lb each and 40 lb baggage each
Crew:	Two pilots and three cabin attendants at 175 lb each and 40 lb baggage each
Range:	1,500 nm (under still air, standard day, ideal conditions) Reserves: 1 hour loiter and 250 nm flight to alternate landing site.
Altitude:	32,000 ft (for the design range)
Cruise:	455 kts (M=0.78) at 32,000 ft

2.0 Aircraft Configuration

Aircraft efficiency can be improved through three methods: improvement of aerodynamic characteristics, reduction of structural weight, and/or improvement of propulsion system. The Avion incorporated each of these methods in its preliminary design. Several initial decisions were made during the proposal period regarding the overall configuration and propulsion system. This section provides a basic overview of the features to be incorporated into the Avion.

2.1 Overall Configuration

All production commercial transports use the conventional wing-tail arrangement. As mentioned in *Section 1.2*, limited effort has been made to deviate from this tradition. This is because the experience and data base accumulated over the past 85 years of successful flight has provided a rather simple and reliable approach to aircraft design. However, especially in recent years, there has been a renewed interest in the canard-wing (or tail-first) design. This is because canards maintain certain inherent advantages:

(1) The trimmed maximum lift coefficient is higher than that for a conventional design.

(2) It is possible to achieve better trimmed lift-to-drag ratios.

(3) Since both the canard and wing produce lift (opposed to negative lift of the tail in a conventional configuration), less wetted area is required for the aircraft, resulting in a substantial decrease in skin friction drag.

However, several matters need special attention in a canard design:

(1) The canard must be designed to stall before the wing, yielding a stable "pitch-break".

(2) The canard must be prevented from stalling during landing to avoid violent pitch-down motions near the ground.

(3) Aerodynamic induction effects of the canard tip vortices and canard downwash on the main wing can cause poor induced drag behavior and adverse structural stresses due to increased wing root bending moment.

These problems were alleviated through three innovative design improvements.

First, the canard stalling problems were solved through the use of both control surfaces and a variable-incidence canard. Also, the canard airfoil was selected such that its lift coefficient would not drop off abruptly at the stall angle.

Second, it was realized that induced aerodynamic effects could be used to an advantage through the use of a forward-swept wing. In this configuration, the canard downwash and vortices compensate for the wing spanwise flow characteristics. Therefore, the forward-swept wing naturally complements the canard arrangement in such a way that the attractions of a canard layout are much more fully achieved that with an aft-swept wing. (Wing design will be discussed further in *Section 5*.)

Third, as an evolutionary hybrid from the conventional and canard configurations, a compromise was reached for the Avion with the three-lifting-surface (or tri-wing) configuration. This configuration retains the tail of the conventional arrangement, but uses it as an additional lifting surface, rather than a stabilizing (down-loading) surface. Among the favorable attributes of the tri-wing configuration are the following:

(1) The tri-wing layout can achieve higher trimmed cruise lift-to-drag ratios than either of the two-surface layouts through minimization of induced drag. This can be achieved at <u>any</u> c.g. location.

(2) The longitudinal primary and trim controls can be incorporated in the horizontal tail as in a conventional configuration.

(3) Trim of flap induced pitching moments can be performed by a flap on the canard which is mechanically geared to the wing flaps.

2.2 Fuselage Configuration

The Avion fuselage carries the crew, passengers, cargo, and most of the systems needed for operation of the aircraft. As Jan Roskam notes:

> "In commercial passenger operations, the interior design reflects a compromise between level of creature comforts and the weights and sizes required to create the creature comforts." (2:45)

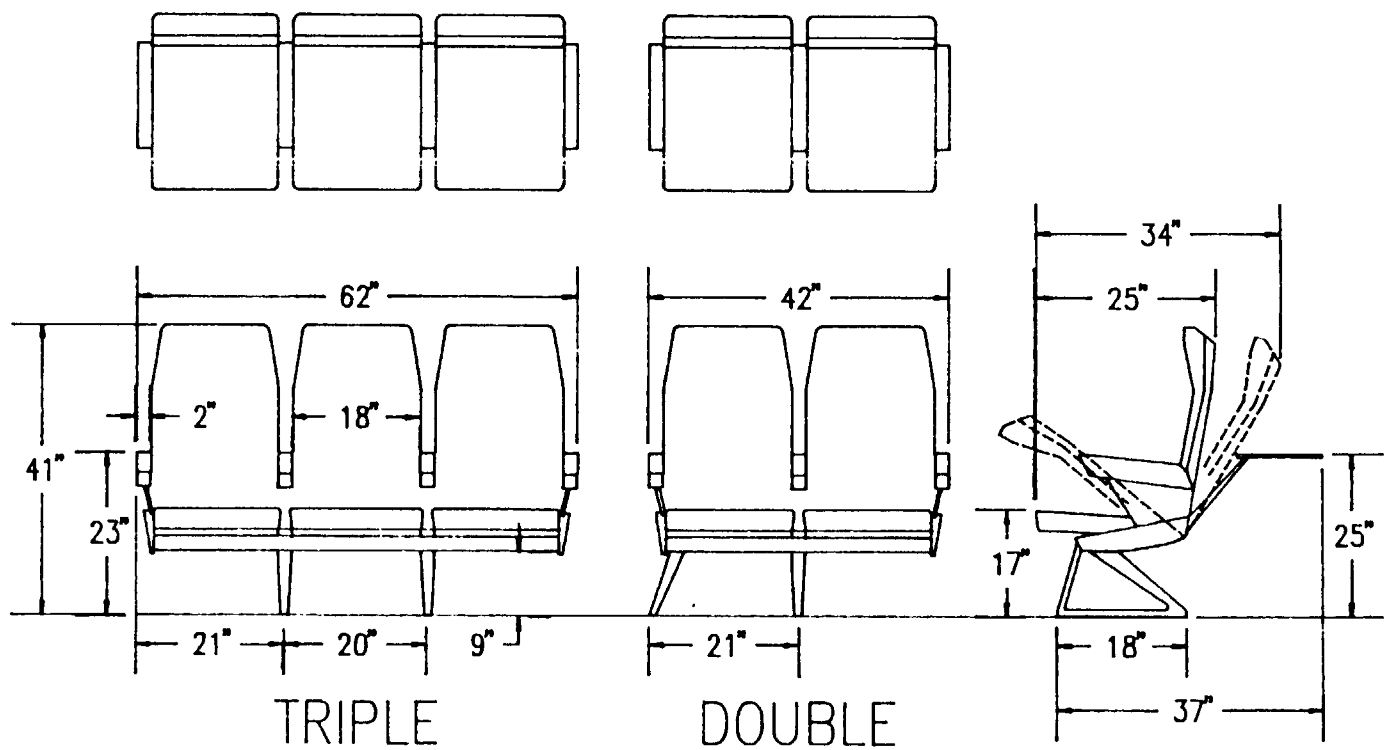
Further, problems associated with servicing, maintenance, and safety dictate where access must be designed into the fuselage. Design for these concerns usually conflicts directly with design for low structural weight, low complexity, and low drag.

Structurally, the most efficient fuselage cross section for a pressurized cabin is the circle. The Avion maintains a fuselage cross section similar to that of the BAC-111. The dimensions and motions of the human body, cargo hold considerations, and structural integrity governed the dimensions and layout of the fuselage cross section. The Avion payload specification called for a five-seat abreast fuselage. Using the seats (Figures 2.1 & 2.2) as the basic building blocks of the fuselage, and abiding with FAR 25 seating requirements, the passenger cabin width was set at a diameter of 128". Structural integrity required a minimum wall thickness of 5", resulting in an overall fuselage width of 138". Using trends from other aircraft (particularly the BAC-111, DC-9, and 727), the dimensions for the Avion fuselage cross section were determined. (See Figure 2.3.)

The Avion seating arrangement allowed for 10 first class passengers and 69 coach or tourist class passengers. Acknowledging industry practice for seat pitch and "creature comforts", the following seat pitches were established:

First Class: 40"

Coach Class: 36"



NOTES:

1. ALL SEATS MUST MEET REQUIREMENTS OF FAR 25, PARTS 561 & 785
2. SEAT PITCH: 36"
3. WEIGHTS: TRIPLE – 66 lb
   DOUBLE – 44 lb

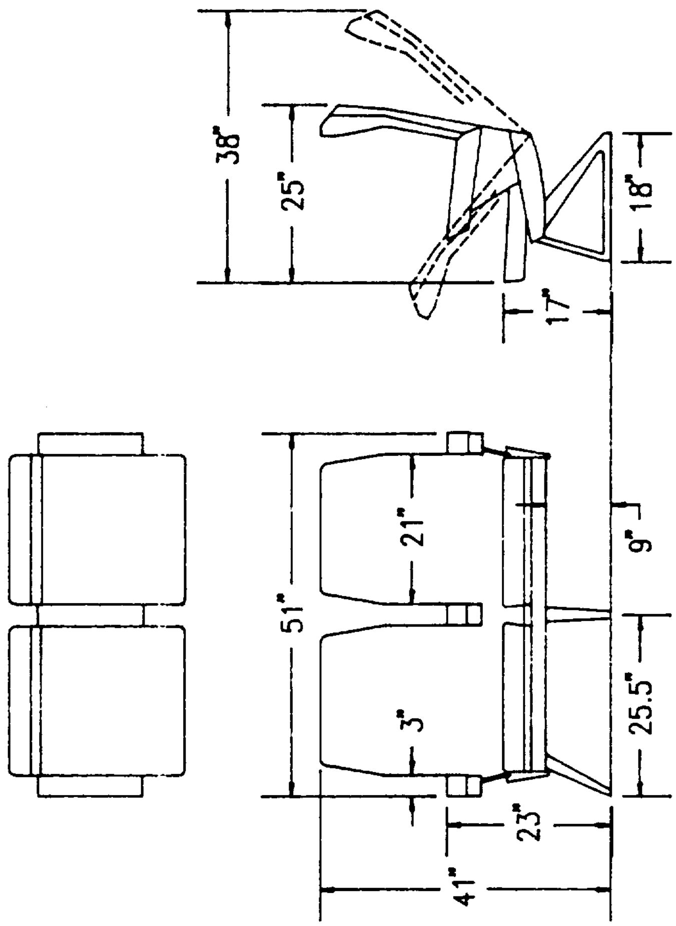
Figure 2.1 Avion Coach Class Seating

AVION DEVELOPMENT COMPANY
AUBURN, ALABAMA

AVION FIXTURES:
COACH SEATING

SIZE	FCSM NO	DWG NO		REV
A	N/A	AVION–XXX		
SCALE 1/2" = 1'			SHEET X OF X	

DRAWN BY: R.M. SCHUESSLER
DATE: 7 SEPTEMBER 1989



NOTES:
1. ALL SEATS MUST MEET REQUIREMENTS OF FAR 25, PARTS 561 & 785
2. SEAT PITCH: 40"
3. WEIGHTS: DOUBLE - 60 lb

Figure 2.2 Avion First Class Seating

AVION DEVELOPMENT COMPANY
AUBURN, ALABAMA

AVION FIXTURES:
FIRST CLASS SEATING

SIZE	FCSM NO	DWG NO	REV
A	N/A	AVION–XXX	
SCALE 1/2" = 1'		SHEET X OF X	

DRAWN BY: R.M. SCHUESSLER
DATE: 7 SEPTEMBER 1989

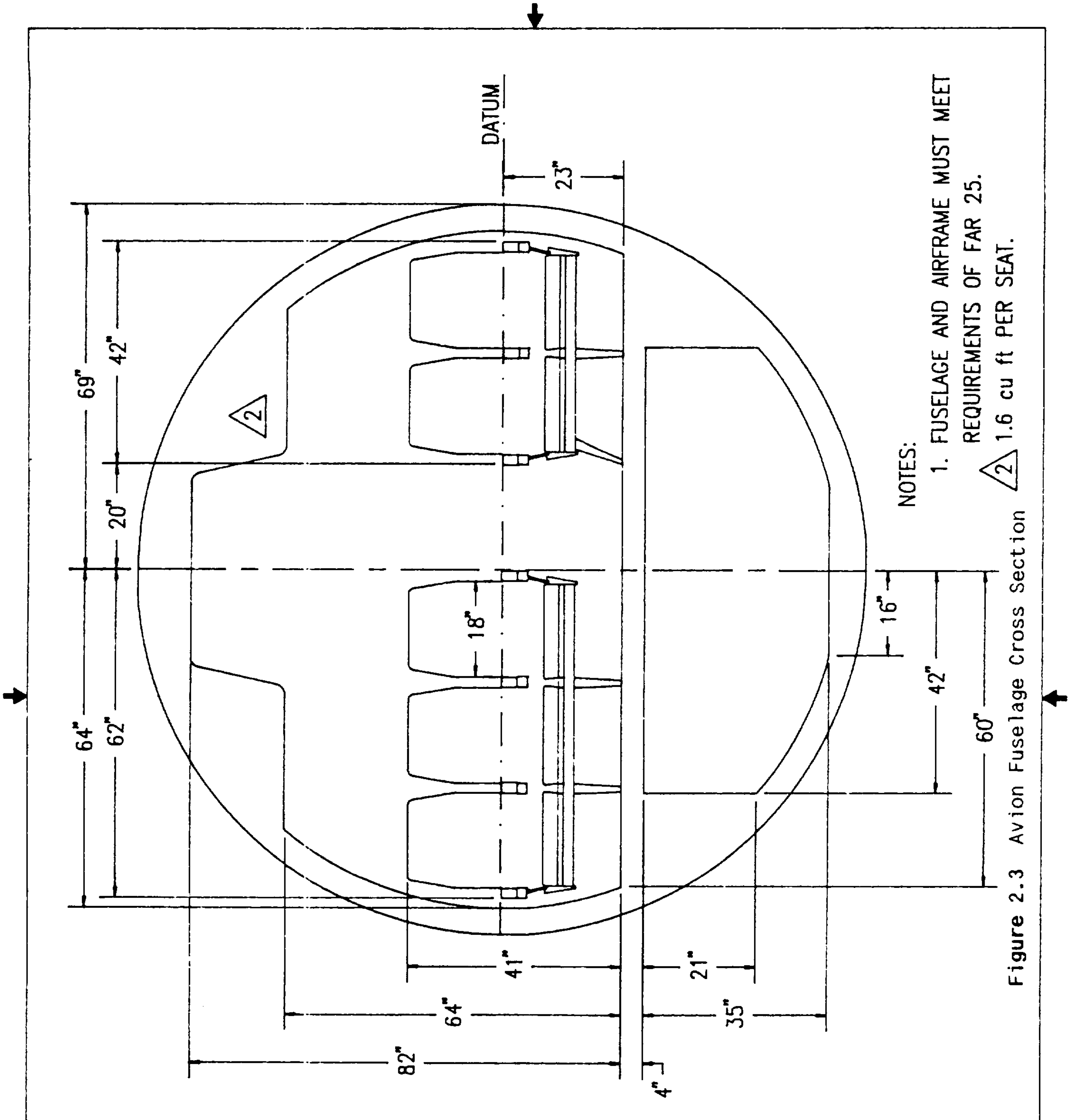


Figure 2.3 Avion Fuselage Cross Section

AVION DEVELOPMENT COMPANY
AUBURN, ALABAMA

AVION LAYOUT:
FUSELAGE CROSS SECTION

SIZE	FCSM NO	DWG NO	REV
A	N/A	AVION–XXX	

SCALE 1/2" = 1'		SHEET X OF X

DRAWN BY: R.M. SCHUESSLER
DATE: 7 SEPTEMBER 1989

Cabin attendant seating also had to be provided. Industry practice called for three cabin attendants for the Avion.

It is important to note that doors, exits, and windows are potential sources for leaks, noise, drag, and excessive weight. FAR's and passenger comfort govern the minimum number and size for doors, exits, and windows. A tradeoff was made between the requirements of safety, comfort, and economics. By FAR 25 Parts 807-813, the Avion needed three types of doors and exits:

(1) Passenger Access Doors (Port Side)

(2) Service Access Doors (Starboard Side)

(3) Emergency Exits

For the Avion, a 79-passenger aircraft, one Type I and one Type III exit had to be provided on each side of the fuselage. The following considerations also had to be made:

(1) FAR 25.807 requirement for a ventral and/or tailcone exit.

(2) Unobstructed access requirements:

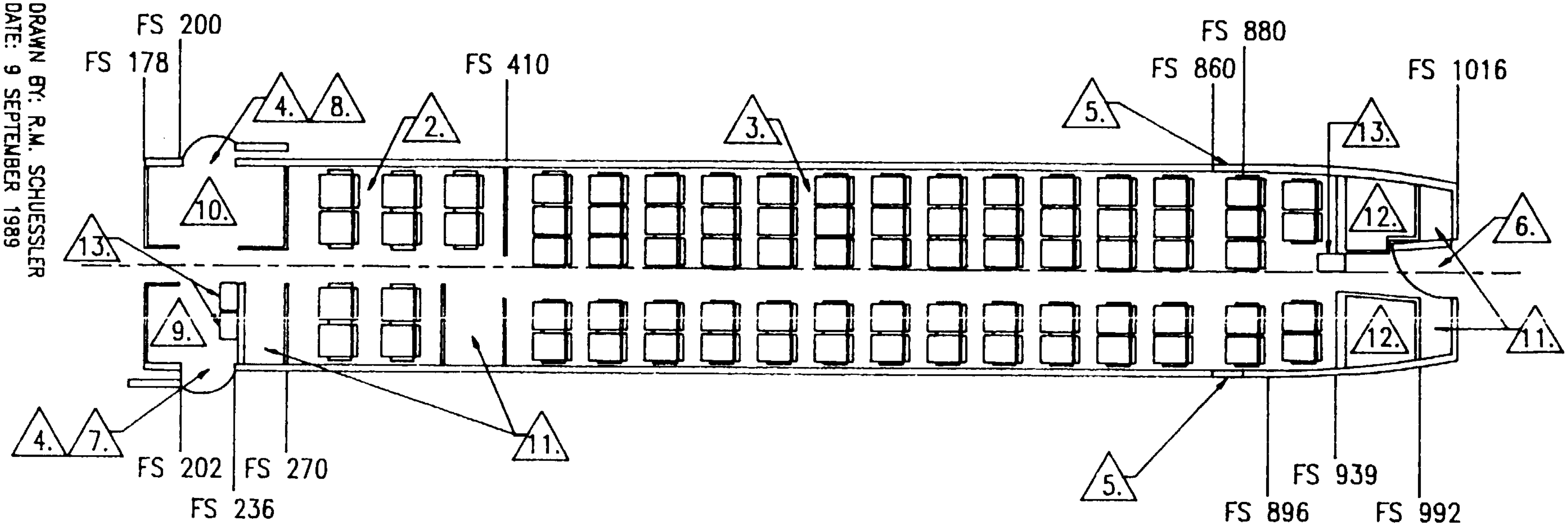
Type I Exit: 36" of access width

Type III Exit: 18" of access width (affects seat pitch)

(3) FAR 25.807 requirement for escape chutes (e.g., Boeing 767-200)

Windows were placed 24" apart. Galleys, lavatories, coat space, and stowage space were laid out in trend with other commercial transports.

Using the aforementioned considerations, the Avion fuselage/seating arrangement was determined. (See Figure 2.4.) The flight deck and aircraft nose length for the Avion was set at 178". Corresponding to Figure 2.4, the passenger cabin was determined to have an overall length of 838". The aft fuselage and tailcone length for the Avion was set at 352" for aerodynamic shaping. The Avion design employed a newly developed tailcone similar to that of the MD-80 which reduces cruise induced drag by 0.5%, translating directly



NOTES:

1. FUSELAGE MUST MEET REQUIREMENTS OF FAR 25, PARTS 807–813.
2. FIRST CLASS SEATING – 40" PITCH
3. COACH CLASS SEATING – 36" PITCH
4. TYPE I EMERGENCY EXIT – 36" UNOBSTRUCTED ACCESS
5. TYPE III EMERGENCY EXIT – 18" UNOBSTRUCTED ACCESS
6. TAILCONE EMERGENCY EXIT – 36" UNOBSTRUCTED ACCESS
7. FORWARD PASSENGER DOOR
8. GALLEY SERVICING DOOR
9. COAT SPACE
10. GALLEY
11. STOWAGE
12. LAVATORY
13. CABIN CREW FOLDING SEAT(S)

Figure 2.4 Avion Fuselage/Seating Layout

DRAWN BY: R.M. SCHUESSLER
DATE: 9 SEPTEMBER 1989

AVION DEVELOPMENT COMPANY
AUBURN, ALABAMA

AVION LAYOUT:
FUSELAGE/SEATING ARRANGEMENT

SIZE	FCSM NO	DWG NO	REV
A	N/A	AVION–XXX	
SCALE 1/8" = 1'		SHEET X OF X	

into long-term fuel savings. Special attention had to be paid to a 15° clearance requirement from the main landing gear to the aft fuselage accounting for aircraft rotation during take-off. This requirement was also critical for propfan blade ground clearance.

