Appendix 10.2

AN EXAMPLE OF AIRPLANE PRELIMINARY DESIGN PROCEDURE - JET TRANSPORT

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An Example of Airplane Preliminary Design Procedure - Jet Transport

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Abstract

The aim of this report, is to present an application of the preliminary design procedure followed in the course entitled “Airplane design(Aerodynamic)”. A 150 seater jet airplane cruising at M = 0.8, at 11 km altitude and having a gross still air range(GSAR) of 4000 km is considered[$]. The presentation is divided into eight sections

- Data collection
- Preliminary weight estimation
- Optimization of wing loading and thrust loading
- Wing design
- Fuselage design, preliminary design of tail surface and preliminary layout
- c.g. calculations
- Revised estimates of areas of horizontal and vertical tails
- Features of designed airplane
- Details of performance estimation

[$] This report was originally prepared in 2007. After the preparation of report, some changes have been incorporated in the text of the course material on “Airplane design (Aerodynamic)”. Hence, there are some differences, in detail, in the procedures followed in this report and those in the course material.

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1 Data collection

1.1 The design philosophy

The conceptual design forms the initial stage of the design process. In spite of the fact that there are numerous airplanes, each having its own special features,one can find common features underlying most of them. For example, the following aspects would dominate the conceptual design of a commercial transport jet.

1.1.1 Types of airplanes and market

The Civil transport jets could be classified in the following manner.

Class	No. of Seats	Typical GSAR (km)	Propulsion
B - 747 type	> 400	> 13000	High bypass Turbofan
B – 757 type	200 – 400	10000	High bypass Turbofan
B – 737 type	100 - 200	5000	Medium bypass Turbofan
Regionals	30 - 100	2000	Turboprop

Table 1: Classification of civil jet airplane

From the values of gross still air range(GSAR) in Table 1, it is clear that intercontinental flights would be restricted to the first two classes while the last two would handle bulk of the traffic in regional routes. The different classes cater to different sections of the market. One decides the range and payload (i.e. passengers) after identifying the target market. In this report, an airplane which caters to the traffic in regional routes is considered. A jet transport airplane with a gross still air range(GSAR) of 4000 km and single-class seating capacity of 150 passengers is considered. It is observed from Table 1, that this airplane belongs to the B-737 class. The data for similar airplanes is collected and used as the basis for making initial estimates.

The aim is to design an airplane that satisfies the following requirements.

Gross still air range = 4,000 km

No. of passengers = 150

Cruise Mach no. = 0.80

Altitude = 11,000 m

1.1.2 Budget and time constraints

Any design team would be required to work with a limited amount of funds and time. These could dictate various aspects of the design process. For example,innovations which could lead to spiralling budget may be shelved. Also, in case of highly competitive markets, the ability to get the airplane ready in the prescribed time frame is very crucial.The design team must ensure that cost and time over-runs are minimized to the extent possible.

1.1.3 Other constraints and standards

Some of the major demands on the design arise from the various mandatory and operational regulations. All commercial airplanes must satisfy the airworthiness requirements of various countries. Typically, each country has its own aviation authority (e.g, DGCA in India, EASA in Europe, FAA in USA). Airworthiness requirements would cover the following aspects of the airplane

1. Flight

   This includes performance items like take-off, climb, cruise, descent, landing and response to rough air. Also included are requirements of stability,controllability and maneuverability.

2. Structural

   Flight loads, ground loads, emergency landing conditions and fatigue evaluation

3. Powerplant

   Fire protection, auxillary power unit,air intake/exhaust,fuel systems,cooling.

4. Other

   Materials quality regulations and bird strike.

Passenger Safety is the primary aim behind these specifications. Additional route-specific constraints may have to be taken into account on a case-by-case basis. e.g, cruise altitude for airplanes flying over the Himalayas must be well over 8 km.

In addition to safety and operational requirements, the design must satisfy the environmental constraints. Two major environmental concerns are noise and emissions :

The Engines are the primary source of noise in an airplane. The airframe could also add to this.Maximum noise is produced during take-off and landing. This can be reduced by opting for a shallower approach, as this reduces the flight time spent near the airport. However, the reduction in noise may not be significant. The development of high-bypass turbofan engines has significantly reduced noise production.

The predominant source of emissions is the engine. The exhaust contains particles, various gases including carbon dioxide(CO_2) , water vapor (H_2O) , various oxides of nitrates(NO_x), carbon monoxide(CO),unburnt hydrocarbons and sulphur dioxide(SO_2). All components except CO_2 and H_2O are considered as pollutants. Again, as in the case with noise, emissions during landing and take-off are of particular concern due to effect on people near airports. Various aviation authorities have set limits on these emissions. The design team must adhere to such constraints.

1.2 Preliminary design

A careful look at the commercial transport jets in use, points out many common features amongst them. Some of these are :

(i) Medium bypass turbofans

This choice regarding the type of engine is due to the following reasons.

In the flight regime of Mach number between 0.6 to 0.85, turbofans give the best efficiency and moreover reduction in thrust output with speed is not rapid. Also, the noise generated by a medium bypass turbofan engine is considerably low. Following this trend a medium bypass turbofan is chosen as the powerplant.

(ii) Wing mounted engines

Though not a rule, wing mounted engines dominate the designs of top aircraft companies like Boeing and Airbus. Alternative designs could be adopted. But,given the experience gained with the wing mounted engines and the large data available a configuration with two wing mounted engines is adopted.

(iii) Swept back wings and a conventional tail configuration is chosen. Again, this choice is dictated by the fact that a large amount of data (is available) for such configurations.

1.2.1 Preliminary weight estimate

Given the number of passengers, the payload can be estimated in the following manner:

1. Include one cabin crew member for 30 passengers. This gives 5 crew members in the present case.
2. Include flight crew of pilot and co-Pilot.Thus the total of passenger + crew is 150+5+2 = 157.
3. Following Ref.4*, chapter 9, allow 110 kgf for each passenger (82 kgf weight per passenger with carry on baggage + 28 kgf of checkin baggage)

Thus, payload (W_{pay}) is 157 × 110 = 17270 kgf.

The gross weight of the airplane (W_g) is now estimated.

From data collection, (Table 2) the following is observed.

* The reference numbers in this report refer to those given at the end of this report (p105)

Aircraft	No. of passengers	Still air range(km)	W_{TO} or W_g (kgf)
737 – 300B	149	4185	60636
737 – 400B	168	3852	64671
737 – 700A	149	2935	60330

Table 2: Take off weight

Based on the above data, an initial weight of 60,000 kgf is chosen.

1.2.2 Wing parameters

To estimate the wing parameters, a value for wing loading (W/S) needs to be chosen.This is one of the most important parameters that not only decides the wing parameters but also plays an important role in the performance of the airplane. After considering similar airplanes, an initial estimate for (W/S) is taken as 5500 N/m^2. The other parameters of the wing are also chosen based on data of similar airplanes.

From aerodynamic considerations, it is desirable to have a large aspect ratio (A). However,structural considerations dictate a moderate value. As the structural design improves, the value of A increases. A value of 9.3. is chosen. Most modern airplanes have a value close to 9 (Table A).The taper ratio(λ) is roughly the same for all airplanes in this category. A value of 0.24 is chosen for λ.The wing quarter chord sweep($\Lambda_{c/4}$) is chosen as 25°.

Considering the above choices the following values are obtained.

$$S = W_g\left(\frac{S}{W}\right) = \frac{60000\times 9.81}{5500} = 107.02\ \text{m}^2 \quad (1)$$

The wing span(b) can be calculated from A and S

$$b = \sqrt{SA} = \sqrt{107.02\times 9.3} = 31.55\ \text{m} \quad (2)$$

The root chord (c_r) and tip chord(c_t) of the wing can be obtained using the following equations.

$$c_r = \frac{2S}{b(1+\lambda)} = \frac{2\times107.02}{31.55(1+0.24)} = 5.47 \text{ m} \tag{3}$$

$$c_t = 5.47\times0.24 = 1.31\,\text{m} \tag{4}$$

1.2.3 Empennage

As explained earlier, a conventional tail configuration has been chosen. The geometric parameters of the horizontal and vertical tails are obtained in this subsection.

The values of S_h/S and S_v/S from the data on similar airplanes(Table A) are:

$$\frac{S_h}{S}=0.31 \text{ and } \frac{S_v}{S} = 0.21$$

Hence,

$S_h = 0.31 \text{ x } 107.02 = 33.18 \text{ m}^2$

$S_v = 0.21 \text{ x } 107.02 = 22.47 \text{ m}^2$

The values of aspect ratios(A_h, A_v) , from the data collection (Table A), are chosen as :

$A_h = 5$ and $A_v = 1.7$. Consequently, the spans (b_h , b_v) are:

$$b_h = \sqrt{A_h S_h} = \sqrt{5\times33.18} = 12.88\,\text{m} \tag{5}$$

$$b_v = \sqrt{A_v S_v} = \sqrt{1.7\times22.47} = 6.18\,\text{m} \tag{6}$$

The chosen values of the taper ratios(λ_h,λ_v), from the data collection(Table A) are $\lambda_h = 0.26$ $\lambda_v = 0.3$. The root chord (c_{rh}, c_{rv}) and tip chords (c_{th}, c_{tv}) are:

$$c_{rh} = \frac{2S_h}{b_h(1+\lambda_h)} = \frac{2\times33.18}{12.88(1+0.26)} = 4.09\,\text{m} \tag{7}$$

$$c_{th} = \lambda_h c_{rh} = 0.26\times4.09 = 1.06\,\text{m} \tag{8}$$

$$c_{rv} = \frac{2S_v}{b_v(1+\lambda_v)} = \frac{2\times22.47}{6.18\times(1+0.3)} = 5.59\,\text{m} \tag{9}$$

$$c_{tv} = \lambda_v c_{rv} = 0.3\times5.59 = 1.68\,\text{m} \tag{10}$$

From the data collection, the quarter chord sweep back angles are: $\Lambda_h = 30^o$ and $\Lambda_v = 35^o$.

1.2.4 Control Surfaces

The following values are chosen based on the three-view drawings of similar airplanes and the data in Table A.

$S_{ele}/S_h = 0.22$

$S_{rud}/S_v = 0.25$

Hence,

$S_{ele} = 7.53\ m^2$

$S_{rud} = 5.8\ m^2$

Trailing edge flaps type : Fowler flaps

Leading edge high lift devices : slats

$S_{flap}/S = 0.17$

$S_{slat}/S = 0.10$

$b_{flap}/b = 0.74$

Area of T.E flaps = 18.98 m^2

Area of L.E slats = 11.60 m^2

$b_{flap} = 23.7$ m

1.2.5 Fuselage

Aerodynamic considerations demand a slender fuselage. But, passenger comfort and structural constraints limit the slenderness. From data collection $l_f/b = 1.05$ and $l_f / d_f = 8.86$ are chosen.

l_f = length of fuselage

d_f = diameter of fuselage

$l_f = 1.05 \times 31.55 = 33.6$ m (11)

$d_f = 33.6/8.86 = 3.79$ m (12)

1.2.6 Engines

Observing the thrust-to-weight ratio (T/W) of similar airplanes, a value of 0.3 is chosen.This implies a thrust requirement of :

$T = 0.3 \times 60000 \times 9.81 \approx 180$ kN or 90 kN per engine

The CFMI FM56-3-B1 model of turbofan engine is noted to be close to this engine requirement.

1.2.7 Landing Gear

A retractable tricycle type landing gear is chosen. It is the most commonly used landing gear in this category of airplanes. It is favoured for the following reasons.

1.During take-off and landing the weight of the plane is taken entirely by the rear wheels.

2. Tricycle landing gear has better lateral stability on ground than the bicycle type landing gear.

A total of 10 wheels - 2 below the nose and two pairs each on the sides(near the wing fuselage junction) are chosen. The location of the wheels is chosen from three-view drawings of similar airplanes.

1.2.8 Overall height

Based on the dimensions of Boeing 737 - 300, 400 and 500, the overall height is chosen as 11.13 m.

The preliminary three-view drawing of the airplane under design, is shown in Fig.4.

A	B	C	D	E	F	G	H	I
Manufacturer	AIR-BUS	AIR-BUS	BOE-ING	BOE-ING	BOE-ING	BOE-ING	BOE-ING	BOE-ING
Type	A319 -	A320 -	727-	737-	737-	737-	737-	737-
Model	100	200	200Adv	300	400	500	600	700
Initial service date	1995	1988	1970	1967	1967	1967	1998	1997
In service (ordered)								
Africa	2(1)	27(6)	58	14(1)	7	17	(7)	(2)
Middle East/Asia/ Pacific	(4)	162 (47)	52	194 (19)	142 (5)	49(2)	-	9(24)
Europe & CIS	48(82)	244 (105)	94	272 (12)	216(4)	145(1)	6(49)	21 (35)
North & South America	57 (264)	237 (146)	799	573 (11)	97(5)	165(1)	-	36 (146)
Total aircraft	107 (351)	670 (304)	1003	1053 (43)	462 (14)	376(4)	6(56)	66 (207)
Engine Manufacturer	CFMI	CFMI	P&W	CFMI	CFMI	CFMI	CFMI	CFMI
Model / Type	CFM 56-5A4	CFM 56-5A3	JTSD-15A	CFM56-3-B1	CFM56-3B-2	CFM56-3-B1R	CFM56 - &B1B	CFM56 -JB20
No.of engines	2	2	3	2	2	2	2	2
Static thrust (kN)	99.7	111.2	71.2	89.0	97.9	82.3	82.0	89.0
Operation-al Items:								
Accomodation								
Max.seats (single class)	153	179	189	149	170	130	132	149
Two class seating	124	150	136	128	146	108	108	128

Table A – Data on 150 seater category airplanes (Contd…)

(Source http://www.bh.com/companions/034074152X/)

A	B	C	D	E	F	G	H	I
Manufacturer	AIR-BUS	AIR-BUS	BOEI-NG	BOEI-NG	BOEI-NG	BOEI-NG	BOEI-NG	BOEI-NG
Type	A319-	A320-	727-	737-	737-	737-	737-	737-
Model	100	200	200Adv	300	400	500	600	700
No.abreast	6	6	6	6	6	6	6	6
Hold volume(m^3)	27.00	38.76	43.10	30.20	38.90	23.30	23.30	30.2
Volume per Passenger	0.18	0.22	0.23	0.20	0.23	0.18	0.18	0.20
Mass (kg):								
Ramp	64400	73900	95238	56700	63050	52620	65310	69610
Max.take-off	64000	73500	95028	56470	62820	52390	65090	69400
Max.landing	61000	64500	72575	51710	54880	49900	54650	58060
Zero-fuel	57000	60500	63318	47630	51250	46490	51480	54650
Max.payload	17390	19190	18597	16030	17740	15530	9800	11610
Max.fuel payload	5360	13500	24366	8705	13366	5280	7831	10996
Design payload	11780	14250	12920	12160	13870	10260	10260	12160
Design fuelload	13020	17940	35944	12441	15580	11170	18390	19655
Operational Empty	39200	41310	46164	31869	33370	30960	36440	37585
Weight Ratios								
Opsempty/ Max. T/O	0.613	0.562	0.486	0.564	0.531	0.591	0.560	0.542
Max. payload/ Max T/O	0.272	0.261	0.196	0.284	0.283	0.296	0.151	0.167
Max. fuel/ Max. T/O	0.295	0.256	0.255	0.281	0.253	0.303	0.316	0.296
Max.landing / Max. T/O	0.953	0.878	0.764	0.916	0.874	0.952	0.840	0.837
Fuel (litres):								
Standard	23860	23860	30622	20105	20105	20105	26024	26024
Optional			40068	23170	23170	23170		

Table A (Contd….)

A	B	C	D	E	F	G	H	I
Manufacturer	AIRBUS	AIRBUS	BOEING	BOEING	BOEING	BOEING	BOEING	BOEING
Type	A319-	A320-	727-	737-	737-	737-	737-	737-
Model	100	200	200Adv	300	400	500	600	700
DIMENSIONS								
Fuselage:								
Length(m)	33.84	37.57	41.51	32.30	35.30	29.90	29.88	32.18
Height(m)	4.14	4.14	3.76	3.73	3.73	3.73	3.73	3.73
Width(m)	3.95	3.95	3.76	3.73	3.73	3.73	3.73	3.73
Finess Ratio	8.57	9.51	7.00	7.40	7.40	7.40	7.40	7.40
Wing:								
Area(m^2)	122.40	122.40	157.90	91.04	91.04	91.04	124.60	124.60
Span(m)	33.91	33.91	32.92	28.90	28.90	28.90	34.30	34.30
MAC(in)	4.29	4.29	5.46	3.73	3.73	3.73	4.17	4.17
Aspect Ratio	9.39	9.39	6.86	9.17	9.17	9.17	9.44	9.44
Taper Ratio	0.340	0.240	0.309	0.240	0.240	0.240	0.278	0.278
Average (t/c)%			11.00	12.89	12.89	12.89		
¼ Chord Sweep(“)	25.00	25.00	32.00	25.00	25.00	25.00	25.00	25.00
High Lift Devices:								
Trailing Edge flaps type	F1	F1	F3	S3	S3	S3	S2	S2
Flap Span/ Wing Span	0.780	0.780	0.740	0.720	0.720	0.720	0.599	0.599
Area (m^2)	21.1	21.1	36.04					
Leading edge Flap type	slats	slats	slats/ flaps	slats/ flaps	slats/ flaps	slats/ flaps	slats/ flaps	slats/ flaps
Area (m^2)	12.64	12.64						
Vertical Tail:								
Area (m^2)	21.50	21.50	33.07	23.13	23.13	23.13	23.13	23.13
Height (m)	6.26	6.26	4.60	6.00	6.00	6.00	6.00	6.00
Aspect Ratio	1.82	1.82	0.64	1.56	1.56	1.56	1.56	1.56
Taper Ratio	0.303	0.303	0.780	0.310	0.310	0.310	0.310	0.310

Table A (Contd….)

A	B	C	D	E	F	G	H	I
Manufacturer	Airbus	Airbus	Boeing	Boeing	Boeing	Boeing	Boeing	Boeing
Model	100	200	200Adv	300	400	500	600	700
¼ Chord Sweep(^{o})	34.00	34.00	53.00	35.00	35.00	35.00	35.00	35.0
Tail Arm (L_v) (m)	10.67	12.53	14.20	13.68	14.90	12.90	13.55	14.7
S_v/S	0.176	0.176	0.209	0.254	0.254	0.254	0.186	0.186
S_vL_v/Sb	0.055	0.065	0.090	0.120	0.131	0.113	0.073	0.080
Horizontal Tail:								
Area (m^2)	31.00	31.00	34.93	31.31	31.31	31.31	32.40	32.40
Span (m)	12.45	12.45	10.90	12.70	12.70	12.70	13.40	13.40
Aspect Ratio	5.00	5.00	3.40	5.15	5.15	5.15	5.54	5.54
Taper Ratio	0.256	0.256	0.380	0.260	0.260	0.260	0.186	0.186
¼ Chord Sweep(^{o})	29.00	29.00	36.00	30.00	30.00	30.00	30.00	30.00
Tail Arm(m) (L_h)	11.67	13.53	20.10	14.78	16.00	14.00	13.58	14.73
S_h/S	0.253	0.253	0.221	0.344	0.344	0.344	0.360	0.260
$S_hL_h/S\bar{c}$	0.689	0.799	0.814	1.363	1.475	1.291	0.847	0.919
Under Carriage:								
Track(m)	7.60	7.60	5.72	5.25	5.25	5.25	5.70	5.7
Wheelbase(m)	12.60	12.63	19.28	12.40	14.30	11.00		12.4
Turning radius (m)	20.60	21.90	25.00	19.50				19.5
No. of wheels (nose, main)	2;4	2;4	2;4	2;4	2;4	2;4	2;4	2;4
Main Wheel Diameter(m)	1.143	1.143	1.245	1.016	1.016	1.016	1.016	1.016
Main Wheel Width(m)	0.406	0.406	0.432	0.362	0.368	0.368	0.368	0.368
Nacelle:								
Length(m)	4.44	4.44	7.00	4.70	4.70	4.70	4.70	4.70
Max.width(m)	2.37	2.37	1.50	2.00	2.00	2.00	2.06	2.06
Spanwise Location	0.338	0.338	-	0.340	0.340	0.340	0.282	0.282

Table A (Contd….)

Manufacturer	AIR-BUS	AIR-BUS	BOE-ING	BOE-ING	BOE-ING	BOE-ING	BOE-ING	BOE-ING
Type	A319-	A320-	727-	737-	737-	737-	737-	737-
Model	100	200	200Adv	300	400	500	600	700
PERFORMANCE								
Loadings:								
Max.power load(kgf/kN)	320.96	330.49	444.89	317.25	320.84	318.29	396.89	389.89
Max wing load (kgf/m^2)	522.88	600.49	601.82	620.28	690.03	575.46	522.39	556.98
Thrust/Weight Ratio	0.3176	0.3084	0.2291	0.3213	0.3177	0.3203	0.2568	0.2615
Take-off(m):								
ISA sea level	1750	2180	3033	1939	2222	1832		
ISA +20^{o}C SL	2080	2590	3658	2109	2475	2003	1878	2042
ISA 5000 ft	2360	2950	3962	2432		2316		
ISA +20^{o}C 5000ft	2870	4390	4176	2637		2649		
Landing (m):								
ISA sea level	1350	1440	1494	1396	1582	1362	1268	1356
ISA +20^{o}C SL	1350	1440	1494	1396	1582	1362	1268	1356
ISA 5000ft	1530	1645	1661	1576	1695	1533		
ISA +20^{o}C 5000 ft	1530	1645	1661	1576	1695	1533		
Speeds (kt*/Mach):								
V_2	133	143	166	148	159	142		
V_{app}	131	134	137	133	138	130		
V_{no}/M_{no}	381/ M 0.89	350/ M 0.82	390/ M 0.90	340/ M 0.82	340/ M 0.82	340/ M 0.82	392/ M 0.84	392/ M 0.84
V_{ne}/M_{ne}	350/ M 0.82	381/ M 0.89	M0.95					
C_{Lmax}(T/O)	2.58	2.56	1.90	2.47	2.38	2.49		
C_{Lmax}(Landing @MLM)	2.97	3.00	2.51	3.28	3.24	3.32		

Table A (Contd….)

