İSTANBUL TECHNICAL UNIVERSITY ★ INSTITUTE OF SCIENCE AND TECHNOLOGY

CONCEPTUAL DESIGN AND SIZING OF JET-POWERED AIRCRAFT USING LOFTIN METHOD

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İSTANBUL TEKNİK ÜNİVERSİTESİ ★ FEN BİLİMLERİ ENSTİTÜSÜ

LOFTİN METODU KULLANILARAK JET MOTORLU BİR UÇAĞIN KONSEPT DİZAYNI VE BOYUTLANDIRILMASI

Yüksek Lisans Tezi
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ŞUBAT 2000

ÖNSÖZ

Üniversiteden mezun olduktan sonra Türk Hava Yolları'nda Yapısal Sistemler Mühendisi olarak uçak bakımı, tamiri ve modifikasyonu konularında çalışıyorum. Mesleğim konusunda kendimi geliştirebilmek ve havacılığın süreçlerine dizayndan başlayarak daha iyi hakim olabilmek için böyle bir çalışmayı yaptım. Bu çalışmanın, bir uçak dizaynında hangi kriterlerin göz önüne alınması gerektiği ve uygulanacak prosedürler konusunda lisans öğrenimim sırasında edinmiş olduğum bilgileri ilerletme konusunda bana çok iyi bir imkan sağladığına inanıyorum.

Çalışmam esnasında benden yardımlarını esirgemeyen değerli bilim adamı hocam Sn. Doç. Dr. Y. Kemal Yıllıkçı'ya teşekkürlerimi sunuyorum. Ayrıca çalışmalarım sırasında beni destekleyen aileme ve eşim Burak Özmenekşe'ye de teşekkürlerimi sunuyorum.

Şubat 2000 Sevil SÜSLÜ ÖZMENEKŞE

CONTENTS

FIGURE LIST

SYMBOLS

a	: Average acceleration of aircraft along ground
A	: Aspect ratio, b^2 / S
B	: Breguet factor
c	: Root chord, ft
C_d	: Total drag coefficient
C_{di}	: Induced drag coefficient
C_{dp}	: Total drag coefficient
C_{df}	: Drag coefficient due to flap deflection
C_{dit}	: Drag coefficient due to landing gear
C_{d0}	: Drag coefficient at zero lift
C_{dm}	: Drag coefficient corresponding to maximum lift.
C_l	: Lift coefficient
$\underline{C_l}$	: Ratio of the lift coefficients
$\underline{C_{lbs}}$	: Ratio of beginning lift coefficient to maximum lift coefficient
$\underline{C_{lbt}}$	: Ratio of ending lift coefficient to maximum lift coefficient
C_{ly}	: Approach lift coefficient
C_{lm}	: Lift coefficient for $(L/D)_{max}$
d	: Fuselage diameter, ft
D	: Drag, lb
e	: Oswald's airplane efficiency factor
F_{or}	: Average value of (L/D)
F_{bs}	: Value of (L/D) at the beginning
F_{bt}	: Value of (L/D) at the cruise end
g	: Acceleration due to gravity, ft/sec^2
K	: Constant value
K_{bs}	: Ratio of F_{bs} to the maximum value of (L/D)
K_{bt}	: Ratio of F_{bt} to maximum (L/D)
l	: Fuselage length, ft
L	: Lift, lb
$\underline{L}$	: Scale parameter
l_{yk}	: The distance required to accelerate to the lift-off speed
M	: Mach number
N	: Constant value
P	: Atmospheric pressure, lb/ft^2
q	: Dynamic pressure, lb/ft^2
r	: Atmospheric density
R	: Range, nmi.
S	: Wing area, ft^2