2.3 Integrated Aircraft Configuration

Figure 1.1 contains the embodiment of the preliminary configuration choices for the Avion.

3.0 Preliminary Weight Estimations

It is a difficult task to obtain an accurate aircraft weight estimate during any stage of the design process; it is even harder to perform during the preliminary stages of the design. This process was compounded by the complexity and unconventional design of the Avion. Aircraft designs must meet certain range, endurance, speed, and cruise requirements while carrying a given payload. It is crucial to obtain a reasonable prediction of the minimum aircraft weight and fuel weight needed for a given mission. Therefore, weight estimation was the most appropriate place to begin the design process for the Avion.

The Avion's mission specification is given in Table 1.1 and Figure 1.1. This section presents a preliminary design method used for estimating the following:

(1) Take-Off Gross Weight

(2) Empty Weight

(3) Mission Fuel Weight

3.1 General Method Outline

The gross take-off weight can be broken down as

$$W_{TO} = W_{OE} + W_F + W_{PL}$$

The operating empty weight is usually written

$$W_{OE} = W_E + W_{tfo} + W_{crew}$$

where W_{tfo} will be assumed 0.5% of W_{TO}.

The empty weight can be further broken down by

$$W_E = W_{ME} + W_{feq}$$

W_{feq} includes avionics equipment, air-conditioning equipment, auxiliary power unit (APU), furnishings and interiors, and other needed operation and mission equipment.

The preliminary sizing process consisted of seven steps:

Step 1. Determination of mission payload weight

Step 2. Guesstimation of take-off weight

Step 3. Determination of mission fuel weight

Step 4. Calculation of tentative operating empty weight by

$$W_{OEtent} = W_{TOguess} - W_F - W_{PL}$$

Step 5. Calculation of tentative empty weight by

$$W_{Etent} = W_{OEtent} - W_{tfo} - W_{crew}$$

Step 6. Determination of empty weight allowable

Step 7. Iteration to a tolerance of 0.5%.

3.2 Determination of Mission Payload Weight and Crew Weight

The mission payload weight was specified by the mission specification. For a passenger transport such as the Avion, this weight consists of passengers and baggage. For passengers in a commercial aircraft, an average weight of 175 lb and 40 lb baggage per person is the standard assumption for short to medium distance flights. Furthermore, the crew for a commercial transport consists of the cockpit crew and the cabin crew. For the Avion, these numbered 2 and 3, respectively.

3.3 Guesstimation of Gross Take-Off Weight

The initial guesstimation of the gross take-off weight is usually obtained by a comparison of the mission specification for the aircraft with the mission capabilities of similar aircraft. For the Avion, this comparative study as well as comparisons of other aircraft parameters can be found in Table 3.1.

Table 3.1 International Market Competition Aircraft (4)

Model Designation	Crew	Passen-gers	Max. Wing Span-ft	Max. Length ft	Max. Height ft	Empty Weight lb	Gross Weight lb	Speed mph
British Aerospace								
BAC 111-400	2	74-89	88.5	93.5	24.5	47,815	87,000	550
BAC 111-475	2	74-89	93.5	93.5	24.5	50,222	98,500	550
BAe 146-100	2	94	86.4	85.8	28.2	49,560	84,000	490
Kawasaki								
C-1	3	60	100.4	95.1	32.9	51,190	85,320	490
Fokker								
Fokker 50	2	50	95.2	82.8	27.8	27,886	45,900	M0.51
Fokker 100	2	100+	92.1	116.5	27.9	53,975	98,000	M0.77

3.4 Determination of Mission Fuel Weight

The mission fuel weight may be estimated from very basic considerations. This weight can be written as follows:

$$W_F = W_{Fused} + W_{Fres}$$

Fuel reserves are normally specified in the mission specification and the FAR's which regulate the operation of commercial passenger transports. For the Avion, the fuel reserves were specified in terms of additional loiter time and additional range so that an alternate airport can be reached.

Jan Roskam's "Fuel-Fraction Method" was used to calculate the fuel weights. In this method, the Avion mission was broken down into a number of phases. (See Figure 1.1.) The fuel used during each phase of flight was found from a simple calculation or estimated on the basis of experience. Each phase, therefore, has a begin weight and an end weight associated with it. The fuel fraction for each phase is defined as the ratio of the end weight to the begin weight. An examination of each mission phase follows:

Phase 1: Engine Start & Warmup
Denoted by W_1/W_{TO}
For commercial jet transports, the suggested value is 0.990

Phase 2: Taxi

Denoted by W_2/W_1

For commercial jet transports, the suggested value is 0.990

Phase 3: Takeoff

Denoted by W_3/W_2

For commercial jet transports, the suggested value is 0.995

Phase 4: Climb to Cruise Altitude and Accelerate to Cruise Speed

Denoted by W_4/W_3

For commercial jet transports, the suggested value is 0.980

Phase 5: Cruise

Denoted by W_5/W_4

This ratio was estimated for the Avion by *Breguet's Range Equation for Jet Aircraft:*

$$R_{cr} = (V/c_j)_{cr}(L/D)_{cr}\ln(W_4/W_5)$$

Based upon suggested values and Avion design considerations, the following values were modestly estimated for use in this equation:

R_{cr} = 1,500 nm (from mission specification)

V_{cr} = 455 kts (M=0.78) @ 32,000 ft

c_j = 0.4 lb/lb/hr

L/D = 16

Phase 6: Loiter

Denoted by W_6/W_5

This ratio was estimated for the Avion by *Breguet's Endurance Equation for Jet Aircraft:*

$$E_{ltr} = (1/c_j)_{ltr}(L/D)_{ltr}\ln(W_5/W_6)$$

Based upon suggested values and Avion design considerations, the following values were modestly estimated for use in this equation:

E_{ltr} = 1 hr

c_j = 0.32 lb/lb/hr

L/D = 19

Phase 7: Descent

Denoted by W_7/W_6

For commercial jet transports, the suggested value is 0.990

Phase 8: Fly to Alternate and Descend
Denoted by W_8/W_7

This ratio was estimated for the Avion by *Breguet's Endurance Equation for Jet Aircraft.* Because of the short distance to fly, an economical cruise altitude would normally not be attainable. It was assumed that for this phase of flight, the following values would be used:

R_{alt} = 250 nm (from mission specification)

V_{alt} = 250 kts max. @≤10,000 ft (FAA Regulations)

c_j = 0.8 lb/lb/hr

L/D = 11

Phase 9: Landing, Taxi, & Shutdown
Denoted by W_9/W_8
For commercial jet transports, the suggested value is 0.992

(3:12,14)

The mission fuel fraction (including fuel used and reserve fuel) was then calculated from the following:

$$M_{ff} = (W_1/W_{TO})\prod_{i=1}^{i=8}(W_{i+1}/W_i)$$

The fuel weight was then found from:

$$W_F = (1 - M_{ff})W_{TO}$$

3.5 Determination of Empty Weight Allowable

It is important to note that a linear relationship exists between $\log_{10}W_E$ and $\log_{10}W_{TO}$. Figure 3.1 demonstrates that this relationship does indeed exist. For a given value of W_{TO}, the allowable value for W_E can be found from the following regression line equation:

$$W_E = \log_{10}^{-1}[\log_{10}W_{TO} - A)/B]$$

For transport jets, the following regression line constants are used:

A = 0.0833

B = 1.0383 (3:47)

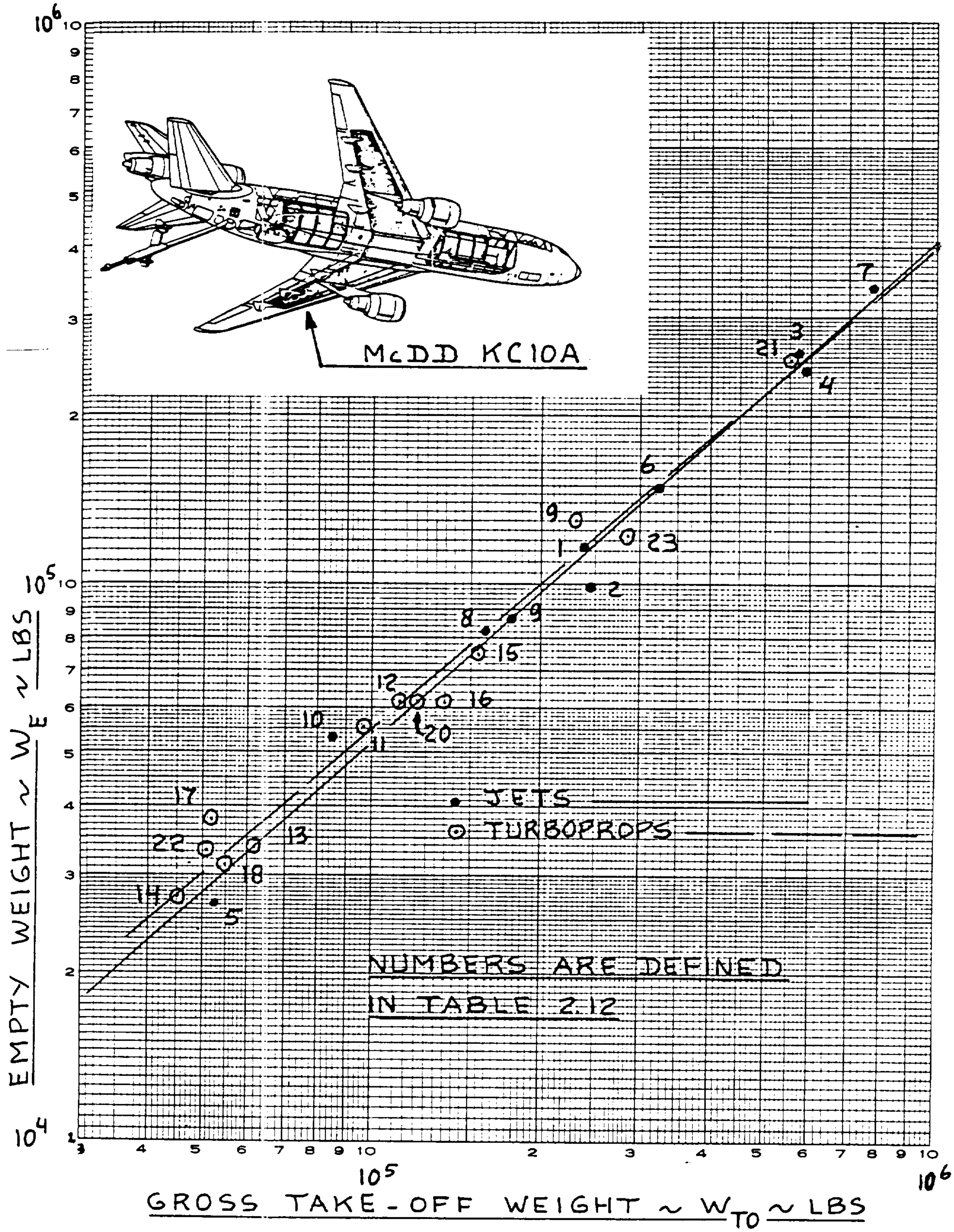


Figure 3.1 Weight Trends for Military Patrol, Bomb, and Transport Aircraft (3:28)

IMPORTANT: The primary structures of most of the airplanes listed in Figure 3.1 are manufactured mainly of metallic materials. The Avion design will incorporate many lighter and stronger composite materials in its design, however, these benefits are offset by the heavier structures of a tri-wing configuration and forward-swept wings.

A spreadsheet iteration analysis using the method outlined in this section produced the results in Table 3.2.

Table 3.2 Spreadsheet Weight Iteration Analysis

W_{PL} = 16,985 lb
W_{crew} = 1,075 lb

Iterated Guesstimation
W_{TO} = 77,000 lb

Cruise

R_{cr}	=	1,500 nm
V_{cr}	=	455 kts
c_j	=	0.40 lb/lb/hr
L/D	=	16.0

Loiter

E_{ltr}	=	1.00 hr
c_j	=	0.32 lb/lb/hr
L/D	=	19.0

Flight to Alternate

R_{alt}	=	250 nm
V_{alt}	=	250 kts
c_j	=	0.80 lb/lb/hr
L/D	=	11.0

Fuel Fraction Method

	Fuel Phase	Fraction
1	Engine Start & Warmup	0.990
2	Taxi	0.990
3	Takeoff	0.995
4	Climb to Cruise	0.980
5	Cruise	0.921
6	Loiter	0.983
7	Descent	0.990
8	Flight to Alternate	0.930
9	Landing, Taxi, Shutdown	0.992
	Mission Fuel Fraction ==>	0.790

W_F = 16,150 lb
$W_{OE:tent}$ = 43,865 lb

$W_{E:tent}$ = 42,405 lb
W_E = 42,268 lb

%DIF = 0.32%

4.0 Performance Design Parameter Estimations

While meeting stringent range, endurance, and cruise speed objectives, the Avion design must meet performance objectives in the following categories:

(A) Stall Speed

(B) Take-Off Field Length

(C) Landing Field Length

(D) Cruise Speed

(E) Climb Rate: AEO - All Engines Operating
OEI - One Engine Inoperative

(F) Time to Climb to Some Altitude

(G) Maneuvering

This section examines and estimates the parameters which have a major impact on these Avion performance categories. These design parameters are

(1) Wing Area

(2) Take-Off Thrust

(3) Maximum Required Lift Coefficient: Clean, Take-Off, & Landing

The calculation methods that will be presented resulted in the determination of a range of values for wing loading, thrust loading, and maximum lift coefficient within which the performance requirements were met. From experience, aircraft which have the highest wing loading and lowest thrust loading while meeting performance requirements result in lower weight and lower cost.

4.1 Sizing to Stall Speed Requirements

It should first be noted that FAR 25 certified aircraft have no requirements for minimum stall speed, but the stall speed must still be known. The power-off stall speed may be calculated from the following equation:

$$V_s = [2(W/S)/\rho C_{L_{max}}]^{\frac{1}{2}}$$

The following maximum lift coefficient values are typical for transport jets:

$$C_{L_{max}} = 1.2 - 1.8$$

$$C_{L_{maxTO}} = 1.6 - 2.2$$

$$C_{L_{maxL}} = 1.8 - 2.8 \qquad (3:91)$$

These values are based on 1984 flap design practice. Considerably higher values may be obtained with more sophisticated flap designs. Maximum lift coefficient values are strongly influenced by the wing and airfoil design, flap type and size, and center of gravity location.

4.2 Sizing to Take-Off Distance Requirements

Take-off distances are affected by the following factors:

(1) Take-Off Weight

(2) Take-Off Speed

(3) Thrust-to-Weight at Take-Off

(4) Aerodynamic Drag and Ground Friction

(5) Pilot Technique

Figure 4.1 illustrates the important take-off quantities of FAR 25.

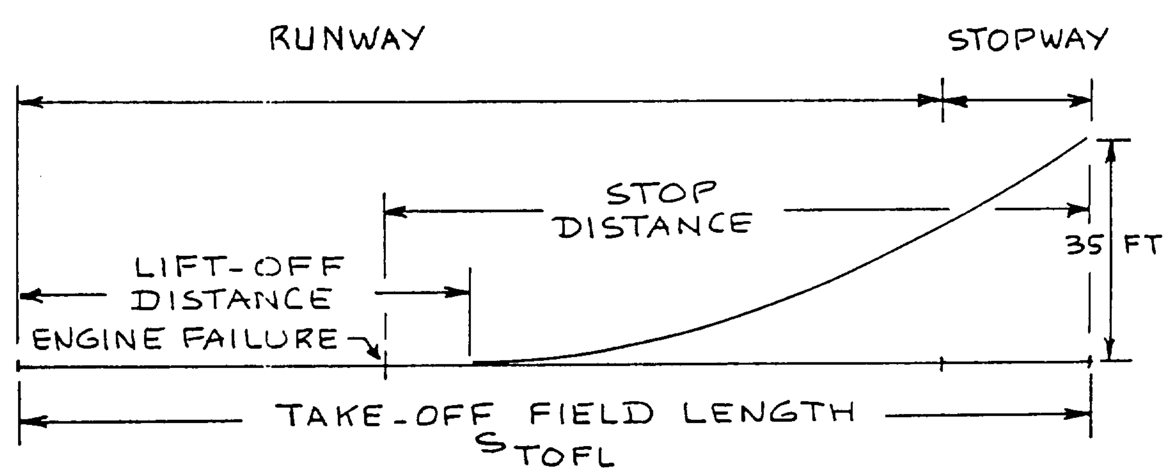


Figure 4.1 Definition of FAR 25 Take-Off Distances (3:99)

Based upon Figu-e 4.2, the following relationship can be obtained:

$s_{TOFL} = 37.5(W/S)_{TO}/(\sigma C_{LmaxTO}(T/W)_{TO}) = 37.5\ TOP_{25}$

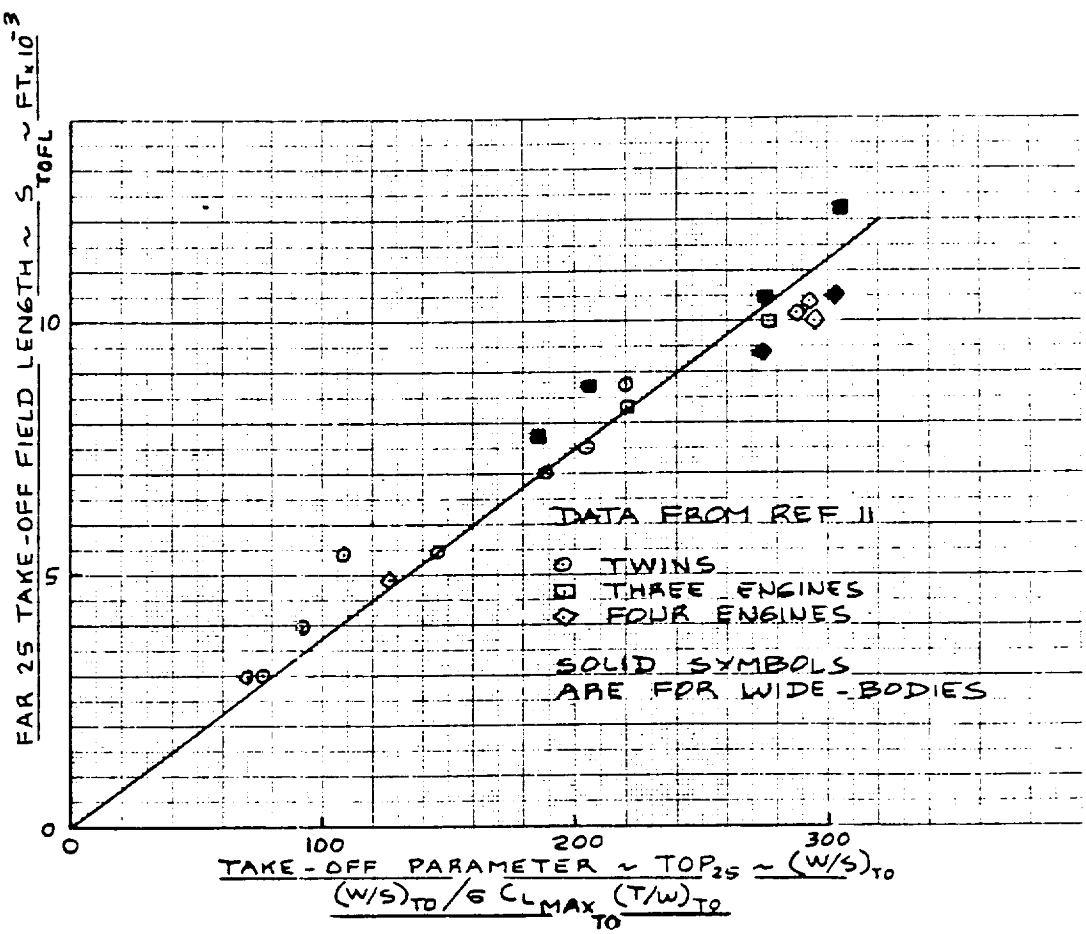


Figure 4.2 Effect of Take-Off Parameter on FAR 25 Take-Off Field Length (3:99)

It was required that the Avion be sized so that the FAR 25 take-off field length is given by

$s_{TOFL} < 6{,}500$ ft @ 8,000 ft, standard atmosphere

Therefore

$TOP_{25} = 6{,}500/37.5 = 173.3\ lb/ft^2$

At 8,000 ft, $\sigma = 0.786$. Therefore

$(W/S)_{TO}/(C_{LmaxTO}(T/W)_{TO}) = 173.3 \times 0.786 = 136.2\ lb/ft^2$

Figure 4.3 illustrates the values for which the field length requirement is met.