* 1 kt = 1.853 kmph

Manufacturer	Airbus	Airbus	Boeing	Boeing	Boeing	Boeing	Boeing	Boeing
Type	A319-	A320	727-	737-	737-	737-	737-	737-
Model	100	200	200Adv	300	400	500	600	700
Max.cruise:								
Speed (kt)*	487	487	530	491	492	492		
Altitude (ft)	33000	28000	25000	26000	26000	26000	41000	41000
Fuel mass consumption (kg/h)	3160	3200	4536	3890	3307	3574		
Long range Cruise:								
Speed (kt)	446	448	467	429	430	429	450	452
Altitude (ft)	37000	37000	33000	35000	35000	35000	39000	39000
Fuel mass consumption (kg/h)	1980	2100	4309	2250	2377	2100	1932	2070
Range(nm*):								
Max payload	1355	637	2140	1578	1950	1360		
Design range	1900	2700	2400	2850	2700	1700	3191	3197
Max fuel (+ payload)	4158	3672		3187	2830	3450	3229	3245
Ferry range								
Design Parameters:								
W/SC_{Lmax} (N/m^2)	1726.69	1962.27	2356.82	1852.54	2090.56	1701.59		
W/SC_{LtoST}	2071.39	2423.85	3918.96	2196.64	2506.93	2024.27		
Fuel/pax/ (kg)	0.0553	0.0443	0.1101	0.0341	0.0395	0.0608	0.0534	0.0480
Seats Range (seats nm)	235600	405000	326400	364800	394200	183600	344628	409216

1 kt = 1.853 kmph ; 1 nm = 1.853 km

Table A – Data on 150 seater category airplanes (Concluded)

(Source http://www.bh.com/companions/034074152X/)

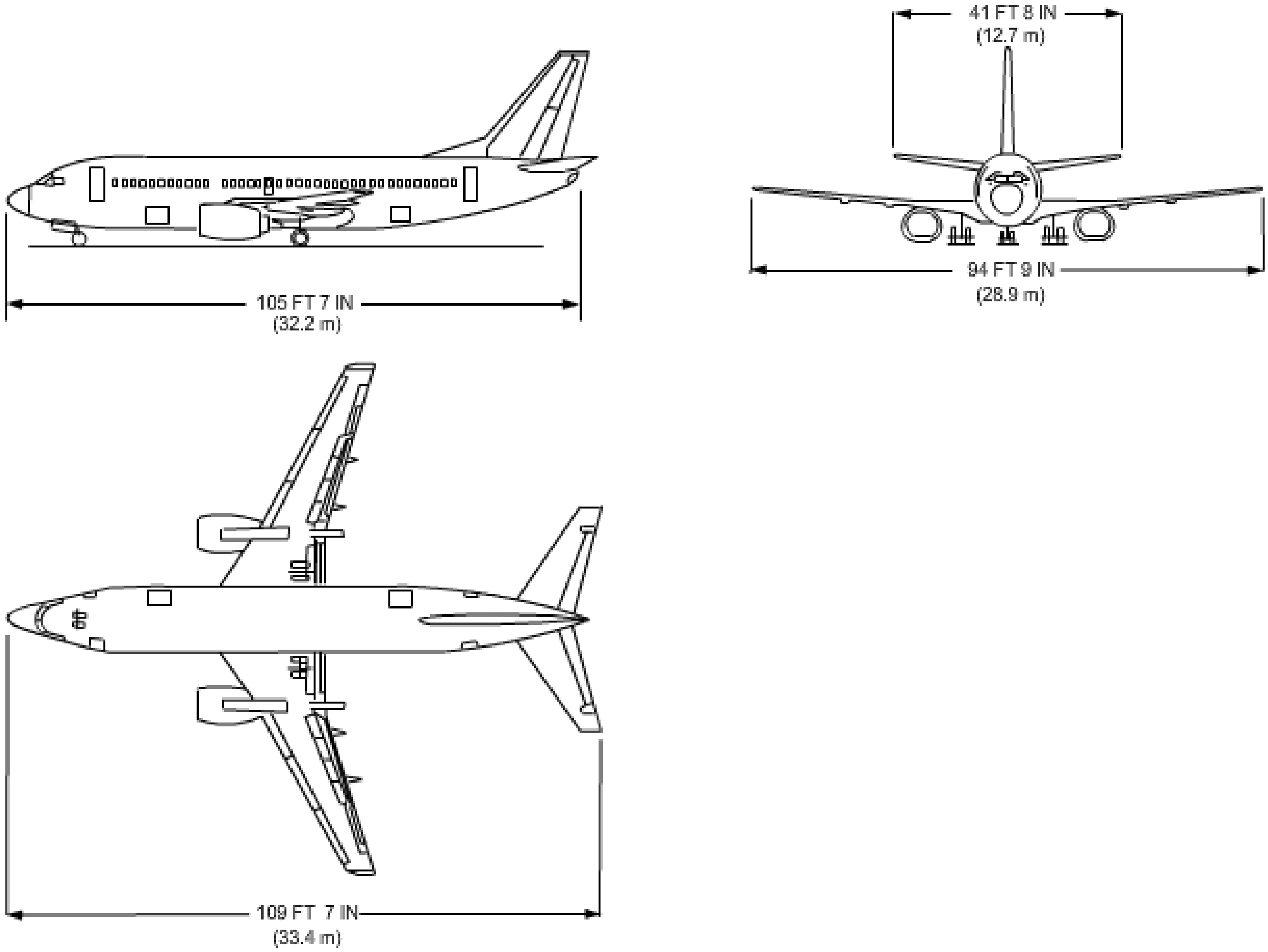

Fig.1 Three-view drawing of Boeing 737-300

Adapted from : http://www.the-blueprints.com

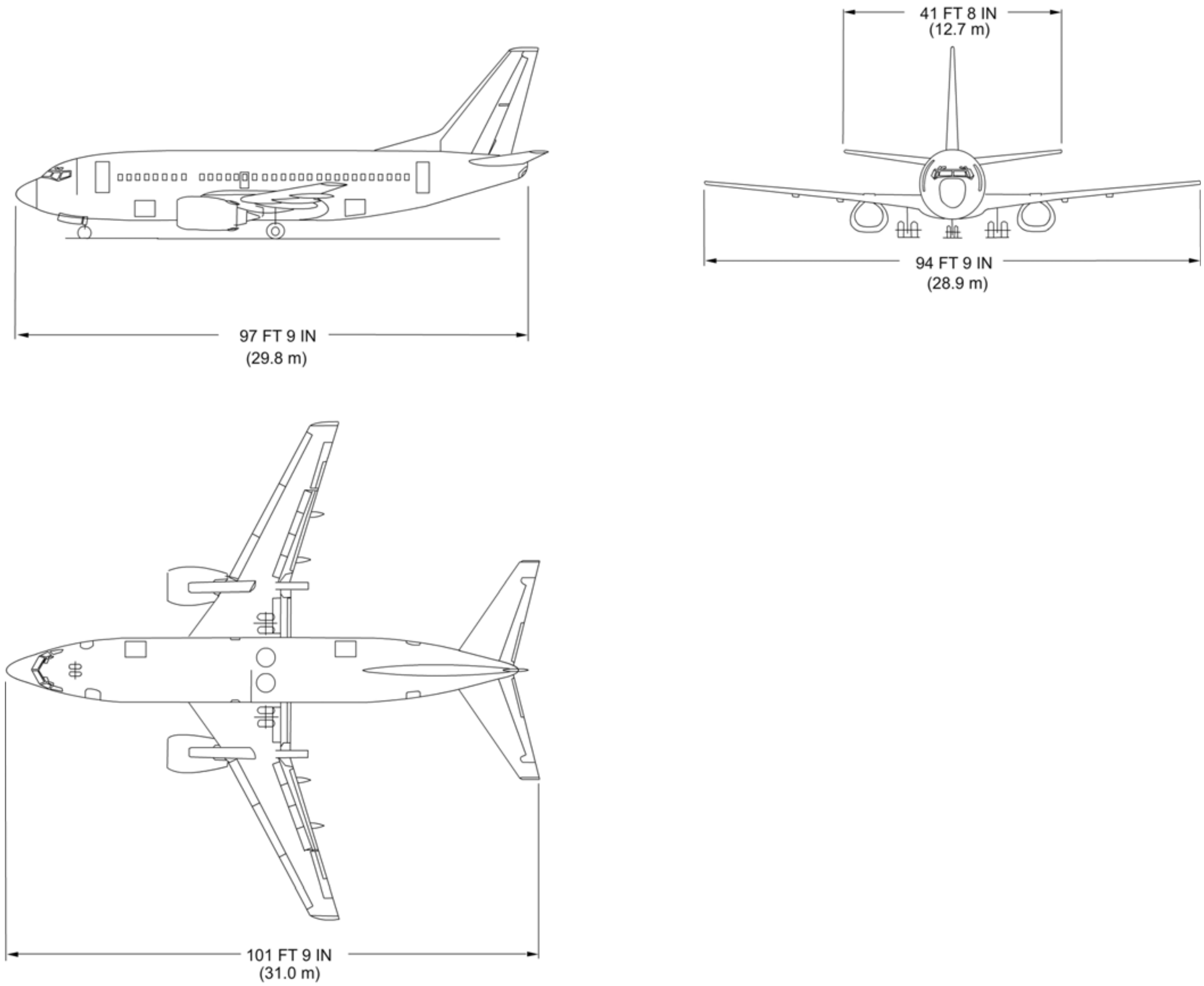

Fig. 2 Three-view drawing of Boeing 737-500

Adapted from : http://www.plans.aerofred.com/

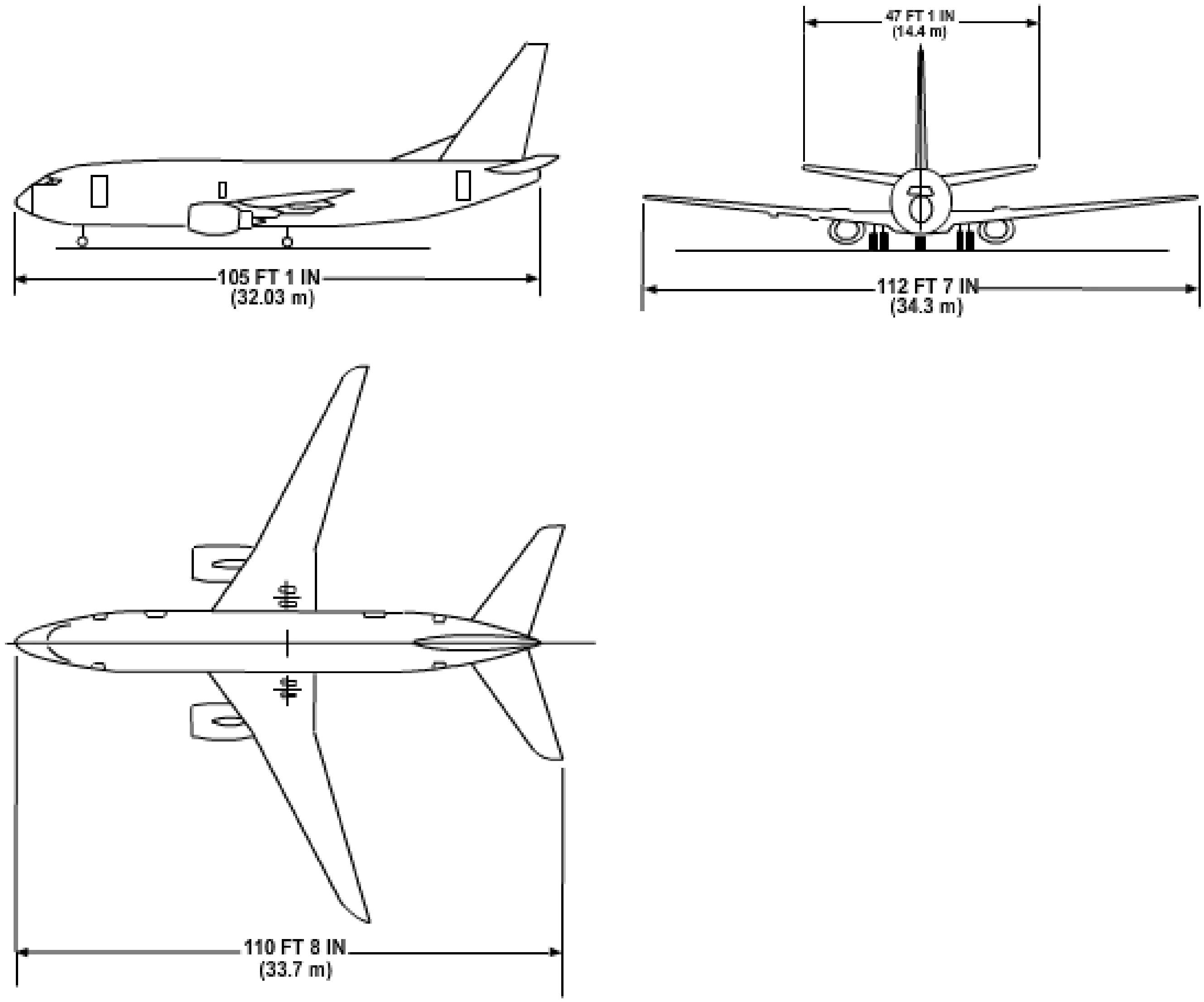

Fig. 3 Three-view drawing of Boeing 737-700

Adapted from : http://www.the-blueprints.com

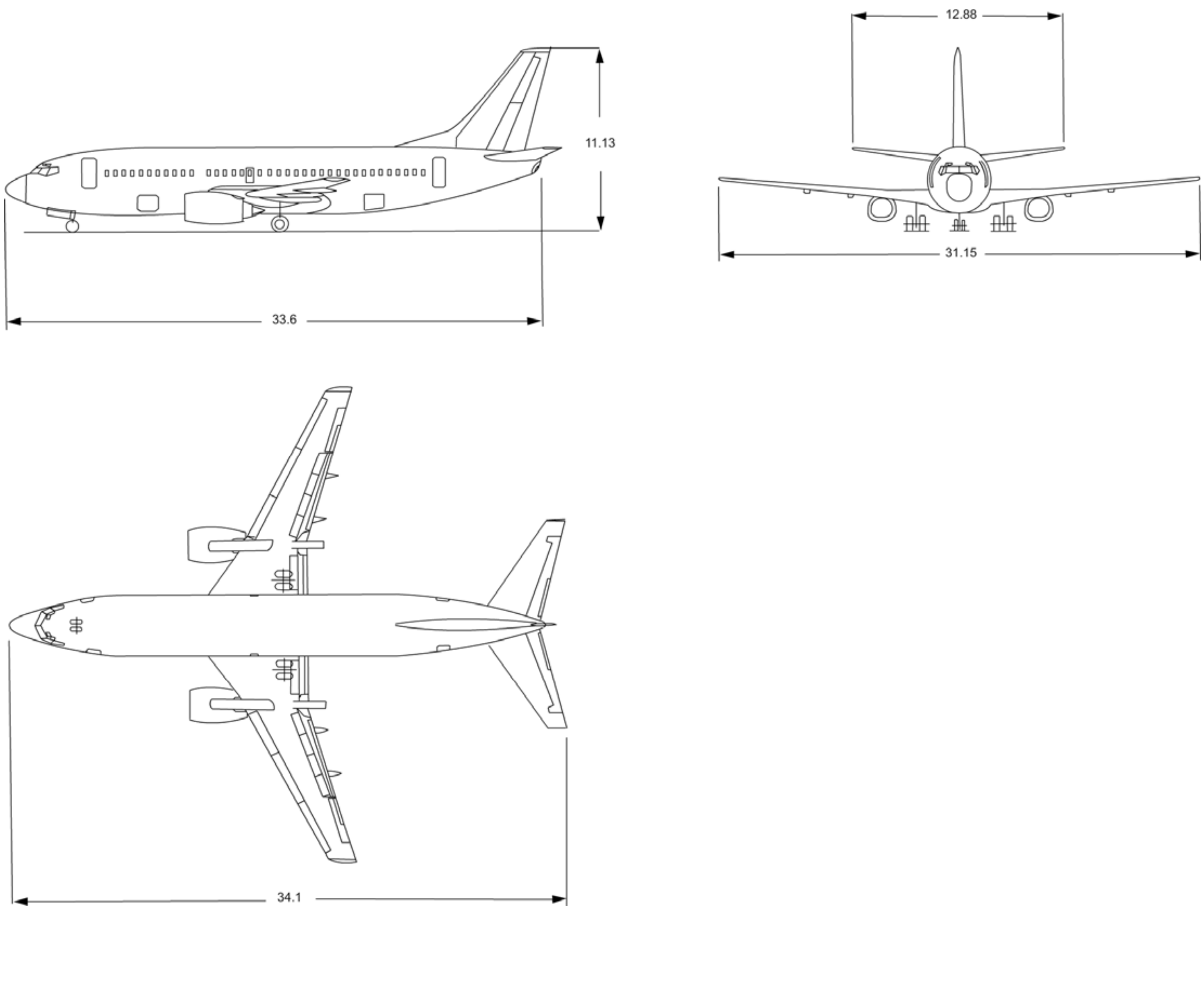

Fig.4 Preliminary three-view of the airplane under design

2 Revised weight estimation

In the previous section, an initial estimate for the aircraft parameters has been carried out. In this section a revised weight is obtained by estimating fuel weight and empty weight. According to Ref.4, chapter 3, the gross weight (W_g or W_0) is expressed as :

$W_g = W_{pay} + W_{crew} + W_f + W_e$; W_f = weight of fuel and W_e = empty weight (12a)

Dividing by W_g gives:

$$1 = \left(\frac{W_{pay} + W_{crew}}{W_g}\right) + \frac{W_f}{W_g} + \frac{W_e}{W_g} \qquad (12b)$$

To obtain revised estimate of W_g, (W_f / W_g) and (W_e / W_g) are estimated in sections 2.1 and 2.2.

Subsequently, $$W_g = \frac{W_{pay} + W_{crew}}{1-\left(\frac{W_f}{W_g} + \frac{W_e}{W_g}\right)} \qquad (12c)$$

2.1 Fuel fraction estimation

The fuel weight depends on the mission profile and the fuel required as reserve. The mission profile for a civil jet transport involves:

Take off

Climb

Cruise

Loiter before landing

Descent

Landing

2.1.1 Warm up and take-off

The weight of the airplane at the start of take-off is W_0 and the weight of the airplane at the end of the take-off phase W_1. The ratio (W_1/W_0) is estimated using the guidelines given in Ref.4, chapter 3.

$$\frac{W_1}{W_0} = 0.97$$

2.1.2 Climb

The weight of the airplane at the start of climb is W_1 and the weight of the airplane at the end of the climb phase is W_2. The ratio (W_2/W_1) for this phase is estimated following the guidelines given in Ref.4, chapter 3.

$$\frac{W_2}{W_1}=0.985$$

2.1.3 Cruise

The weight of the airplane at the start of cruise is W_2 and the weight of the airplane at the end of the cruise phase is W_3. The ratio (W_3/W_2) for the cruise phase of flight is calculated using the following expression from Ref.4, chapter 3.

$$\frac{W_3}{W_2} = \exp\left[\frac{-RC}{V_{cruise}(L/D)_{cruise}}\right];\ C = \text{TSFC during cruise} \qquad (13)$$

Gross still air range (GSAR) of the airplane is prescribed as 4000 km. Hence, the

Safe range during cruise $\approx \frac{GSAR}{1.5} = \frac{4000}{1.5} = 2667\,\text{km}$

$(L/D)_{max}$ is taken as 18 from Fig. 3.6 of Ref.4. This value corresponds to an average value for civil jet airplanes.

As prescribed in Ref.4, chapter 3,

$$(L/D)_{cruise} = 0.866(L/D)_{max} \qquad (14)$$

Hence, $(L/D)_{cruise} = 0.866 \times 18 = 15.54$

The allowances for (a) additional distance covered due to head wind during cruise and (b) provision for diversion to another airport in the event of landing being refused at the destination, are obtained as follows.

Head wind is taken as 15 m/s.

The cruise is at M = 0.8 at 11 km altitude. The speed of sound at 11 km altitude is 295.2 m/s. Hence, V_{cruise} is 236 m/s or 849.6 kmph

The time to cover the cruise range of 2667 km at V_{cr} of 849.6 km/hr is :

$$\text{Time} = \frac{2667}{849.6} = 3.13 \text{ hours}$$

Therefore, with a head wind of 15 m/s or 54 km/hr the additional distance needed to be accounted for is :

Additional distance = 54 × 3.13 = 169 km

The allowance for diversion to another airport is taken as 400 km.

The total extra distance that is to be accounted for : 169 + 400 = 569 km.

The total distance during the cruise (R) = 2667 + 569 = 3236 km.

Following guidelines from Ref.4 chapter 3, TSFC during cruise is taken as $0.6\ hr^{-1}$.

Substituting the various values in Eq.(13) yields,

$$\frac{W_3}{W_2} = \exp\left[\frac{-3236 \times 0.6}{849.6 \times 15.59}\right] = 0.863$$

2.1.4 Loiter

Sometimes the permission to land at the destination is not accorded immediately and the airplane goes around in circular path above the airport. This phase of flight is called loiter. The weight of the airplane at the end of loiter is W_4. The ratio (W_4 / W_3) is calculated using the following expression from Ref.4, chapter 3.

$$\frac{W_4}{W_3} = \exp\left[\frac{-E \times TSFC}{(L/D)}\right] \tag{15}$$

During Loiter, the airplane usually flies at a speed corresponding to $(L/D)_{max}$. Hence, that value is used in Eq.(15). The loiter time (E) is taken as 30 minutes (0.5 hr). TSFC is taken as $0.6\ hr^{-1}$.

Hence,

$$\frac{W_4}{W_3} = \exp\left[\frac{-0.5 \times 0.6}{18}\right] = 0.983$$

2.1.5 Landing

The weight of the airplane at the start of landing is W_4 and the weight of the airplane at the end of the landing is W_5. Following the guidelines specified by Ref.4, chapter 3, the ratio W_5 / W_4 is taken as :

$$\frac{W_5}{W_4} = 0.995$$

Finally,

$$\frac{W_5}{W_g} = \frac{W_5}{W_0} = 0.97 \times 0.985 \times 0.863 \times 0.983 \times 0.995 = 0.806$$

Allowing for a reserve fuel of 6%, the fuel fraction for the flight, ζ or (W_f/ W_g) is given by :

$$\frac{W_f}{W_g} = \zeta = 1.06\left(1-\frac{W_5}{W_0}\right) = 0.205$$

2.2 Empty weight fraction

The ratio (W_e/ W_g) is called empty weight fraction. To determine this fraction, the method given in Ref.4, chapter 3, is used. The relationship between W_e/W_g and W_g for a jet transport is as follows.

$$\frac{W_e}{W_g} = 1.02(2.202W_g)^{-0.06} \text{ ; where, } W_g \text{ is in kgf} \tag{16}$$

From subsection 1.2.1, (W_{pay} + W_{crew}) is 17270 kgf.

Substituting in Eq.(12a) :

$$W_g = \frac{W_{pay} + W_{crew}}{1-\left(W_f / W_g\right)-\left(W_e / W_g\right)} = \frac{17270}{1 - 0.205 - 1.202(2.202W_g)^{-0.06}} \tag{16a}$$

Both the sides of Eq.(16a) involve W_g. The solution of this equation is obtained by an iterative procedure. A guessed value of gross weight, (W_g (guess)), is substituted on the right hand side of Eq.(16a) and value of W_g is obtained. If the obtained value and the guest value are different, a new guest value is used. The steps are repeated till the guest value and the obtained value are same. The procedure is presented in Table 3.

W_g (guess)	W_e/W_g(from eq.(16))	W_g(from eq.(16a))
60000	0.50274	59090
59090	0.50320	59184
59184	0.50315	59174
59174	0.50316	59175
59175	0.50316	59175

Table 3: Iterative procedure for W_g

Hence, the gross weight W_g is obtained as:

W_g = 59, 175 kgf

The important weight ratios are:

$$\frac{W_e}{W_g} = 0.503\,;\ \frac{W_f}{W_g} = 0.205\,;\ \frac{W_{pay} + W_{crew}}{W_g} = 0.292$$

3 Wing loading and thrust loading

The thrust-to-weight ratio (T/W) and the wing loading(W/S) are the two most important parameters affecting aircraft performance. Optimization of these parameters forms a major part of the design activities conducted after initial weight estimation. For example, if the wing loading used for the initial layout is low, then the wing area would be large and there would be enough space for the landing gear and fuel tanks. However, it results in a heavier wing. Wing loading and thrust-to-weight ratio are interconnected for a number of critical performance items, such as take-off distance, maximum speed, climb, range etc. These two are often the design drivers. A requirement for short take-off can be met by using a large wing (low W/S) with a relatively low T/W. On the other hand, the same take-off distance could be met with a high W/S along with a higher T/W.

In this section, different criteria are used to optimize the wing loading and thrust loading.

Wing loading affects stalling speed, climb rate, take-off and landing distances, minimum fuel required for range and turn performance.

Similarly, a higher thrust loading would result in more cost which is undesirable. However, it would also lead to enhanced climb performance.

Hence, a trade-off is needed while choosing W/S and T/W. Optimization of W/S and T/W based on various considerations is carried out in the following subsections.

3.1 Landing distance consideration

To decide the wing loading from landing distance consideration the landing field length needs to be specified. Based on data collection of similar airplanes in Table A, the landing field length is chosen to be 1425 m.

s_{land} = 1425 m

Next, the C_{Lmax} of the airplane is chosen. The Maximum lift coefficient depends upon the wing geometry, airfoil shape, flap span and geometry, leading edge slot or slat geometry, Reynolds number, surface texture and interference from other parts of the airplane such as the fuselage, nacelles or pylons. Reference 4, chapter 5 provides a chart for C_{Lmax} as a function of $\Lambda_{c/4}$ for different types of high lift devices. For the airplane under design, it is decided to use Fowler flap and slat as the high lift devices. This gives C_{Lmax} of 2.5 for a $\Lambda_{c/4} = 25^o$.

To calculate W/S based on landing considerations, the following formula is used.

$$\frac{W}{S} = \frac{1}{2}\rho V_s^2 C_{Lmax} \qquad (17)$$

The stalling speed V_s is estimated in the following manner.

(a) s_{land} is prescribed as = 1425 m = 4675.2 feet

(b) According to Ref.4, chapter 3, the approach speed (V_a) in knots is related to the landing distance(s_{land}) in feet as,

$$V_a(\text{in knots}) = \sqrt{\frac{s_{land}(\text{in feet})}{0.3}}$$

(c) $V_a = 1.3\ V_s$

In the present case

(V_a) (in knots) = $\sqrt{\dfrac{4675.2}{1.3}}$ = 124.84 kts = 64.25 m/s

Hence, $V_s = \dfrac{64.25}{1.3} = 49.4\ ms^{-1}$ (18)

Now, using this value for V_s in Eq.(17),

$$\left(\frac{W}{S}\right)_{land} = 3743\ Nm^{-2}$$

From Ref.4, chapter 3, for this type of airplane, W_{land} = 0.85 W_{TO}. Hence, W/S based on W_{TO} is :

$$\left(\frac{W}{S}\right)_{TO} = \frac{1}{0.85}\left(\frac{W}{S}\right)_{Land} = 4403\ Nm^{-2}$$

Allowing $\pm$ 10 % variation in V_s, gives the following range of wing loadings for which the landing performance is near optimum:

$3639 < p < 5328\ N/m^2$

3.2 Maximum speed (V_{max}) consideration

The optimization of wing loading from consideration of V_{max} is carried out through the following steps.

(I) For jet transport airplanes, V_{max} is sometimes decided based on the value of maximum Mach number (M_{max}). In turn M_{max} is taken as :

$M_{max} = M_{cruise} + 0.04$

Hence,

M_{max} = 0.80 + 0.04 = 0.84 and V_{max} = 0.84 x 295.2 = 248 m/s

(II) The estimation of drag polar and its alternate representation are carried out as follows. It may be noted that the procedure given below is slightly different from that given in subsection 4.4.1 of Ref.5.

The drag polar is expressed as :

$$C_D = C_{DO} + KC_L^2 \tag{19}$$

where,

$$K = \frac{1}{\pi A e}; \quad e = \text{Oswald efficiency factor} \qquad (20)$$

Following Ref.4, chapter 2, C_{DO} is given as :

$$C_{DO} = C_{fe} \times \frac{S_{wet}}{S}, \qquad (21)$$

where, C_{fe} = equivalent skin friction drag coefficient ; S_{wet} = Wetted area of the airplane.

From Fig 2.5 of Ref.4, S_{wet}/S = 6.33.

The estimation of K is carried out next and then the value of C_{DO} is deduced using the earlier assumption that $(L/D)_{max}$ is 18.