S_{ts}	: Wetted area, ft^2
S_{ky}	: Sum of vertical and horizontal tail area, ft^2
S_{mb}	: Nacelle wetted area, ft^2
SFC	: Engine specific fuel consumption (liber of fuel per liber of thrust per hour)
TOP	: Take-off parameter
T_0	: Maximum sea-level static thrust, lb
T_c	: Cruise thrust, lb
u	: Span loading parameter
$\underline{U}$	: Useful load fractions
V	: Speed, knots
V_y	: Approach speed, knots
V_{yk}	: The aircraft starts from zero and reaches the take-off speed
W	: Weight, lb
W_{bv}	: Empty weight, lb
W_g	: Maximum take-off gross weight, lb
W_i	: Landing weight, lb
W_{mxi}	: Maximum landing weight, lb
W_p	: Payload weight, lb
W_y	: Fuel weight, lb
ρ	: Atmospheric density, slugs/ft^3
ρ_0	: Atmospheric density at sea level
σ	: Density ratio, ρ/ρ_0
γ	: Ratio of specific heats or flight-path angle

Üniversitesi	**: İstanbul Teknik Üniversitesi**
Enstitüsü	**: Fen Bilimleri**
Anabilim Dalı	**: Uçak Mühendisliği**
Programı	**: Uçak Mühendisliği**
Tez Danışmanı	**: Doç. Dr. Y. Kemal YILLIKÇI**
Tez Türü ve Tarihi	**: Yüksek Lisans – Şubat 2000**

ÖZET

LOFTİN METODU KULLANILARAK JET MOTORLU BİR UÇAĞIN KONSEPT DİZAYNI VE BOYUTLANDIRILMASI

Sevil SÜSLÜ ÖZMENEKŞE

Birinci ve ikinci kısımlarda ön bilgi verme amacıyla açıklama yapılmıştır. Üçüncü kısımda yöntemin ana hatları açıklanmıştır. Yöntemin uygulanışına daha başka kriterleri de katmak mümkündür. Sonuç şeklin oluşturulmasındaki amaç misyon şartlarını sağlayan ve sağlamayan bölgeleri görebilmektedir. Örneğin kalkış pist uzunluğu için çizilen eğrinin altında kalan bölge içinde çözüme uygun bir nokta yoktur. Diğer iki eğri içinde aynı durum söz konusudur. Bunlardan farklı olarak belirli bir kanat yüklemesi değer aralığı tanımlanmışsa küçük olan değeri ifade eden soldaki dikey doğrunun solundaki alan içinde herhangi bir çözüm söz konusu olamayacaktır. Tabii yöntemi uygularken belirli bir kanat yüklemesi önşartı kullanılmayıp her üç eğri içinde dörder veya beşer kanat yüklemesi – çekme yüklemesi ikilisi oluşturulmuştur. Yorumlar 4. ve 5. kısımlarda sunulmuştur. Bir tablolama programı olan Excel' de hazırlanan örnek uygulamada ödevde bulunmaktadır.

Anahtar Kelimeler: Uçak Dizaynı, Misyon Şartları, Performans Karakteristikleri

Bilim Dalı Sayısal Kodu: 618.01.00

University	**: İstanbul Technical University**
Institute	**: Institute of Science and Technology**
Science Programme	**: Aeronautical Engineering**
Programme	**: Aeronautical Engineering**
Supervisor	**: Assoc. Prof. Dr. Y. Kemal YILLIKÇI**
Degree Awarded and Date	**: Msc – February 2000**

SUMMARY

CONCEPTUAL DESIGN AND SIZING OF JET POWERED AIRCRAFT USING LOFTIN METHOD

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Brief information was given in the first and second chapters. The third chapter includes the main principles of the method. Some other criteria could be included in the method. The final figure was formed in order to clearify the zones satisfying the mission requirements. For instance, there is no matching point satisfying the mission requirements under take-off field length curve. For the other two curves, there are no matching points either. If a wing loading value has been defined, there will be no matching point at the left-hand side of the left vertical line. For the three curves, four or five wing loading-thrust loading couples were formed instead of making assumption for the wing loading. Comments have been presented in chapters 4 and 5. Also a sample performed in MS Excel has been presented in this study.