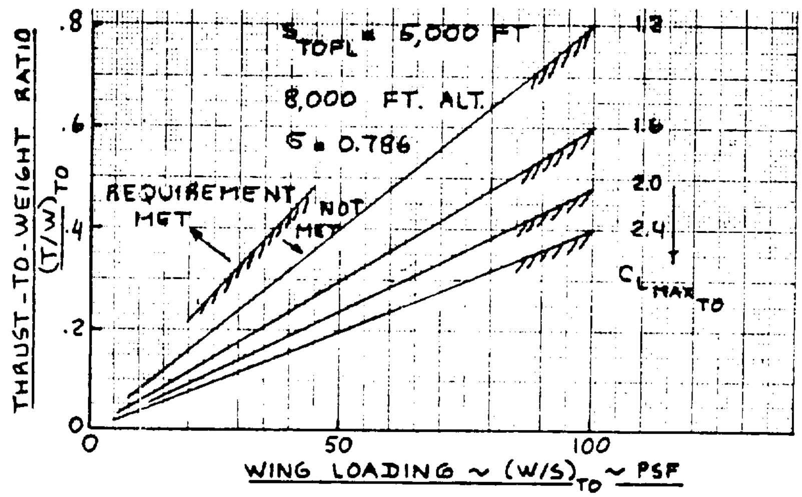


Figure 4.3 Effect of Take-Off Wing Loading and Maximum Take-Off Lift Coefficient on Take-Off Thrust-to-Weight Ratio (3:100)

Choosing $(W/S)_{TO}$ = 100 and C_{LmaxTO} = 2.4 for the Avion yielded $(T/W)_{TO}$ = 0.31.

4.3 Sizing to Landing Distance Requirements

Landing distances are affected by the following factors:

(1) Landing Weight

(2) Approach Speed

(3) Deceleration Method Used

(4) Aircraft Flying Qualities

(5) Pilot Technique

The typical values for landing weight to take-off weight ratio for transport jets are as follows:

W_L/W_{TO}	Minimum:	0.65
	Average:	0.84
	Maximum:	1.00

(3:107)

Figure 4.4 illustrates the important landing quantities of FAR 25.

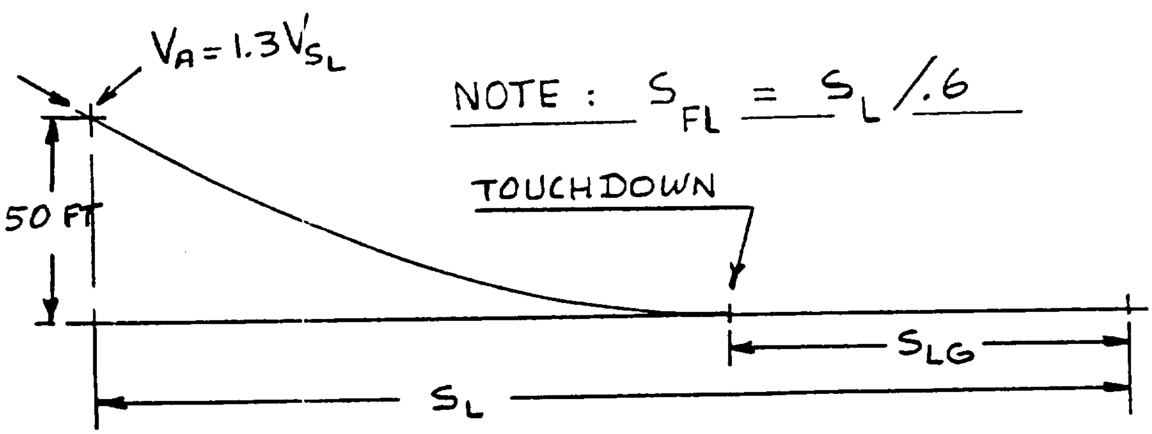


Figure 4.4 Definition of FAR 25 Landing Distances (3:112)

The FAR 25 field length is correlated with the approach speed of the aircraft, which is defined by

$V_A = 1.3\ V_{sL}$

Figure 4.5 illustrates how the FAR 25 field length is related to the approach speed through the following relationship

$s_{FL} = 0.3\ V_A^2$

It was required that the Avion be sized so that the FAR 25 field length is given by

$s_{FL} < 5{,}000$ ft @ sea level on a standard day

Therefore

$V_A = (5{,}000/0.3)^{1/2} = 129.1$ kts

$V_{sL} = 129.1/1.3 = 99.3$ kts

$2(W/S)_L/0.0023769\ C_{LmaxL} = (99.3 \times 6080/3600)^2 = 28{,}100\ ft^2/sec^2$

$(W/S)_L = 33.4\ C_{LmaxL}$

Assuming $W_L = 0.87 W_{TO}$

$(W/S)_{TO} = (33.4/0.87) C_{LmaxL} = 38.4\ C_{LmaxL}$

Choosing $(W/S)_{TO} = 100$ for the Avion yielded $C_{LmaxL} = 2.6$.

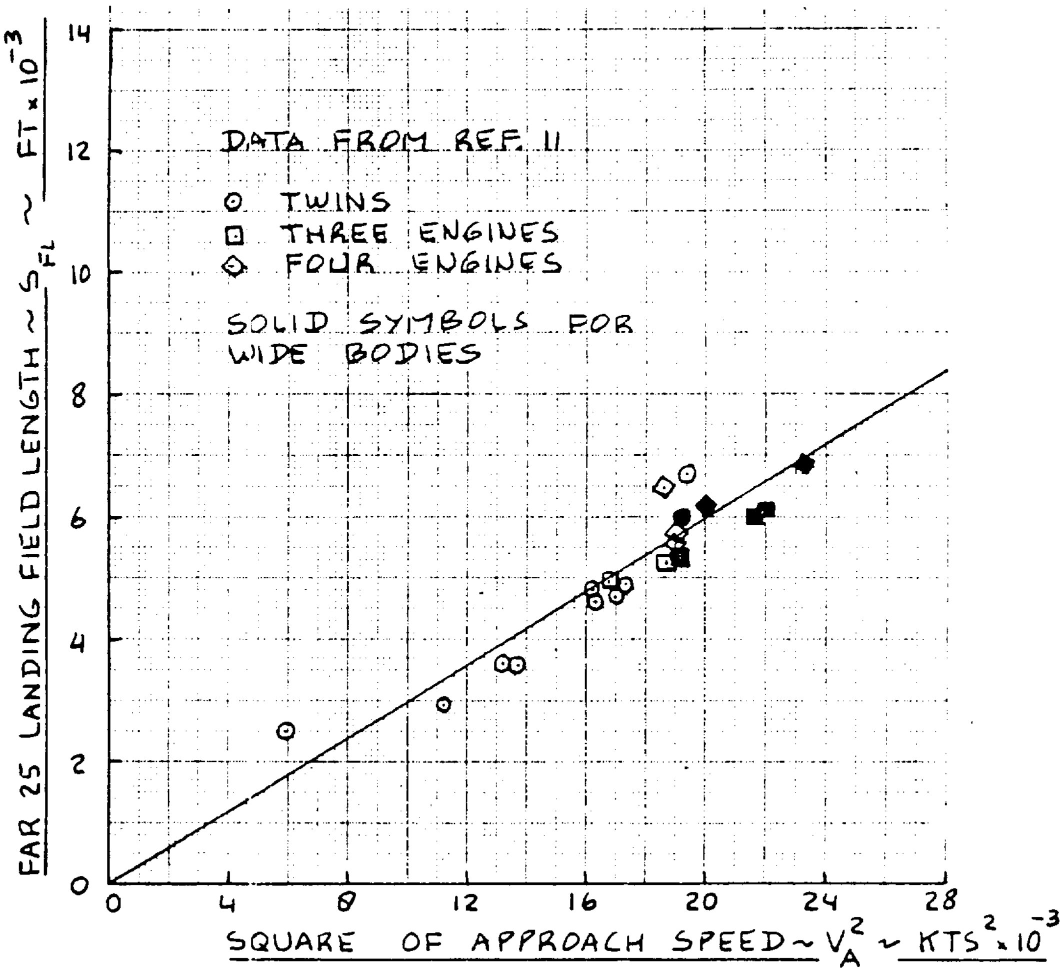


Figure 4.5 Effect of Square of Approach Speed on FAR 25 Field Length (3:112)

Figure 4.6 illustrates the values for which the field length requirement is met.

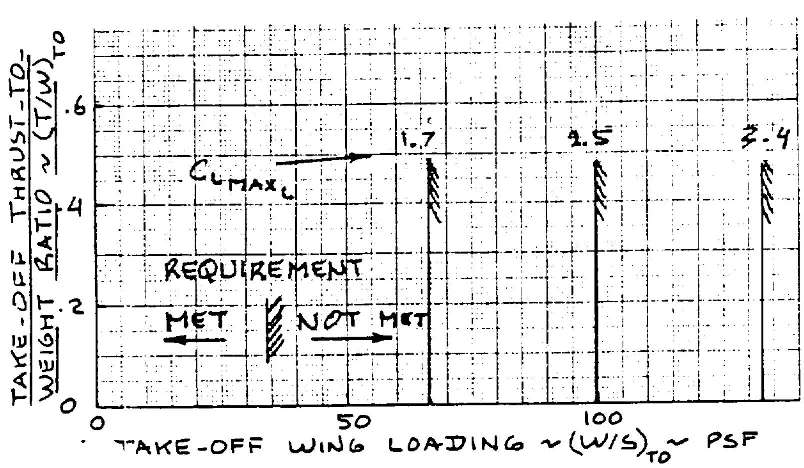


Figure 4.6 Allowable Wing Loading to Meet Field Length Requirement (3:114)

4.4 Drag Polar Estimations

The Avion must meet certain climb rate or climb gradient requirements; however, to size to these requirements, it was necessary to have an estimate for the drag polars.

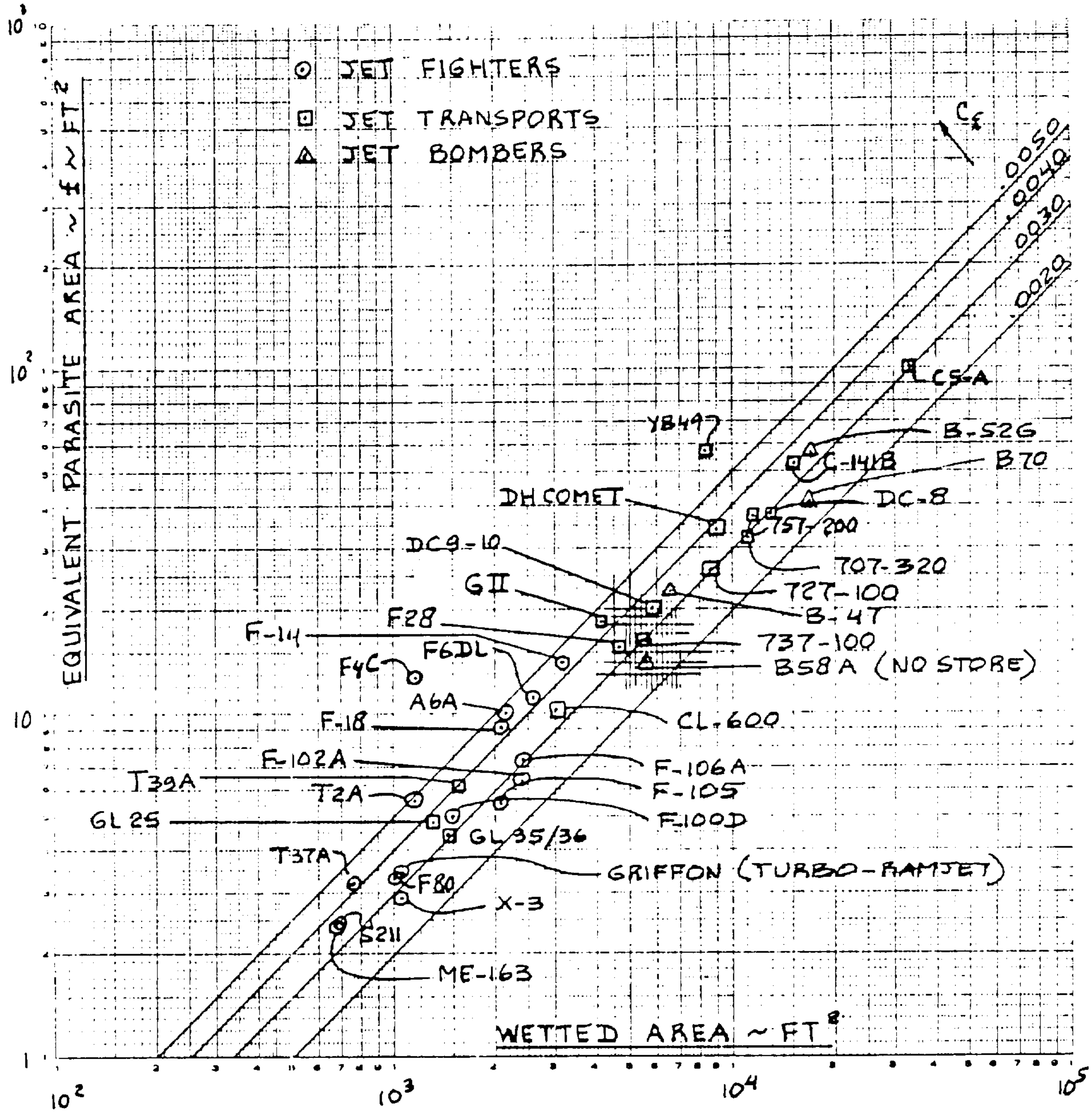


Figure 4.7 Effect of Equivalent Skin Friction and Wetted Area on Equivalent Parasite Area for Jet Aircraft (3:120)

Assuming a parabolic drag polar, the following relationship exists:

$$C_D = C_{Do} + C_L^2/\pi Ae$$

The zero-lift drag coefficient may be expressed by

$$C_{Do} = f/S$$

The relationship between equivalent parasite area and wetted area is illustrated in Figure 4.7. This linear relationship may be expressed by the following:

$$\log_{10} f = a + b \log_{10} S_{wet}$$

The correlation coefficients a and b are a function of the equivalent skin friction of the aircraft. This is determined by the smoothness and streamlining of the design.

Examination of Figure 4.7 resulted in a reasonable prediction of c_f = 0.0030. This yielded constants

$$a = -2.5229$$

$$b = 1.0000 \qquad (3:122)$$

Obviously, the method for estimating drag depends upon the ability to predict a realistic value for the wetted area. Fortunately, the wetted area correlates well with the take-off weight for transport jets. From Figure 4.8, an initial estimation of wetted area was made. The following relationship is implied:

$$\log_{10} S_{wet} = c + d \log_{10} W_{TO}$$

For transport jets, the regression line coefficients are given by

$$c = 0.0199$$

$$d = 0.7531 \qquad (3:122)$$

Using W_{TO} = 77,000 lb from the Avion preliminary sizing, the following calculations were made:

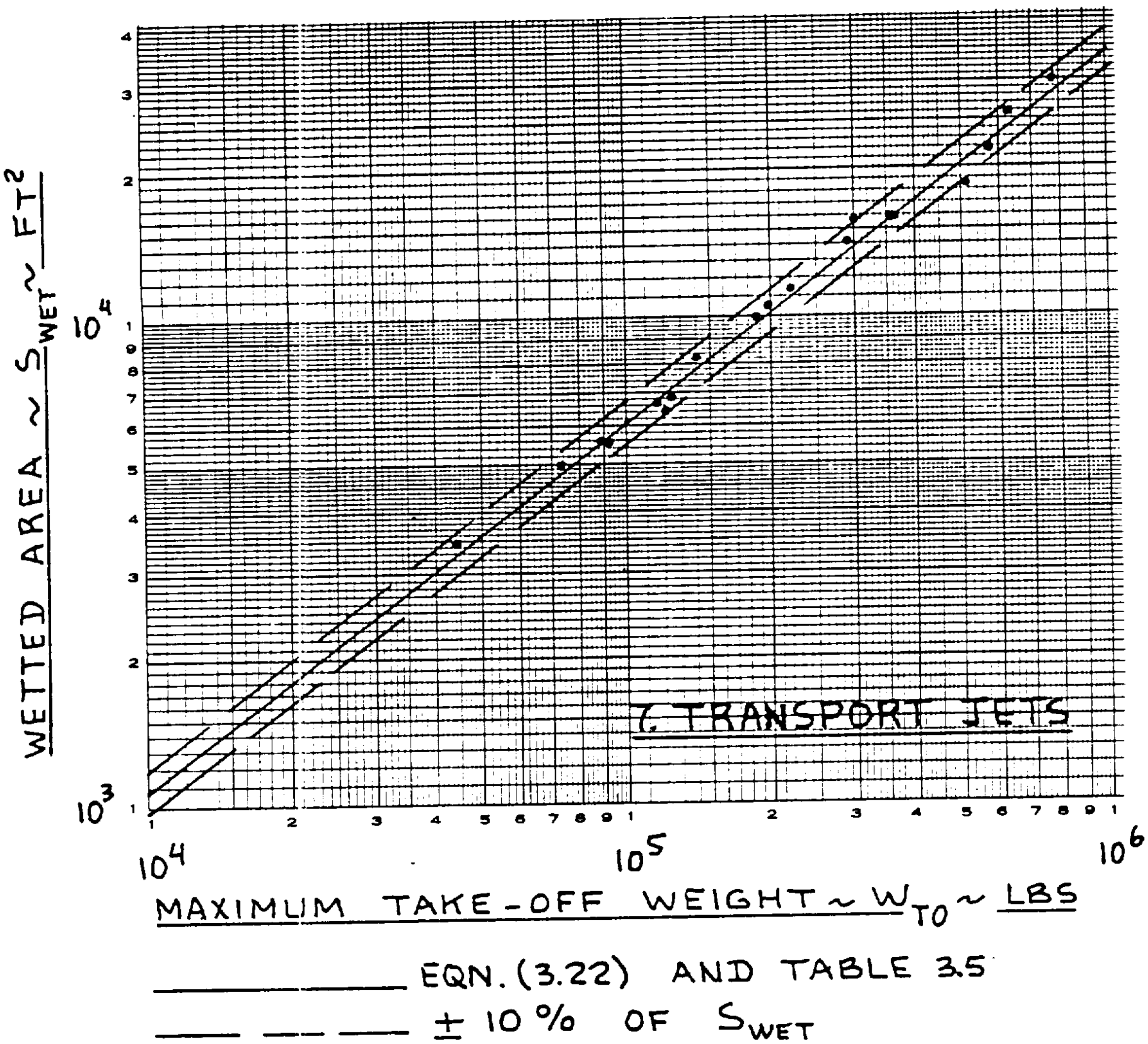


Figure 4.8 Correlation Between Wetted Area and Take-Off Weight for Transport Jets (3:124)

$$\log_{10}S_{wet} = 0.0199 + 0.7531 \log_{10}(77{,}000)$$

$$S_{wet} = 5{,}010 \text{ ft}^2$$

and

$$\log_{10}f = -2.5229 + 1.0000 \log_{10}(5010)$$

$$f = 15.0 \text{ ft}^2$$

Furthermore,

$$S = W_{TO}/(W/S)_{TO} = 77{,}000/100 = 770 \text{ ft}^2$$

$$C_{Do} = 15.0/770 = 0.0195$$

Flap and landing gear effects needed to be accounted for in the drag polars. The magnitudes of the added zero-lift coefficients due to these devices are dependent upon the size and type of these items. Typical values may be found in Table 4.1.