Estimation of K:

The value of Oswald efficiency factor (e) is estimated from Ref.6, chapter 2 as :

$$\frac{1}{e} = \frac{1}{e_{wing}} + \frac{1}{e_{fuse}} + 0.05 \qquad (22)$$

From Ref.12, chapter 1,

e_{wing} = 0.84 for an unswept wing of A = 9.3 and λ = 0.24.

From Ref.9, chapter 7, e_{wing} for a swept wing is :

$$\left(e_{wing}\right)_{\Lambda} = \left(e_{wing}\right)_{\Lambda=0} \cos\left(\Lambda - 5\right)$$

Hence, in the present case, $e_{wing} = 0.84 \cos(25-5) = 0.7893$ (23)

From Ref.6, chapter 2 $\frac{1}{e_{fuse}} = 0.1$

Finally, $\frac{1}{e} = \frac{1}{0.7893} + 0.1 + 0.005$

or $e = 0.707$

$$K = \frac{1}{\pi \times 9.3 \times 0.707} = 0.0482$$

To get C_{DO}, it is recalled that $(L/D)_{max} = 18$ was assumed in subsection 2.1.3.

Further,

$$(L/D)_{max} = \frac{1}{2\sqrt{C_{D0}K}} \tag{24}$$

Hence,

$$C_{D0} = \frac{1}{4K(L/D)^2_{max}} = \frac{1}{4\times 0.0482\times 18^2} = 0.0161$$

Using this value of C_{D0} in Eq.21, gives C_{fe} as :

$$C_{fe} = \frac{0.0161}{6.33} = 0.00254 \tag{25}$$

Hence, the drag polar is :

$$C_D = 0.0161 + 0.0482\, C_L^2 \tag{25a}$$

Alternate representation of drag polar :

To obtain the optimum W/S based on maximum speed, the method given in Ref.7, is followed. It is also described in section 4.4 of Ref.5. The drag polar is expressed as :

$$C_D = F_1 + F_2\,p + F_3\,p^2 \tag{26}$$

where,

$$F_1 = C_{fe}\left(1 + \frac{S_{ht}}{S} + \frac{S_{vt}}{S}\right)\left(\frac{S_{wet}}{S}\right)_w = C_{fe}K_t \tag{27}$$

$$F_2 = \frac{(C_{D0} - F_1)}{W/S} \tag{28}$$

$$F_3 = \frac{K}{q^2} \tag{29}$$

Two values of F_1, F_2 and F_3 are obtained as follows.
From section 1.2 and subsections 1.2.1 to 1.2.7:

$$\frac{S_{ht}}{S} = 0.31$$

$$\frac{S_{vt}}{S} = 0.21$$

Hence,

$$K_t = 1 + \frac{S_{ht}}{S} + \frac{S_{vt}}{S} = 1 + 0.31 + 0.21 = 1.52$$

$$(C_{Do})_W = C_{fe}\left(\frac{S_{wet(exposed)}}{S}\right)_W \tag{30}$$

To calculate ($S_{wet(exposed)}/S$), the dimensions of the exposed wing are obtained as follows. From subsection 1.2.2, the parameters of the equivalent trapezoidal wing (ETW)are :

$S = 107.02\ m^2$
$\lambda = 0.24$
$A = 9.3$
$b = 31.55\ m$
$c_r = 5.47\ m$
$c_t = 1.31\ m$
$\Lambda_{c/4} = 25^o$

Hence, for ETW, the chord distribution is givenby :

$$c(y) = c_r - \frac{c_r - c_t}{b/2} = 5.47 - 0.264y$$

Taking fuselage diameter of 3.79 m, the chord at y = 1.895 m is the root chord of the exposed wing, ($c_{r(exposed)}$) i.e.

$c_{r(exposed)} = = 5.47 - 0.264 \times 1.895 = 4.97$ m

Semi-span of the exposed wing is $= \frac{31.55}{2} - \frac{3.79}{2} = 13.89$ m

Hence, $S_{exposedwing} = \frac{1}{2}(4.97 + 1.31) \times 13.89 \times 2 = 87.23\ m^2$

The wetted area of the exposed wing ($S_{wet(exposed\ wing)}$) is approximated as :

$$S_{wet(exposed\ wing)} = 2S_{exposedwing}\left\{1 + 1.2(t/c)_{avg}\right\} \tag{31}$$

Assuming $(t/c)_{avg}$ of 12.5%,

$$S_{wet(exposed\ wing)} = 2 \times 87.23 \times \left\{1 + 1.2 \times (0.125)\right\} = 200.63\ m^2$$

Hence,

$$(C_{D0})_W = 0.0025 \times \frac{200.63}{107.02} = 0.004687$$

$F_1 = 1.52 \times 0.004687 = 0.007124$

$$F_2 = \frac{C_{Do} - F_1}{W/S} = \frac{0.0161 - 0.007124}{5500} = 1.632 \times 10^{-6} \, m^2/N$$

The above drag polar will not be valid at M greater than the M_{cruise}.

Hence, the drag polar (values of C_{DO} and K) at M_{max} needs to be estimated .

The drag divergence Mach number (M_D) for the aircraft is fixed at M = 0.82 which is 0.02 greater than M_{cruise}. This would ensure that there is no wave drag at M_{cruise} of 0.80. To estimate the increase in C_{DO} from M = 0.80 to M = 0.84, a reasonable assumption is that the slope of the C_{DO} vs M curve remains constant in the region between M = 0.82 and M = 0.84.

The value of this slope is 0.1 at M = 0.82. Hence, the increase in C_{DO} is estimated as 0.02 × 0.1 = 0.002.

From the data on B 787 available in website [Ref.2], it is observed that the variation in K is not significant in the range M = 0.82 to 0.84. Hence, the value of K is retained as in subcritical flow. However, better estimates are used in performance calculations presented later (subsection 9.3.2).

Consequently, the drag polar that is valid at M_{max} is estimated as :

$$C_D = 0.0181 + 0.0482 C_L^2 \tag{32}$$

The change in the C_{DO} is largely due to change in the zero lift drag of the wing, horizontal tail and vertical tail. This means that the change in C_{DO} affects the value of F_1 alone.

Hence, at M_{max} , F_1 = 0.009124

The value of F_3 depends on the dynamic pressure at V_{max}.

$$V_{max} = M_{max} \times \sqrt{\text{speed of sound at } h_{cruise}} = 0.84 \times 295.2 = 248 \, m/s$$

$$q_{max} = \frac{1}{2} \rho V_{max}^2 = 0.5 \times 0.364 \times 248^2 = 11200.95$$

$$F_3 = \frac{0.0482}{11200.95^2} = 3.84 \times 10^{-10} \, m^4/N^2$$

The optimum value of W/S, from V_{max} consideration, is the wing loading (p) which minimises thrust required for V_{max} . The relation between the thrust required for V_{max} and p is:

$$\bar{t}_{Vmax} = q_{max}\left(\frac{F_1}{p} + F_2 + F_3 p\right) \; ; \; \bar{t}_{Vmax} = \text{(thrust required for } V_{max}\text{) / W} \qquad (33)$$

Differentiating Eq.(33) and equating to zero gives the optimum wing loading ($p_{optimum}$) i.e.

$$\frac{\partial \bar{t}_{vmax}}{\partial p} = q_{max}\left(\frac{F_1}{p^2} + F_3\right) = 0$$

$$\text{Or } p_{optimum} = \sqrt{\frac{F_1}{F_3}}$$

Hence, in the present case : $p_{optimum} = \sqrt{\dfrac{0.009124}{3.84 \times 10^{-10}}} = 4873.31\, N/m^2$

The value of $\bar{t}_{Vmax}$ with $p = p_{optimum}$ is found from Eq.(33) as :

$$\bar{t}_{Vmax} = 11200.95\left(\frac{0.009124}{4873.31} + 1.632 \times 10^{-6} + 3.84 \times 10^{-10} \times 4873.31\right) = 0.06022 \; (33a)$$

If the value of $\bar{t}_{Vmax}$ can be permitted to be 5 % higher than the minimum in Eq.(33a), i.e. 0.0632, Eq. (33) gives two values of p viz.

$p_1 = 3344\ Nm^{-2}$

$p_2 = 7101\ Nm^{-2}$

Thus, any value of p between 3344 and 7101 would be acceptable from V_{max} considerations with a maximum of 5% deviation from the optimum.

3.3 $(R/C)_{max}$ consideration

The value for $(R/C)_{max}$ at sea level is chosen as 700 m/min (11.67 m/s) which is typical for passenger airplanes.The thrust required for climb at chosen flight speed(V) is related to (R/C) in the following manner(section 4.6 of Ref.5).

$$\bar{t}_{R/c} = \frac{R/C}{V} + \frac{q}{p} C_D \; ; \; \bar{t}_{R/c} = \text{(Thrust required for climb) / W} \qquad (34)$$

However, C_D can be represented as :

$$C_D = F_1 + F_2\, p + F_3 p^2 \qquad (35)$$

$$q = \frac{1}{2}\rho_0\sigma V^2 \qquad (36)$$

Hence,

$$\overline{t}_{R/c} = \frac{R/C}{V} + \frac{1}{2}\rho_0\sigma\frac{V^2}{p}\left(F_1 + F_2 p + F_3 p^2\right) \qquad (37)$$

The flight speed for optimum climb performance is generally not high and value of F_1 corresponding to its value for $M < M_{cruise}$ is appropriate. However, F_3 is a function of the dynamic pressure and depends on chosen flight velocity.
The aim here is to find the minimum sea level static thrust ($\overline{t}_{SR/C}$) for various values of V and then choose the minimum amongst the minima. For a chosen V , differentiating Eq.(37) with p and equating to 0 gives the optimum wing loading as

$$p_{opt} = \sqrt{\frac{F_1}{F_3}} \qquad (37a)$$

Choosing different values of V, the values of p_{opt} and $\overline{t}_{R/C}$ are obtained using Eqs.(37a) and (37). These values are presented in first 3 columns of Table 4. However, the aim of the optimization is to obtain an engine with minimum sea level static thrust (T_s). Since, thrust output varies with flight speed the values of $\overline{t}_{R/C}$ need to be converted to $\overline{t}_{SR/C}$. Where, $\overline{t}_{SR/C}$ is (T_s / W). The value of T_s is chosen for the climb setting. The values of T at different velocities are obtained from Ref.8, chapter 9 for an engine with bypass ratio of 6.5. The value of $\overline{t}_{R/C}$ multiplied by (T_s/T) gives $\overline{t}_{SR/C}$. The values of $\overline{t}_{SR/C}$ are also given in Table 4. It is observed that $\overline{t}_{R/c}$ has a minimum of 0.2469 at p = 4615. However, the values of $\overline{t}_{sR/c}$ are only slightly different from the minimum for values of V from 120 to 170 m/s. Hence, for wingloading (p) between 3391 to 6805 N/ m^2 the thrust loading would be close to the optimum.

V (m/s)	p_{opt}	$\bar{t}_{R/c}$	$\bar{t}_{sR/c}$
80	1507	0.1893	0.2868
100	2355	0.1637	0.2641
120	3391	0.1487	0.2507
140	4615	0.14	0.2469
150	5298	0.1373	0.2483
160	6028	0.1356	0.2510
170	6805	0.1346	0.2554
180	7629	0.1343	0.2617
190	8500	0.1345	0.2691
200	9419	0.1354	0.2780

Table 4: Variation of $\bar{t}_{sR/c}$ with different values of V

3.4 Minimum fuel for range (W_{fmin}) consideration

In cruise, the weight of the fuel (W_f) used is to cover the range (R) is related to the wing loading (p) as given below (section 4.7 of Ref.5).

$$\bar{W}_f = \frac{R}{3.6}\sqrt{\frac{\rho_0}{2}}\, TSFC\sqrt{\sigma q}\left(\frac{F_1}{p} + F_2 + F_3 p\right) \quad (38)$$

The values of F_1, F_2 and F_3 corresponding to cruise conditions are as follows.

$F_1 = 0.007124$

$F_2 = 1.632 \times 10^{-6}$

$V_{cruise} = M_{cruise} \times 295.2 = 0.8 \times 295.2 = 236.3$ m/s

$q_{cruise} = 0.5 \times \rho \times V^2_{cruise} = 0.5 \times 0.364 \times 236.3^2 = 10159.59\ N/m^2$

Hence, $F_3 = \dfrac{0.0482}{10159.59^2} = 4.67 \times 10^{-10}\ m^4/N^2$

Differentiating Eq.(38) by p and equating to 0 gives the optimum wing loading from range consideration as :

$$\left(p_{optimum}\right)_R = \sqrt{\frac{F_1}{F_3}} \tag{39}$$

In this case, $\left(p_{optimum}\right)_R = \sqrt{\frac{0.007124}{4.67\times10^{-10}}} = 3905.84\ N/m^2$

Using this value of p, in Eq.(38) along with R = 4000 km and

TSFC = 0.6 hr^{-1}, yields :

$\overline{W}_{fmin} = 0.1514$

Allowing an excess fuel of 5 % i.e. $\overline{W}_{f\,min} = 0.1590$ and using Eq.(38) gives

two values p_1 and p_2 as :

$p_1 = 2676\ N/m^2$

$p_2 = 5700\ N/m^2$

Thus, a value of p within p_1 and p_2 would be acceptable from the point of view of minimizing $\overline{W}_f$.

3.5 Absolute ceiling consideration

At absolute ceiling (H_{max}), the flight is possible at only one speed. Observing the trend of H_{max} as h_{cruise} + 0.6 km the absolute ceiling H_{max} is chosen as 11.6 km. To find the $\overline{t}_{Hmax}$, the following two equations are solved (section 4.5 of Ref.5).

$$\overline{t}_h = \sqrt{4K(F_1 + F_2 p)} \tag{40}$$

$$\overline{t}_h = 2q_{hmax}\left(\frac{F_1}{p} + F_2\right) \tag{41}$$

The values of F_1 and F_2 corresponding to this case are :

$F_1 = 0.007124$

$F_2 = 1.632 \times 10^{-6}$

In the absence of a prescribed velocity at H_{max}, the velocity corresponding to flight at $(L/D)_{max}$ is assumed to calculate q_{Hmax} . The value of C_L corresponding to $(L/D)_{max}$ is given by :

$$C_L = \sqrt{\frac{C_{Do}}{K}} = \sqrt{\frac{0.0161}{0.048}} = 0.577 \qquad (42)$$

$$q_{Hmax} = \frac{(W/S)}{C_L} = \frac{5500}{0.577} = 9532.06$$

The solution for p_{opt} is obtained by solving Eqs.(40) and (41).

p_{opt} = 5500 Nm^{-2} as it should be.

$\bar{t}_{Hmax}$ corresponding to $p_{optimum}$ is :

$\bar{t}_{Hmax}$ = 0.05581

Allowing a 5 % variation in $\bar{t}_{Hmax}$, gives the following two values.

$\bar{t}_{Hmax1}$ = 0.05302

$\bar{t}_{Hmax2}$ = 0.05860

The solutions to Eq.(40) with the new $\bar{t}_{Hmax}$ values are:

$p_1 p_1$ = 4567 Nm^{-2}

p_2 = 6547 Nm^{-2}

Similarly, using in Eq.(41), the two values are :

p_1 = 4942 Nm^{-2}

p_2 = 6201 Nm^{-2}

From the above four values, the lower and upper bounds for values of p from the ceiling considerations are

p_1 = 4942 Nm^{-2}

p_2 = 6201 Nm^{-2}

Hence, for p between 4942 and 6201 N/m^2 would be acceptable from consideration of ceiling.

3.6 Choice of optimum wing loading

The range of wing loading which give near optimum results in various cases discussed above are tabulated in Table 5.

Performance criteria	Allowable range of W/S in (Nm^{-2})
s_{land}	3639 - 5328
V_{max}	3344 - 7101
$(R/C)_{max}$	3391 - 6805
$\bar{W}_f$	2676 - 5700
h_{max}	4942 - 6201

Table 5: Allowable range of W/S in various cases

From the above table, it is observed that the range of wing loading in which all the criteria are satisfied is 4942 to 5328 N/m^2

3.7 Consideration of wing weight (W_w)

The weight of the wing depends on its area. According to Ref.4, chapter 15, for passenger airplanes, the weight of the wing is proportional to $S^{0.649}$[9]. Thus, a wing with lower area will be lighter and for wing area to be lower, the W/S should be higher. However, for a passenger the consideration of fuel required for range is also an important consideration. From section 3.4 it is noted that the wing loading should be 3906 for $\bar{W}_f$ to be minimum. At the same time with a wing loading of 5700 N / m^2, the value of $\bar{W}_f$ is only 5 % higher than $\bar{W}_{fmin}$ Hence, the advantage of choosing a wing loading higher than that indicated by $\bar{W}_{fmin}$ requirement, is examined.

It may be pointed out that the weight of wing structure is about 12% of W_g. The optimum W/S from range consideration is 3906 N/m^2 whereas with a 5% increase in W_f, the wing loading could go up to 5700 N/m^2. If the wing loading of 5700 N/m^2 is chosen, instead of 3906 N/m^2, the weight of the wing would decrease by a factor of :

$$\left(\frac{3906}{5700}\right)^{0.649} = 0.782$$

With weight of the wing as 12 % of W_g, the saving in the wing weight when higher wing loading is chosen is :

12 x (1-0.782) = 2.6 %.

However, this higher wing loading results in an increase in fuel by 5 % of W_f. In the present case, W_f is around 20% and hence a 5% in W_f means an increase in the weight by 0.05 × 0.2 = 1%.

Thus, by increasing W/S from 3906 to 5700 N/m^2, the saving in W_g would be 2.6 - 1 = 1.6%. Thus, it is advantageous to have higher wing loading W/S.

3.8 Final choice of W/S

It is observed from Table 5 that a wide range of wing loading is permissible which still satisfies various requirements with permissible deviations from the optimum. To arrive at the final choice, the take-off requirement is considered. The highest wing loading which would permit take-off within permissible distance without excessive (T/W) requirement is the criteria. From data collection, the balanced field length for take-off is assumed to be 2150 m. From Fig.5.4 of Ref.4, the take-off parameter $\{(W/S)/\sigma\ C_{LTO}\ (T/W)\}$ for this field length is 180. With (W/S) in lb/ft^2. The take-off is considered at sea level or $\sigma = 1$. The value of C_{LTO} is 0.8 x C_{Lmax} = 0.8 × 2.5 = 2. Generally the airplanes in this category have (T/W) of 0.3. Substituting these values gives,

p_{final} = 108.2 lb/ft^2 = 5195 Nm^{-2}

It is reassuring that this value of p lies within the permissible values summarized in Table 5.

3.9 Thrust requirements

After selecting the W/S for the aircraft, the thrust needed for various design requirements is obtained. These requirements decide the choice of engine.

3.9.1 Requirement for V_{max}

The chosen value of p i.e. 5195 Nm^{-2} is substituted in the following equation :

$$\bar{t}_{V\max} = q_{\max}\left(\frac{F_1}{p} + F_2 + F_3 p\right) \tag{43}$$

$$= 11200.95\left(\frac{0.009124}{5195} + 1.632\times10^{-6} + 3.84\times10^{-10}\times5195\right) = 0.0602 \tag{44}$$

Referring to engine charts in Ref.8, chapter 9, for a turbo fan engine with bypass ratio of 6.5, the thrust loading based on sea level static thrust is :

$$\frac{T}{W} = \frac{0.0602}{0.18} = 0.334 \tag{45}$$

In the present case, this would mean a thrust requirement of

T_{req} = 0.334 x 59175 x 9.81 = 193.9 kN

3.9.2 Requirements for (R/C)max

The following equation is used

$$\bar{t}_{R/c} = \frac{R/C}{V} + \frac{1}{2}\rho_0\sigma\frac{V^2}{p}(F_1 + F_2 p + F_3 p^2) \tag{46}$$

Substituting appropriate values, yields :

$$\left(\frac{T}{W}\right)_{R/C} = 0.252 \tag{47}$$

In the present case, this implies a thrust requirement of 146.3 kN

3.9.3 Take-off thrust requirements

The value of (T/W) for take-off has been taken as 0.3.This implies a thrust requirement of 0.3 x 59175 x 9.81 = 174.2 kN

3.10 Engine choice

From the previous section, it is observed that the maximum thrust requirements occurs from V_{max} consideration i.e. T_{max} = 193.9 kN

As a twin engine configuration has been adopted, the above requirement implies thrust per engine of 96.95 kN/engine.

An engine which supplies this thrust and has a TSFC of 0.6 hr^{-1} and bypass ratio of around 6.5 is needed. Some of the engines which perform close to these requirements are taken from Ref.8, chapter 9 and website 1.
Finally, CFM56-2B model of turbofan with a sea level static thrust of 97.9 kN is chosen.

3.11 Engine characteristics

For calculation of the performance of the airplane, the variations of thrust and TSFC with speed and altitude are required. Reference 8, chapter 9 has given non-dimensional charts for turbofan engines with different bypass ratios. Choosing the charts for bypass ratio = 6.5 and sea level static thrust of 97.9kN, the engine curves are calculated and presented in Figs.5 and 6.

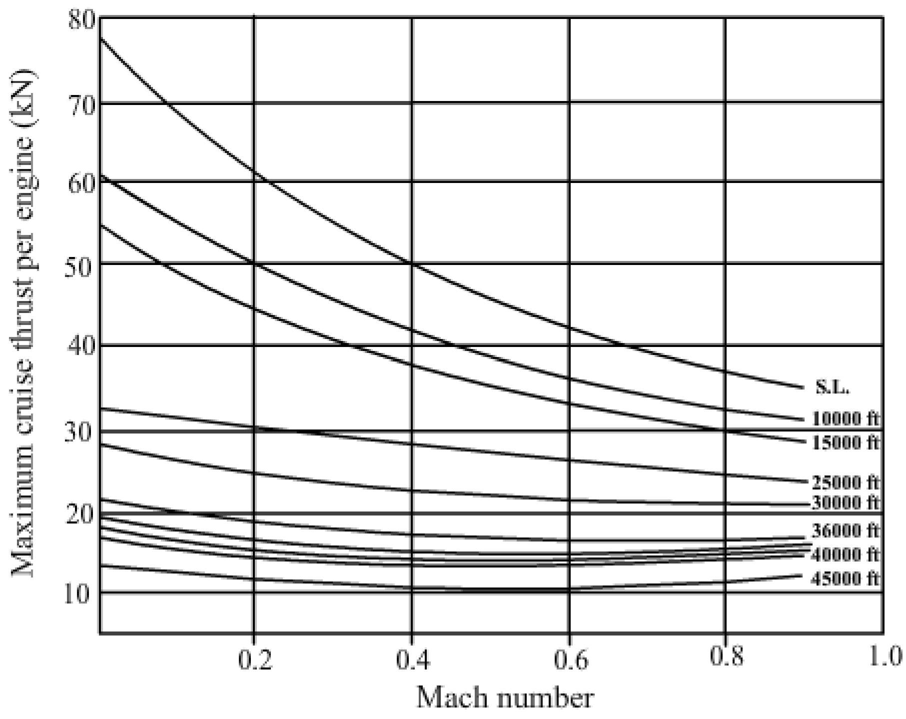

Fig. 5 Variations of thrust with Mach number at different altitudes with cruise setting of engine.

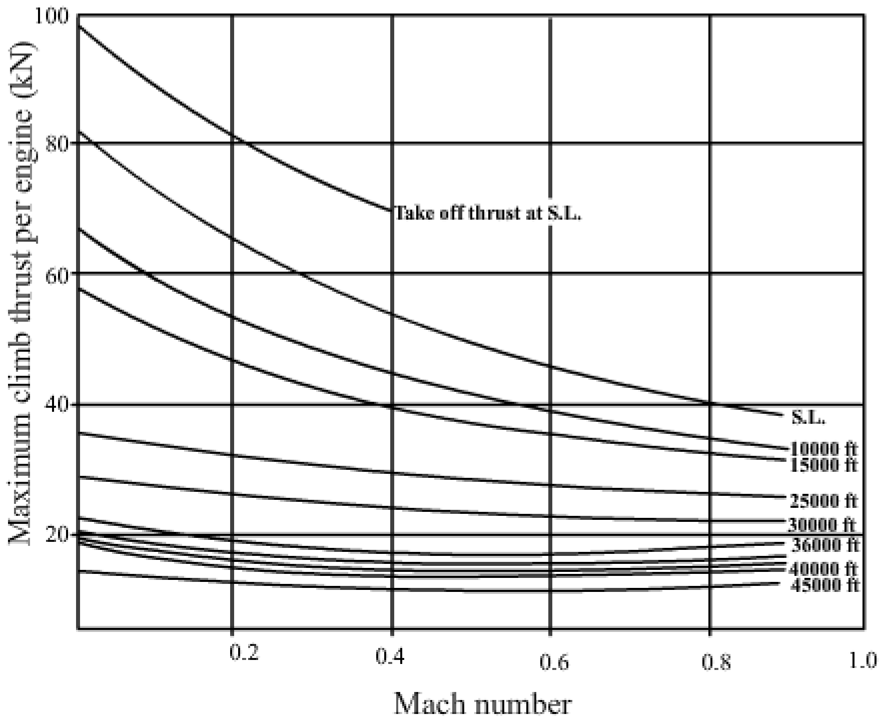

Fig. 6 Variations of thrust with Mach number (a) at sea level with take-off setting and (b) at various altitudes with climb setting of engine

4 Wing Design

4.1 Introduction

The weight and the wing loading of the airplane have been discussed in sections 2 and 3 as 59175 kgf (579915 N) and 5195 N/m^2 . These give wing area as 111.63 m^2 . The wing design involves choosing the following parameters.