Keywords: Aircraft Design, Mission Requirements, Performance Characteristics
Science Code: 618.01.00

1. INTRODUCTION

The aim of this project is to adapt Loftin Jet Aircraft Conceptual design method to an MS Excel database. During the project, previous studies on Loftin Method were investigated. In this project, Loftin Method was studied and developing an Excel database, a sample design has been made.

Conceptual design means determining the major dimensions of an aircraft starting from the selected or given performance and mission criteria of the aircraft. The mission of an aircraft is to transport a certain amount of payload to the highest possible range, in the shortest time, in a safe and cost effective way. Maximum range, load, speed and safety are the key criteria for design. Design, development, test, manufacturing and maintenance of an aircraft are subjected to some rules set by regulations and specifications.

Issues like airport specifications, engine noise limits, mission characteristics are all standardized in those international regulations. ICAO is an international regulation. FAR is national regulation of the U.S. The U.K. uses BCAR regulation that is quite similar to FAR.

The first step to design an aircraft is to determine the demand considering the amount of the total budget of the project. At the end of the demand analysis a tangible target (such as " an inter-city taxi jet aircraft will be designed with 6 people capacity and 800 aircraft will be sold within 15 years time period and spare parts and maintenance will be provided") should be set.

If the demand is sufficient to design a new aircraft, conceptual design phase starts. In this phase Loftin or some other methods like Roskam, Raymer etc. can be used. The general features and dimensions of the required aircraft is determined with the conceptual design.

Performance means flight characteristics. Range, speed, altitude, maximum payload, cost efficiency are the key criteria in defining the performance. Conceptual design methods provide only some performance and mission criteria. In order to determine the cost of development manufacturing and operation of the aircraft, different methods should be used. In this study, related equations to design an aircraft using Loftin Method are created in Macintosh Excel environment. A conceptual design has been presented made by the developed database software.

When Loftin Method is used, the input data used in the beginning phase of the method must be logical to provide correct output data.

For instance, if landing field length is used as 3.300 ft. during the simulation and in order to satisfy other mission characteristics we found that landing distance must be 3.800 ft. It means that the aircraft will need 15 % longer landing distance. Application steps of design have been shown in the study. Statistical data and graphics of modern business jet aircraft were added to the study. As an example, a design was added to the study as well. Flow charts were also provided and the charts used for performance calculations were taken from reference [1].

2. CONCEPTUAL DESIGN

2.1 Introduction of the Method

The mission will be defined first and then using the method major dimensions and characteristics of the aircraft will be calculated such as gross weight, empty weight, fuel weight, engine thrust, wing area, length and fuselage maximum diameter. If output data is not as expected, mission requirements are changed and calculations are repeated. Using interpolation methods, some data such as maximum cruising altitude, non-dimensional lift coefficients, wing loading and thrust-loading ratio can be calculated.

The characteristics of the aircraft utilized in developing the various correlation includes various versions of a given design and encompasses long medium and short-range commercial transport aircraft as well as the smaller business jet aircraft. Using the method, conceptual design of aircraft gross weights extended from 10.500 lb. to 800.000 lb. can be done with a sufficient precision.

If the velocity of the reference aircraft (in knot) is multiplied by the root chord (in feet), we get 16.300. We will use 16.300 as Reynolds coefficient. The 0 zero lift drag coefficient of this aircraft is 0.0131. We will make an assumption using the both values. If we name it RFC (Reynolds Factor); RFC will be taken as 0.451 using

$$RFC = (VELC_{root})/16,300$$

equation for the given velocity and root chord. If a smaller and modern turbofan aircraft is accepted as reference, RFC can be taken as 0.33 and assumed as constant. It is the same for root chord C_{root}. Another assumption is made by taking Oswald coefficient which shows wing efficiency as 0.7 and constant. In reality the wing efficiency of these aircraft is 0.8 but for safety reasons it will be used as 0.7 in this study.

3. LOFTIN METHOD

3.1 Performance Objectives

Modern jet-powered cruising aircraft are designed to meet, as a minimum, the following performance criteria:

1. Airport Performance

FAR landing field length; missed approach requirement

FAR balanced take-off field length; second-segment climb gradient requirement

2. Cruise Performance

Cruising speed usually expressed in terms of Mach number.