Table 4.1 First Estimates for ΔC_{Do} and e With Flaps and Gear Down (3:127)

Configuration	ΔC_{Do}	e
Clean	0	0.80 - 0.85
Take-Off Flaps	0.010 - 0.020	0.75 - 0.80
Landing Flaps	0.055 - 0.075	0.70 - 0.75
Landing Gear	0.015 - 0.025	N/A

Assuming A = 10.0 and e = 0.85, the clean drag polar for low speeds was predicted as

$$C_D = 0.0195 + C_L^2/(\pi \times 10.0 \times 0.85)$$

$$C_D = 0.0195 + 0.0374\ C_L^2$$

Considering the use of flaps and landing gear, the following values were estimated:

ΔC_{Do} due to:

Take-Off Flaps = 0.015 with e = 0.80
Landing Flaps = 0.065 with e = 0.75
Landing Gear = 0.017

The Avion drag polars are now summarized as follows:

Low Speed, Clean	$C_D = 0.0195 + 0.0374\ C_L^2$
Take-Off, Gear Up	$C_D = 0.0345 + 0.0398\ C_L^2$
Take-Off, Gear Down	$C_D = 0.0515 + 0.0398\ C_L^2$
Landing, Gear Up	$C_D = 0.0845 + 0.0424\ C_L^2$
Landing, Gear Down	$C_D = 0.1015 + 0.0424\ C_L^2$

4.5 Sizing to Climb Requirements

The FAR 25 climb requirements are given for two flight conditions: take-off and balked landing. These requirements must be met with the available thrust minus losses caused by accessory operations. For turbine powered aircraft, the engine thrust must be that for 34% humidity and 50°F above standard temperature.

The FAR 25 take-off climb and landing climb requirements as pertains to the Avion are summarized as follows:

For Take-Off Climb:

FAR 25.111 (OEI) CGR > 0.012
Configuration: gear up, take-off flaps, take-off thrust on remaining engines, ground effect, 1.2 V_{sTO}.

FAR 25.121 (OEI) CGR > 0
Configuration: gear down, take-off flaps, take-off thrust on remaining engines, ground effect, speed between V_{LOF} and 1.2 V_{sTO}.

FAR 25.121 (OEI) CGR > 0.024
Configuration: gear up, take-off flaps, no ground effect, take-off thrust on remaining engines, 1.2 V_{sTO}.

FAR 25.121 (OEI) CGR > 0.012
Configuration: gear up, flaps up, en route climb altitude, maximum continuous thrust on remaining engines, 1.25 V_s.

For Landing Climb:

FAR 25.119 (AEO) CGR > 0.032
Configuration: gear down, landing flaps, take-off thrust on all engines, maximum design landing weight, 1.3 V_{sL}.

FAR 25.121 (OEI) CGR > 0.021
Configuration: gear down, approach flaps, take-off thrust on remaining engines, 1.5 V_{sA}.

FAR 25.111 (OEI) (gear up, take-off flaps)

$(T/W)_{TO}$ = 2(1/(L/D) + 0.012), at 1.2 V_{sTO}.

Using $C_{L_{max}TO}$ = 2.4, the actual lift coefficient due to the stall speed factor was given by

$C_L = 2.4/(1.2)^2 = 1.67$

The following were calculated from the drag polar:

$C_D = 0.0345 + 0.0398(1.67)^2 = 0.1451$

$(L/D) = C_L/C_D = 1.67/0.1451 = 11.51$

$(T/W)_{TO} = 2(\ 1/(11.51) + 0.012) = 0.1977$

This, however, had to be corrected for the +50°F temperature effects on turbofan engines. The ratio of maximum thrust at this temperature differential is 0.80. Therefore

$(T/W)_{TO} = 0.1977/0.80 = 0.25$

<u>FAR 25.121 (OEI)</u> (gear down, take-off flaps)

$(T/W)_{TO} = 2(\ 1/(L/D) + 0.0)$, between V_{LOF} and V_2.

Assuming $V_{LOF} = 1.1\ V_{sTO}$ and using $C_{LmaxTO} = 2.4$, the actual lift coefficient was given by

$C_L = 2.4/(1.1)^2 = 1.98$

The following were calculated from the drag polar:

$C_D = 0.0515 + 0.0398(1.98)^2 = 0.2081$

$(L/D) = C_L/C_D = 1.98/0.2081 = 9.53$

$(T/W)_{TO} = 2(\ 1/(9.53) + 0.0) = 0.2098$

Corrected for the +50°F temperature differential:

$(T/W)_{TO} = 0.2098/0.80 = 0.26$

For $V_2 = 1.2\ V_{sTO}$, the actual lift coefficient was given by

$C_L = 2.4/(1.2)^2 = 1.67$

The following were calculated from the drag polar:

$C_D = 0.0515 + 0.0398(1.67)^2 = 0.1621$

$(L/D) = C_L/C_D = 1.67/0.1621 = 10.28$

$(T/W)_{TO}$ = 2(1/(10.28) + 0.0) = 0.1945

Corrected for the +50°F temperature differential:

$(T/W)_{TO}$ = 0.1945/0.80 = 0.24

<u>FAR 25.121 (OEI)</u> (gear up, take-off flaps)

$(T/W)_{TO}$ = 2(1/(L/D) + 0.024), at 1.2 V_{sTO}.

Using C_{LmaxTO} = 2.4, the actual lift coefficient was given by

$C_L = 2.4/(1.2)^2 = 1.67$

The following were calculated from the drag polar:

$C_D = 0.0345 + 0.0398(1.67)^2 = 0.1451$

$(L/D) = C_L/C_D = 1.67/0.1451 = 11.49$

$(T/W)_{TO}$ = 2(1/(11.49) + 0.024) = 0.2221

Corrected for the +50°F temperature differential:

$(T/W)_{TO}$ = 0.2221/0.80 = 0.28

<u>FAR 25.121 (OEI)</u> (gear up, clean)

$(T/W)_{TO}$ = 2(1/(L/D) + 0.012), at 1.25 V_s.

Using C_{Lmax} = 1.4 for the clean configuration, the actual lift coefficient was given by

$C_L = 1.4/(1.25)^2 = 0.90$

The following were calculated from the drag polar:

$C_D = 0.0195 + 0.0374(0.90)^2 = 0.0495$

$(L/D) = C_L/C_D = 0.90/0.0495 = 18.09$

$(T/W)_{TO}$ = 2(1/(18.09) + 0.012) = 0.1345

Corrected by 0.94 for maximum continuous thrust and by 0.80 for the +50°F temperature differential:

$(T/W)_{TO}$ = 0.1345/0.94/0.80 = 0.18

<u>FAR 25.119 (AEO)</u> (balked landing)

$(T/W)_L = 1/(L/D) + 0.032$, at 1.3 V_{sL}.

Using $C_{LmaxL} = 2.6$, the actual lift coefficient was given by

$C_L = 2.6/(1.3)^2 = 1.54$

The following were calculated from the drag polar:

$C_D = 0.1015 + 0.0424(1.54)^2 = 0.2019$

$(L/D) = C_L/C_D = 1.54/0.2019 = 7.62$

$(T/W)_L = 1/(7.62) + 0.032 = 0.1632$

Since the design landing weight is given by

$W_L = 0.92\ W_{TO} = 0.92(77{,}000) = 70{,}840$ lb

this translated into the following take-off requirement (including temperature effects):

$(T/W)_{TO} = 0.1632(70{,}840/77{,}000)/0.80 = 0.19$

<u>FAR 25.121 (OEI)</u> (balked landing)

$(T/W)_L = 2(\ 1/(L/D) + 0.021)$, at 1.5 V_{sA}.

Using $C_{LmaxA} = 2.5$ (halfway between C_{LmaxTO} and C_{LmaxL}), the actual lift coefficient was given by

$C_{LA} = 2.5/(1.5)^2 = 1.11$

The following were calculated from the drag polar:

$C_D = 0.0765 + 0.0424(1.11)^2 = 0.1288$

$(L/D) = C_L/C_D = 1.11/0.1288 = 8.62$

$(T/W)_L = 2(\ 1/(8.62) + 0.021) = 0.2739$

With weight and temperature effects:

$(T/W)_{TO} = 0.2739(70{,}840/77{,}000)/0.80 = 0.32$

This last requirement was the most critical for the Avion design.

5.0 Wing Planform Design

This section serves to provide a preliminary wing planform design for the Avion. The following planform design characteristics were determined:

(1) Wing Area

(2) Aspect Ratio

(3) Sweep Angle

(4) Thickness Ratio

(5) Airfoil

(6) Taper Ratio

(7) Incidence Angle

(8) Dihedral Angle

(9) Lateral Control Surfaces

5.1 General Design

Table 5.1 contains the wing geometries for several jet transport aircraft. As previously stated, the overall configuration of the Avion is the tri-wing configuration. The Avion utilizes a cantilever wing since braced (or strutted) wings are generally only used on low speed aircraft. Above 200 kts, the profile and interference drag increment dominates the wing weight advantages of the strutted arrangement. (5:142)

The Avion wing is mounted in a low position on the fuselage for structural advantages. Most jet transports utilize a low wing design.

Because of the Avion's high-speed, subsonic cruise requirement, several decisions needed to be made about sweep angle and thickness ratio. The Avion utilizes forward (or negative) sweeping since forward swept wings have significant stall characteristic advantages over aft swept wings. This is due to the fact that lateral control surfaces mounted on the outboard stations of

Table 5.1 Jet Transports: Wing Geometric Data (5:146)

Type	Dihedral Angle, Γ_w, deg.	Incidence Angle, i_w, root/tip deg.	Aspect Ratio, A	Sweep Angle, $\Lambda_{c/4}$, deg.	Taper Ratio, λ_w	Max. Speed, V_{max}, kts	Wing Type
BOEING							
727-200	3	2	7.1	32	0.30	549(22K)	ctl/low
737-200	6	1	8.8	25	0.34	462(33K)	ctl/low
737-300	6	1	8.0	25	0.28	462(33K)	ctl/low
747-200B	7	2	7.0	37.5	0.25	523(30K)	ctl/low
747SP	7	2	7.0	37.5	0.25	529(30K)	ctl/low
757-200	5	3.2	7.9	25	0.26		ctl/low
767-200	6	4.3	7.9	31.5	0.27		ctl/low
McDONNELL DOUGLAS							
DC-9 Super 80	3	1.3	9.6	24.5	0.16	500	ctl/low
DC-9-50	1.5	NA	8.7	24	0.18	537	ctl/low
DC-10-30	5.3/3	+/-	7.5	35	0.25	530(25K)	ctl/low
AIRBUS							
A300-B4	5	NA	7.7	28	0.35	492(25K)	ctl/low
A310	11.1/4.1	5.3	8.8	28	0.26	483(30K)	ctl/low
Lockh.1011-500	7.5/5.5	NA	7.0	35	0.30	525(30K)	ctl/low
Fkr F28-4000	2.5	NA	8.0	16	0.31	390	ctl/low
Rombac 111-495	2	2.5	8.5	20	0.32	470(21K)	ctl/low
BAe 146-200	-3	3.1/0	9.0	15	0.36	420(26K)	ctl/high
Tupolev Tu154	0	NA	7.0	35	0.27	526(31K)	ctl/low

ctl = cantilever (30K) = 30,000 ft altitude

the wing maintain their effectiveness well into the stall since the wing root stalls first. (See Figure 5.1.) Furthermore, Figure 5.2 illustrates that sweep angle has a very favorable effect on the compressibility drag.

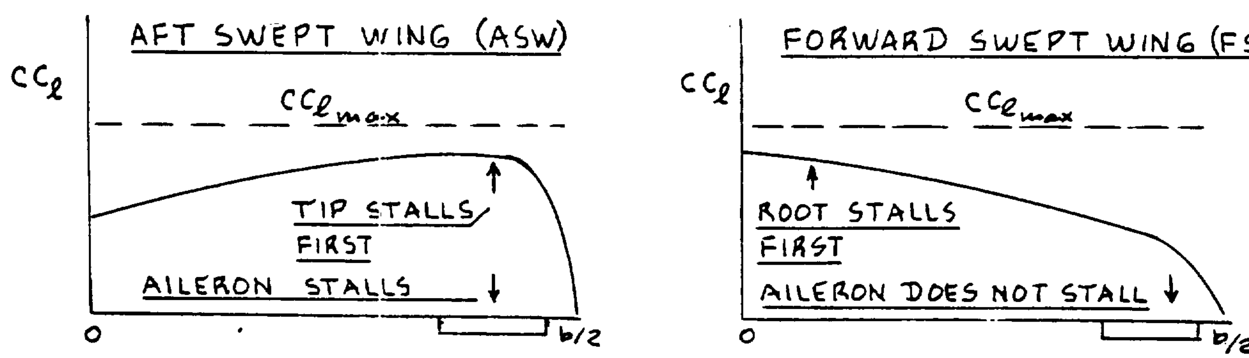


Figure 5.1 Effect of Sweep on Stall Behavior (2:173)

Forward swept wings, however, do possess several disadvantages. First there is a substantial weight penalty associated with forward swept wings (above that of aft swept wings). (See Figure 5.3.) As Jan Roskam notes:

> "The reason is the structural divergence phenomenon associated with forward sweep. By tailoring the ratio of bending to torsion stiffness it is possible to make the weight penalty associated with forward swept wings quite acceptable. Such tailoring is inherently possible with composite structures." (2:175)

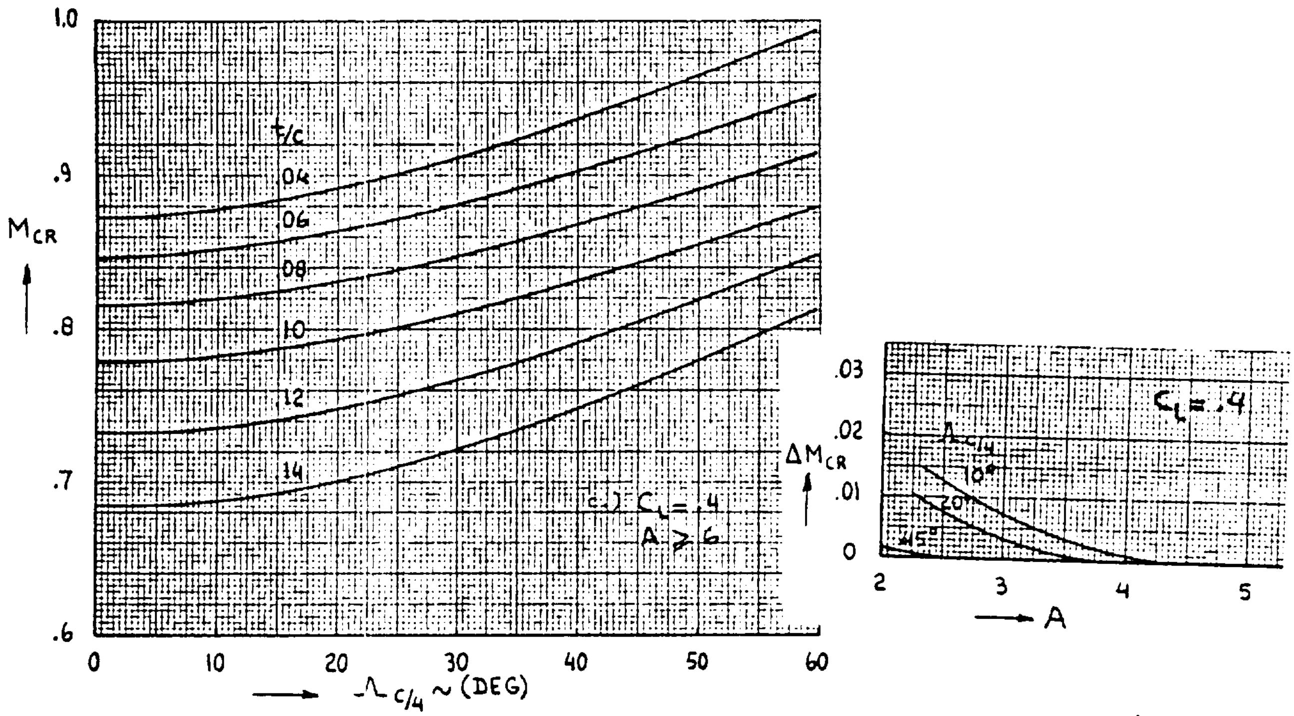


Figure 5.2 Effect of Thickness Ratio and Sweep Angle on Critical Mach Number (5:150-151)

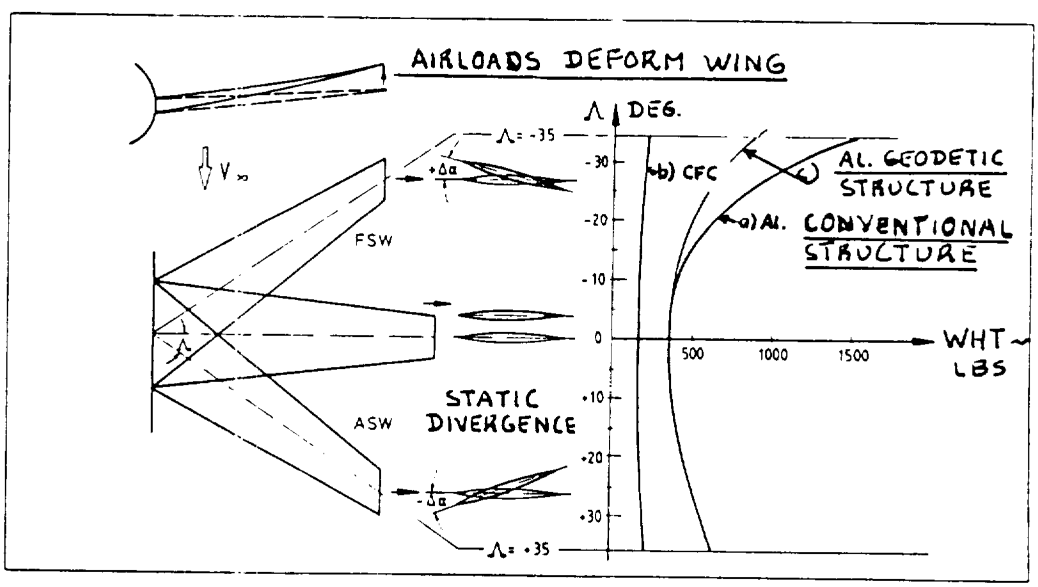


Figure 5.3 Effect of Sweep on Wing Weight (2:173)

A severe problem with forward sweeping is its effects on aircraft stability. Even unswept wing designs may encounter trouble with the c.g. location being too far aft. Usually aft sweeping corrects this problem since this has the effect of moving the aircraft a.c. aft faster than the aircraft c.g. The Avion avoids this anomaly through the following:

(1) A relatively high fuselage fineness ratio allowing the fuselage to be long enough to manipulate the c.g.

(2) Only modest forward sweeping of the wing.

(3) Far aft positioning of the wing.

(4) Tri-wing configuration effects on c.g. and a.c.

(5) Leading edge extension (LEX) fuel storage.

5.2 Design Parameters

Using the cruise Mach number of 0.78 at 32,000 ft, the cruise lift coefficient was estimated by

$$C_{Lcr} = (W_{TO} - 0.4\ W_F)/qS$$

$$C_{Lcr} = (77{,}000 - 0.4 \times 16{,}150)/(\tfrac{1}{2} \times (0.348 \times 0.0023769) \times (0.78 \times 0.883 \times 1116)^2 \times 770 = 0.38$$

Using Figure 5.2 and a quarter chord sweep of -20°, a thickness ratio of 0.12 was chosen. Based on this information, the airfoil selected for the Avion was a supercritical derivative of the NACA 64_A412 airfoil. From Table 5.1, the taper ratio was selected as 0.30 and the wing dihedral angle as 3° for the Avion. From other aircraft estimations, the wing incidence angle was selected as 2°.

Assuming an aspect ratio of 10 to minimize induced drag, and a wing area of 770 ft², the wingspan was calculated:

$$b = (A\ S)^{\frac{1}{2}} = (10 \times 770)^{\frac{1}{2}}\ /\ 1{,}050 \text{ in}$$

The characteristics of each wing were then determined:

$c_{av} = (S/2)/l = (770 \times 12^2) / 1050 = 106$ in

$c_r = 106 / 0.65 = 163$ in

$c_t = 0.30 \times 163 = 49$ in

5.3 High Lift Devices and Lateral Control Surfaces

Now that the initial choices have been made for the wing design parameters, it must be verified that the chosen wing planform can provide a C_{LmaxW} consistent with the C_{Lmax} clean of 1.4.