1. Airfoil selection
2. Aspect ratio
3. Sweep
4. Taper ratio
5. Twist
6. Incidence
7. Dihedral
8. Vertical location

In the following subsections, the factors affecting the choice of these parameters are mentioned and then the choices are effected.

4.2 Airfoil Selection

The airfoil shape influences C_{Lmax} , C_{Dmin}, C_{Lopt} , C_{mac} and stall pattern. These in turn influence stalling speed, fuel consumption during cruise, turning performance and weight of the airplane.

For high subsonic airplanes, the drag divergence Mach number(M_D) is an important consideration. It may be recalled that (M_D) is the Mach number at which the increase in the drag coefficient is 0.002 above the value at low subsonic Mach numbers. A supercritical airfoil is specially designed to increase M_D. NASA has carried out tests on several supercritical airfoils and recommends the use of NASA-SC(2) series airfoil with appropriate thickness ratio and camber.

4.2.1 Design lift coefficient

The C_{Lopt} of an airfoil is the lift coefficient at which the drag coefficient is minimum. For passanger airplanes, the airfoil is chosen in such a way that C_{Lopt} equals $C_{Lcruise}$.

$$C_{Lcruise} = \frac{(W/S)}{q_{cruise}} \tag{48}$$

Using the value of (W/S) = 5195 Nm^{-2} and q corresponding to M = 0.8 at 11 km altitude, gives :

$$C_{Lcruise} = 0.512 \tag{49}$$

C_{Lopt} is taken as 0.5 for choosing airfoil thickness ratio and wing sweep.

4.2.2 Airfoil thickness ratio and wing sweep

Airfoil thickness ratio (t/c) has a direct influence on drag, maximum lift, stall characteristics, structural weight and critical Mach number. A higher t/c implies a lower critical Mach number but also a lower wing weight. Thus, an optimum t/c for the airfoil needs to be chosen.

C_{Lopt} = 0.5 has been chosen and the cruise Mach number is 0.8. In order to ensure that the drag divergence Mach number is greater than M_{cruise}, M_D is chosen as 0.82. This is based on the consideration that there should be no increase in drag at M_{cruise} . It may be recalled that ΔC_{Dwave} is 0.002 at M_D and the slope of the C_D Vs M curve around M_D is 0.1.

Reference 3 gives experimental results for several super-critical airfoils with different values of (t/c) and C_{Lopt}. Curves for C_{Lopt} = 0.4, 0.7, 1.0 are available in the aforesaid report.The curve corresponding to C_{Lopt} = 0.5 is obtained by interpolation.

The M_D for the wing is estimated in the following manner.

$$M_D = (M_D)_{airfoil} + \Delta M_A + \Delta M_\Lambda \tag{50}$$

where, ΔM_A and ΔM_Λ are corrections for influences of the aspect ratio and the sweep.

The change in M_D with A is almost zero for A > 8. Since, A = 9.3, has been chosen the second term in the above equation will not contribute to M_D. Further, from Ref.[9], chapter 15, the change in M_D due to sweep is given as :

$$1 - \frac{\Lambda}{90} = \frac{1 - M_{D\Lambda}}{1 - M_{D\Lambda=0}} \tag{51}$$

The supercritical airfoil with (t/c) = 14% has M_D = 0.74 at C_{Lopt} of 0.5. Using this in Eq.(51) gives Λ which would give $M_{D\Lambda}$ of 0.82 i.e.

$$1 - \frac{\Lambda}{90} = \frac{1-0.82}{1-0.74}$$

Or $\Lambda = 27.7^o$

The average thickness has been chosen as 14 %. However, to reduce the structural weight, the (t/c) at wing root is increased and the (t/c) at wing tip is decreased, Considering the features for Airbus A310 and Boeing B 767 which have M_{cruise} = 0.8 and similar values of $\Lambda_{1/4}$, it is decided that the variation of (t/c) along the span be such that (t/c) is 15.2% at root, 11.8% at spanwise location of the thickness break and 10.3% at the tip. Thickness break location is the spanwise location upto which the trailing edge is straight. From the data collection this location is at 34% of semi-span.

4.3 Other parameters

4.3.1 Aspect ratio

The aspect ratio affects $C_{L\alpha}$, C_{Di} and wing weight. The value of $C_{L\alpha}$ decreases as A decreases. For example, in the case of an elliptic wing,

$$C_{L\alpha} = \frac{A}{A+2}\left(C_{l\alpha}\right)_{airfoil} \tag{52}$$

The induced drag coefficient can be expressed as

$$C_{Di} = \frac{C_L^2}{\pi A}(1+\delta) \tag{53}$$

where, δ depends on A, λ and Λ. A high value of A increases the span of the wing which in turn requires more hanger space. A higher aspect ratio would also result in poor riding quality in turbulent weather. All these factors need careful optimization. However, at the present stage of design, A = 9.3 is chosen based on trends indicated by data collection.

Correspondingly, the wing span would be

$b = \sqrt{AS} = \sqrt{9.3 \times 111.63} = 32.22$ m

4.3.2 Taper ratio

Wing taper ratio is defined as the ratio between the tip chord and the root chord.

Taper ratio affects :

Induced drag

Weight of wing and

Tip stalling

Induced drag is low for taper ratios between 0.3 - 0.5. Lower the taper ratio, lower is the weight. A swept wing also has higher structural weight than an unswept wing. Since, the present airplane has a swept wing, a taper ratio of

0.24 has been chosen based on the trends of current swept wing airplanes.

4.3.3 Root and tip chords

Root chord and tip chord of the equivalent trapezoidal wing can now be evaluated.

$$c_r = \frac{2S}{b(1+\lambda)} = \frac{2\times 111.63}{32.22(1+0.24)} = 5.59 \text{ m}$$

$$c_t = c_r\lambda = 5.59\times 0.24 = 1.34 \text{ m}$$

$$\text{Mean aerodynamic chord (mac)} = \bar{c} = \frac{2}{3}\frac{(1+\lambda+\lambda^2)}{(1+\lambda)}c_r = 3.9 \text{ m}$$

The location of the quarter chord of the mac from leading edge of the root chord is calculated as 4.76 m.

4.3.4 Dihedral

The dihedral (Γ) is the angle of the wing with respect to the horizontal plane when seen in the front view. Dihedral of the wing affects the lateral stability of the airplane. Since, there is no simple technique for arriving at the dihedral angle that takes all the considerations into effect, the dihedral angle is chosen based on data collected (Table A). Hence, a value of $\Gamma = 5^\circ$ is chosen.

4.3.5 Wing twist

A linear twist of 3° is chosen tentatively.

4.4 Cranked wing design

An observation of the design of current high subsonic airplanes, indicates that the trailing edge is straight for a part of the span, in the inboard region. This results in a larger chord in the inboard section as compared to a normal swept wing which is trapezoidal in shape. A larger chord in the inboard region has following advantages.

1. More space for fuel and landing gear.

2.The lift distribution is changed such that more lift is produced in the inboard region of wing, which reduces the bending moment at the root. This type of design is called a wing with cranked trailing edge. The value of the span upto which the trailing edge is straight has to be obtained by optimization considering drag and weight of wing. However, at the present stage of design, based on the current trends, the trailing edge is made unswept till 35% of the semi span in the present case, the semi-span of the wing portion with unswept trailing edge is:

$0.35 \times (32.22/2) = 5.64$ m

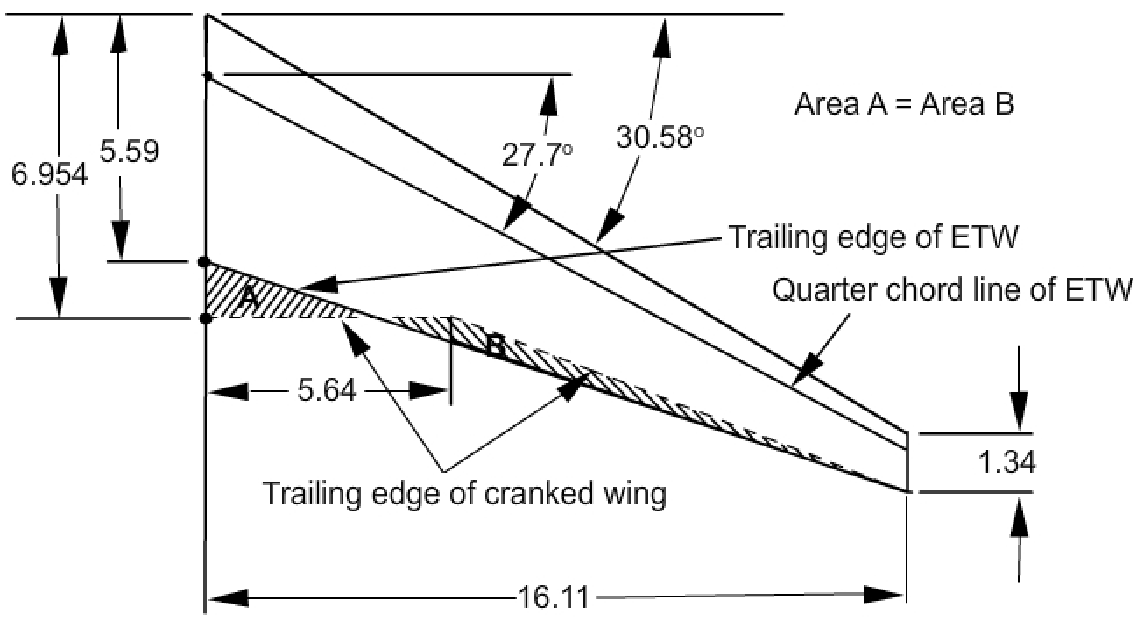

Fig. 7 Plan forms of ETW and cranked wing

The cranked wing and the equivalent trapezoidal wing (ETW) are shown in Fig.7. The planform of the cranked wing is obtained as follows.

(i) The area of the cranked wing = area of ETW = 111.63 m^2

(ii) The span of the cranked wing = span of ETW = 32.22 m

(iii) Tip chord of the cranked wing (c_{te}) = tip chord of ETW (c_{tc}) = 1.34 m

(iv) Leading edge sweep of cranked wing = leading edge sweep of ETW = 30.58°

(v) Rootchord of ETW = c_{re} = 5.59 m. The root chord of the cranked wing is obtained in the next two steps.

(vi) As mentioned above, the straight portion with unswept trailing edge of the cranked wing extends upto 5.64 m on either side of root chord. Because of this choice, the leading edge of the chord at a spanwise location (y) of 5.64 m is:

5.64 x tan 30.58 = 3.33 m behind the leading edge of the root chord.

(vii) Let c_{rc} be the root chord of the cranked wing. Considering the shape of the cranked wing in Fig.7 and noting that the area of the cranked wing is 111.63 m^2, gives the following equation for c_{rc}.

$$2\left\{\left(\frac{c_{rc}+c_{rc}-3.33}{2}\right)\times 5.64+\left(\frac{c_{rc}-3.33+1.34}{2}\right)\times(16.11-5.64)\right\} = 111.63$$

Or $c_{rc} = 6.954$ m.

4.5 Wing incidence(i_w)

The wing incidence is the angle between wing reference chord and fuselage reference line. Wing incidence is chosen to minimize the drag at some operating conditions,usually cruise.The wing incidence is chosen such that when the wing is at the correct angle of attack for the selected design condition, the fuselage is at the angle of attack for minimum drag(usually at zero angle of attack). The wing incidence is finally set using wind tunnel data. However, an initial estimate, of i_w, for preliminary design purpose, is obtained as follows.

$$C_{Lcruise} = C_{L\alpha}\left(i_w - \alpha_{0L}\right) \tag{54}$$

In the present case,

$C_{Lcruise} = 0.512$

$C_{L\alpha}$ is calculated using the following formula in Ref.4, chapter 12,

$$C_{L\alpha} = \frac{2\pi A}{2+\sqrt{4+\frac{A^2\beta^2}{\eta^2}\left(1+\frac{\tan^2\Lambda_{max}}{\beta^2}\right)}}\left(\frac{S_{exposed}}{S_{ref}}\right)(F) \tag{55}$$

where,

$\beta^2 = 1 - M^2$

$\eta = 1$

$F = 1.07\left(1+\frac{d}{b}\right)^2$

$S_{exposed}$ = area of exposed wing

Λ_{max} = sweep angle of the maximum thickness line of airfoil

Substituting various values, gives

$C_{L\alpha} = 6.276\ rad^{-1} = 0.1095\ deg^{-1}$

The zero lift angle α_{0L} for the airfoil was calculated using camber line of the supercritical airfoil with 14% thickness ratio. The value is -5.8^o. Substituting the values yields a value of i_w which is negative. This can be attributed to the fact that the value of $C_{L\alpha}$ as estimated above is high. It may be pointed out that in Appendix 'C' of Ref.14, the stability derivatives of Boeing 747 are evaluated. There also the calculated value of $C_{L\alpha}$ at $M = 0.8$ is higher than the experimental value. This is because the estimated value is for a rigid wing. The actual wing is flexible and the theoretical increase in $C_{L\alpha}$ due to Mach number, is not realizied.

$i_w = 1^o$ is chosen. This value is recommended in Ref.4, chapter 4.

4.6 Vertical location of wing

The wing vertical location for the airplane under design is chosen as low wing configuration. This is typical for similar airplanes.

4.7 Areas of flaps and ailerons

These areas are chosen based on the data on similar airplanes.

1.Trailing edge : Fowler flaps.

2.Leading edge : full span slats.

$$\frac{S_{flap}}{S} = 0.17,\ \frac{S_{slat}}{S} = 0.1,\ \frac{\text{flap span}}{\text{wing span}} = 0.74$$

5 Fuselage and tail layout

5.1 Introduction

The fuselage layout is important in the design process as the length of the airplane depends on this.The length and diameter of the fuselage are related to the seating arrangement.

The Fuselage of a passenger airplane can be divided into four basic sections viz. nose, cockpit, payload compartment and tail fuselage. In this section, the fuselage design is carried out by choosing the parameters of these sections.

5.2 Initial estimate of fuselage length

Observing the values of (l_f /b) for similar airplanes, a value of 1.05 is chosen.

Using b = 32.22 m as obtained from wing design, the fuselage length is :

32.22 x 1.05 = 33.83 m.

Ref.4, chapter 6 provides the following relation between gross weight and length of fuselage.

$$l_f = a W_o^c \tag{56}$$

where, W_o is in lbs and l_f in ft.

For a jet transport airplane, a = 0.67 and c = 0.43.

Using W_o = 59175 × 2.205 lbf, an l_f of 31.83 m is obtained.

This is in good agreement of the value obtained based on data collection.

5.3 Nose and cockpit - front fuselage

The front fuselage accommodates the forward looking radar in the nose section, the flight deck with associated windscreen, and the nose undercarriage. Anthropometric data for flight crews provide the basis for the arrangement of pilot’s seats, instruments and controls. Development of electronic displays has transformed the traditional layout of the flight deck. The airplane must be capable of being flown from either pilot’s seat ; therefore the wind screen and front geometry is symmetric about the aircraft longitudinal center line. Further, modern ’glass’ cockpit displays and side stick controllers have transformed the layout of the flight deck from the traditional aircraft configuration. The front fuselage profile presents a classical design compromise between a smooth shape for low drag and the need to have flat sloping windows to give good visibility. The layout of the flight deck and the specified pilot window geometry is often the starting point of the overall fuselage layout.

For the present design, the flight decks of similar airplanes are considered and the value of l_{nose}/l_f is chosen as 0.03.

For the cockpit length ($l_{cockpit}$), standard values are prescribed by Ref.4 chapter 9. The length of cockpit for the two member crew is chosen as 100 inches (2.5 m).

5.4 Passenger cabin layout

Two major geometrical parameters that specify the passenger cabin are cabin diameter and cabin length. These are in turn decided by more specific details like number of seats, seat width, seating arrangement (number abreast), seat pitch, aisle width and number of aisles.

5.4.1 Cabin cross section

The shape of the fuselage cross section is dictated by the structural requirements for pressurization. A circular shell resists the internal pressure loads by hoop tension. This makes the circular section efficient and therefore lowest in structural weight. However, a fully circular section may result in too much unusable volume above or below the cabin space. This problem is overcome by the use of several interconnecting circular sections to form the cross-sectional layout. The parameters for the currently designed airplane are arrived at by considering similar airplanes(Table A).

A circular cross section for the fuselage is chosen here.

The overall size must be kept small to reduce aircraft weight and drag, yet the resulting shape must provide a comfortable and flexible cabin interior which will appeal to the customer airlines. The main decision to be taken is the number of seats abreast and the aisle arrangement.The number of seats across will fix the number of rows in the cabin and thereby the fuselage length.Design of the cabin cross section is further complicated by the need to provide different classes like first class, business class, economy class etc.

5.4.2 Cabin length

Following the trend displayed by current airplanes, a two class seating arrangement is chosen viz economy class and business class.The total number of seats(150) is distributed as 138 seats in the economy class and 12 seats in the business class.

Cabin parameters are chosen based on standards for similar airplanes. The various parameters chosen are as follows.

Parameter	Economy class	Business class
Seat / pitch (in inches)	32	38
Seat width (in inches)	20	22
Aisle width (in inches)	22	24
Seats abreast	6	4
Number of aisles	1	1
Max. height (in m)	2.2	2.2

Since, the business class has a 4 abreast seating arrangement,the number of rows required is 3 and the economy class has 23 rows.The cabin length is found out by using the seat pitch for each of the classes.

Class	No. of seats	No. of rows	Seat Pitch (in)	Cabin length(m)
Economy	138	23	32	18.4
Business	12	3	38	2.85

Hence,the total cabin length will be 18.4 + 2.85 = 21.25 m.

5.4.3 Cabin diameter

Using the number of seats abreast,seat width,aisle width the internal diameter of the cabin is calculated as:

$d_{finternal}$ = 22 × 1 + 19 × 6 = 136 in = 3.4 m

According to the standards prescribed by Ref[4], chapter 9, the structural thickness in inches is given by :

t = 0.02 $d_{finternal}$ + 1 = 0.02 × 136 + 1 = 3.72 in = 0.093 m

Therefore, the external diameter of the fuselage is obtained as :

3.4+0.093 × 2 = 3.59 m.

5.5 Rear fuselage

The rear fuselage profile is chosen to provide a smooth, low drag shape which supports the tail surfaces. The lower side of the profile must provide adequate clearance for the airplane, when it is in rotation during take off. The rear fuselage should also house the auxillary power unit(APU).

Based on data collected for similar airplanes the ratio l_{tail}/l_f is chosen as 0.25.

5.6 Total fuselage length

The cabin length and cockpit length have been decided to be 21.25 m and 2.5 m respectively. The ratios of nose and tail length with respect to l_f have been chosen as 3% and 25%. Thus, cabin and cockpit length form 72% of l_f.

Hence, the fuselage length is calculated as 23.75/0.72 = 33 m.The lengths of various parts of the fuselage are indicated below.

Nose length = 1 m

Cockpit length = 2.5 m

Cabin length = 21.25 m

Rear length = 8.25 m

Total = 33 m

It may be noted that the revised value of l_f is nearly same as the earlier estimate.

The details like galleys, toilets, cabin crew seats, doors and emergency exits have not been worked out. However, the ratio l_f/b is in the range of values for similar airplanes. Hence, it is assumed that aforesaid items can be suitably accommodated.

5.7 Tail surfaces

The type and area of the tail surfaces are important from the point of view of stability of the airplane. A conventional tail arrangement is chosen. Some of the important parameters that decide the aerodynamic characteristics of the tail are (a) area ratios (S_h/S) and ((S_v/S)) , (b) tail volume ratios (V_H and V_V), (c) tail arm and (d) tail span. All these parameters need to be decided for both the horizontal and vertical tails.

From the data of similar airplanes, the following values are chosen.

Parameter	Horizontal Tail	Vertical Tail
Area ratio (S_h/S), (S_v/S)	0.31	0.21
Aspect ratio	5	1.7
Taper ratio	0.26	0.31

The Areas of the horizontal and vertical tails (S_h and S_v) are:

$S_h = 0.31 \times 111.63 = 34.61\ m^2$
$S_v = 0.21 \times 111.63 = 23.44\ m^2$

The spans of the horizontal and vertical tails (b_h and b_v) are :

$$b_h = \sqrt{A_h S_h} = \sqrt{5 \times 34.61} = 13.15\ m \qquad (57)$$

$$b_v = \sqrt{A_v S_v} = \sqrt{1.7 \times 23.44} = 6.31\ m \qquad (58)$$

The chord lengths of the horizontal and vertical tails are :

$$c_{rh} = \frac{2S_h}{b_h(1+\lambda_h)} = \frac{2 \times 34.61}{13.15 \times (1+0.26)} = 4.18\ m$$

$$c_{rv} = \frac{2S_v}{b_v(1+\lambda_v)} = \frac{2 \times 23.44}{6.31 \times (1+0.31)} = 5.67\ m$$

$$c_{th} = \lambda_h c_{rh} = 0.26 \times 4.18 = 1.09\ m$$

$$c_{tv} = \lambda_v c_{rv} = 0.31 \times 5.67 = 1.76\ m$$

Tail arm

Tail arm is the distance between the aerodynamic center of the wing and the aerodynamic center of horizontal tail (l_h) or vertical tail(l_v). The values of the tail arm are chosen based on the data collection as :

l_h = 45 % of l_f and l_v = 42% of l_f i.e.

l_h = 0.45 x 33 =14.85 m

l_v = 0.42 x 33 = 13.86 m

$$V_H = \frac{S_h l_h}{S\bar{c}} = \frac{34.61 \times 14.85}{111.63 \times 3.9} = 1.18 \qquad (59)$$

$$V_V = \frac{S_V l_V}{Sb} = \frac{23.44 \times 13.86}{111.63 \times 32.22} = 0.09 \qquad (60)$$

5.8 Engine location

The type of engine mounting and it's location play a major role in deciding the overall drag coefficient of the airplane. A conventional wing mounted engine is chosen as it facilitates periodic engine maintenance. This is important in airline industry where an unscheduled downtime could mean considerable loss to the company. The engines are attached to the lower side of the wing using pylons to reduce drag. The other reason for choosing a wing mounted engine is that the fuel is stored in the wing and this reduces the length of the fuel lines.

From the data collection of similar airplanes, the engine location is fixed at 34% of the semi span.

5.9 Landing gear arrangement

One of the principal moving parts on the aircraft is the landing gear. This must be light, small, provide smooth ride during taxing and safe energy absorption at touch down. It must be retractable to reduce drag during flight. Housing of the landing gear is a space constraint. A conventional tricycle landing gear is chosen based on the trend followed by similar airplanes. The important parameters of this type of landing gear are wheel track, wheel base and turning radius. The values of the parameters(shown below) are based on data collection.

Parameter	Value
Wheel base (in m)	13.2
Track length (in m)	5.8
Turning radius (in m)	19.3

6 Estimation of component weights and c.g location

Airplane weight is a common factor which links different design activities namely aerodynamics, structures, propulsion, layout, airworthiness, economic and operational aspects. Hence, at each stage of the design, a check is made on the expected total mass of the completed airplane. The design bureau has a department which assees and controls the weight. In the preliminary design stage, the estimates of the weights of major components of the airplane are based on statistical data. As the components are manufactured and the airplane prototype reaches completion it is possible to cross check the accuracy of the estimates by weighing each component and where necessary initiate weight reduction programmes.

6.1 Airplane mass statement

The weight of the entire airplane can be sub-divided into empty weight and useful load. The empty weight can be further subdivided into weights of :

Structures group

Propulsion group

Equipment group

DCPR(Defense Contractor Planning Report) weight is taken as the weight obtained after deducting weights of wheels, brakes, tyres, engines, starters, batteries, equipments, avionics etc from the empty weight. DCPR weight is important for cost estimation, and can be viewed as the weight of the parts of the airplane that the manufacturer makes as opposed those of items bought and installed.

It is a normal practice in aircraft design to list the various components of airplane mass in a standard format.The components are grouped in convenient subsections as given below.

6.1.1 Structures group

1. Wing (including control surfaces)
2. Tail (horizontal and vertical including controls)
3. Body (or fuselage)
4. Nacelles
5. Landing gear (main and nose units)
6. Surface controls

6.1.2 Propulsion group

1. Engine(s) (dry weight)
2. Accessory gearbox and drives
3. Induction system
4. Exhaust system
5. Oil system and cooler
6. Fuel system
7. Engine controls
8. Starting system
9. Thrust reversers

6.1.3 Fixed equipment group

1. Auxiliary power unit
2. Flight control systems (sometimes included in structural group)
3. Instruments and navigation equipment
4. Hydraulic systems
5. Electrical systems
6. Avionics systems
7. Furnishing

8. Air conditioning and anti-icing

9. Oxygen system

10. Miscellaneous (e.g.fire protection and safety systems)

6.2 Weights of various components

After making the classification between various groups and listing the components in each group, the next step is to determine the weights of these components.

In the preliminary design stage, it is not possible to know the size of individual aircraft components in great detail but it is possible to use prediction methods that progressively become more accurate as the airplane geometry develops. Most airplane design textbooks contain a set of equations empirically derived based on existing airplanes. In the present case, the weights of the various individual components are calculated using the equations given in Ref.16, chapter 8.