Range

Payload

Considering these criteria following parameters are determined:

Gross weight

Fuel weight

Empty weight

Wing area and wing loading

Engine thrust and thrust loading.

The cruising altitude is given by the analysis. For simplicity, noise constraints are not considered in the conceptual design phase.

3.2 Sizing Procedure

The landing field length and approach lift coefficients are utilized to calculate wing loading. The approach lift coefficient depends upon the type of high-lift system and is chosen on the basis of statistical data for current aircraft. Using take-off field length and lift coefficient, thrust-to-weight ratio is calculated as a function of wing loading. The lift coefficient is again determined on the basis of statistical data for current aircraft.

The difficulty in satisfying missed approach and take-off climbing gradient requirements is a result of the necessity for providing the same performance with loss of an engine in these situations . In the cruise matching analysis thrust-to-weight ratio is determined as a function of wing loading. The defined thrust loading is sufficient for each wing loading to permit steady flight at the specified cruise Mach number and at the design lift coefficient which are usually near that for maximum lift-drag ratio. The altitude for cruise also comes from this analysis. The inputs to the cruise matching analysis are aircraft lift-drag ratio L/D, engine performance, cruise Mach number M, and characteristics of the atmosphere.

Rapid methods are presented for estimating the lift-drag ratio in terms of certain geometric characteristics of the aircraft. The appropriate characteristics of the atmosphere are presented in a form which relates wing loading and altitude to Mach number and lift coefficient. The characteristics of four representative turbofan engines are presented in a convenient non dimensional form that allows easy scaling of pertinent parameters to different engine thrust levels.

The outputs of the analysis will be utilized to determine other flight characteristics such as wing loading and thrust loading. The output values of wing loading and thrust loading are utilized in the weight analysis. The specific value of the ratio of payload weight to aircraft gross weight is calculated. The fuel fraction is determined from the Brequet range equation for the specified range and cruise Mach number. The payload weight is a specified value. Using the formulas, first empty weight, then fuel weight and finally gross weight is calculated. Such quantities as wing area, total thrust, fuel load, and operating weight empty may then be calculated.

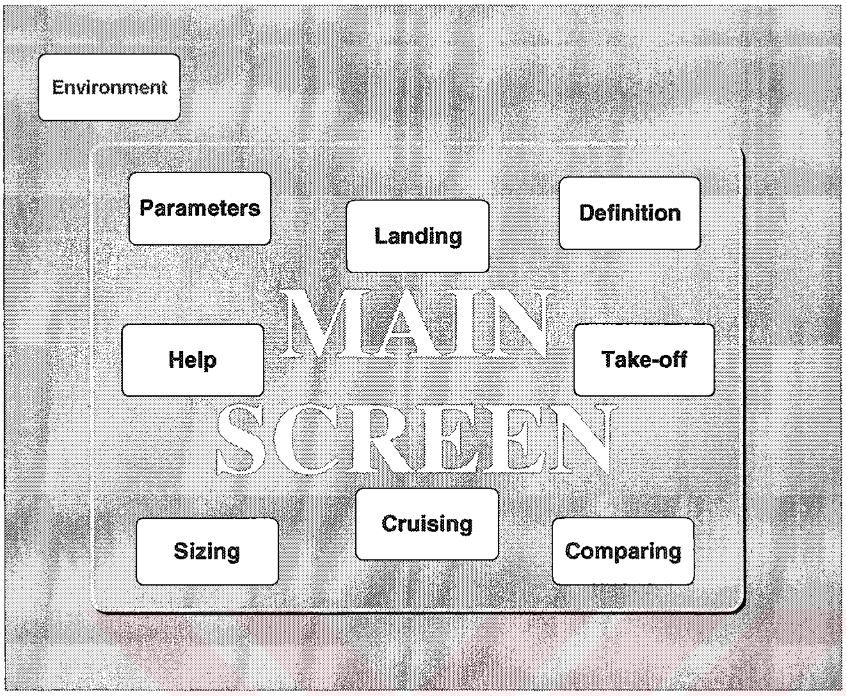


Figure 3.1 Main Screen

3.2.1 Landing Field Length

The landing field length is defined by ICAO and by the Federal Air Regulations for transport category aircraft. These regulations define landing field length and landing distance.