$C_{LmaxW} = 1.1\ C_{Lmax} = 1.1 \times 1.4 = 1.54$

where the factor 1.1 accounts for the tail and canard interference on the wing.

The lift coefficient was corrected for sweep by

$C_{LmaxWus} = C_{LmaxW}/\cos\Lambda_{c/4} = 1.54 / \cos(-20^\circ) = 1.64$

The equation

$C_{LmaxW} = K_\lambda\ (C_{Lmaxr} + C_{Lmaxt})/2$

where $K_\lambda = 0.95$ must be used to verify that the wing can produce the required C_{LmaxW}.

Figure 5.4 was used to obtain the section C_{Lmax} at the root and at the tip. The Reynolds numbers for these sections were found first:

$R_{Nr} = \rho V_{TO} c_r/\mu = 0.0023769 \times 225 \times 163/12 / 3.737\text{x}10^{-7} = 19.5\text{x}10^6$

$R_{Nt} = 0.3\ R_{Nr} = 0.3 \times 19.5\text{x}10^6 = 5.8\text{x}10^6$

Using Figure 5.4, the section C_{Lmax} were found to be

$C_{Lmaxr} = 1.9$

$C_{Lmaxt} = 1.6$

Calculating C_{Lmax} for the unswept wing:

$C_{Lmaxus} = K_\lambda\ (C_{Lmaxr} + C_{Lmaxt})/2 = 0.95 \times (1.9 + 1.6)/2 = 1.66$

Correcting for sweep and interference:

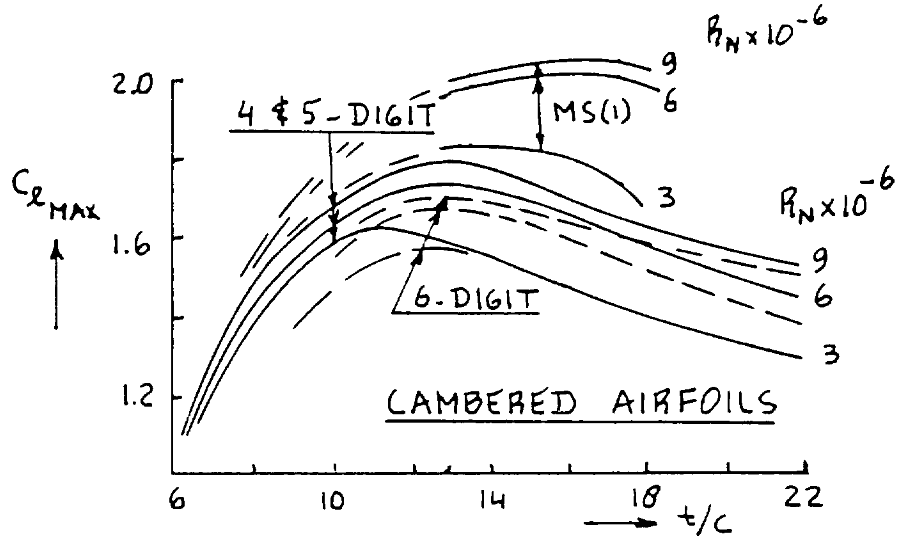


Figure 5.4 Effect of Thickness Ratio and Reynolds Number on Section Maximum Lift Coefficient (5:169)

$C_{LmaxW} = C_{LmaxWus} \cos\Lambda_{c/4} = 1.66 \cos(-20^\circ) = 1.56$

$C_{Lmax} = C_{LmaxW}/1.1 = 1.56/1.1 = 1.42$

This verified that the wing could produce the required value for C_{Lmax} of 1.4.

The incremental values of C_{Lmax} which need to be produced by any high lift devices that are utilized are calculated by

Take-off: $\Delta C_{LmaxTO} = 1.05\ (C_{LmaxTO} - C_{Lmax}) = 1.05\ (2.4 - 1.4) = 1.05$

Landing: $\Delta C_{LmaxL} = 1.05\ (C_{LmaxL} - C_{Lmax}) = 1.05\ (2.6 - 1.4) = 1.26$

where the factor 1.05 accounts for the additional trim penalties incurred by the use of flaps.

Using the above calculations and a study of high-lift devices used on jet transports, the Avion design employs Fowler flaps on the trailing edge and slats on the leading edge of the wing. These devices are used to obtain the highest C_{Lmax} as well as the highest lift-to-drag ratio at take-off.

Leading edge slats are used to provide camber and boundary layer energy improvement. Historically, leading edge slats are the most effective method of high-lift used on jet transports.

All large transport aircraft use slotted flaps. The Fowler flaps combine the benefits of slots with an increase in flap performance. Slotted flaps improve the energy of the upper surface boundary layer by bringing high energy air from the lower surface. Therefore, the Avion will depend on Fowler flaps to increase its lift performance.

Ailerons are used to raise the lift on one side of the wing and lower it on the other, resulting in a roll condition about the longitudinal axis. Ailerons will only be included on the outboard wing stations, as forward swept wings maintain lateral control characteristics deep into a stall based upon the location of these devices.

The use of spoilers disturbs the flow over the wing and reduces the lift to obtain the following conditions:

(1) To create drag and increase the rate of descent.

(2) To aid in the rolling process if operated on one side only.

(3) To get more load on the wheels during a braked ground run.

5.4 Conclusions

The exact value of the parameters critical to wing design can only be determined after a more complete and in-depth study of the aircraft design. This includes many calculations, model simulations, and wind tunnel testing.

6.0 Empennage Planform Design

The empennage is comprised of the horizontal and vertical tail surfaces. The process used to make decisions concerning the empennage is similar to that used for the wing. The Avion employs a horizontal and vertical tail in a T-tail arrangement. In this form, the horizontal surface acts like an end plate and increases the lift-curve slope of the vertical tail. The disadvantage of this arrangement is the imposition of some weight penalties. However, this can be alleviated by sweeping the vertical tail slightly aft. This increases the moment arm of the horizontal tail, and thus reduces the surface area and weight of this surface.

During preliminary sizing, approximations were used to obtain the empannage moment arms. By examing the detailed fuselage drawings of the Avion, values for x_h and x_v were decided upon:

$$x_h = 655 \text{ in} \qquad x_v = 480 \text{ in}$$

Surface volume coefficients of similar aircraft can be used during the sizing of the empennage. The horizonatal and vertical tail volume coefficient are defined as

$$V_h = x_h S_h/Sc_{av} \qquad V_v = x_v S_v/Sb$$

By comparing values for similar aircraft (see Tables 6.1 - 6.3), the values for the surface volume ccefficients for the Avion were chosen:

$$V_h = 1.15 \qquad V_v = 0.079$$

By rearranging the tail volume coefficient equations, the tail surface areas were calculated:

$$S_h = V_h Sc_{av}/x_h = (1.15)(770)(106/12)/(655/12) = 143.3 \text{ ft}^2$$

$$S_v = V_v Sb/x_v = (0.079)(770)(1050/12)/(480/12) = 133.1 \text{ ft}^2$$

Table 6.1 Jet Transports: Horizontal Tail Volume and Elevator Data (5:197)

Type	Wing Area S ft^2	Wing mgc $\bar{c}$ ft	Wing Airfoil root/tip	Hor. Tail Area S_h ft^2	S_e/S_h	x_h ft	$\bar{V}_h$	Elevator Chord root/tip $fr.c_h$
BOEING								
727-200	1,700	18.0	BAC	376	0.25	67.0	0.82	.29/.31
737-200	980	11.2	BAC	321	0.27	43.8	1.28	.30/.32
737-300	1,117	10.9	BAC	330	0.24	49.7	1.35	.24/.34
747-200B	5,500	38.0	BAC	1,470	0.24	104.5	0.74	0.29
747SP	5,500	38.0	BAC	1,534	0.21	72.9	0.54	.32/.20
757-200	1,951	14.9	BAC	585	0.25	56.9	1.15	.29/.38
767-200	3,050	19.8	BAC	836	0.23	67.6	0.94	.30/.25
McDONNELL-DOUGLAS								
DC-9 S80	1,270	15.7	N.A.	314	0.34	61.4	0.96	.39/.38
DC-9-50	1,001	11.8	N.A.	276	0.38	56.8	1.32	.41/.47
DC-10-30	3,958	24.7	N.A.	1,338	0.22	65.9	0.90	.25/.30
AIRBUS								
A300-B4	2,799	19.2	N.A.	748	0.26	80.4	1.12	0.35
A310	2,357	19.3	N.A.	689	0.26	72.0	1.09	.33/.30
Lockheed L1011					geared elevator			
-500	3,541	24.5	N.A.	1,282	0.19	55.9	0.83	stabilator
Fokker F-28								
-4000	850	10.9	N.A.	210	0.20	47.2	1.07	.34/.33
Rombac/British Aerospace								
1-11 495	1,031	11.8	N.A.	258	0.27	40.7	0.86	.41/.35
British Aerospace								
146-200	832	10.2	N.A.	276	0.39	45.3	1.48	.42/.44
Tu-154	2,169	16.8	N.A.	436	0.18	58.9	0.71	.27/.25

Table 6.2 Jet Transports: Vertical Tail Volume and Rudder Data (5:197)

Type	Wing Area S ft^2	Wing Span b ft	Vert. Tail Area S_v ft^2	S_r/S_v	x_v ft	$\bar{V}_v$	Rudder Chord root/tip $fr.c_v$	S_a/S	Inb'd Ail. Span in/out $fr.b/2$	Inb'd Ail. Chord in/out $fr.c_w$
BOEING										
727-200	1,700	108	422	0.16	47.4	0.110	.29/.28	0.034	.38/.46	.17/.24
737-200	980	93.0	233	0.24	40.7	0.100	.25/.22	0.024	none	none
737-300	1,117	94.8	239	0.31	45.7	0.100	.26/.50	0.021	none	none
747-200B	5,500	196	830	0.30	102	0.079	0.30	0.040	.38/.44	.17/.25
747-SP	5,500	196	885	0.27	69.5	0.057	.31/.34	0.040	.38/.44	.17/.25
757-200	1,951	125	384	0.34	54.2	0.086	.35/.33	0.027	none	none
767-200	3,050	156	497	0.35	64.6	0.067	.33/.36	0.041	.31/.40	.23/.20
McDONNELL-DOUGLAS										
DC-9 S80	1,270	108	168	0.39	50.5	0.062	.49/.46	0.030	none	none
DC-9-50	1,001	93.4	161	0.41	46.2	0.079	.45/.44	0.038	none	none
DC-10-30	3,958	165	605	0.18	64.6	0.060	0.35	0.047	.32/.39	.20/.25
AIRBUS										
A300-B4	2,799	147	487	0.30	79.5	0.094	.35/.36	0.049	.29/.39	.23/.27
A310	2,357	144	487	0.35	68.5	0.098	.33/.35	0.027	.32/.40	.23/.27
Lockheed L1011										
-500	3,541	164	550	0.23	58.2	0.055	.29/.26	0.051	.40/.49	.22/.23
Fokker F-28										
-4000	850	82.3	157	0.16	37.9	0.085	.29/.31	0.034	none	none
Rombac/British Aerospace										
1-11 495	1,031	93.5	117	0.28	31.6	0.038	.39/.37	0.030	none	none
British Aerospace										
146-200	832	86.4	224	0.44	38.9	0.12	0.29	0.046	none	none
Tu-154	2,169	123	341	0.27	43.3	0.055	0.37	0.036	none	none

Table 6.3 Jet Transports: Vertical Tail Volume, Rudder, Aileron, and Spoiler Data (5:198)

Type	Outb'd Ail. Span in/out fr.b/2	Outb'd Ail. Chord in/out fr.c_w	Inb'd Spoiler Span Loc. in/out fr.b/2	Inb'd Spoiler Chord in/out fr.c_w	Inb'd Spoiler Hinge Loc. in/out fr.c_w	Outb'd Spoiler Span Loc. in/out fr.c_w	Outb'd Spoiler Chord in/out fr.c_w	Outb'd Spoiler Hinge Loc. in/out fr.c_w
BOEING								
727-200	.76/.93	.23/.30	.14/.37	.09/.14	.79/.69	.48/.72	.16/.20	.65/.63
737-200	.74/.94	.20/.28	.40/.66	.14/.18	.66/.67	none	none	none
737-300	.72/.91	.23/.30	.38/.64	0.14	.64/.70	none	none	none
747-200B	.70/.95	.11/.17	.46/.67	.12/.16	0.71	none	none	none
747-SP	.70/.95	.11/.17	.46/.67	.12/.16	0.71	none	none	none
757-200	.76/.97	.22/.36	.41/.74	.12/.13	.73/.69	none	none	none
767-200	.76/.98	.16/.15	.16/.31	.09/.11	.85/.78	.44/.67	.12/.17	.74/.71
McDONNELL-DOUGLAS								
DC-9 S80	.64/.85	.31/.36	.35/.60	.10/.08	.69/.65	none	none	none
DC-9-50	.78/.95	.30/.35	.35/.60	.10/.08	.69/.65	none	none	none
DC-10-30	.75/.93	.29/.27	.17/.30	.05/.06	.78/.74	.43/.72	.11/.16	.75/.70
AIRBUS								
A300-B4	.83/.99	.32/.30	.57/.79	.16/.22	.73/.72	none	none	none
A310	none	none	.62/.83	.16/.22	.69/.66	none	none	none
Lockheed L1011								
-500	.77/.98	.26/.22	.13/.39	.08/.12	.82/.73	.50/.74	.14/.14	.67/.67
Fokker F-28								
-4000	.66/.91	.29/.28	no lateral control spoilers					
Rombac/British Aerospace								
1-11 495	.72/.92	0.26	.37/.68	.06/.11	.68/.63	none	none	none
British Aerospace								
146-200	.78/1.0	.33/.31	.14/.70	.22/.27	.76/.68	none	none	none
Tu-154	.76/.98	.34/.27	.43/.70	.14/.20	.62/.60	none	none	none

The following values are typical of jet transport aircraft:

Horizontal Tails

Dihedral Angle	0° - +11°
Incidence Angle	Variable
Aspect Ratio	3.4 - 6.1
Sweep Angle	18° - 37°
Taper Ratio	0.27 - 0.62

Vertical Tails

Aspect Ratio	0.7 - 2.0
Sweep Angle	33° - 53°
Taper Ratio	0.26 - 0.73

(5:207)

Based on these typical ranges, the Avion empennage surface parameters were chosen. (See Table 6.4.)

Table 6.4 Avion Empennage Surface Parameters

Parameter	Horizontal Tail Surface	Vertical Tail Surface
Aspect Ratio	5.7	1.7
Sweep Angle	28°	40°
Taper Ratio	0.4	0.4
Thickness Ratio	0.11	0.13
Dihedral Angle	0°	N/A
Incidence Angle	Variable	N/A
Span	28.6 ft	15.0 ft
Airfoil	NACA 0011	NACA 0013
c_{av}	5.0 ft	8.9 ft
c_r	7.1 ft	11.8 ft
c_t	2.8 ft	4.7 ft

From Tables 6.1 & 6.2, values for the control surface size ratios were obtained:

$S_e/S_h = 0.25$ $\qquad$ $S_r/S_v = 0.35$

The corresponding elevator and rudder areas were than calculated:

$S_e = 35.8\ ft^2$ $\qquad$ $S_r = 46.6\ ft^2$

These values are extremely important because of the effect these surfaces have on the aircraft. The vertical tail provides directional control and lateral stability, while the horizontal tail provides longitudinal control and stability.

The values determined in this section are only estimations. The exact value of the parameters critical to empennage design can only be determined after a more complete and in-depth study of the aircraft design. This includes many calculation, model simulations, and wind tunnel testing.

7.0 Propulsion System Integration

Propfans are one of the most promising developments for raising propulsive efficiencies at high subsonic Mach numbers. These powerplants combine the efficiency of a propeller with the speed capabilities of a jet engine. For these reasons, propfans appear to be an excellent choice for use in the Avion propulsion system. The demonstration of this new technology on test aircraft has shown that propfans are clearly superior to current turbofan engines in the area of efficiency while still meeting the rigid FAR requirements.

Integration of powerplant systems into the Avion requires not only the choice of engine type, but also the size and placement of the engines. These choices will each have an effect on some aspect of the Avion's performance and must be considered carefully.

7.1 Thrust Requirement

Using the predetermined W_{TO} of 77,000 lb and FAR regulated $(T/W)_{TOmin}$ of 0.32, the Avion's minimum required thrust at sea level was determined to be

$$T_{TOmin} = (T/W)_{TOmin} W_{TO} = 0.32 \times 77{,}000 = 24{,}640 \text{ lb}$$

The required thrust for the Avion was then set at 25,000 lb (2 x 12,500 lb). It is important to note that the greatest efficiency improvements over turbofan engines have been obtained for engines in this thrust level regime.

7.2 Noise

High noise levels have been a major concern of airframe manufacturers since they began to consider propfans as an alternative powerplant. The challenge is to have an acceptable sonic fatigue life and a quiet cabin without a large weight penalty. The main parameters determining propeller source noise are power loading and helical tip speed, both of which increase as flight speed increases.

Helical tip speed can be kept to around 650 ft/sec at Mach 0.7, which would enable open rotors to be placed near the wing. At Mach 0.8, however, tip Mach numbers of 1.1 to 1.15 can be expected. The supersonic acoustical disturbances these Mach numbers cause may require the fans to be shrouded or the engines to be moved to a position behind the wing where they would not cause significant cabin noise. (9:142)

In a flight development program, McDonnell Douglas had two different design teams develop aircraft to use the GE UDF and the IAE V2500 Superfan (the most advanced and efficient turbofan in development). The GE UDF was found to be more efficient than the Superfan and had similar noise levels. When the GE UDF was flight tested on a 727-100, the approach, sideline, and departure noise showed that the engine could meet FAR 36 stage 3 noise regulations. (13:66)

The noise problems are being countered with effective new technology. Research in varying pitch and rotor speeds as well as changing blade attack angle have led to further improvements. Acoustical damping of the fuselage is also being studied to eliminate excessive cabin noise. There are even experiments to determine the ability of sound waves to travel through laminar and turbulent boundary layers at the fuselage skin. From the above considerations, there appears to be little doubt that acceptable noise levels can be met for the propfan's commercial use.

7.3 Placement

The placement of the engines is important to any aircraft design because of its effects on weight, stability, exhaust/slipstream interference, and maintenance/accessibility. Engine placement is *critical* in the case of the Avion because of the additional noise, vibration, and safety difficulties associated with propfan engines.

The original Avion proposal favored over-the-wing engine mounting to achieve both engine and wing performance improvements; however, this option proved to be unacceptable for the following reasons:

(1) Industry concerns over blade separation trajectories possibly impacting the pressure bulkhead or other critical components of the aircraft. (6)

(2) FAR 25.807 over-the-wing emergency escape routes could not be met.

(3) Heavy structure or lack of structural integrity of engine pylons.

(4) Blade tip shock concerns which include:
  (a) Interference of air flow over the wing.
  (b) Sonic fatigue of aircraft structures.

(5) Preliminary design difficulties in verification of performance improvements.

(6) Excessive noise levels due to engine proximity to fuselage.

After consideration of all factors, the decision was made to mount the engines on pylons at the rear of the fuselage. This positioning is the most appropriate for a pusher profan configuration.** The rear pylon mounting (coupled with the T-tail empennage) alleviates the difficulties of exhaust and slipstream interference while maintaining excellent accessibility of the engines for maintenance and repair. Further, since the plane of rotation of the blades is behind the aft pressure bulkhead, the cabin noise and vibration problems are reduced.

It is important that the engines meet FAR requirements for engine mounting. These regulations stipulate that no blade tip may make contact with the ground in a tires deflated or gear up emergency landing condition. This regulation can be easily met through the wide range of mounting heights that are possible with this configuration.

** NOTE: The Avion will probably use the unshrouded, contrarotating, geared, pusher propfan since it is considered to be the most efficient and convenient configuration available from current technology.

7.4 Preliminary Sizing

Propfan propulsion is still an evolving technology. Currently, there is no propfan that meets the thrust and configuration requirements of the Avion. Therefore, it must be assumed that a powerplant will be developed specifically for the Avion. There are two available methods for predicting engine specifications.