6.3 C.G location and c.g. travel

6.3.1 Wing location along length of fuselage

The longitudinal location of wing is decided based on the consideration that the c.g. of the entire airplane with full payload and fuel is around the quarter chord of the m.a.c of wing. For this purpose, the weights and the c.g locations of various components are tabulated. Then applying moment equilibrium about the nose of the airplane, the distance of the leading edge of root chord of the wing from the nose (X_{le}) is calculated to satisfy the aforesaid requirement. The steps to obtain X_{le} are given below.

As regards the c.g. locations of wing, horizontal tail and vertical tail it is assumed that the c.g. is at 40% of the respective m.a.c.The fuselage c.g. is taken to be at 42% of it's length.The engine c.g. location is taken to be at 40% of it's length.For this purpose the distance of the engine c.g. from the root chord is measured for various airplanes and a distance of 2 m is chosen. All other components (equipments, furnishings etc.) are assumed to have their combined c.g. location at 42% of the fuselage length.The tabulated values are given below. The nose wheel is placed at 14% of the fuselage length and the main landing gear position is determined using the wheelbase from section 5.9.

Remarks:

(i) Using the data on equivalent trapezoidal wing in section 4.3.3, the locations of the wing a.c. and c.g. are respectively 4.76 m and 5.34 m behind the leading edge of the root chord of the wing.

(ii) Noting that the tail arm is 14.85 m and that the c.g of tail is 15 % behind its a.c., the distance of the c.g. of the horizontal tail from the leading edge of root chord of wing is 20.05 m. In a similar manner, the c.g. of the vertical tail is at 19.56 m behind the leading edge of the root chord of wing.

The weights of various components and the c.g. locations are given in table below.

Component	Weight (kgf)	c.g. location from nose (m)
Wing	5855.41	X_{le}+5.34
Fuselage	6606.60	13.86
Horizontal tail	1160.94	X_{le}+20.05
Vertical tail	746.22	X_{le}+19.56
Engine group	5659.19	X_{le}+2
Nose wheel	363.18	4.62
Main landing gear	1961.25	17.82
Fixed equipment total	7421.09	13.86
Fuel	12130.88	X_{le}+4.76
Payload	17270	14.13
Gross Weight	59175	X_{le}+4.76

Applying moment equilibrium about the nose of the airplane, X_{le} is obtained as 9.85 m from the nose of the airplane. Consequently, the location of the c.g. of the airplane from the nose is at : 9.85 + 4.76 = 14.61 m.

6.4 C.G travel in critical cases

The movements of the c.g. under various loading conditions are examined below.

6.4.1 Full payload and no fuel

For the case of full payload and no fuel, the fuel contribution to the weight is not present. However, it has been assumed that the fuel tanks are located such that the c.g of the fuel is at the quarter chord of m.a.c. of wing. Since the c.g. of the entire airplane is also at the quarter chord of wing m.a.c., there is no shift in the c.g. when the fuel has been consumed. Hence, the C.G shift is 0%.

6.4.2 No payload and no fuel

For this case, the fuel as well as the payload contributions are not present.Since the c.g of payload is not at the c.g of the entire airplane,the c.g is bound to shift by a certain amount in this case.The moment calculations are performed and the new c.g location is obtained at 14.93 m from the nose.Therefore, the c.g shift : is 14.93 - 14.63 = 0.3 m i.e. 7.28 % of m.a.c.

6.4.3 No payload and full fuel

For this case,since there is no payload, the c.g shifts. On performing calculations, the new c.g. location is obtained at 14.84 m.Therefore, the c.g. shift is :14.84 - 14.63 = 0.21 m i.e. + 5.7 % .

Hence, the c.g shift is +5.17% of the m.a.c.

6.4.4 Payload distribution for 15% c.g travel

Sometimes the c.g. shift is calculated for hypothetical cases like (a) only half the pay load concentrated in the front half of the passenger cabin and (b) only half the pay load concentrated in the rear half of the passenger cabin. These cases result in large shift in c.g. Hence, an alternate strategy is suggested.

According to Ref.7, a total c.g shift of 15% is acceptable for commercial airplanes. To ensure this, as a first step the maximum payload that can be concentrated in the front portion of the passenger cabin is calculated such that a c.g shift of only 7.5% is obtained.

It is assumed that the percentage of payload is “x “and also the payload c.g of to be at x % of the passenger cabin length. Performing the c.g. calculations yields the value of x to be 90%.

As a second step, similar calculations are performed, such that the maximum payload that can be concentrated at the rear half of the passenger cabin resulting in a c.g shift of only 7.5 %. On performing the calculation, a value of 70% is obtained for x .

Hence, the c.g locations for various critical cases and payload distributions have been calculated.

6.4.5 Summary of c.g. calculation

Wing location (leading edge of root of trapezoidal wing) : 9.85 m

c.g location with full payload and full fuel : 14.61 m

c.g travel for no payload and no fuel : +7.28 % of m.a.c.

c.g travel for no payload and full fuel : + 5.17% of m.a.c.

For a c.g travel of 7.5% on either side of original c.g location: 90% of passengers can be concentrated in the front or 70% in the rear.

7 Revised estimates of areas of horizontal and vertical tails

7.1 Stability and controllability

The ability of a vehicle to maintain its equilibrium is termed stability and the influence which the pilot or control system can exert on the equilibrium is termed its controllability.The basic requirement for static longitudinal stability of any airplane is a negative value of dC_{mcg} /dC_L. Dynamic stability requires that the vehicle be not only statically stable,but also that the motions following a disturbance from equilibrium be such as to restore the equilibrium.

Even though the vehicle might be statically stable, it is possible that the oscillations following a disturbance might increase in magnitude with each oscillation,thereby making it impossible to restore the equilibrium.

7.2 Static longitudinal stability and control

7.2.1 Specifications

The horizontal tail must be large enough to insure that the static longitudinal stability criterion, dC_{mcg}/dC_L is negative for all anticipated center of gravity positions.

An elevator should be provided so that the pilot is able to trim the airplane (maintain $C_m = 0$) at all anticipated values of C_L .

The horizontal tail should be large enough and the elevator powerful enough to enable the pilot to rotate the airplane during the take-off run, to the required angle of attack.This condition is termed as the nose wheel lift-off condition.

7.2.2 Revised estimate of the area of horizontal tail

In chapter 9 of Ref.5 procedures are indicated (a) to verify that the neutral point stick-free is beyond aft most location of c.g. and (b) to verify that the elevator is adequate to trim the airplane during landing.

Here, the following simpler approach is used to obtain the area of horizontal tail (h.tail).

(i)Chapter 16 of Ref.4 presents a curve for $(C_{m\alpha})_{\text{stick-free}}$ vs M for different types of airplanes. From this curve the appropriate value of $C_{m\alpha}$ is obtained. From this value of $C_{m\alpha}$, the static margin stick-free is calculated.

(ii) The contribution of wing, fuselage, power and h.tail are worked out for cruise flight condition with airplane weight equal to the design gross weight. It may be pointed out that (a) the contribution of wing is zero in this case as the c.g. is at the a.c. of the wing (b) the contribution of h.tail should provide the required value of static margin. This gives the required value of tail volume ratio V_H .Subsequently the area of h.tail can be calculated.

The steps are as follows.

$$\frac{dC_m}{dC_L} = \left(\frac{dC_m}{dC_L}\right)_{wing} + \left(\frac{dC_m}{dC_L}\right)_{fuselage} + \left(\frac{dC_m}{dC_L}\right)_{nacelle} + \left(\frac{dC_m}{dC_L}\right)_{power} + \left(\frac{dC_m}{dC_L}\right)_{h.tail} \quad (61)$$

Contribution of fuselage $\left(\frac{dC_m}{dC_L}\right)_{Fus}$:

From Ref.4 chapter 16, an approximate expression for this contribution is :

$$\left(\frac{dC_m}{dC_L}\right)_{Fus} = \frac{K_f\, W_f^2\, l_f}{Sca_w} \quad (63)$$

where,

W_f = width of fuselage = 3.59 m

l_f = length of fuselage = 33 m

S = wing area = 111.63 m^2

c = mean aerodynamic chord of wing = 3.9 m

a_w = slope of lift curve of a wing = 6.276 rad^{-1} = 0.1095 deg^{-1}

The value of K_f is obtained as 0.0119 from Fig.16.14 of Ref.4.

From section 45 : a_w = 6.276 /radian = 0.1095 /degree

From Fig.16.14 of Ref.4, the value of K_f is 0.0119.

Hence, $\left(\frac{dC_m}{dC_L}\right)_{fus} = \frac{0.0119 \times 3.59^2 \times 33}{111.63 \times 3.9 \times 0.1095} = 0.1036$

The contribution of nacelle to (dC_m/dC_L) is neglected.

Contribution of power $\left(\frac{dC_m}{dC_L}\right)_{power}$:

From Ref.11, Chapter 5,

$$\left(\frac{dC_m}{dC_L}\right)_{power} = \frac{T\,t_p}{Wc} \quad (65)$$

where, T = thrust, W = weight of airplane, t_p is the distance of the thrust line from c.g. (the distance is measured perpendicular to the thrust line).

For the airplane under design, t_p is estimated as 0.19 m. At the cruise altitude, (T/W) is chosen as 0.06.

Hence,

$$\left(\frac{dC_m}{dC_L}\right)_{power,cruise} = \frac{0.06 \times 0.19}{3.9} = 0.00292$$

Contribution of h.tail in stick-free case $\left(\frac{dC_m}{dC_L}\right)_{h.tail}$ is (Ref.14, chapter 3):

$$\left(\frac{dC_m}{dC_L}\right)_{h.tail} = -\frac{a_t}{a_w} V_H \eta_t \left(1-\frac{d\varepsilon}{d\alpha}\right)\left(1-\tau\frac{C_{h\alpha t}}{C_{h\delta e}}\right)$$

a_t = slope of lift-curve of h.tail

a_w = slope of lift-curve of wing

η_t = tail efficiency

V_H = Tail volume ratio

ε = down wash angle

$\tau = dC_{Lt} / d\delta_e / dC_{Lt} / d\alpha$

C_{Lt} = lift coefficient of tail

δ_e = elevator deflection

α_t = angle of attack of h.tail

$C_{h\alpha t} = \partial C_h / \partial \alpha_t$

$C_{h\delta e} = \partial C_h / \partial \delta_e$

C_h = hinge moment coefficient

Using Eq.(55), a_t = 0.0828 deg^{-1}

From section 4.5, a_w = 0.1095 deg^{-1}

η_t is assumed to be 0.95

$d\varepsilon / d\alpha$ is estimated using the following approximate formula :

$$\frac{d\varepsilon}{d\alpha} = \frac{114.6 \times a_w}{\pi A} \tag{64}$$

Or $\frac{d\varepsilon}{d\alpha} = \frac{114.6 \times 0.1095}{\pi \times 9.3} = 0.43$

τ : In the present case, from section 1.2.4, $(S_{ele}/S_h) = 0.22$

From Ref.11, Fig.5.33, for this value of $S_{ele}/S_{h,}$ τ = 0.43

$C_{h\alpha t}$ & $C_{h\delta e}$:

Boeing airplanes generally have sealed internal balance. Reference 15, chapter 12 gives the values of $C_{h\alpha}$ and $C_{h\delta e}$ for various values of c_b / c_f ; c_b and c_f are lengths of elevator ahead and behind the hinge.

From this reference $C_{h\alpha} = 0.0044$ deg^{-1} and $C_{h\delta e} = 0.0068$ deg^{-1}.

Hence, contribution of h.tail is :

$$-\frac{0.0828}{0.1095} \times V_H \times 0.95 \times (1\text{-}0.43)\left(1\text{-}0.43 \times \frac{0.0044}{0.0065}\right) = 0.296\ V_H$$

From chapter 16 of Ref.4, the recommended value of $C_{m\alpha}$, for a jet transport flying at a Mach no. of 0.8 is - 1.15.

Hence,

$$\left(\frac{dC_m}{dC_L}\right) = \frac{C_{m\alpha}}{a_w} = \frac{-1.15}{6.276} = -0.183$$

Substituting in Eq.(61), yields:

$$-0.183 = 0.1036 + 0.00292 - 0.296\ V_H$$

Or $\quad V_H = 0.98$

Noting that $V_H = \frac{S_h}{S}\frac{l_h}{c}$

Or $S_h = \frac{V_H cS}{l_h}$

$= \frac{0.98 \times 3.9 \times 111.63}{14.86} = 28.71\, m^2$; note : c = 3.9 m, S = 111.63 m^2 and

$l_h = 14.86$ m

Remark:

Keeping in view the large number of approximations involved in calculation of parameters during landing and take-off, the cross checks for forward c.g. location and nose wheel lift-off conditions are not carried out at this stage.

7.3 Lateral stability and control

7.3.1 Specifications

The directional static stability criterion, $dC_n / d\beta$, should be positive for any flight speed greater than 1.2 times the stalling speed.

The yawing moment control (rudder) must be powerful enough to counteract the yawing moment encountered (a) in roll (adverse yaw), (b) in cross-wind landing or takeoff, (c) when one engine is inoperative for multi-engine airplanes. The spin recovery is also effected primarily by the rudder.

7.3.2 Equations for directional stability

The equation for directional stability can be derived as (Ref.14, chapter 5):

$$\frac{dC_n}{d\beta} = \left(C_{n\beta}\right)_{(wing)} + \left(C_{n\beta}\right)_{(Fuselage)} + \left(C_{n\beta}\right)_{(power)} + \left(C_{n\beta}\right)_{(V.Tail)} \tag{72}$$

7.3.3 Revised estimate of area of vertical tail

In the preliminary analysis of directional static stability, the contributions of wing, power and interference effects are ignored. It is further assumed that the contributions to $C_{n\beta}$ due to wing sweep and low wing position cancel each other.

$\left(C_{n\beta}\right)_{fuselage}$:

An approximate formula from Ref.10 chapter 1-9 is :

$$\left(C_{n\beta}\right)_{(Fuselage)} = \frac{-k_n V_n}{28.7 S b} \tag{73}$$

where k_n = a factor which depends on fineness ratio of fuselage

V_n = Volume of fuselage ,

b, S = wing span, area.

Reference 4, chapter 16 gives a slightly different formula for $\left(C_{n\beta}\right)_{(Fuselage)}$.

From Figure 1:9-2 of Ref.10, k_n is 0.95,

Hence, $\left(C_{n\beta}\right)_{(Fuselage)} = \dfrac{0.95 \times 217.86}{28.7 \times 111.63 \times 32.22} = -0.002005$

$C_{n\beta(V.tail)}$:

$$C_{n\beta(V.tail)} = a_v \frac{S_v}{S}\frac{l_v}{b}, \tag{74}$$

a_v = slope of lift curve of v.tail = 0.0378 deg^{-1}

$V_V = \dfrac{S_v}{S}\dfrac{l_v}{b}$

Hence, $C_{n\beta(V.tail)} = 0.0378 \times V_V$

The value of $C_{n\beta(desirable)}$ is given by Ref.11, chapter 8 is :

$$C_{n\beta(desirable)} = 0.0005\left[\frac{W}{b^2}\right]^{1/2}, \tag{75}$$

W = weight of airplane in lbs

b = span of wing in ft

Hence,

$$C_{n\beta(desirable)} = 0.0005\left\{\frac{59175 \times 2.2046}{(32.22/0.3048)^2}\right\}^{\frac{1}{2}} = 0.001709$$

From Eq.(72),

$$C_{n\beta(desirable)} = C_{n\beta(fuse)} + C_{n\beta(V.tail)} \tag{76}$$

Substituting various values : $0.001709 = -0.002005 + 0.0378 \times \bar{V}_V$ (77)

Or $V_V = 0.098$

This value is almost the same as that obtained in the initial tail sizing.

Hence, the vertical tail area is :

$$S_V = \frac{V_V \times Sb}{l_v} = \frac{111.63 \times 32.22 \times 0.098}{13.86} = 25.43 \text{ m}^2 \text{ , note } l_v = 13.86 \text{ m}$$

8 Features of the designed airplane

8.1 Three-view drawing

The three-view drawing of the designed airplane is given in Fig.8

8.2 Overall dimensions

Length : 34.32

Wing Span : 32.22 m

Height above ground : 11.17

Wheel base : 13.2 m

Wheel track : 5.8 m

8.3 Engine details

Similar to **CFM 56 - 2B**

Seal Level Static Thrust : 97.9 kN

By pass ratio : 6.5 (For which the characteristics are given in Ref.8, chapter 9)

SFC : At M = 0.8, h = 10972 m (36 000 ft), SFC is taken as 0.6 hr^{-1}

8.4 Weights

Gross weight : 59175 kgf

Empty weight : 29706 kgf

Fuel weight : 12131 kgf

Payload : 17338 kgf

Maximum landing weight : 50296 kgf

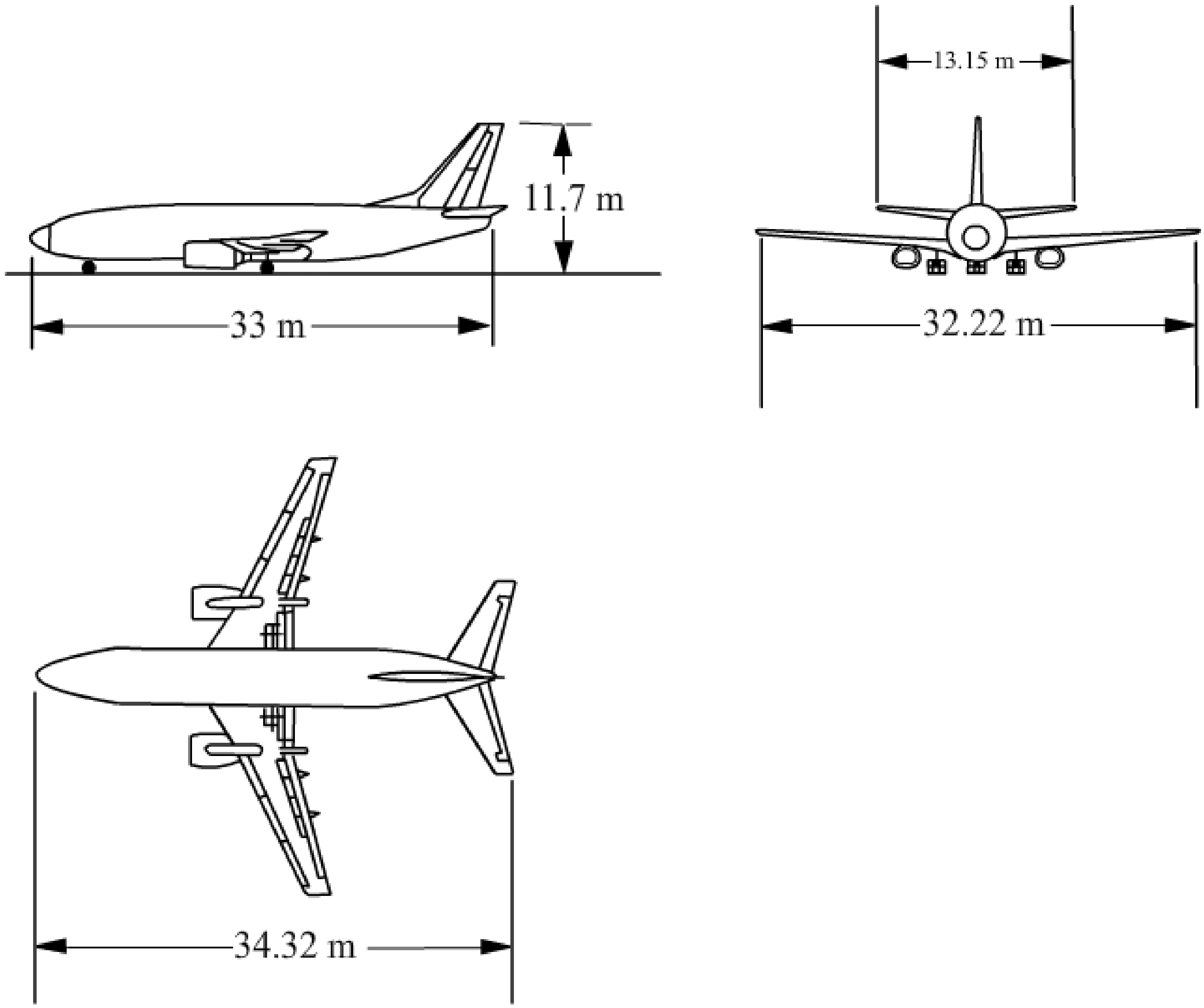

Figure 8: Three-view drawing of the airplane

8.5 Wing geometry

Planform shape : Cranked wing

Area : 111.63 m^2

Span : 32.22 m

Airfoil : NASA - SC(2) series, t/c = 14%, C_{lopt} = 0.5

Root chord : 5.59 m (Equivalent trapezoidal wing)

Tip chord : 1.34 m (Equivalent trapezoidal wing)

Root chord of cranked wing : 6.954 m

Portion of wing with straight trailing edge : 5.64 m on either wing half

Mean aerodynamic chord : 3.9 m

Quarter chord sweep : 27.7^{o}

Dihedral : 5^{o}

Twist : 3^{o}

Incidence : 1^{o}

Taper ratio : 0.24 (Equivalent trapezoidal wing)

Aspect ratio : 9.3

8.6 Fuselage geometry

Length : 33 m

Maximum diameter : 3.59 m

8.7 Nacelle geometry

No. of nacelles : 2

Nacelle diameter : 1.62 m

Cross sectional area : 2.06 m^2

Length of nacelle : 3.3 m (based on B737 nacelle)

8.8 Horizontal tail geometry

Area : 28.71 m^2

Span : 11.98 m

Mean aerodynamic chord : 2.67 m

Quarter chord sweep : 32^{o}

Root chord : 3.80 m

Tip chord : 0.99 m

Taper ratio : 0.26

Aspect ratio : 5

8.9 Vertical tail geometry

Area : 25.43 m^2

Span : 6.58 m

Root chord : 5.90 m

Tip chord : 1.83 m

Mean aerodynamic chord : 4.22 m

Quarter chord sweep : 37^o

Taper ratio : 0.31

Aspect ratio : 1.70

8.10 Other details

C_{Lmax} without flap : 1.4

C_{Lmax} with landing flaps : 2.7

Maximum load factor n_{max} : 3.5

C_{Lmax} with T.O flaps : 2.16

8.11 Crew and payload

Flight crew : 2 (pilot and co-pilot)

Cabin crew : 5

Passenger seating : 138 economy and 12 business class

8.12 Performance

The detailed performance estimation is given in section 9. The highlights are as follows.

The performance is calculated for a gross weight of 59175 kgf and wing loading of 5195 Nm^{-2} except for landing where the landing weight is taken as 85% of take-off weight.

Maximum Mach no. at 36000 ft : 0.859 with cruise thrust and 0.874 with climb thrust.

Maximum still air range: 5602 km at M = 0.81 and h = 36000 ft.

Maximum rate of climb at sea level : 1087 m/min with climb thrust

Absolute ceiling : 11.88 km

Service ceiling, $(R/C)_{max}$ of 30 m/min : 11.68 km ;

Take-off distance over 50 ft : 860 m(2820 ft) and balanced field length

: 1830 m(6000 ft)

Landing distance from 15 m : 1140 m(3740 ft)

Remark :

The designed airplane meets the requirements set out in the specifications. The seating arrangement takes care of the passenger comfort and the choice of engine reflects low level of noise.

9 Performance estimation

The details regarding overall dimensions, engine details, weights, geometric parameters of wing, fuselage, nacelle, horizontal tail, vertical tail and other details like C_{Lmax} in various conditions and maximum load factor are given in sections 8.2 - 8.10.

The details of flight condition for estimation of drag polar are as follows.

Altitude : 10972 m = 36000 ft

Mach number : 0.8

Kinematic viscosity : $3.90536 \times 10^{-5} m^2/s$

Density : $0.3639 \ kg/m^3$

Speed of sound : 295.07 m/s

Flight speed : 236.056 m/s

Weight of the airplane : 59175 kgf

9.1 Estimation of drag polar

The drag polar is assumed to be of the form:

$$C_D = C_{DO} + \frac{C_L^2}{\pi A e}$$

The quantity C_{Do} is assumed to be given by:

$$C_{DO} = (C_{DO})_{WB} + (C_{DO})_V + (C_{DO})_H + (C_{DO})_{Misc} \quad (78)$$

where suffices WB, V, H, Misc denote wing-body combination, vertical tail, horizontal tail, and miscellaneous contributions respectively.

9.1.1 Estimation of $(C_{DO})_{WB}$

Initially, the drag polar is obtained at a Mach number of 0.6 as suggested by Ref.6. $(C_{DO})_{WB}$ is given as :

$$(C_{DO})_{WB} = (C_{DO})_W + (C_{DO})_B \frac{S_B}{S_{ref}}$$

The suffix B denotes fuselage and S_B is the maximum frontal area of fuselage. $(C_{DO})_W$ is given as :

$$(C_{DO})_W = C_{fw}\ [1 + L\left(\frac{t}{c}\right)]\left(\frac{S_{wet}}{S_{ref}}\right)_{wing}$$

Here, the Reynolds number used to determine the turbulent flat plate skin friction coefficient is based on the mean aerodynamic chord $\bar{c}_e$ of the exposed wing. $(S_{wet})_e$ is the wetted area of the exposed wing.