First, an existing propfan engine may be scaled to fit the thrust requirements of the Avion. This method generates basic data regarding the size of the propfan to be developed. Furthermore, the only significant engine data obtainable is from the propfan forerunners. These pioneering engines have been antiquated by recently developed propfans with increased performance levels.

The Pratt & Whitney/Allison 578-DX is one of the most highly developed propfans. It is designed in a 6-blade, pusher, contrarotating configuration applicable to the Avion. This engine has undergone almost a decade of technology development in order to maximize its efficiency. Major advances have been made with the gearbox in particular. This device allows the blades to turn at an ideal rate, keeping the blade tip speed constant while allowing the core engine to operate at its peek RPM efficiency. Early gearboxes could not endure the high loading of the new, more powerful engines. Light weight, high efficiency gearboxes have since been developed to accommodate these higher loadings.

Scaling certain parameters of the performance of this engine for application on the Avion yielded the following results:

Thrust:	12,500 lb
Blade Dia.:	8.0 ft
Power:	6,800 hp
Weight:	4,000 lb

The second method uses existing research data on a specific configuration. It is then assumed that the Avion propfan will be similar, but with modestly improved performance characteristics. Examples of calculations for propfans can be found in *Reference 7*. This report deals with the Large Scale Advanced Prop-Fan (LAP) and covers calculations for performance, acoustic, and weight estimation. Since the technology revealed in this reference is not current, example calculations are omitted from this report.

7.5 Design Challenge

In conclusion, the Avion is expected to make use of the most advanced technology of its time to produce a safe, reliable, and highly efficient propulsion system. The challenge to competing engine manufacturers will be to produce a pusher propfan engine to meet the following requirements:

Thrust = 12,500 lb

Weight = 5,000 lb (or less)

The engine should be as efficient as possible while maintaining the reliability and maintainability of today's aircraft engines.

8.0 Component Weight & Balance Estimations

This section serves to provide an estimation of the Avion component weight and balance. Preliminary sizing weight estimation methods rely on the assumption that major aircraft component weights can be expressed as a percentage of gross take-off weight or empty weight.

The component weight list contains the following items:

I. Structure Weight

1. Wing
2. Canard
3. Empennage
4. Fuselage
5. Nacelles
6. Landing Gear

II. Powerplant Weight

1. Engines
2. Propellers
3. Fuel System
4. Propulsion System

III. Fixed Equipment Weight

1. Flight Control System
2. Hydraulic and Pneumatic System
3. Electrical System
4. Instrumentation, Avionics, and Electronics
5. Air Conditioning, Pressurization, Anti-Icing and De-Icing System
6. Oxygen System
7. Auxiliary Power Unit
8. Furnishings
9. Operational Items
10. Flight Test Instrumentation
11. Paint
12. Other Weight

The Avion empty weight can be expressed by

$$W_E = W_{struct} + W_{pwr} + W_{feq}$$

For preliminary sizing methods, aircraft of similar mission specifications were examined and their weight fractions averaged. Table 8.1 contains comparisons of the McDonnell Douglas DC-9-30 & MD-80 and Boeing 727-100 & 737-200 aircraft.

Table 8.1 Aircraft Weight Comparison Spreadsheet

Weight Item	McDonnell Douglas DC-9-30 lb	McDonnell Douglas DC-9-30 %	McDonnell Douglas MD-80 lb	McDonnell Douglas MD-80 %	Boeing 727-100 lb	Boeing 727-100 %	Boeing 737-200 lb	Boeing 737-200 %	Average %	Avion lb	Avion %
Take-Off Gross Weight	108,000		140,000		160,000		115,500			77,000	
Empty Weight	58,070	53.77%	78,950	56.39%	88,300	55.19%	60,210	52.13%	54.37%	38,843	50.45%
Wing Group	11,400	10.56%	15,560	11.11%	17,764	11.10%	10,613	9.19%	10.49%	7,495	9.73%
Empennage Group	2,780	2.57%	3,320	2.37%	4,133	2.58%	2,718	2.35%	2.47%	1,765	2.29%
Fuselage Group	11,160	10.33%	16,150	11.54%	17,681	11.05%	12,108	10.48%	10.85%	7,752	10.07%
Nacelle Group	1,430	1.32%	2,120	1.51%	3,870	2.42%	1,392	1.21%	1.62%	1,154	1.50%
Landing Gear Group	4,170	3.86%	5,340	3.81%	7,211	4.51%	4,354	3.77%	3.99%	2,849	3.70%
Structure Total	30,940	28.65%	42,490	30.35%	50,659	31.66%	31,185	27.00%	29.42%	21,015	27.29%
Engines	6,410	5.94%	8,820	6.30%	9,325	5.83%	6,217	5.38%	5.86%	4,188	5.44%
Exhaust and Thrust Reverser System	1,240	1.15%	1,540	1.10%	1,744	1.09%	1,007	0.87%	1.05%	752	0.98%
Fuel System	600	0.56%	640	0.46%	1,143	0.71%	575	0.50%	0.56%	397	0.52%
Propulsion Install.	0	0.00%	0	0.00%	250	0.16%	378	0.33%	0.12%	86	0.11%
Power Plant Total	8,250	7.64%	11,000	7.86%	12,462	7.79%	8,177	7.08%	7.59%	5,423	7.04%
Avionics & Instrum.	1,450	1.34%	2,130	1.52%	756	0.47%	625	0.54%	0.97%	693	0.90%
Surface Controls	1,620	1.50%	2,540	1.81%	2,996	1.87%	2,348	2.03%	1.80%	1,289	1.67%
Hydraulic System & Pneumatic System	760	0.70%	830	0.59%	1,418	0.89%	873	0.76%	0.73%	525	0.68%
Electrical System	1,330	1.23%	1,720	1.23%	2,142	1.34%	1,066	0.92%	1.18%	843	1.10%
Electronics					1,591	0.99%	956	0.83%	0.46%	325	0.42%
APU	820	0.76%	840	0.60%	60	0.04%	836	0.72%	0.53%	379	0.49%
Oxygen System	150	0.14%	220	0.16%							
Air Cond. System & Anti-Icing System	1,600	1.48%	2,130	1.52%	1,976	1.24%	1,416	1.23%	1.37%	976	1.27%
Furnishings	8,450	7.82%	11,400	8.14%	10,257	6.41%	6,643	5.75%	7.03%	5,024	6.52%
Operating Items	2,700	2.50%	3,650	2.61%							
Miscellaneous					85	0.05%	124	0.11%	0.04%	29	0.04%
Fixed Equip. Total	18,880	17.48%	25,460	18.19%	21,281	13.30%	14,887	12.89%	15.46%	11,048	14.35%
Max. Fuel Capacity	28,746	26.62%	39,362	28.12%	48,353	30.22%	34,718	30.06%	28.75%	20,542	26.68%
Max. Payload	28,930	26.79%	43,050	30.75%	29,700	18.56%	34,790	30.12%	26.56%	18,972	24.64%
									107.78%		

The Avion preliminary component weight estimations were based upon the average values obtained from the aircraft compared in Table 8.1. The "Average" column percentages (totaling 107.78%) had to be normalized to a total of 100% before being applied to the Avion components. This normalization process yielded the percentages found under the "Avion %" column. Recalling Table 3.2 results:

W_{TO} = 77,000 lb $\qquad$ W_E = 42,405 lb

W_F = 16,150 lb $\qquad$ W_{PL} = 16,985 lb

W_{tfo} = 385 lb $\qquad$ W_{crew} = 1,075 lb

The gross take-off weight was then used as the base value in Table 8.1 to estimate the individual component weights. The chief component values were transferred to Table 8.2 for adjustment.

Table 8.2 Avion Preliminary Sizing Component Weight Estimation Spreadsheet

Component	First Estimate	Adjustment	Material Adjustment	Totals
Wing	7,495	1,000	-849	7,645
Canard	0	900	-90	810
Empennage	1,765	-400	-136	1,228
Fuselage	7,752	2,000	-975	8,777
Nacelle	1,154	500	-165	1,489
Landing Gear	2,849	0	0	2,849
Power Plant	5,423	4,596	0	10,019
Fixed Equipment	11,048	-1,460	0	9,588
Empty	*37,486*	*7,136*	*-2,217*	*42,405*
Trapped Fluids		385		385
Crew		1,075		1,075
Operating Empty		*8,596*		*43,865*
Fuel	20,542	-4,392		16,150
Payload	18,972	-1,987		16,985
Gross Take-Off	*77,000*	*2,217*	*-2,217*	*77,000*

Upon initial inspection of the first estimate data in Table 8.2, it became readily obvious that the fuel and payload weights obtained were unsatisfactory. The values obtained from the weight fraction method exceeded those predicted by the conservative preliminary sizing. The excess fuel and payload weights were designated to be transferred to other components of the aircraft which were deemed lacking in appropriate weight. The crew and trapped fluid weights also needed to be accounted for, and were therefore designated weight appropriately.

The first necessary weight increase occurred with the powerplant component weight estimation. Early weight predictions for an appropriately sized propfan engine suggested a 10,000 lb (2 x 5,000 lb) propulsion system for the Avion. Weight was added to the first estimate to facilitate this requirement. The engine nacelle (mounting pylon) was also adjusted for increased structural strength in the Avion's aft-mounted propfan design.

Because of major structural design differences between the Avion and other transport aircraft, significantly higher weights will be required to obtain the structural integrity needed for the forward-swept, tri-wing configuration. Furthermore, for seating comfort, structural, and stability purposes, the Avion fuselage length was modestly enlarged past that expected of a 79-passenger, 5-seat abreast aircraft.

The remaining excess weight from the previous calculations did not appear sufficient to meet the standard design requirements for an aluminum structure. Therefore, lithium/aluminum and carbon-based composite materials were considered for use as the primary structural material for the Avion. A reasonable assumption was to apply a 10% weight reduction to the wing, canard, empennage, fuselage, and nacelle. These component weights were then increased appropriately and adjusted to the target take-off weight of 77,000 lb.

After satisfactory determination of the weight and location of the major components of the Avion, a preliminary moment analysis was performed. This allowed for the determination of c.g. location and maximum c.g. travel during flight conditions. Moments were taken about the nose of the aircraft. Table 8.3 contains the preliminary moment and c.g. analysis for the Avion.

Table 8.3 Avion Preliminary Moment and C.G. Analysis Spreadsheet

Component	Weight (lb)	c.g. (in)	Moment (in-lb)
Wing	7,645	900	6.88E+06
Canard	810	325	2.63E+05
Empennage	1,228	1,300	1.60E+06
Fuselage	8,777	575	5.05E+06
Nacelle	1,489	1,050	1.56E+06
Landing Gear	2,849	780	2.22E+06
Power Plant	10,019	1,050	1.05E+07
Fixed Equipment	9,588	455	4.36E+06
Empty	*42,405*	*765*	*3.25E+07*
Trapped Fluids	385	750	2.89E+05
Crew	1,075	319	3.43E+05
Operating Empty	*43,865*	*754*	*3.31E+07*
Fuel	16,150	825	1.33E+07
Payload	16,985	675	1.15E+07
Gross Take-Off	*77,000*	*752*	*5.79E+07*
Operating Empty			
- with Fuel	*60,015*	*773*	*4.64E+07*
- with Payload	*60,850*	*732*	*4.46E+07*
	Max c.g. Travel:	*41 in.*	
		0.39 c_{av}	

From Table 8.3, the critical c.g. location are as follows:

(1) Operating Empty Weight

(2) Operating Empty Weight + Fuel Weight

(3) Operating Empty Weight + Payload Weight

(4) Gross Take-Off Weight

These c.g. locations must all lie close to each other in order to minimize c.g. travel during flight.

The industry trend for c.g. travel ranges of jet transports are as follows:

C.G. Range: 26-91 in 0.12-0.32 c_{av} (5:243)

The results of c.g. and moment analysis of Table 8.3 revealed a maximum c.g. travel of 41" or 0.39 c_{av}. This value for c.g. travel is within the FAR limits and near industry trends for commercial aircraft. The chord fraction value is slightly high due to the high aspect ratio of the wing, yielding a relatively small c_{av}. The governing component of c.g. travel is the fuel c.g. location. Movement of the fuel c.g. location forward would result in two favorable conditions:

(1) Minimization of c.g. travel during flight conditions.

(2) Movement of aircraft flight c.g.'s forward improving static longitudinal stability.

Table 8.4 reflects the improved c.g. positions and travel values that would be obtained if the fuel c.g. location were moved forward. The c.g. travel is minimized at the fuel c.g. location corresponding to 795 inches. There are three proposed methods for forward movement of the fuel c.g. location.

First, the leading edge extensions (LEXes or strakes) of the Avion, which are already planned to be used for most of the fuel storage, could be extended forward approaching the canard. This would allow more fuel storage forward of the present fuel c.g. location and less fuel storage required in the wing.

Table 8.4 Effects of Fuel C.G. Location on Avion C.G. Travel

Fuel C.G.	Operating Empty with Fuel C.G.	Gross Take-Off C.G.	Max. C.G. Travel	
825 in	773 in	752 in	41 in	0.39 c_{av}
820	772	751	40	0.38
815	771	750	38	0.36
810	769	748	37	0.35
805	768	747	36	0.34
800	767	746	34	0.32
795	765	745	33	0.31

Second, a controversial solution would be to provide fuel storage in the fuselage. Commercial aircraft of this size are currently prohibited from carrying highly flammable fluids in this area. With the emergence of high flash point commercial jet fuels (similar to the fuel used in the SR-71), these restrictions may be removed in the future.

Third, during flight, a fuel management system may be used to pump fuel to various fuel chambers to control c.g. travel and to provide another method for trimming the aircraft. The system is presently in use on many large commercial transports that have problems with c.g. travel.

9.0 Aircraft Inertia Estimations

This section serves to provide an estimation of the Avion aircraft inertias. The analysis of this section relies on the assumption that the radii of gyration may be determined and used in the following equations:

$I_{xx} = R_x^2 W/g$

$I_{yy} = R_y^2 W/g$

$I_{zz} = R_z^2 W/g$

Based on Roskam methods, the non-dimensional radius of gyration is related to each R component through the following:

$\underline{R}_x = 2R_x/b$

$\underline{R}_y = 2R_y/L$

$\underline{R}_z = 2R_z/e$, where $e = (b + L)/2$

Since aircraft of the same mission orientation and size tend to have similar values for their non-dimensional radii of gyration, the Avion based its values on the McDonnell Douglas DC9-10:

GW = 74,000 lb	$\underline{R}_x$ = 0.242		
b = 89.4 ft	$\underline{R}_y$ = 0.360		
L = 104.3 ft	$\underline{R}_z$ = 0.435		
e = 96.9 ft	Engines: 2 on Fuselage		(8:201)

The Avion moment of inertias were calculated from the following:

L = 1405 in = 117.1 ft

b = 1050 in = 87.5 ft

e = (1405 + 1050)/(2 * 12) = 102.3 ft

$I_{xx} = b^2 W \underline{R}_x^2/4g$

$I_{yy} = L^2 W \underline{R}_y^2/4g$

$I_{zz} = e^2 W \underline{R}_z^2/4g$

At take-off:

$$I_{xx} = (87.5)^2(77{,}000)(0.242)^2/(4 \times 32.174) = 268{,}270 \text{ slug-ft}^2$$

$$I_{yy} = (117.1)^2(77{,}000)(0.360)^2/(4 \times 32.174) = 1{,}063{,}272 \text{ slug-ft}^2$$

$$I_{zz} = (102.3)^2(77{,}000)(0.435)^2/(4 \times 32.174) = 1{,}184{,}828 \text{ slug-ft}^2$$

At operating empty:

$$I_{xx} = (87.5)^2(44{,}000)(0.242)^2/(4 \times 32.174) = 153{,}297 \text{ slug-ft}^2$$

$$I_{yy} = (117.1)^2(44{,}000)(0.360)^2/(4 \times 32.174) = 607{,}584 \text{ slug-ft}^2$$

$$I_{zz} = (102.3)^2(44{,}000)(0.435)^2/(4 \times 32.174) = 677{,}045 \text{ slug-ft}^2$$

10.0 Conclusions

The scope of this detailed report includes the preliminary sizing, configuration design, performance parameter estimations, planform design, propulsion integration, component weight estimation, and aircraft inertia aspects of the Avion design process. This section summarizes the initial speculations and feasibility studies of the conceptual design process. Based upon the overall conclusions drawn to this point, the entire Avion development process can now focus upon a more detailed, "Class II" design.

The Avion has evolved from its initial conception into a promising aircraft design. Based upon this preliminary research, the first impressions of the Avion's most important characteristics were developed and sized. Figure 1.1 illustrates the embodiment and detailed layout of these features. It must be brought to the attention of the reader, that the Avion is still in the very preliminary stages of the design process. In order to bring this design to fruition, a continued effort of research and development must take place in the future. Several areas which need further attention and were not addressed properly by this report due to time constraints are as follows:

(1) Continued sizing to performance parameters

(2) Further propfan powerplant development and integration

(3) Control surface sizing

(4) Landing gear sizing

(5) Stability and control analysis

(6) Drag analysis and prediction

(7) Internal structural design

(8) Canard & winglet layout and sizing

(9) Aircraft systems (e.g., fuel, hydraulic, electrical)

(10) Aerodynamic force and moment (airloads) analysis

The Avion preliminary design process contained within the pages of this report only touches upon the research and design necessary for development of an aircraft. Within the scope of the Auburn University Senior Design sequence, the members of the Avion design team have discovered the true meaning behind the process known as "engineering". Through trade studies, advanced conceptual design, problem identification & resolution, design verification & feasability, economic analysis, and design presentation, the Avion members have developed an appreciation and deeper understanding of the scope and processes involved with aerospace engineering, and engineering in general.