Now c_r = 5.59 m, c_t = 1.34 m, (b/2) = 16.11 m and d_{fus} = 3.59 m. Hence,

$$\text{Root chord of the exposed wing} = c_{re} = 5.59 - \frac{5.59 - 1.34}{16.11} \times \frac{3.59}{2} = 5.116\ \text{m}$$

$$\text{Taper ratio of exposed wing} = \lambda_e = \frac{1.34}{5.116} = 0.262$$

$$\text{Hence, } \bar{c}_e = \frac{2}{3}\left[5.116\left(\frac{1 + 0.262 + 0.262^2}{1 + 0.262}\right)\right] = 3.596\ \text{m}$$

$$\text{Semi-span of exposed wing} = (b/2)_e = 16.11 - 1.795 = 14.315\ \text{m}$$

$$S_{exposedwing} = 14.314\left(\frac{5.116 + 1.341}{2}\right) \times 2 = 92.41\ \text{m}^2$$

M = 0.6, a = 295.07 m/s, V = 0.6 x 295.07 = 177.12 m/s, υ = 3.90536 × 10^{-5} m^2/s.

Hence,

$$\text{Re} = \frac{177.12 \times 3.596}{3.90536 \times 10^{-5}} = 16.31 \times 10^{6}$$

Average height of roughness = k = 1.015×10^{-5} m, corresponds to standard camouflage paint, application (Ref.6).

Hence, $\frac{l}{k} = \frac{3.596}{1.015 \times 10^{-5}} = 3.543 \times 10^{5}$

The $R_{ecutoff}$ corresponding to the above value of l/k is 30×10^{6}.

The value of C_{fw} corresponding to $R_{ecutoff}$ is obtained from Ref.6 as : 0.00265

$(t/c)_{avg}$ = 14% and $(t/c)_{max}$ at x/c > 0.3 gives L = 1.2.

Hence, $(S_{wet})_{wing} = 2 \times 92.41(1+1.2 \times 0.14) = 215.8 \text{ m}^2$

$(C_{DO})_B$ is given as:

$(C_{DO})_B = (C_{Df})_B + (C_{Dp})_B + C_{Db}$

$$(C_{DO})_B = C_{fB}\left[1+\frac{60}{(l_b/d)3}+0.0025\left(\frac{l_b}{d}\right)\right]\left(\frac{S_{wet}}{S_B}\right)_{fus} + C_{Db}\frac{S_{base}}{S_{ref}}$$

l_f = 33.0 m and d_{max} = 3.59 m

$$R_{eb} = \frac{177.12 \times 33}{3.905 \times 10^{-5}} = 149.6 \times 10^{6}$$

k = 1.015×10^{-5} m corresponds to standard camouflage paint, application.

Hence, $\frac{l}{k} = \frac{33}{1.015 \times 10^{-5}} = 32.51 \times 10^{5}$

The $R_{ecutoff}$ corresponding to this l/k is 2.6×10^{8}.

The value of C_{fB} from Ref.6 is 0.00019

$$(S_{wet})_{fus} \approx 0.75 \times \pi \times 3.59 \times 33 = 279 \text{ m}^2$$

$$S_B = \frac{\pi}{4} \times 3.59^2 = 10.12 \text{ m}^2$$

Hence, $(C_{Df})_B = 0.0019 \times \frac{279}{10.12} = 0.0524$

$$(C_{Dp})_B = 0.0019\left[\frac{60}{(33/3.59)^3} + 0.0025 \times (33/3.59)\right]\frac{279}{10.12} = 0.00524$$

C_{Db} is assumed to be zero, since base area is almost zero.

Hence, $(C_{Do})_B = 0.0524 + 0.00524 + 0 = 0.0576$

$(\Delta C_D)_{canopy}$ is taken as 0.002. Hence, $(C_{Do})_B = 0.0596$

Finally, $(C_{Do})_{WB} = 0.00598 + 0.0596 \frac{10.12}{111.63} = 0.01138$

9.1.2 Estimation of $(C_{DO})_H$ and $(C_{DO})_V$

The estimation of $(C_{DO})_H$ and $(C_{DO})_V$ can be done in a manner similar to that for the wing. However, the details regarding the exposed tail area are needed. In the absence of the detailed data on the shape of fuselage at rear, a simplified approach given in Ref.6 is adopted, wherein $C_{Df} = 0.0025$ is used for both horizontal and vertical tails and S_{wet} equals $2(S_h + S_v)$.

Hence,

$$(C_{Do})_{hv} = 0.0025 \times 2 \times (28.71 + 25.43)\frac{1}{111.63} = 0.0024 \qquad (79)$$

9.1.3 Estimation of misc. drag - Nacelle

For calculating drag due to the nacelles the short cut method is used : (Ref.6).

$$(C_{DO})_{nacelle} = 0.006 \times \frac{S_{wet}}{S_{ref}}$$

where, S_{wet} is the wetted area of nacelle. Here, $S_{wet} = 16.79 m^2$. Since, there are two nacelles, the total wetted area is twice of this. Finally :

$$(C_{DO})_{nacelle} = 0.006 \times \frac{16.79}{111.63} \times 2 = 0.0018$$

9.1.4 C_{DO} of the airplane

Taking 2% as the interference drag (Ref.6), the C_{DO} of the airplane is :

$$C_{DO} = 1.02\ [0.01138 + 0.0024 + 0.0018] = 0.0159 \qquad (80)$$

9.1.5 Induced drag

The expression for induced drag includes Oswald's efficiency factor (e). This quantity is estimated by adding the effects of all the aircraft components on induced drag.

A rough estimate of e is :

$$\frac{1}{e} = \frac{1}{e_{wing}} + \frac{1}{e_{fuselage}} + \frac{1}{e_{other}}$$

From Ref.9, chapter 7

$$e_{wing} = (e_w)_{\Lambda=0} \cos(\Lambda - 5)$$

where Λ is the wing sweep. From Ref.12 $(e_{wing})_{\Lambda=0}$ = 0.97 for AR = 9.3, λ = 0.24.

Hence, e_{wing} = 0.97 × cos (27.7− 5) = 0.8948. Also $\frac{1/e_{fus}}{(S_f/S)} = 0.8$ for a round fuselage. Hence,

$$\frac{1}{e_{fus}} = 0.8 \times \frac{10.12}{111.63} = 0.0725$$

Finally, $e = \frac{1}{0.8948^{-1} + 0.0725 + 0.05} = 0.8064$

Hence,

$$K = \frac{1}{\pi A e} = \frac{1}{\pi \times 9.3 \times 0.8064} = 0.04244$$

9.1.6 Final drag polar

$$C_D = 0.0159 + 0.04244 \times C_L^2 \qquad (81)$$

The drag polar is shown in Fig.9 .

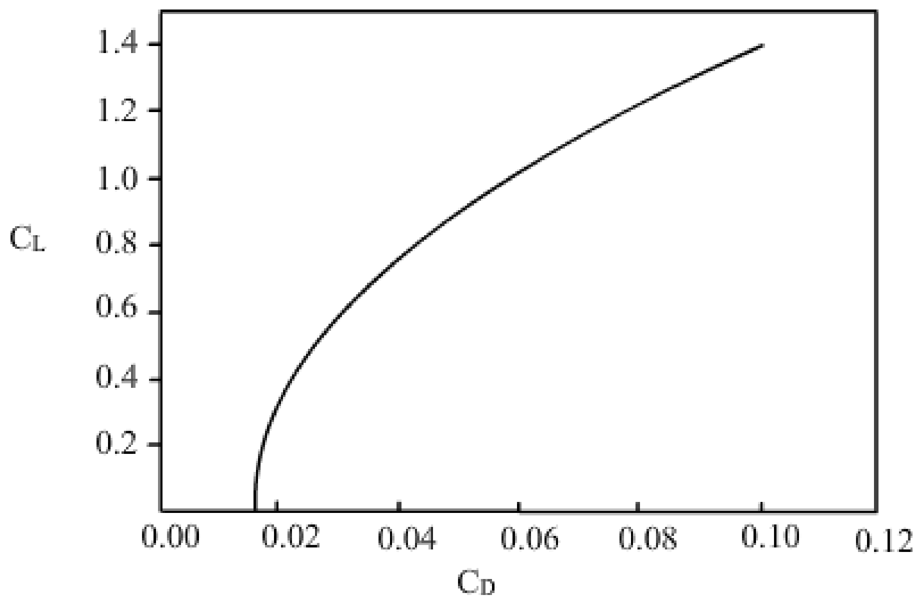

Figure 9: Subsonic drag polar

Remarks:

(i)The polar given by Eq.(81) is valid at subcritical Mach numbers. The increase in C_{DO} and K at higher Mach numbers is discussed in sub section 9.3.2.

(ii)The maximum lift to drag ratio ($(L/D)_{max}$) is given by:

$$(L/D)_{max} = \frac{1}{2\sqrt{C_{DO}K}} \tag{81a}$$

Using Eq.(81a), $(L/D)_{max}$ is 19.25, which is typical of modern jet transport airplanes.

(iii) It may be noted that the parabolic polar is an approximation and is not valid beyond C_{Lmax}. It is also not accurate close to $C_L = 0$ and $C_L = C_{Lmax}$.

9.2 Engine characteristics

To calculate the performance, the variations of thrust and SFC with speed and altitude are needed. Chapter 9 of Ref.8 contains these variations for turbofan engines with various bypass ratios. The thrust variations vs Mach number with altitude as parameters are given in non-dimensional form for take-off, cruise and climb ratings. These values were obtained from the curves and later smoothened. The values multiplied by 97.9 kN, the sea level static thrust rating for the chosen engine, are shown in Figs. 10 and 11. Figure 10 also contains (a) the variation of thrust with Mach number at sea level with take-off rating and (b) variations of climb thrust with Mach number at h = 38000 and 39000 ft; these are obtained by interpolating the values at 36000 and 40000 ft and are used for computations of performance at these altitudes. The SFC variation is also given in Ref.8, but is taken as $0.6 hr^{-1}$ under cruise conditions based on the value recommended in Ref.4 chapter 3.

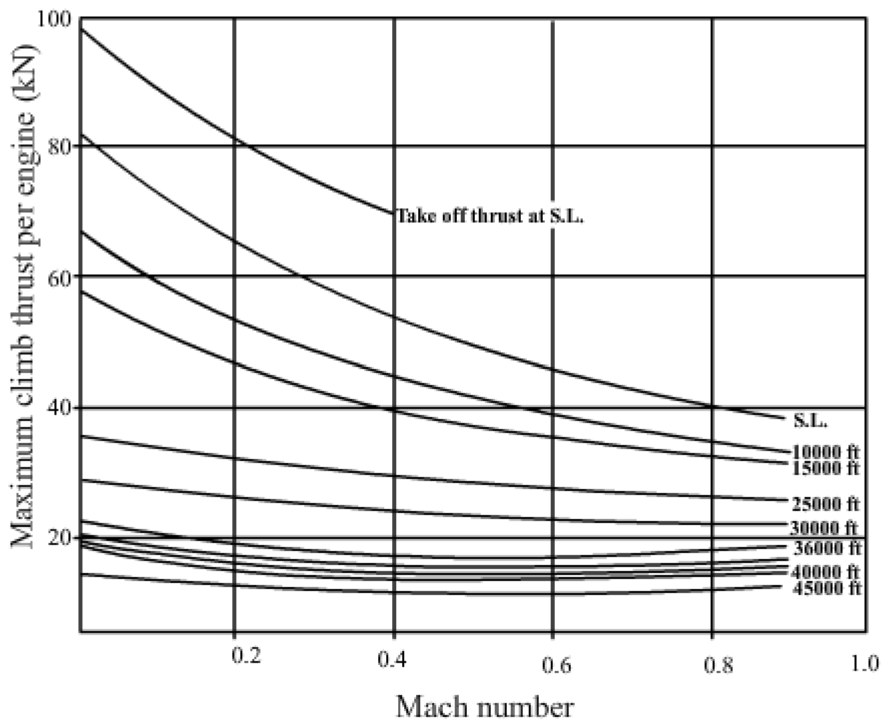

Fig.10 Variations of thrust with Mach number (a) at sea level with take-off setting and (b) at various altitudes with climb setting of engine

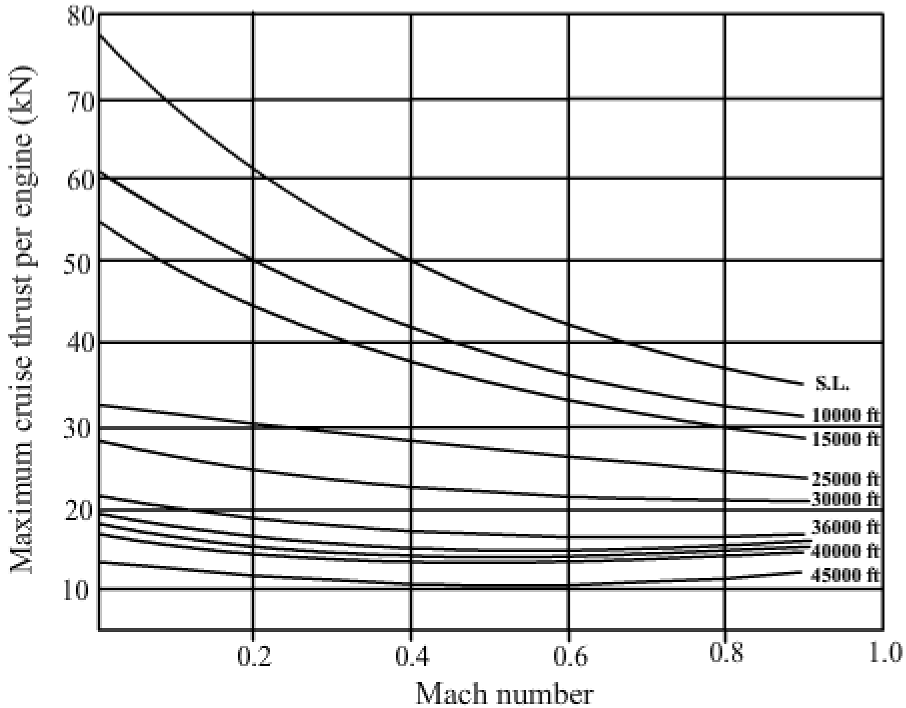

Fig.11 Variations of thrust with Mach number at different altitudes with cruise setting of engine.

9.3 Level flight performance

In steady Level flight, the equations of motion, in standard notations, are

$$T - D = 0 \quad (82)$$

$$L - W = 0 \quad (83)$$

$$L = W = \frac{1}{2}\rho V^2 S C_L \quad (84)$$

$$T = D = \frac{1}{2}\rho V^2 S C_D \quad (85)$$

9.3.1 Stalling speed

In level flight,

$$V = \sqrt{\left(\frac{2W}{\rho S C_L}\right)} \quad (86)$$

Since, C_L cannot exceed C_{Lmax}, there is a flight speed below which the level flight is not possible. The flight speed at $C_L = C_{Lmax}$ is called the stalling speed and is denoted by V_s.

$$V_S = \sqrt{\left(\frac{2W}{\rho S C_{L_{max}}}\right)} \tag{87}$$

Since, ρ decreases with altitude, V_s increases with height. It is noted that $W/S = 5195\ N/m^2$, $C_{Lmax} = 2.7$ with landing flaps and $C_{Lmax} = 1.4$ without flaps. The values of stalling speed at different altitudes and flap settings are tabulated in Table 6 and shown in Fig.12.

h (m)	ρ (kg/m^3)	$V_s(C_{L_{max}} = 1.4)$ (m/s)	$V_s(C_{L_{max}} = 2.7)$ (m/s)
0	1.225	77.83	56.04
2000	1.006	85.86	61.83
4000	0.819	95.18	68.54
6000	0.659	106.06	76.37
8000	0.525	118.87	85.59
10000	0.412	134.09	96.56
11000	0.363	142.80	102.83
12000	0.310	154.52	111.27

Table 6: Variation of stalling speed with altitude

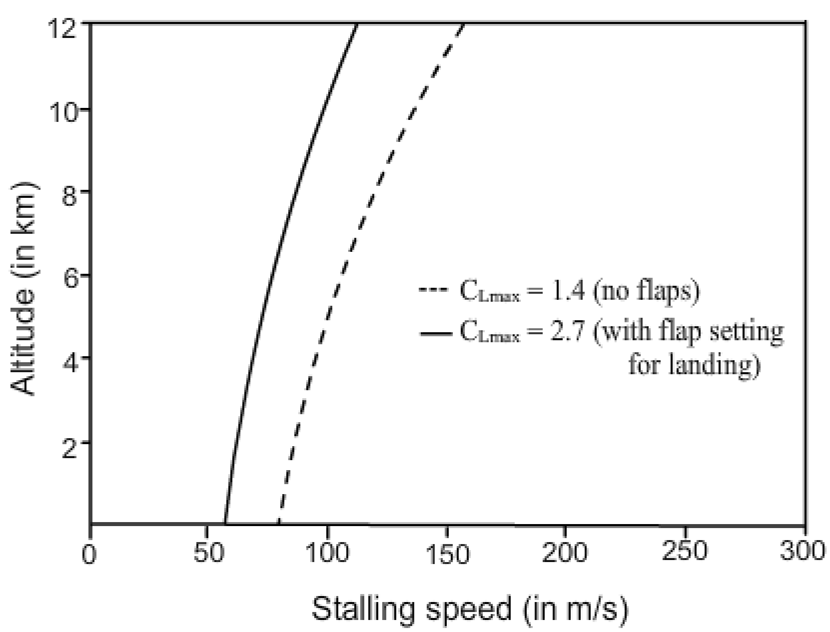

Fig.12 Stalling speed vs altitude

9.3.2 Variations of V_{min} and V_{max} with altitude

To determine V_{min} and V_{max} at each altitude, the following procedure is adopted.

(i)The engine thrust (T_{avail}) as a function of velocity (or Mach number) at each altitude, is obtained from the smoothened data.

(ii)The drag at each altitude is found as a function of velocity using the drag polar and the level flight formulae given below.

$$C_L = \frac{2 \times (W/S)}{\rho V^2} \tag{88}$$

$$C_D = C_{DO} + KC_L^2 \; ; \tag{89}$$

$$Drag = \frac{1}{2}\rho V^2 S C_D \tag{90}$$

T_{avail} depends on Mach number or $T_{avail} = f(M)$ (91)

The values of C_{DO} = 0.0159 and K = 0.04244 are valid at subcritical Mach number. However, the cruise Mach number (M_{cruise}) for this airplane is 0.8. Hence, C_{DO} and K are expected to become functions of Mach number above M_{cruise}. To get some guidelines about variations of C_{DO} and K, The drag polars of B-727 given in volume 6, chapter 5 of Ref.13 are considered. These drag polars are shown in the Fig.13 as discrete points.

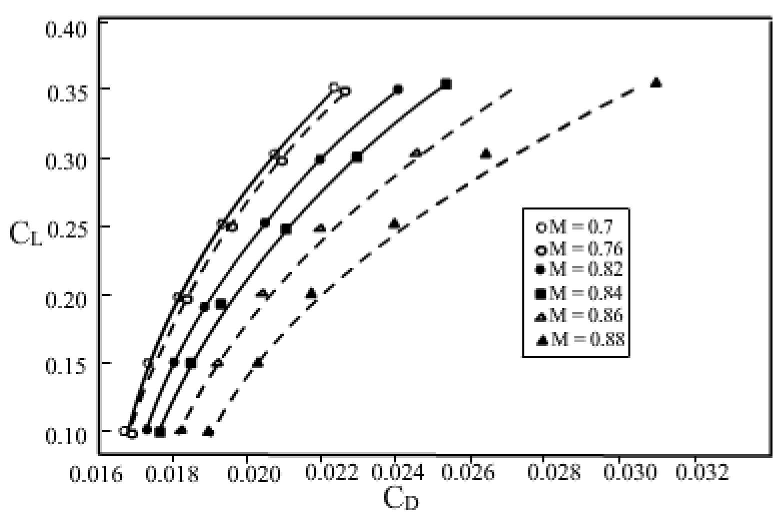

Fig. 13 Drag polars at different Mach numbers for B727-100; Symbols are data from Ref.13 and solid lines are the parabolic fits

These polars are approximated by the parabolic polar expression namely $C_D = C_{DO} + K \times C_L^2$. The values of C_{DO} and K at various Mach numbers are given in Table 7. The parabolic fits are also shown in Fig. 13.

M	C_{DO}	K
0.7	0.01631	0.04969
0.76	0.01634	0.05257
0.82	0.01668	0.06101
0.84	0.01695	0.06807
0.86	0.01733	0.08183
0.88	0.01792	0.103

Table 7: Variations of C_{DO} and K with Mach number (Parabolic fit)

The variations in C_{DO} and K with Mach number are plotted in the Figs. 14 and 15. It is seen that there is no significant increase in C_{DO} and K upto M = 0.76. This is expected to be the cruise Mach number for the airplane (B727-100). Following analytical expressions are found to closely represent the changes in C_{DO} and K from M = 0.76 to M = 0.86.

$$C_{DO} = 0.01634 - 0.001 \times (M - 0.76) + 0.11 \times (M - 0.76)^2 \quad (92)$$

$$K = 0.05257 + (M - 0.76)^2 + 20.0 \times (M - 0.76)^3 \quad (93)$$

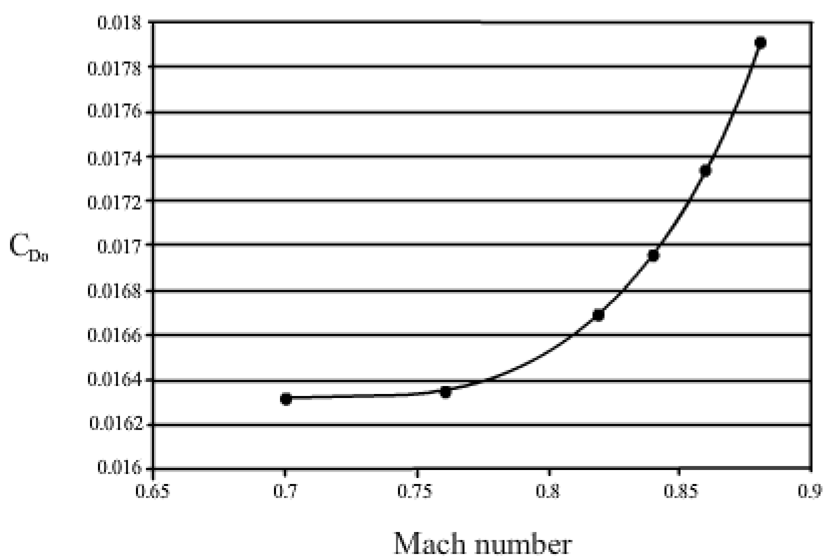

Fig.14 Variation of C_{DO} with Mach number

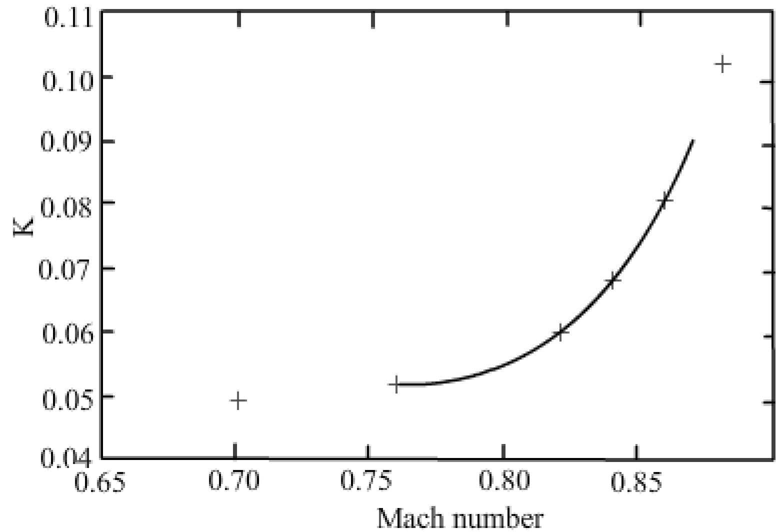

Fig. 15 Variation of K with Mach number

In the case of the present airplane, the cruise Mach number is 0.8. The variations of C_{DO} and K above M_{cruise} and upto M = 0.9, based on B727-100 data are taken as follows.

$$C_{DO} = 0.0159 - 0.001 \times (M - 0.8) + 0.11 \times (M - 0.8)^2 \quad (94)$$

$$K = 0.04244 + (M - 0.8)^2 + 20.0 \times (M - 0.8)^3 \quad (95)$$

(iii)The thrust available and thrust required curves are plotted at each altitude as a function of velocity. The points of intersection give the $(V_{min})_e$ and V_{max} at each altitude (Fig.21); $(V_{min})_e$ is the minimum speed from thrust available consideration. To arrive at V_{min}, the minimum speed of the airplane at an altitude, the stalling speed (V_s) also needs to be taken into account. V_{min} is higher of $(V_{min})_e$ and V_s. Since, the drag polar is not valid below V_s , in Figs 16 to 21 the thrust required curves are plotted only when $V > V_s$; V_s is taken for C_{Lmax} without flaps.The calculations are carried out for h = 0, 10000, 15000, 25000, 30000 and 36000 ft, i.e S.L, 3048, 4572, 7620, 9144 and 10972.8 m using T_{avail} as climb thrust and cruise thrust. The plots are presented only for climb thrust case.

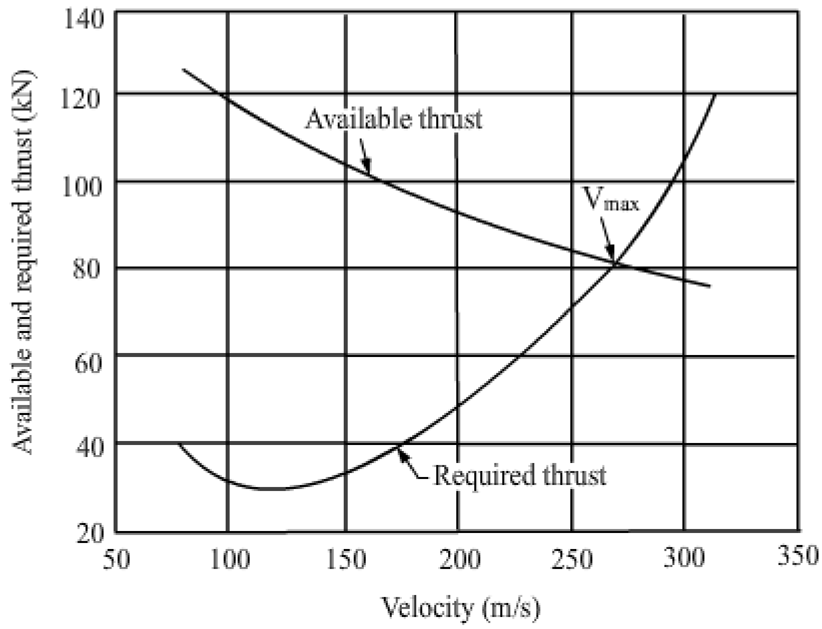

Fig.16 Available and required thrust at s.l

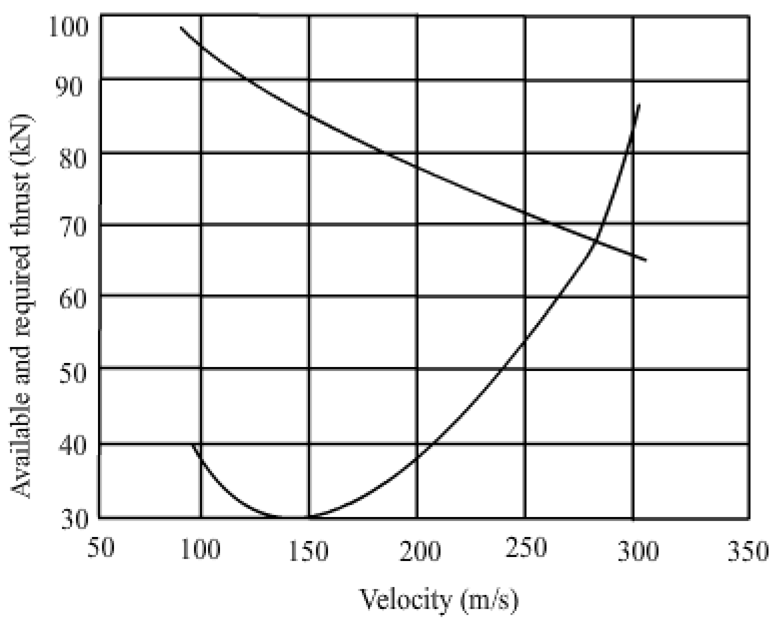

Fig. 17 Available and required thrust at h = 3048 m

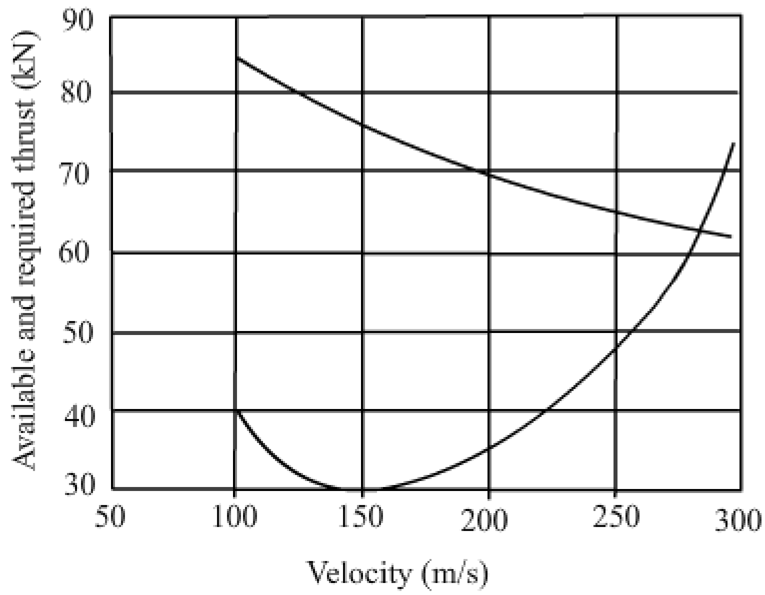

Fig. 18 Available and required thrust at h = 4572 m

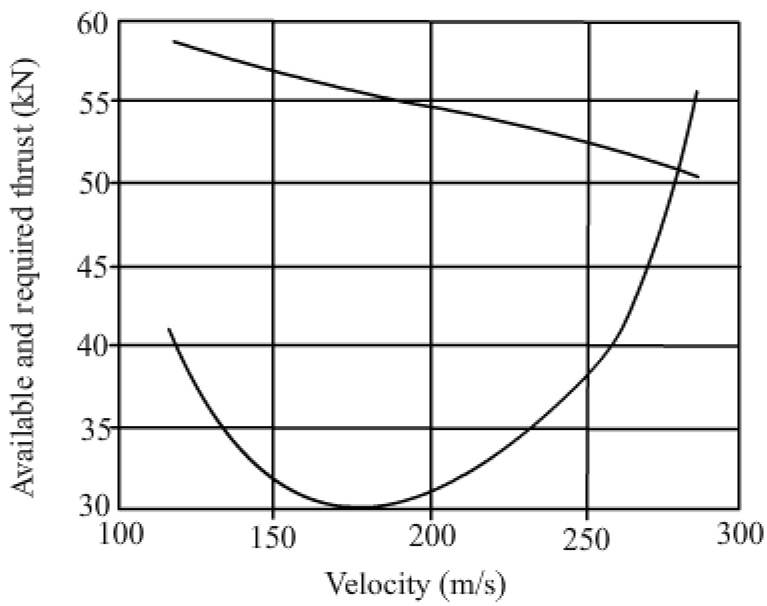

Fig.19 Available and required thrust at h = 7620 m

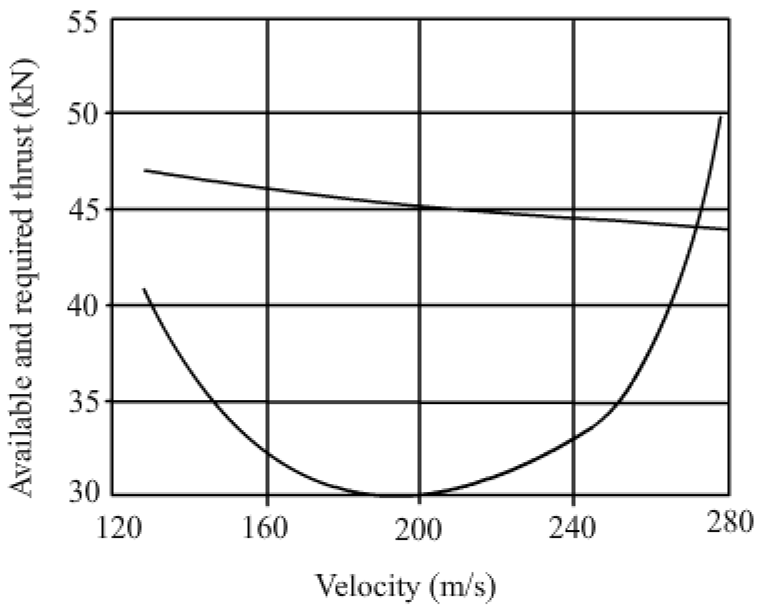

Fig. 20 Available and required thrust at h = 9144 m

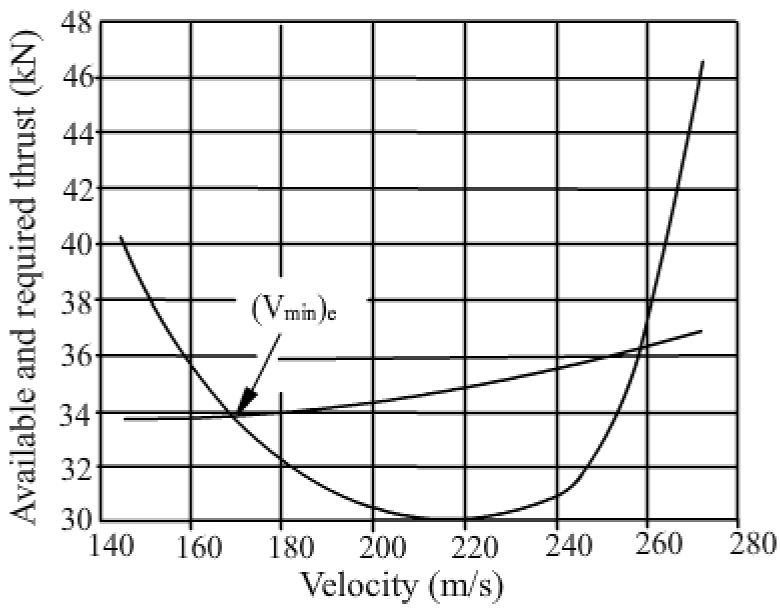

Fig. 21 Available and required thrust at h = 10972 m

The variations of V_s, $(V_{min})_e$ and V_{max} are tabulated in Table 8 and presented in Fig.22.

h (ft)	h (m)	V_s (m/s)	$(V_{min})_e$ (m/s) $T = T_{cr}$	$(V_{min})_e$ (m/s) $T = T_{climb}$	V_{max} (m/s) $T = T_{cr}$	V_{max} (m/s) $T = T_{climb}$	V_{max} (kmph) $T = T_{climb}$
S.L	0	77.833	< V_s	< V_s	258.711	269.370	969.7
10000	3048	90.579	< V_s	< V_s	272.060	280.595	1010.1
15000	4572	98.131	< V_s	< V_s	275.613	283.300	1019.9
25000	7620	116.292	< V_s	< V_s	272.929	279.291	1005.4
30000	9144	127.278	< V_s	< V_s	267.854	271.755	978.3
36000	10973	142.594	176.054	169.071	253.671	258.154	929.4
38000	11582	149.557	217.386	200.896	243.676	248.630	895.1
38995	11884	153.159	235.471	229.865	235.48	238.649	859.1
39220	11954	153.950	-	236.40	-	236.40	851.04

Table 8: Variations of V_{min} and V_{max}

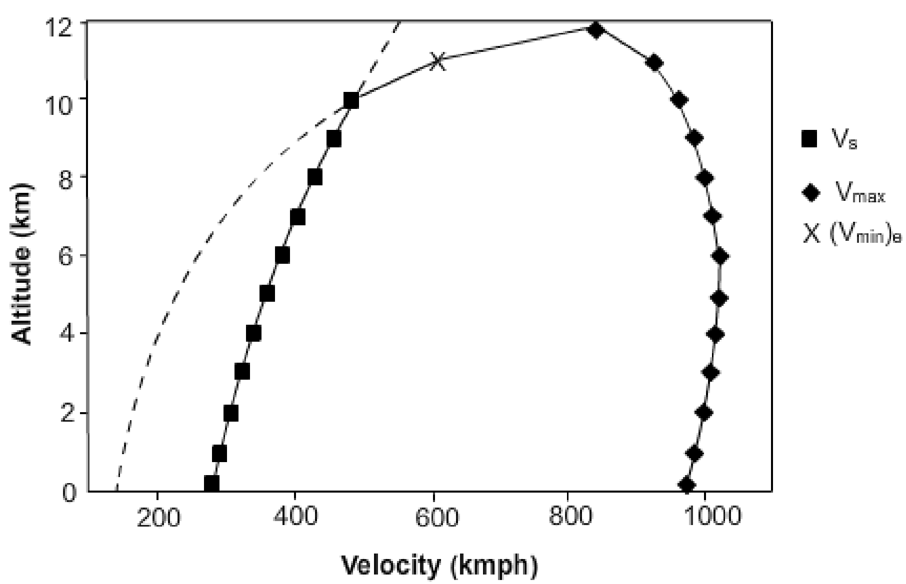

Fig. 22 Variations of V_{min} and V_{max} with altitude

9.4 Steady climb

In this flight, the c.g. of the airplane moves along a straight line inclined to the horizontal at an angle γ . The velocity of flight is assumed to be constant during the climb. Since, the flight is steady, acceleration is zero and the equations of motion can be written as:

$$T - D - W \sin \gamma = 0 \quad (96)$$

$$L - W \cos \gamma = 0 \quad (97)$$

To calculate the variation of rate of climb with flight velocity at different altitudes, the following procedure is adopted.

(i) Choose an altitude.

(ii) Choose a flight speed.

Noting that $C_L = 2\,W\cos\gamma\ / \rho S V^2$, gives :

$$C_D = C_{DO} + K\left(\frac{2W\cos\gamma}{\rho S V^2}\right)$$

Also

$$V_c = V\sin\gamma$$

$$\cos\gamma = \sqrt{1-\frac{V_c^2}{V^2}}$$

Using above equations yields the following equation for (V_c/V),

$$A\left(\frac{V_c}{V}\right)^2 + B\left(\frac{V_c}{V}\right) + C = 0 \qquad (98)$$

$$A = \frac{kW^2}{\frac{1}{2}\rho V^2 S};\ B = -W;\ C = T_{avail} - \frac{1}{2}\rho V^2 S C_{DO} - \frac{2kW^2}{\rho V^2 S} \qquad (99)$$

Since, altitude and flight velocity have been chosen, the thrust available is read from the climb thrust curves in Fig.10. Further the variation of C_{DO} and K with Mach number is taken as in Eqs. (94) and (95).

(iii) Equation (98) gives 2 values of V_c/V. The value which is less than or equal to 1.0 is chosen as $\sin\gamma$ cannot be greater than unity. Hence,

$$\gamma = \sin^{-1}(V_c / V) \qquad (100)$$

$$V_c = V\sin\gamma \qquad (101)$$

(iv)This procedure is repeated for various speeds between V_{min} and V_{max}.

The entire procedure is then repeated for various altitudes.

The variations of (R/C) and γ_{max} with velocity and with altitude as parameters are shown in Figs. 23 and 25. The variations of $(R/C)_{max}$ and γ_{max} with altitude are shown in Figs. 24 and 26. The variations of $V_{(R/C)max}$ and $V_{\gamma max}$ with altitude are shown in Figs. 27 and 28. A summary of results is presented in Table 9.

h (ft)	h (m)	$(R/C)_{max}$ (m/min)	$V_{(R/C)max}$ (m/s)	γ_{max} (degrees)	$V_{\gamma max}$ (m/s)
0	0.0	1086.63	149.7	8.7	88.5
10000	3048.0	867.34	167.5	6.0	111.6
15000	4572.0	738.16	174.0	4.7	125.7
25000	7620.0	487.41	198.2	2.6	164.1
30000	9144.0	313.43	212.2	1.5	188.0
36000	10972.8	115.57	236.1	0.5	230.2
38000	11582.4	41.58	236.9	0.2	234.0
38995	11885.7	1.88	235.8	0.0	235.8

Table 9: Climb performance

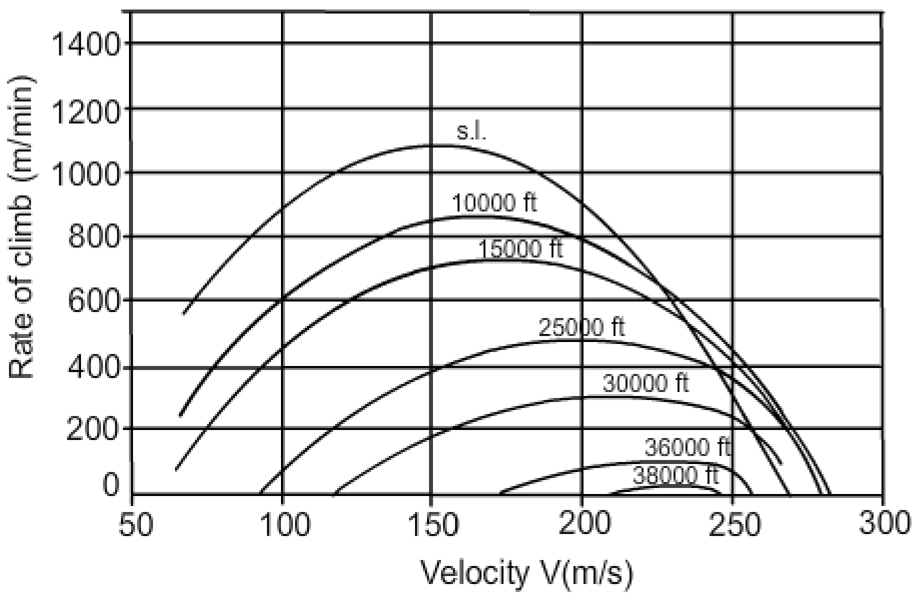

Fig.23 Rate of climb vs velocity for various altitudes

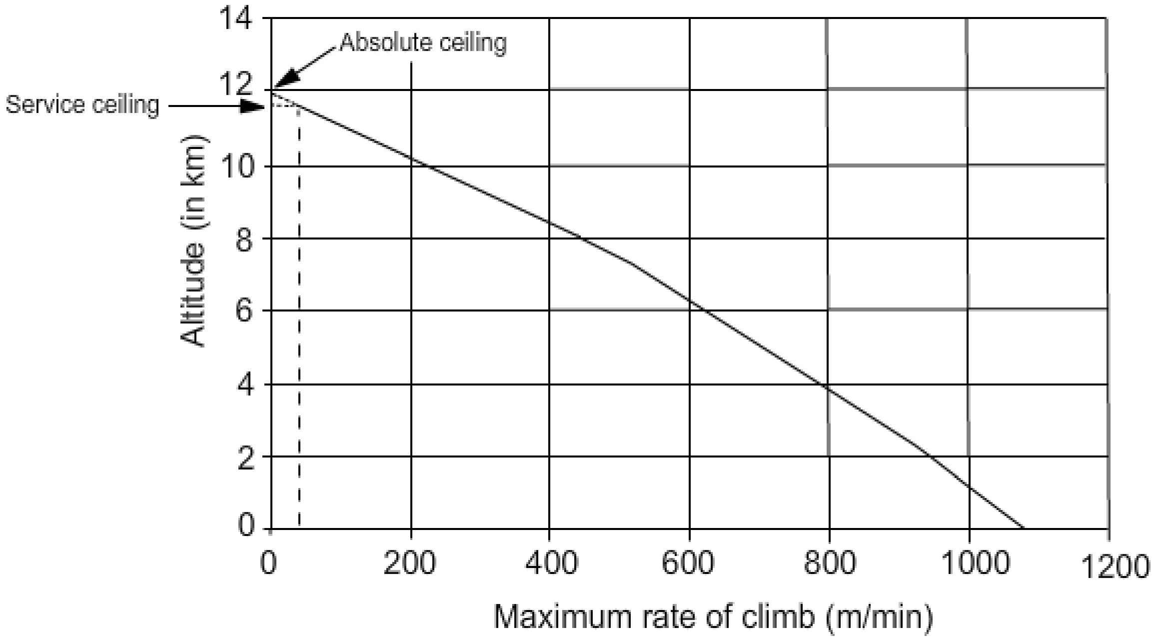

Fig.24 Maximum rate of climb vs altitude

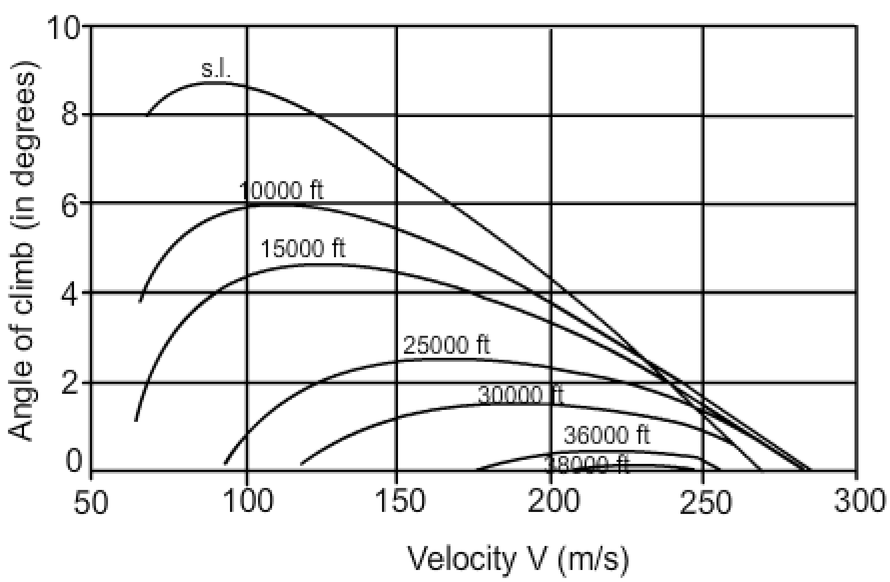

Fig. 25 Angle of climb vs velocity for various altitudes

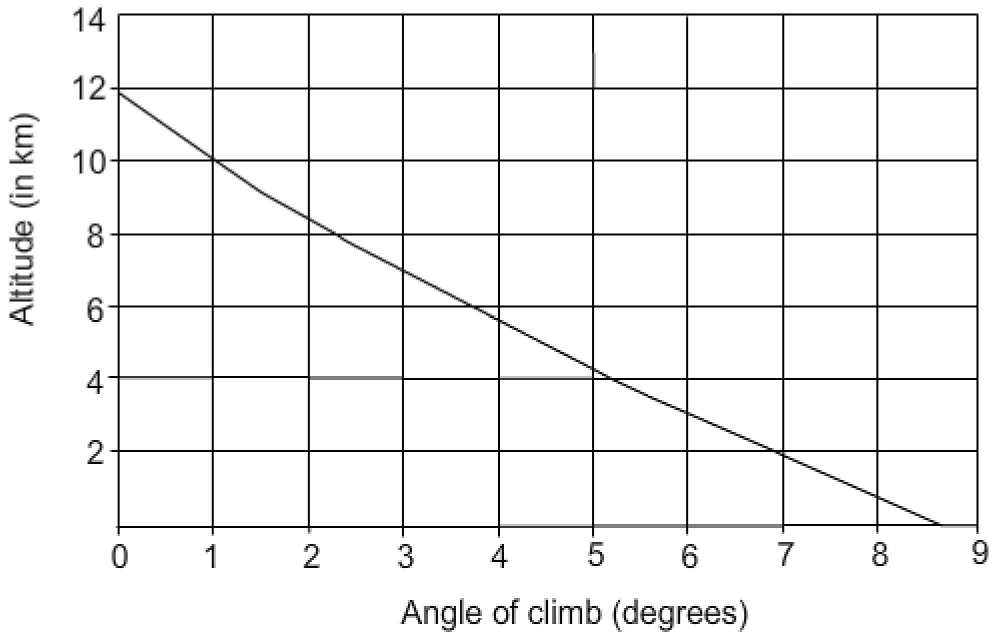

Fig. 26 Maximum angle of climb vs altitude

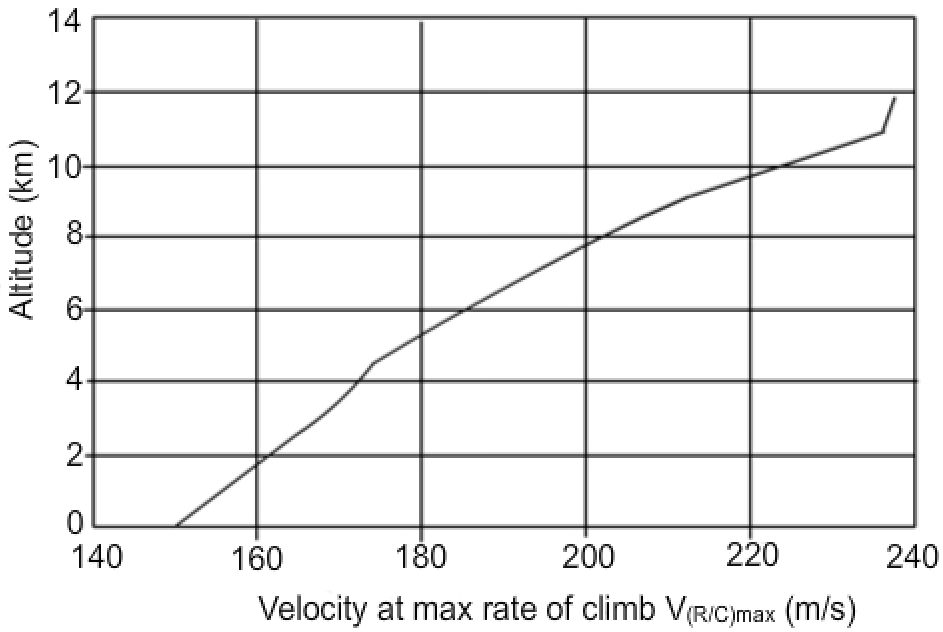

Fig. 27 Velocity at maximum rate of climb vs altitude

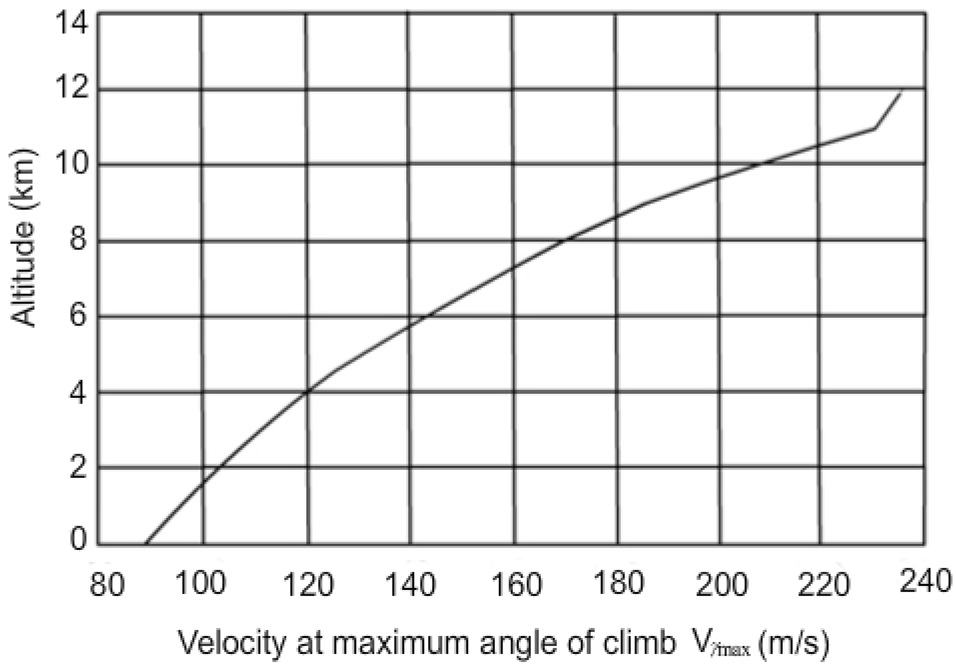

Fig.28 Velocity at maximum angle of climb vs altitude

Remarks:

(i)The discontinuity in slope in Figs. 27 and 28 at high velocities are due to the change in drag polar as the Mach number exceeds 0.8.