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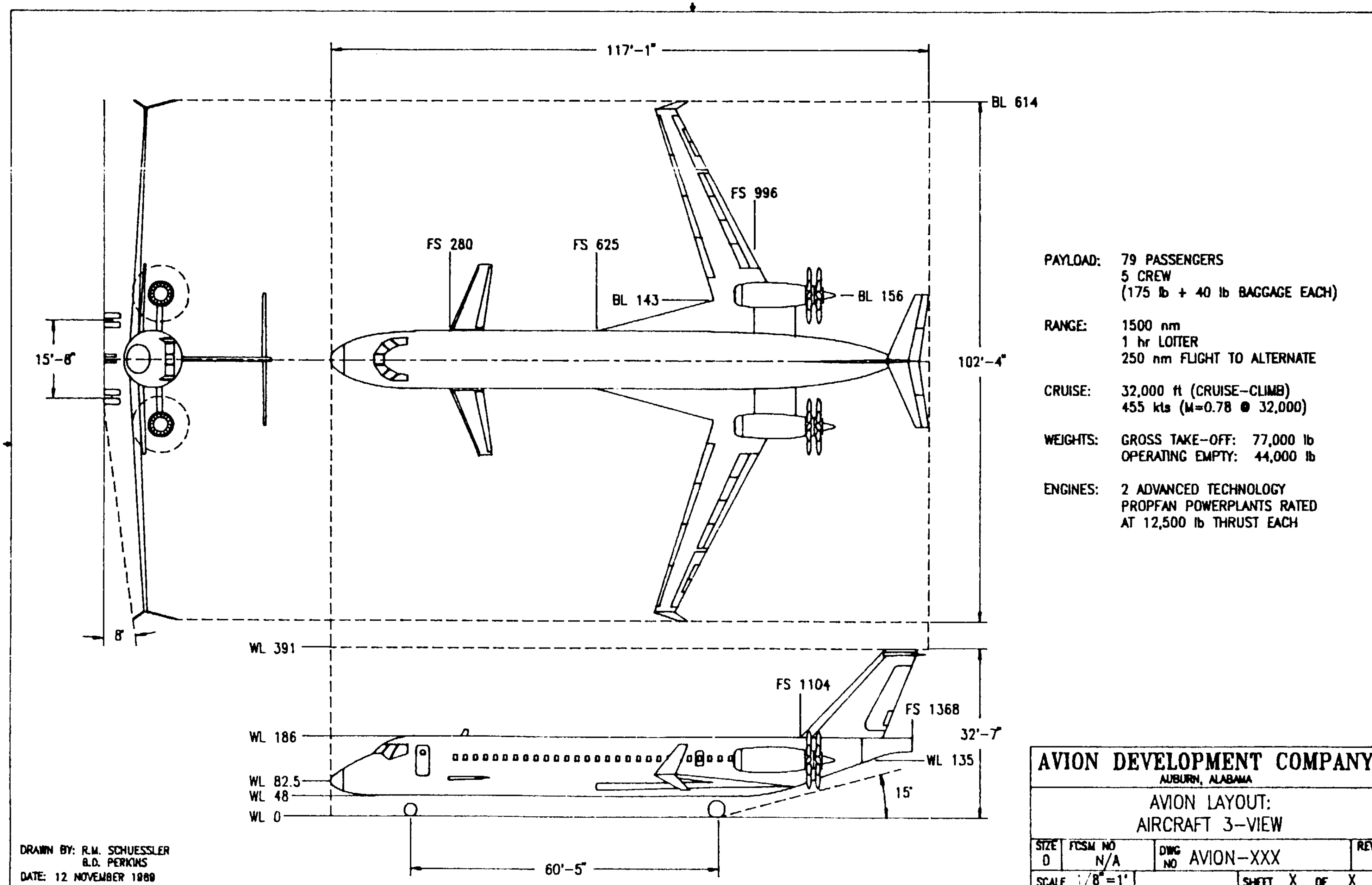


Figure 1. Avion Configuration 3-View

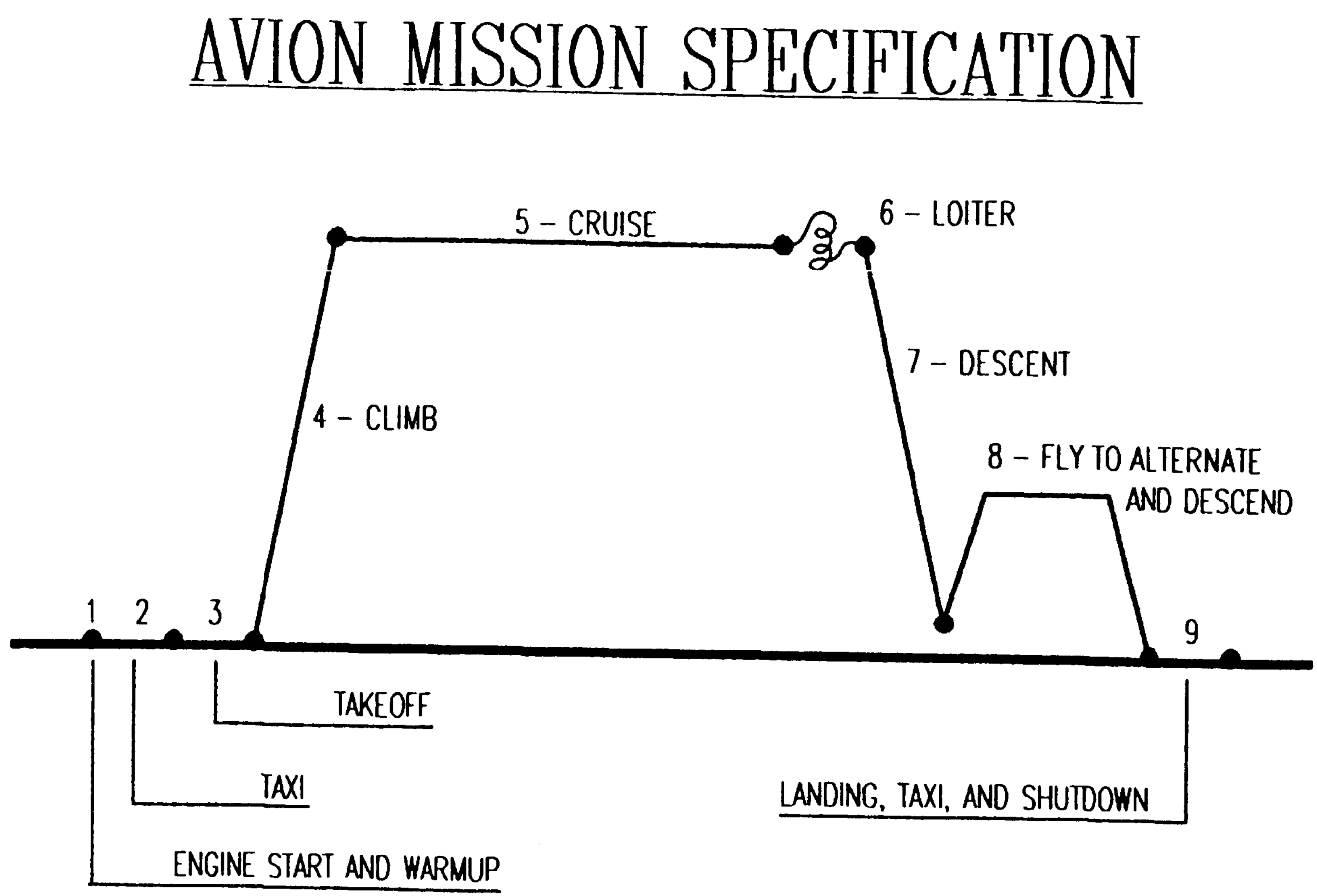
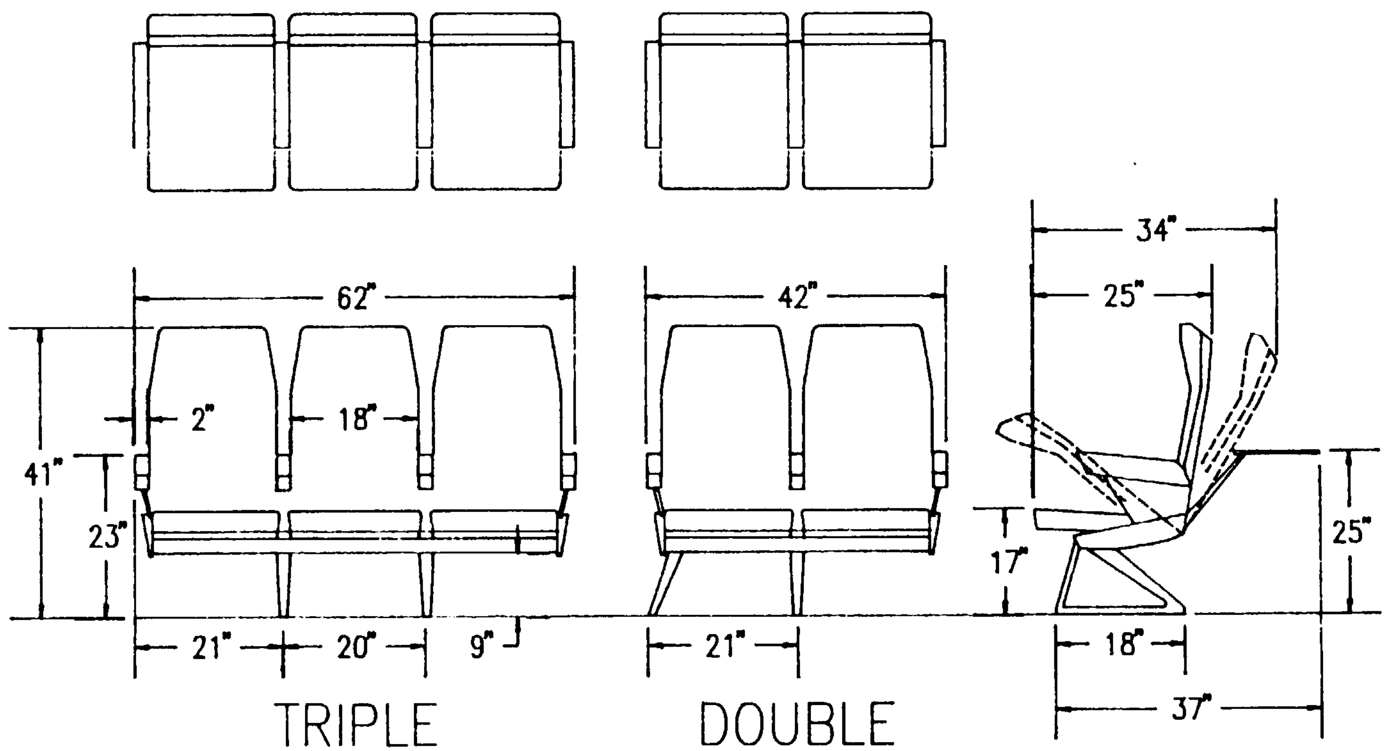


Figure 2. Avion Mission Specification



NOTES:

1. ALL SEATS MUST MEET REQUIREMENTS OF FAR 25, PARTS 561 & 785
2. SEAT PITCH: 36"
3. WEIGHTS: TRIPLE – 66 lb
   DOUBLE – 44 lb

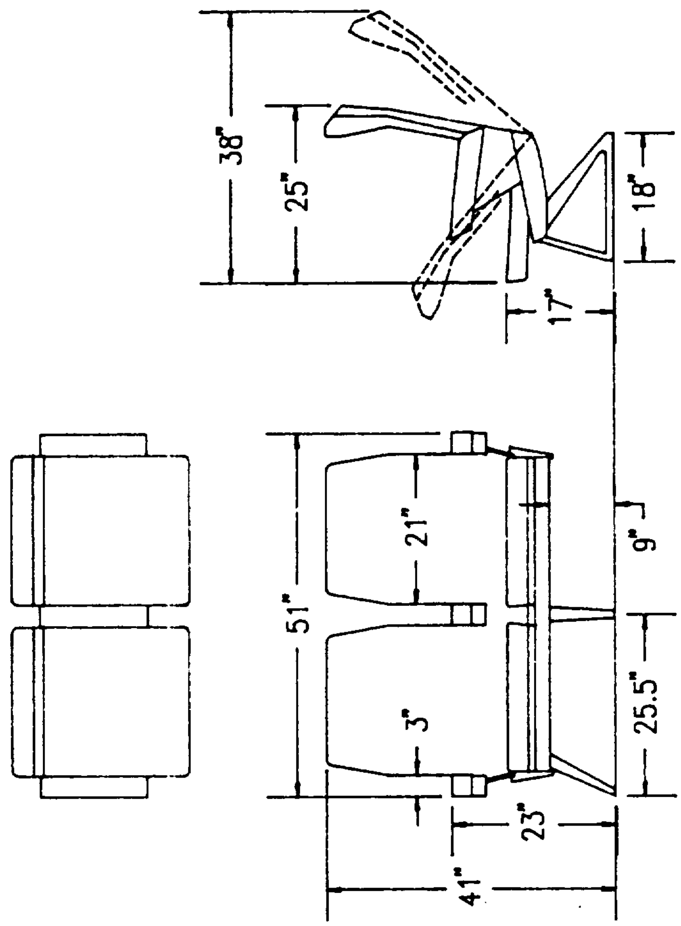
Figure 3. Avion Coach Class Seating

AVION DEVELOPMENT COMPANY
AUBURN, ALABAMA

AVION FIXTURES:
COACH SEATING

SIZE	FCSM NO	DWG NO	REV
A	N/A	AVION–XXX	
SCALE 1/2" = 1'		SHEET X OF X	

DRAWN BY: R.M. SCHUESSLER
DATE: 7 SEPTEMBER 1989



NOTES:
1. ALL SEATS MUST MEET REQUIREMENTS OF FAR 25, PARTS 561 & 785
2. SEAT PITCH: 40"
3. WEIGHTS: DOUBLE - 60 lb

Figure 4. Avion First Class Seating

AVION DEVELOPMENT COMPANY
AUBURN, ALABAMA

AVION FIXTURES:
FIRST CLASS SEATING

SIZE	FCSM NO	DWG NO	REV
A	N/A	AVION-XXX	

SCALE 1/2" = 1'		SHEET X OF X

DRAWN BY: R.M. SCHUESSLER
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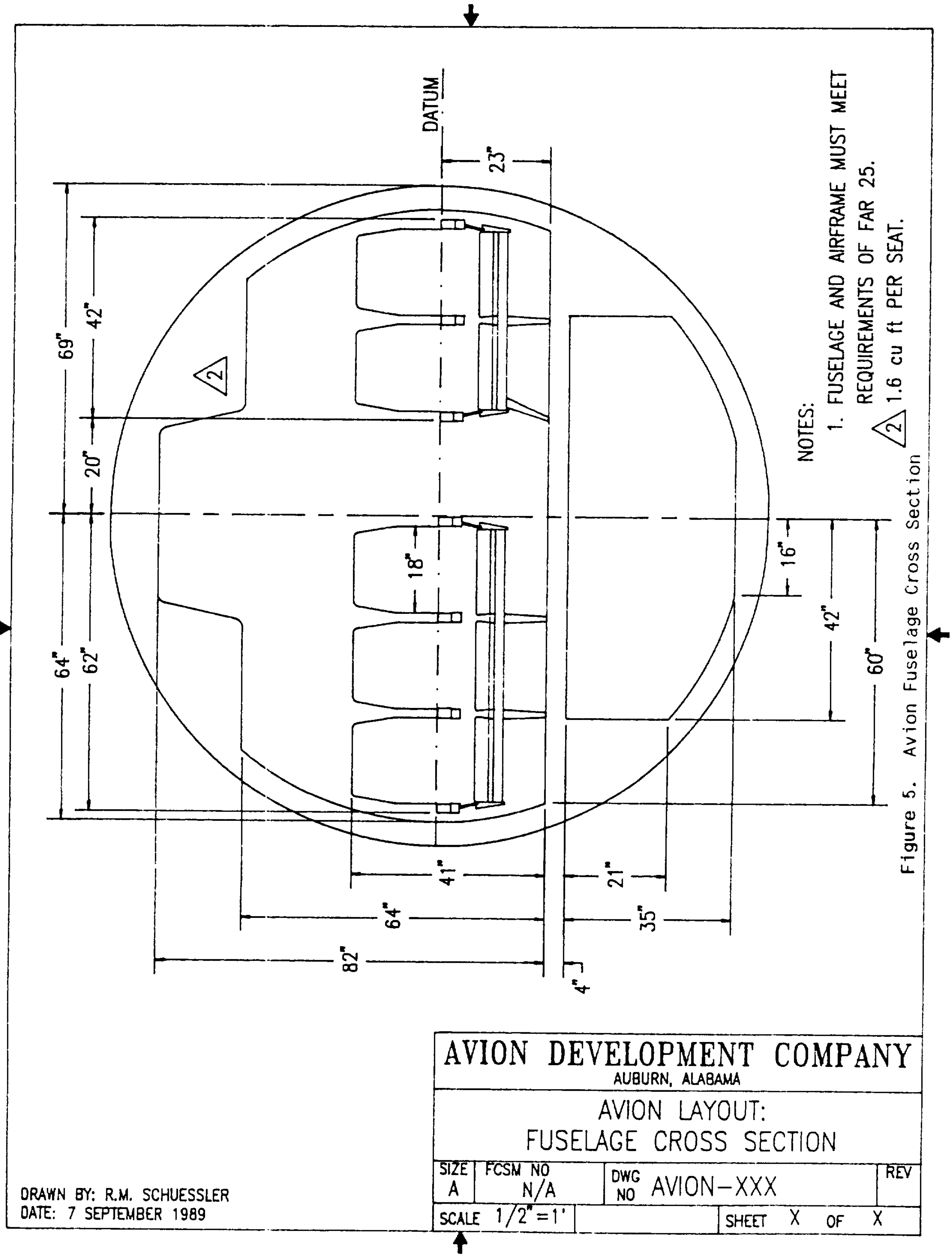
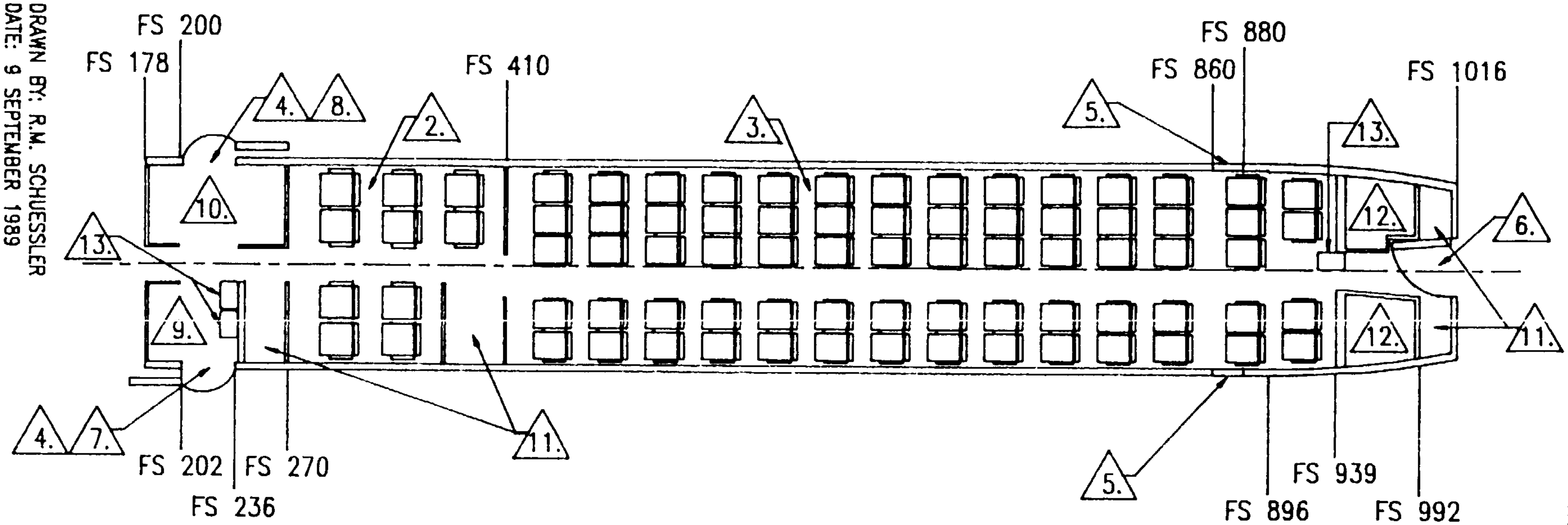


Figure 5. Avion Fuselage Cross Section



NOTES:
1. FUSELAGE MUST MEET REQUIREMENTS OF FAR 25, PARTS 807-813.
2. FIRST CLASS SEATING – 40" PITCH
3. COACH CLASS SEATING – 36" PITCH
4. TYPE I EMERGENCY EXIT – 36" UNOBSTRUCTED ACCESS
5. TYPE III EMERGENCY EXIT – 18" UNOBSTRUCTED ACCESS
6. TAILCONE EMERGENCY EXIT – 36" UNOBSTRUCTED ACCESS
7. FORWARD PASSENGER DOOR
8. GALLEY SERVICING DOOR
9. COAT SPACE
10. GALLEY
11. STOWAGE
12. LAVATORY
13. CABIN CREW FOLDING SEAT(S)