(ii) From Fig. 24, the absolute ceiling (at which $(R/C)_{max}$ is zero) is 11.88 km. The service ceiling at which $(R/C)_{max}$ equals 30 m/min is 11.68 km

9.5 Range and endurance

In this section, the range of the aircraft in a constant altitude and constant velocity cruise is studied. Range is given by the formula:

$$R = \frac{3.6\ V}{TSFC\sqrt{KC_{DO}}}\left[\tan^{-1}\frac{2W_1}{\rho V^2 S}\sqrt{\frac{K}{C_{DO}}} - \tan^{-1}\frac{2W_2}{\rho V^2 S}\sqrt{\frac{K}{C_{DO}}}\right] \qquad (102)$$

where, W_1 is the weight of the airplane at the start of the cruise and W_2 is the weight of the airplane at the end of the cruise.

The cruising altitude is taken as h = 10972 m. TSFC is taken to be constant as $0.6hr^{-1}$. The variation of drag polar above M = 0.8 is given by Eqs. (94) and (95).

$W_1 = W_0 = 59175 \times 9.81N$

$W_f = 0.205 \times W_1$

Allowing 6% fuel as trapped fuel, W_2 becomes

$W_2 = W_1 - 0.94 \times W_f$

The values of endurance (in hours) are obtained by dividing the expression for range by 3.6V where V is in m/s. The values of range (R) and endurance(E) in flights at different velocities are presented in Table 10 and are plotted in Figs.29 and 30.

M	V (in m/s)	C_{DO}	K	R (in km)	E (in hours)
0.50	147.53	0.0159	0.04244	2979.0	5.61
0.55	162.29	0.0159	0.04244	3608.0	6.18
0.60	177.04	0.0159	0.04244	4189.6	6.57
0.65	191.79	0.0159	0.04244	4691.7	6.80
0.70	206.54	0.0159	0.04244	5095.6	6.85
0.75	221.30	0.0159	0.04244	5396.5	6.77
0.80	236.05	0.0159	0.04244	5599.8	6.59
0.81	239.00	0.0159	0.04256	5619.7	6.53
0.82	241.95	0.01592	0.04300	5621.6	6.45
0.83	244.90	0.01597	0.04388	5597.7	6.35
0.84	247.85	0.01604	0.04532	5544.1	6.21
0.85	250.80	0.01613	0.04744	5460.4	6.05
0.86	253.75	0.01624	0.05036	5349.3	5.86
0.87	256.71	0.01637	0.05420	5210.1	5.64
0.88	259.66	0.01652	0.05908	5051.1	5.40

Table 10: Range and endurance in constant velocity flights at h = 10972 m (36000 ft)

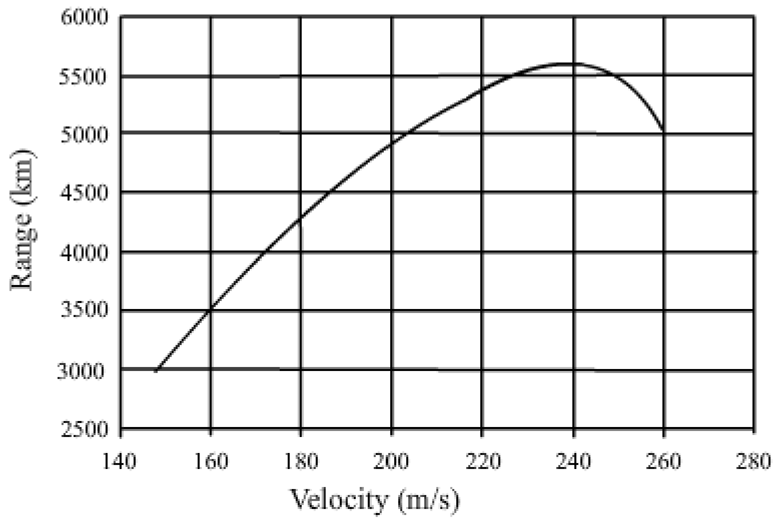

Fig.29 Range in constant velocity flights at different flight speeds (h = 10972 m)

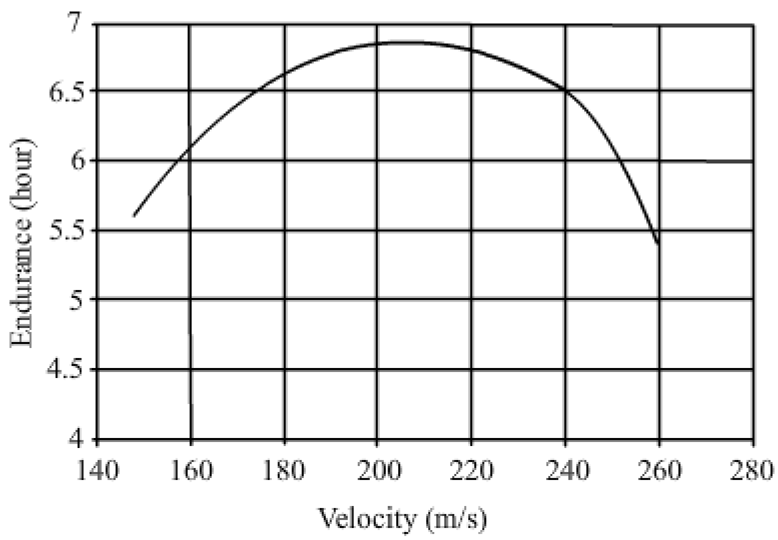

Fig.30 Endurance in constant velocity flights at different flight speeds (h = 10972 m)

Remarks:

(i) It is observed that the maximum range of 5600 km is obtained at a velocity of 239 m/s (860 kmph). Corresponding Mach number is 0.81. which is slightly higher than the Mach number beyond which C_{DO} and K increase. This can be explained based on two factors namely

(a) the range increases as the flight speed increases (b) after M_{cruise} is exceeded, C_{DO} and K increase thus reducing $(L/D)_{max}$.

(ii) The range calculated above is the gross still air range. The safe range would be about two-thirds of this. In the present case, the safe range would be 3733 km.

(iii)The maximum endurance of 6.85 hours occurs in a flight at V = 206 m/s. (742 kmph). It can noted that the endurance is roughly constant over a speed range of 190 m/s to 230 m/s.

9.6 Turning performance

In this section, the performance of the airplane in a steady, co-ordinated, level turn is studied. The equations of motion in this case are:

$$T - D = 0$$

$$W - L\cos\phi = 0$$

$$L\sin\phi = \frac{W}{g}$$

where, ϕ is the angle of bank.

These equations give:

$$r = \frac{V^2}{g\tan\phi}$$

$$\dot{\psi} = \frac{V}{r} = \frac{g\tan\phi}{V}$$

$$\text{Load factor } (n) = \frac{L}{W} = \frac{1}{\cos\phi}$$

where, $\dot{\psi}$ is the rate of turn and r is the radius of turn.

The following procedure is used to obtain r_{min} and $\dot{\psi}_{max}$

1.A flight speed and altitude are chosen and the level flight lift coefficient C_{LL} is obtained as :

$$C_{LL} = \frac{2(W/S)}{\rho V^2}$$

2. If $C_{Lmax} / C_{LL} < n_{max}$, where n_{max} is the maximum load factor for

which the aircraft is designed, then the turn is limited by C_{Lmax} and $C_{LT1} = C_{Lmax}$. However, if $C_{Lmax} / C_{LL} > n_{max}$, then the turn is limited by n_{max}, and $C_{LT1} = n_{max}C_{LL}$.

3. From the drag polar, C_{DT1} is obtained corresponding to C_{LT1} . Then,

$$D_{T1} = \frac{1}{2}\rho V^2 S\, C_{DT1}$$

If $D_{T1} > T_a$, where, T_a is the available thrust at that speed and altitude, then the turn is limited by the engine output. In this case, the maximum permissible value of C_D in turning flight is obtained from :

$$C_{DT} = \frac{T_a}{\frac{1}{2}\rho V^2 S}$$

From the above relation, the value of C_{LT} is calculated as

$$C_{LT} = \sqrt{\frac{C_{DT} - C_{DO}}{K}}$$

However, if $D_{T1} < T_a$, then the turn is not limited by the engine output and the value of C_{LT} calculated in step (2) is retained.

4. Once C_{LT} is known, the load factor during the turn is determined as

$$n = \frac{C_{LT}}{C_{LL}}$$

Once n is known, the values of ϕ, r and $\dot{\psi}$ can be calculated using the equations given above.

The above steps are repeated for various speeds and altitudes. A typical turning flight performance estimation is presented in Table 11. In these calculations, C_{Lmax} = 1.4 and n_{max} = 3.5 are assumed. The variation of turning flight performance with altitude is shown in Table. 12. Figures 31, 32, 33, 34 respectively present (a) radius of turn with velocity and with altitude as parameter, (b) minimum radius of turn with altitude, (c) rate of turn with velocity and with altitude as parameter and (d) maximum rate of turn with altitude.

V (m/s)	n	C_{LT}	ϕ (degrees)	r (m)	$\dot{\psi}$ (rad/s)
78.83	1.026	1.4000	12.892	2767.70	0.0285
98.83	1.612	1.4000	51.670	787.21	0.1255
118.83	2.331	1.4000	64.596	683.63	0.1738
138.83	2.813	1.2376	69.173	747.41	0.1858
158.83	2.993	1.0062	70.482	911.60	0.1742
178.83	3.089	0.8192	71.112	1115.38	0.1603
198.83	3.080	0.6607	71.053	1383.50	0.1437
218.83	2.930	0.5189	70.045	1772.43	0.1235
238.83	2.573	0.3826	67.132	2452.36	0.0974
241.83	2.494	0.3617	66.363	2609.20	0.0927

Table 11 Typical turning flight performance at sea level

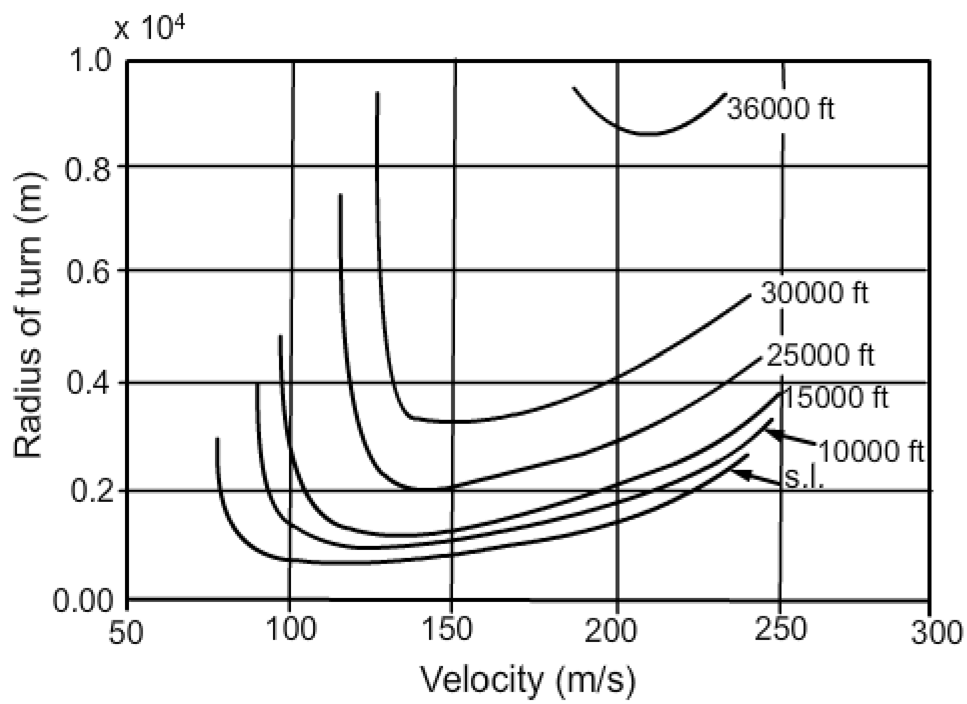

Fig.31 Radius of turn vs velocity at various altitudes

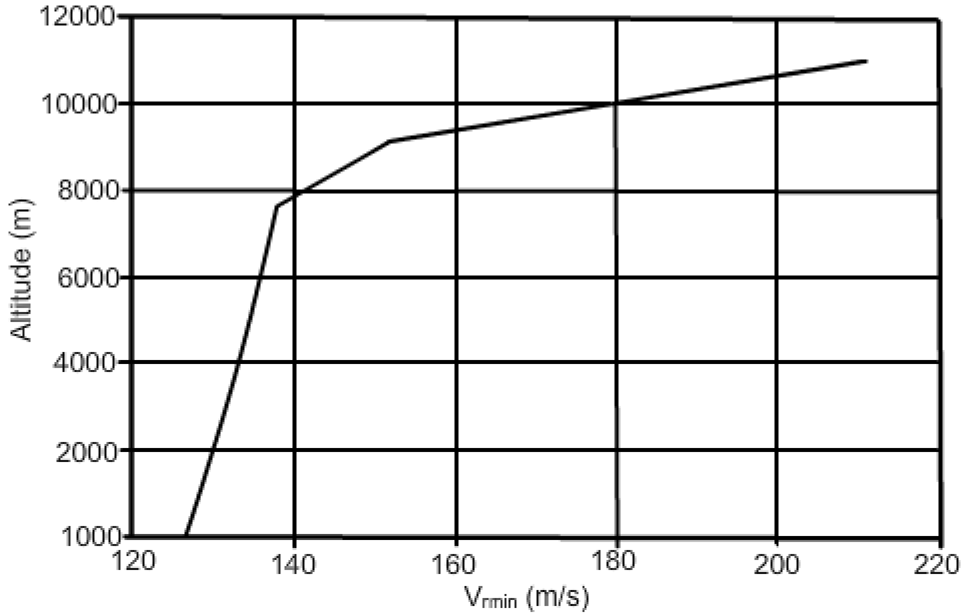

Fig.32 Velocity at r_{min} vs altitude

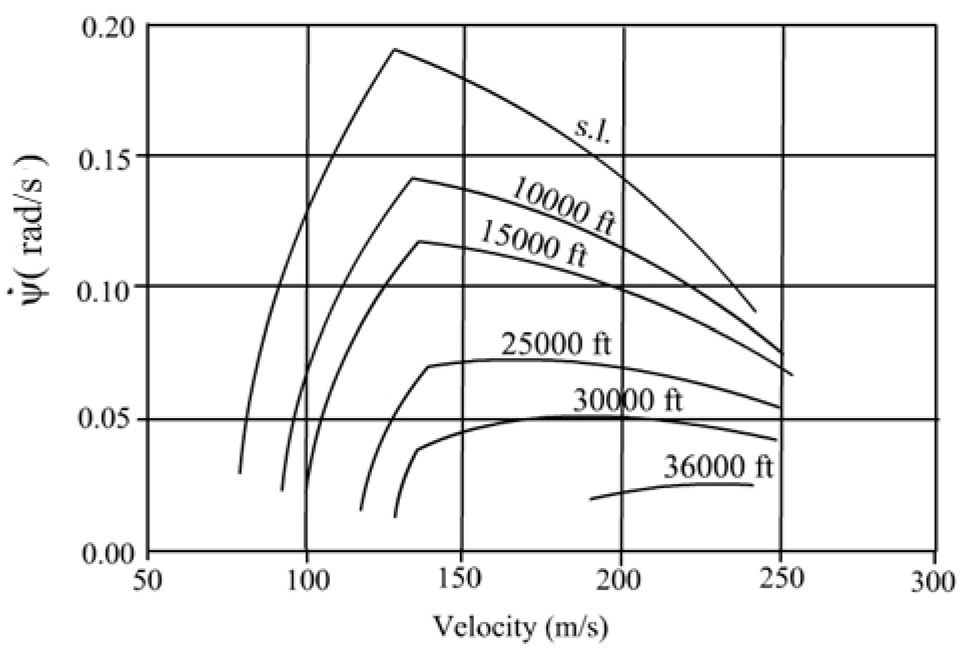

Fig. 33 $\dot{\psi}$ vs speed at various altitudes

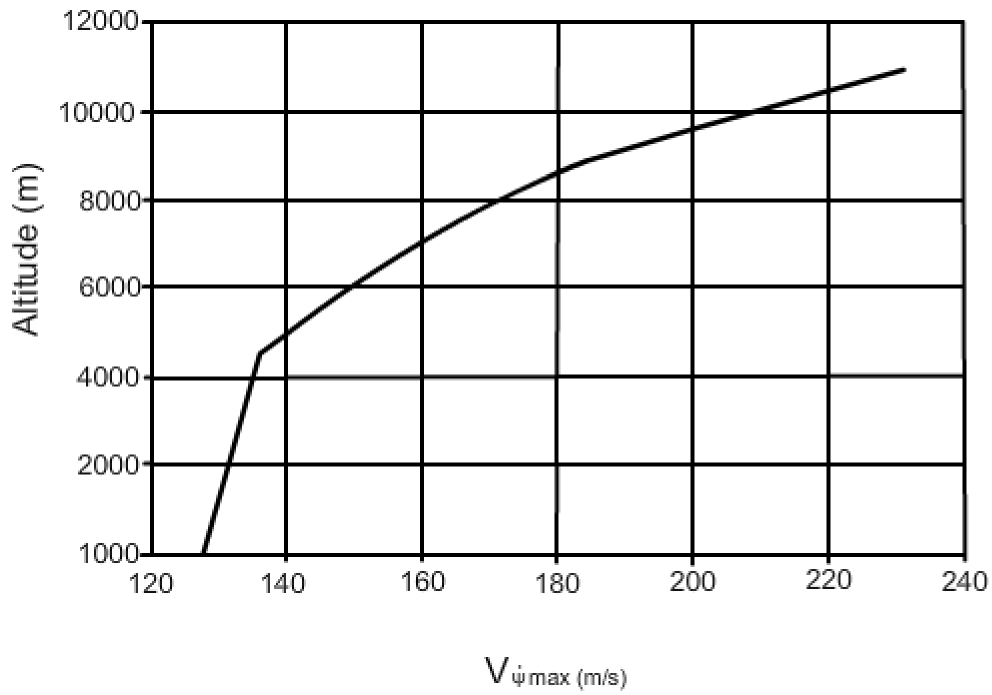

Fig. 34 Velocity at $\dot{\psi}_{max}$ vs altitude

h (m)	r_{min} (m)	V_{rmin} (m/s)	$\dot{\psi}_{max}$ (rad/s)	$V_{\dot{\psi}max}$ (m/s)
0.0	666	126.8	0.1910	127.8
3048.0	945	132.6	0.1410	133.6
4572.0	1155	135.1	0.1170	136.1
7620.0	1971	138.3	0.0731	165.3
9144.0	3247	151.3	0.0513	187.3
10973.8	8582	211.0	0.0256	231.0

Table 12 Turning flight performance

Remarks:

1. The maximum value of $\dot{\psi}_{max}$ is 0.191 and occurs at a speed of 127.8 m/s at sea level.
2. The minimum radius of turn is 666 m and occurs at a speed of 126.8 m/s at sea level.
3. The various graphs show a discontinuity in slope when the criterion, which limits the turn, changes from n_{max} to thrust available.

9.7 Take-off distance

In this section, the take off performance of the airplane is evaluated. The take-off distance consists of take-off run, transition and climb to screen height. Rough estimates of the distance covered in these phases can be obtained by writing down the appropriate equations of motion. However, the estimates are approximate and Ref.4 chapter 5 recommends the following formulae for take-off distance and balance field length based on the take-off parameter.

This parameter is defined as:

$$\text{Take-off parameter} = \frac{W/S}{\sigma C_{LTO}(T/W)} \tag{103}$$

where, W/S is wing loading in lb/ft^2, C_{LTO} is $0.8 \times C_{Lland}$ and σ is the density ratio at take-off altitude.

In the present case:

$$\frac{W}{S} = 5195\ N/m^2 = 108.2\ lb/ft^2;\ C_{LTO} = 0.8 \times 2.7 = 2.16;\ \sigma = 1.0\ (\text{sea level})$$

$$\text{and } \frac{T}{W} = \frac{2 \times 97.9\ kN}{59175 \times 9.81} = 0.3373$$

Hence,

$$\text{Take-off parameter} = \frac{108.2}{1.0 \times 2.16 \times 0.3373} = 148.86 \tag{104}$$

From Ref.4, chapter 5, the take off distance, over 50', is $2823'$ or 861 m. The balance field length for the present case of two engined airplane is $6000'$ or 1829 m.

Remark:

It may be noted that the balance field length is more than twice the take off distance.

9.8 Landing distance

In this section the landing distance of the airplane is calculated. From Ref.4 chapter 5, the landing distance for commericial airliners is given by the formula

$$S_{land} = 80\left(\frac{W}{S}\right)\frac{1}{\sigma C_{Lmax}} + 1000 \text{ ft} \tag{105}$$

where, W/S is in lbs/ft^2. In the present case:

$(W/S)_{land} = 0.85 \times (W/S)_{takeoff} = 0.85 \times 108.5 = 92.225 lb/ft^2$

$C_{Lmax} = 2.7$

$\sigma = 1.0$

Hence,

$$S_{land} = 80 \times 92.225\frac{1}{1.0 * 2.7} + 1000 = 3732 \text{ ft} = 1138 \text{ m} \tag{106}$$

9.9 Concluding remarks

1. Performance of a typical commercial airliner has been estimated for stalling speed, maximum speed, minimum speed, steady climb, range, endurance, turning, take-off and landing.
2. The performance approximately corresponds to that of B737-200.
3. Figure 35 presents the variation with altitude of the characteristic velocities corresponding to :

   stalling speed, V_s

   maximum speed, V_{max}

   minimum speed as dictated by thrust, $(V_{min})_e$

   maximum rate of climb, $V_{(R/C)max}$

   maximum angle of climb, $V_{\gamma max}$

   maximum rate of turn, $V_{\dot{\psi}max}$

   minimum radius of turn, V_{rmin}

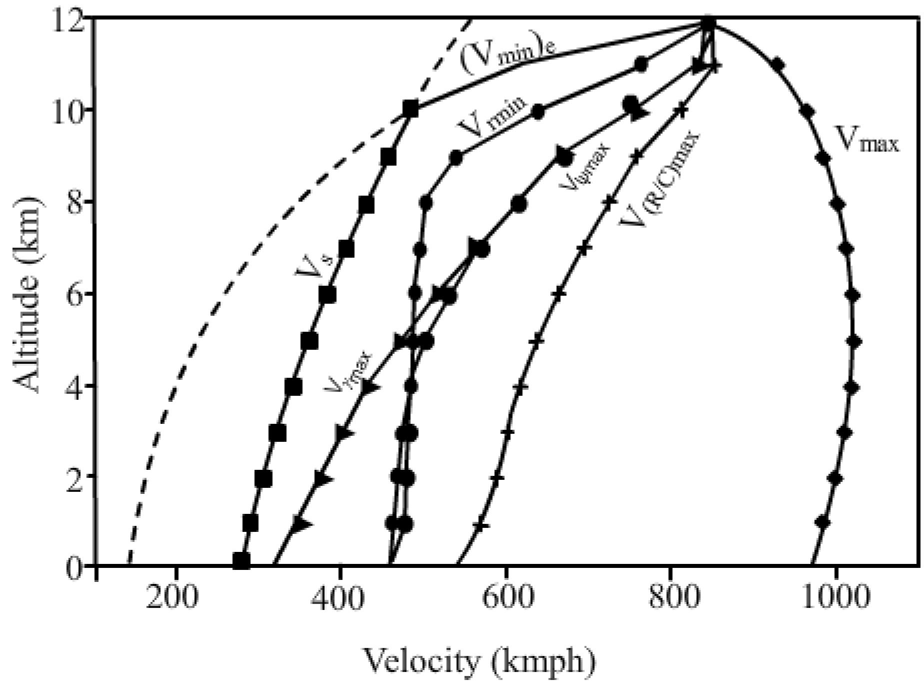

Fig. 35 Variations of characteristic velocities with altitude

10 Acknowledgements

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