DRAWN BY: R.M. SCHUESSLER
DATE: 9 SEPTEMBER 1989

AVION DEVELOPMENT COMPANY
AUBURN, ALABAMA

AVION LAYOUT:
FUSELAGE/SEATING ARRANGEMENT

SIZE	FCSM NO	DWG NO	REV
A	N/A	AVION-XXX	

SCALE 1/8" = 1' SHEET X OF X

Figure 6. Avion Fuselage/Seating Layout

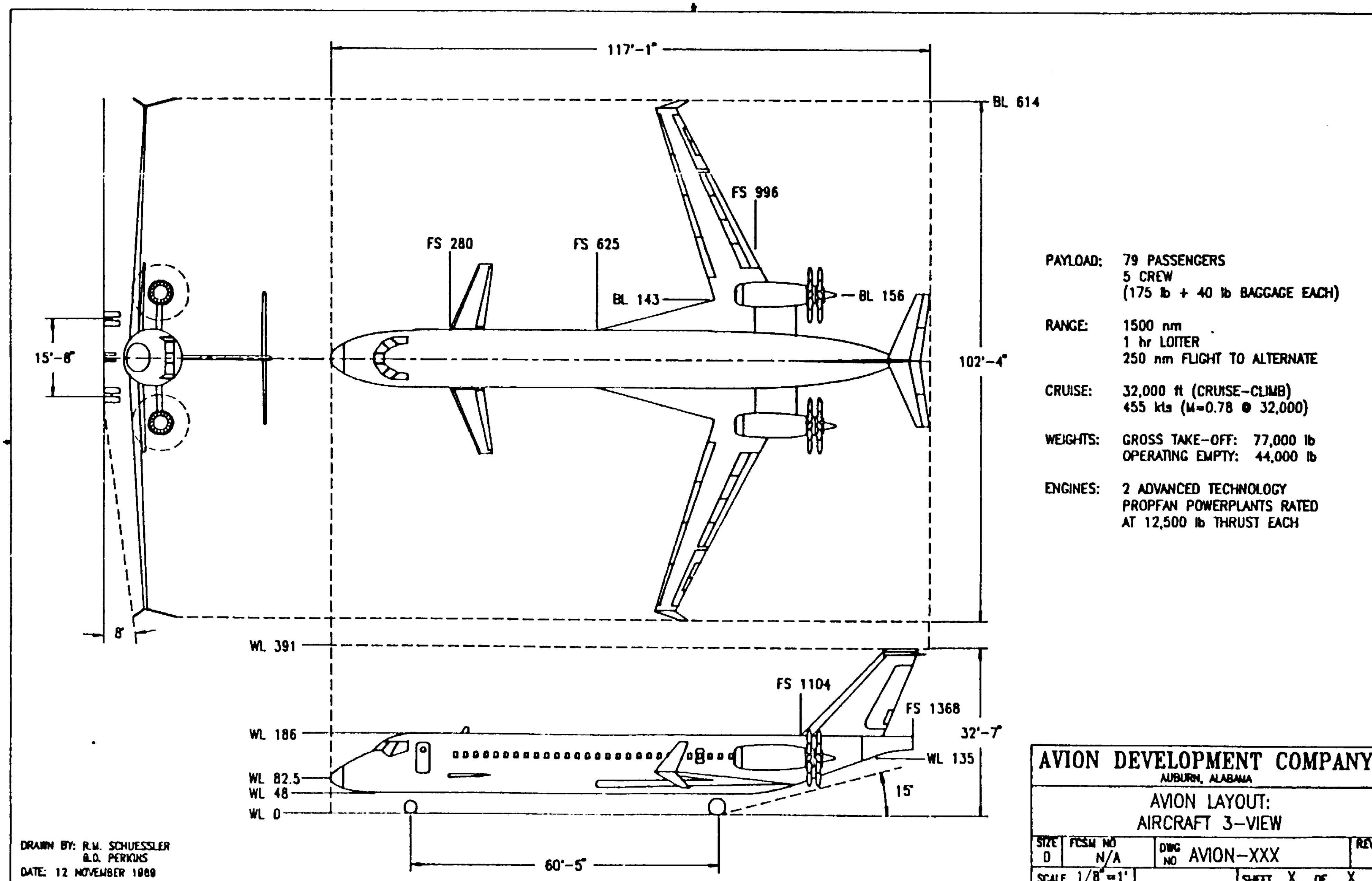


Figure 1. Avion Configuration 3-View

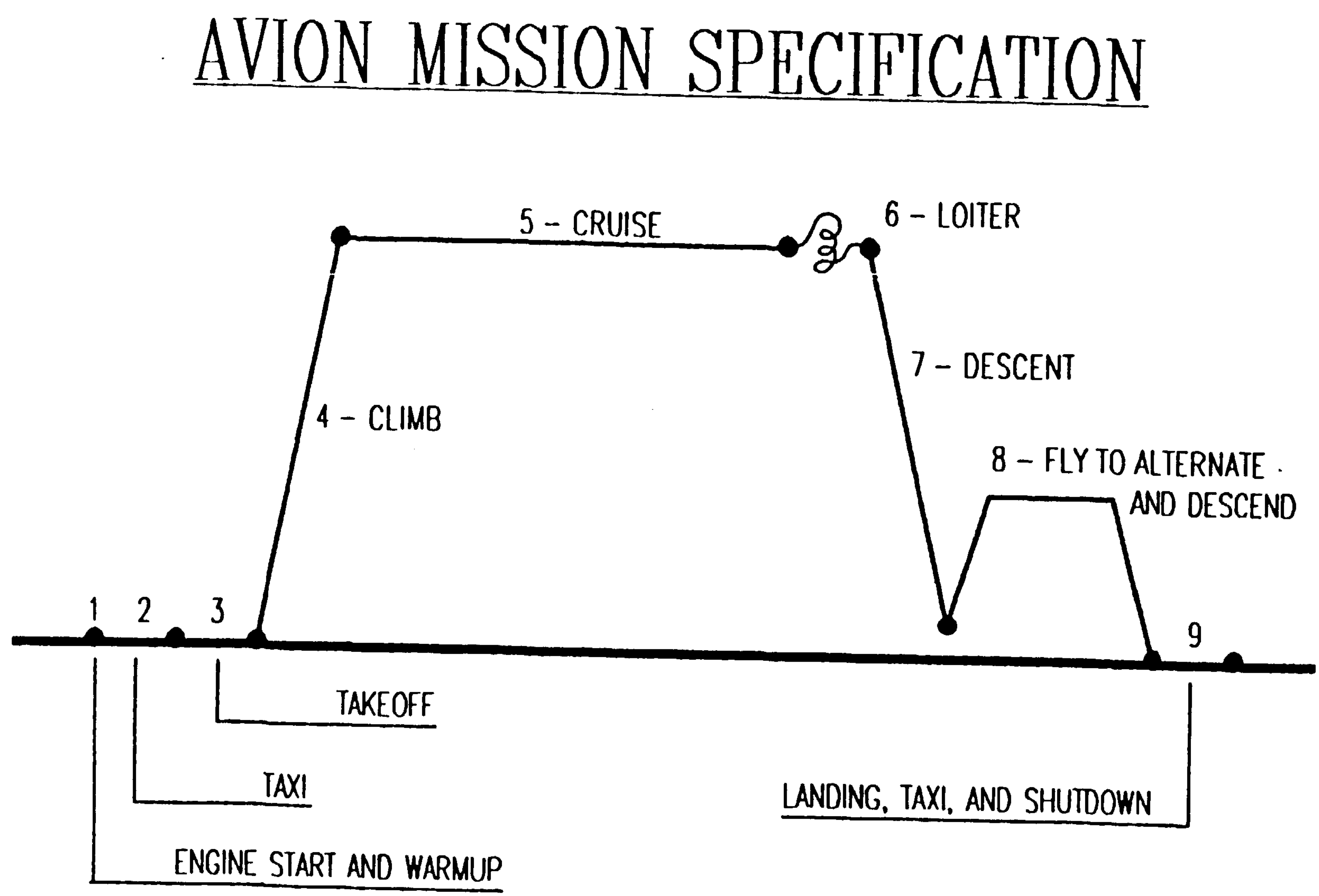
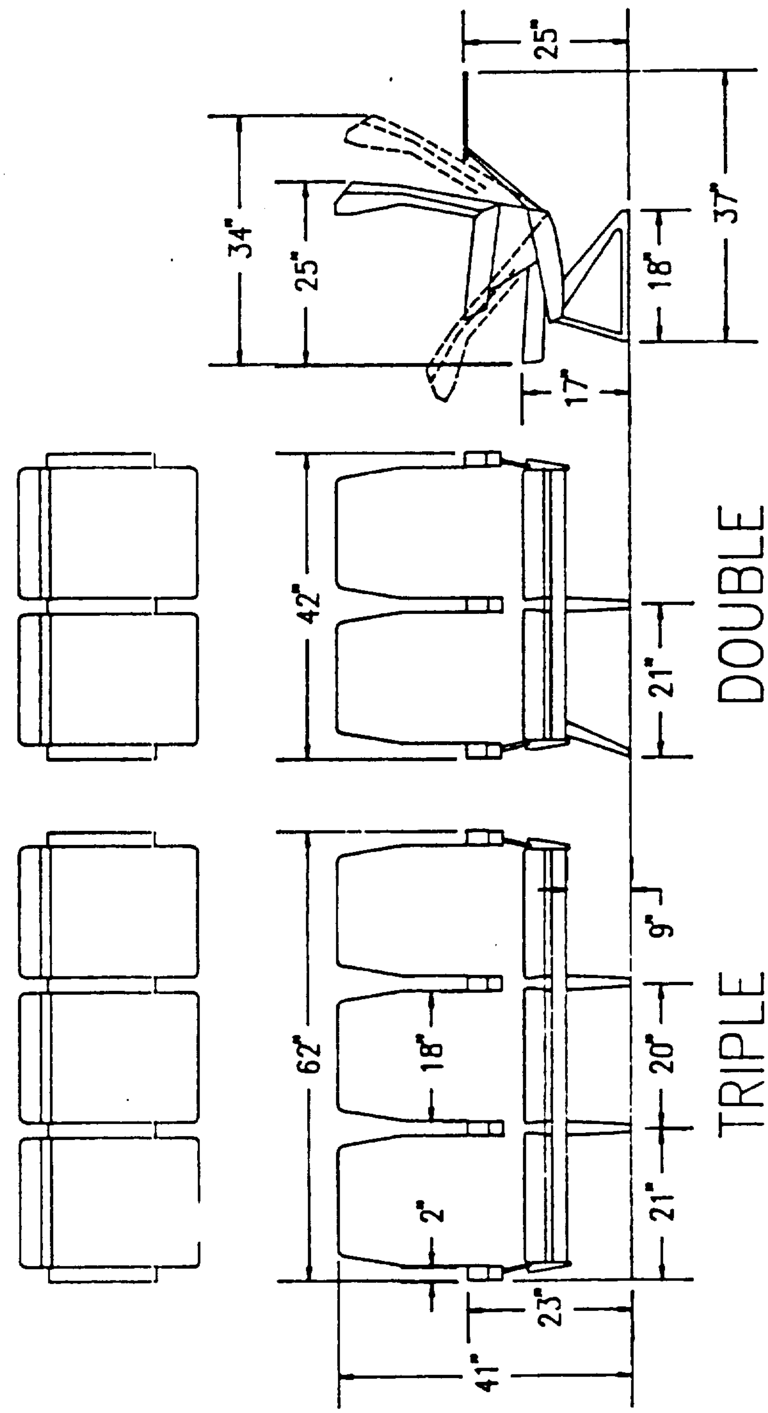


Figure 2. Avion Mission Specification

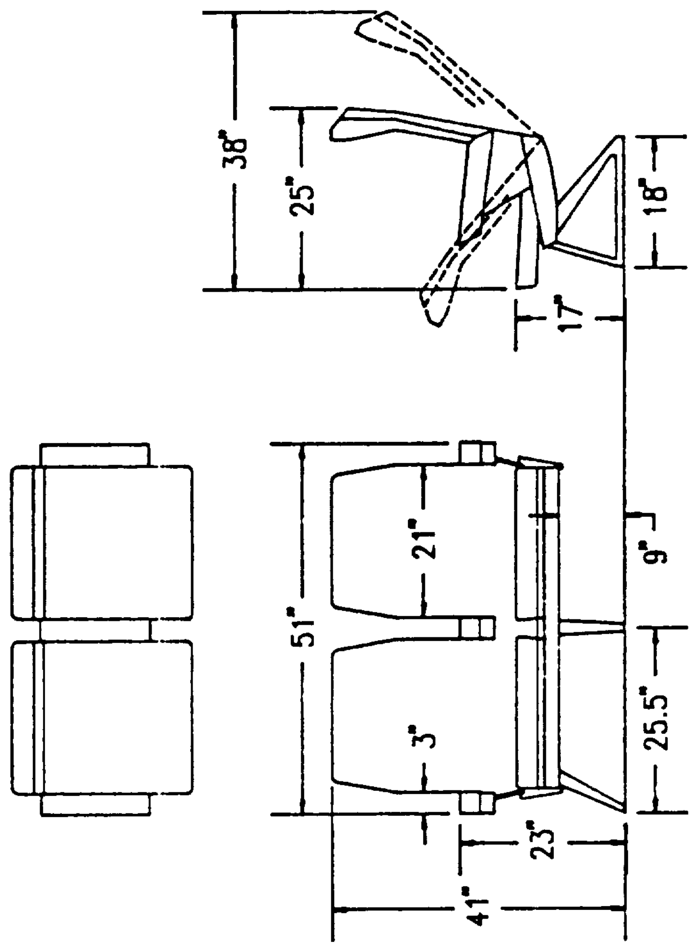


NOTES:
1. ALL SEATS MUST MEET REQUIREMENTS OF FAR 25, PARTS 561 & 785
2. SEAT PITCH: 36"
3. WEIGHTS: TRIPLE – 66 lb
DOUBLE – 44 lb

Figure 3. Avion Coach Class Seating

AVION DEVELOPMENT COMPANY AUBURN, ALABAMA			
AVION FIXTURES: COACH SEATING			
SIZE A	FCSM NO N/A	DWG NO AVION–XXX	REV
SCALE 1/2" = 1'		SHEET X OF X	

DRAWN BY: R.M. SCHUESSLER
DATE: 7 SEPTEMBER 1989



NOTES:
1. ALL SEATS MUST MEET REQUIREMENTS OF FAR 25, PARTS 561 & 785
2. SEAT PITCH: 40"
3. WEIGHTS: DOUBLE – 60 lb

Figure 4. Avion First Class Seating

AVION DEVELOPMENT COMPANY AUBURN, ALABAMA			
AVION FIXTURES: FIRST CLASS SEATING			
SIZE A	FCSM NO N/A	DWG NO AVION–XXX	REV
SCALE 1/2" = 1'		SHEET X OF X	

DRAWN BY: R.M. SCHUESSLER
DATE: 7 SEPTEMBER 1989

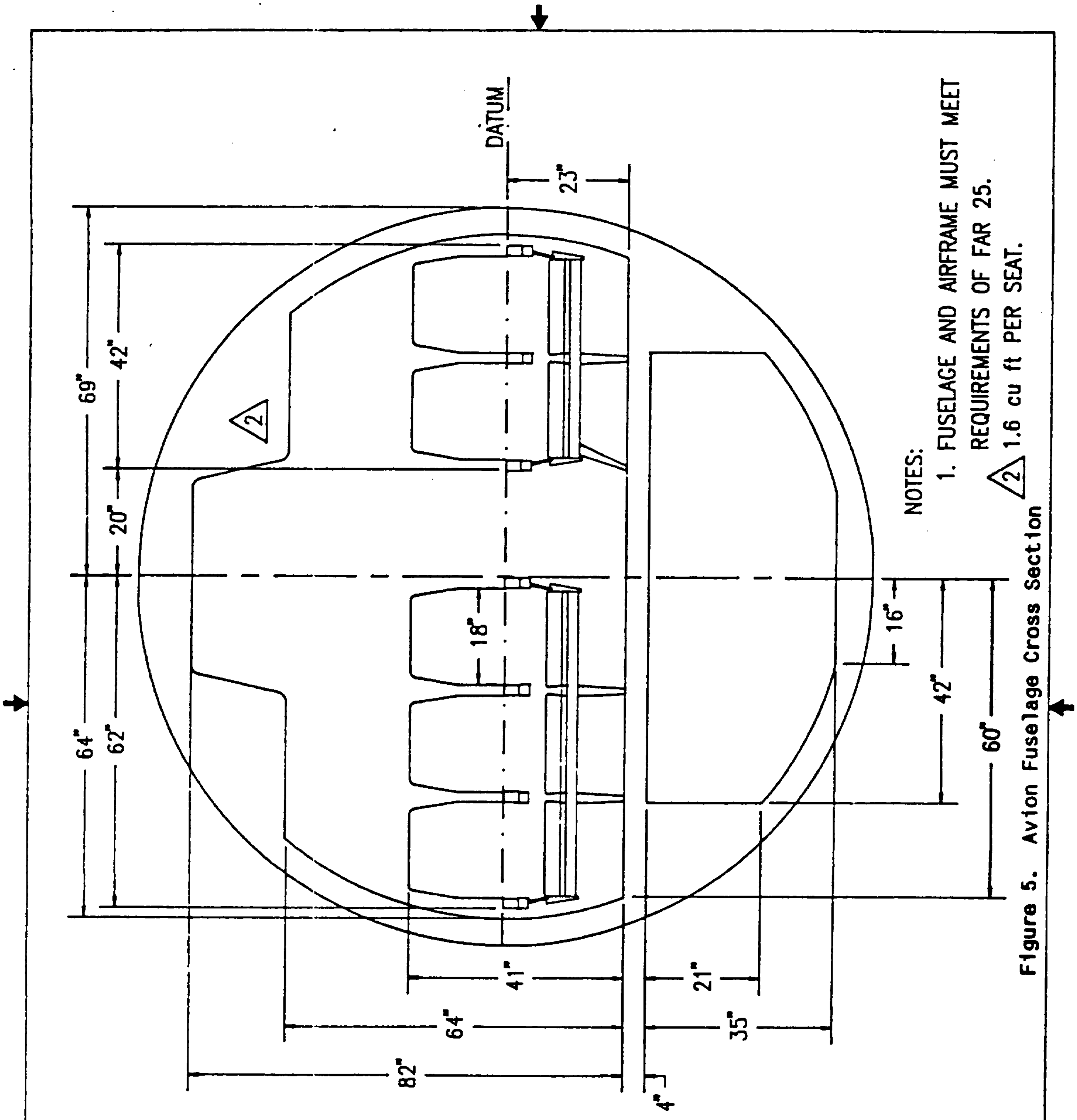
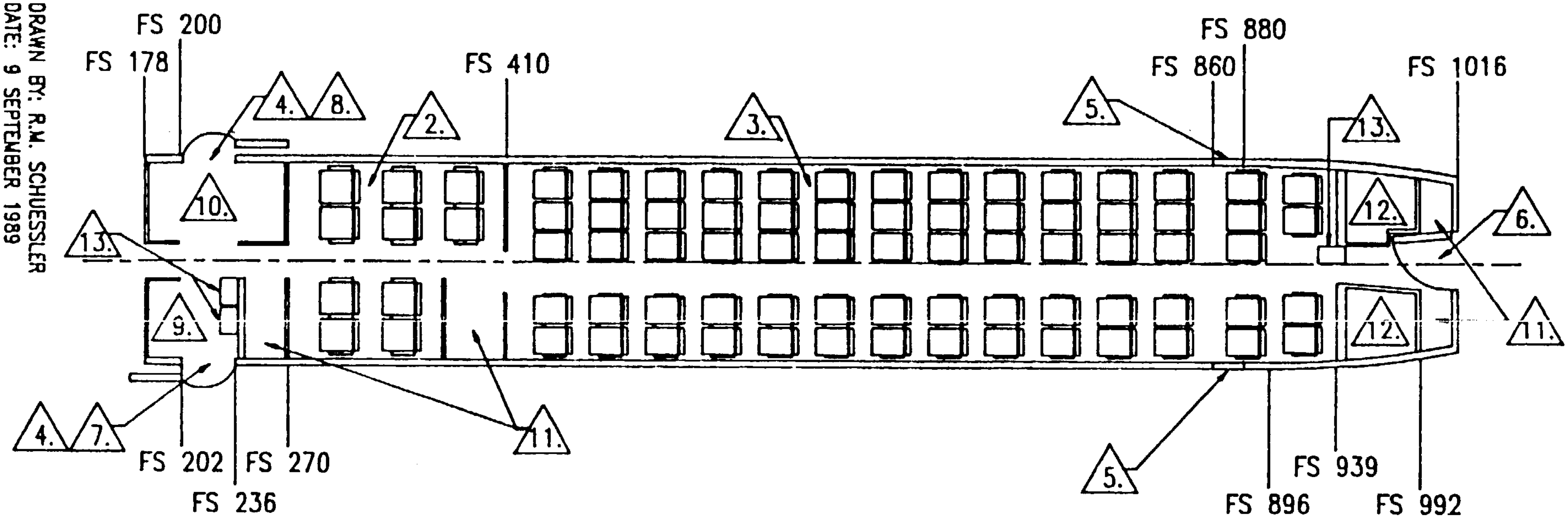


Figure 5. Avion Fuselage Cross Section

AVION DEVELOPMENT COMPANY AUBURN, ALABAMA			
AVION LAYOUT: FUSELAGE CROSS SECTION			
SIZE A	FCSM NO N/A	DWG NO AVION–XXX	REV
SCALE 1/2" = 1'		SHEET X OF X	

DRAWN BY: R.M. SCHUESSLER
DATE: 7 SEPTEMBER 1989



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DRAWN BY: R.M. SCHUESSLER
DATE: 9 SEPTEMBER 1989

AVION DEVELOPMENT COMPANY AUBURN, ALABAMA			
AVION LAYOUT: FUSELAGE/SEATING ARRANGEMENT			
SIZE A	FCSM NO N/A	DWG NO AVION–XXX	REV
SCALE 1/8" = 1'		SHEET X OF X	

Figure 6. Avion Fuselage/Seating Layout