The landing distance is measured horizontally from the point at which the aircraft 50 ft above the surface, to the point at which the aircraft is brought to a complete stop on a hard, dry, smooth runway surface.

The landing field length is obtained by dividing the measured landing distance by 0.6 in order to account for the possibility of variations in approach speed, touchdown point, and other deviations from standard procedures. The procedure for estimating the FAR landing field length is divided into two steps. The first step relates the approach speed to the wing loading and the lifting capability of the wing and flap system; the second step expresses the actual FAR landing field length as a function of the approach speed. During the landing, the angle from the point at which the aircraft

50 ft above the surface, to the point at which the aircraft comes to touchdown point can be considered as zero. So in the equation wing loading can be considered as equal to gravity and thrust is equal to drag. But in reality W_{mxi}, is greater than lift Land drag force D is greater than thrust T. The equation for lift in metric system is;

$$L = W_{mxi} = C_{ly} \rho V_y^2 S / 2 \tag{3.1}$$

But the equation will not be used in metric system. We can take L as equal to maximum landing weight, W_{mxi} is maximum landing weight, C_{ly} is approach lift coefficient, V_y is approach speed. If we substitude the value of ρ,

$$L = W_{mxi} = C_{ly} \sigma V_y^2 S / 294 \tag{3.2}$$

and

$$V_y = 17.15 \sqrt{\frac{W_i / S}{C_{ly} \sigma}} \tag{3.3}$$

In this equation, L and W are in lb, S is wing area in square feet, s is non-dimensional atmosphere density ratio, V is speed in knots. The last equation shows that approach speed is a function of wing loading and lift coefficient. The lift coefficient is related to high-lift system and there is a linear relation between landing field length and approach speed.

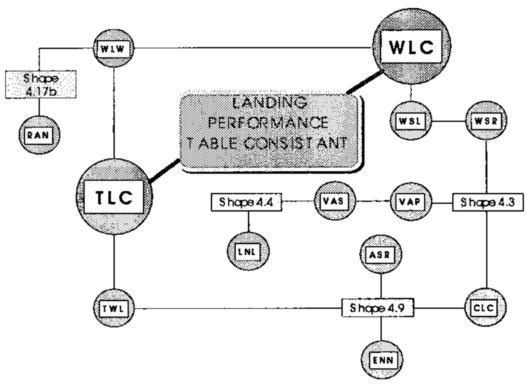


Figure 3.2 Landing Performance Table

At the missed approach and one engine inoperative conditions, the equation for climb gradient becomes

$$T/W_i = \frac{N}{N-1}\left(\frac{1}{L/D} + \gamma\right) \qquad (3.4)$$

In this equation N is the number of engines. The specified climb gradients are 2.7 percent for four-engine aircraft, 2.4 percent for three-engine aircraft, and 2.1 percent for two-engine aircraft. The change of the landing thrust loading with aspect ratio and lift coefficient is presented in fig. 3.4.

L/D is given

$$L/D = C_l / C_d = \frac{C_l}{C_{dp} + C_{di}} = \frac{C_l}{C_{dp} + \frac{C_l^2}{\pi A e}} \qquad (3.5)$$

$$C_{di} = \frac{C_l^2}{\pi A e} \qquad (3.6)$$

$$C_{dp} = C_{do} + \Delta C_{df} + \Delta C_{dit} \qquad (3.7)$$

In this equation, C_{di} is induced drag coefficient, C_{dp} is total drag coefficient except C_{di}, C_{df} is drag coefficient due to flap deflection, C_{dit} is drag coefficient due to landing gear, A is aspect ratio and "e" is Oswald's airplane efficiency factor. Figure 3.2 presents the relation between the landing parameter in and the approach speed. Minimum operating weight is calculated using the formulas.

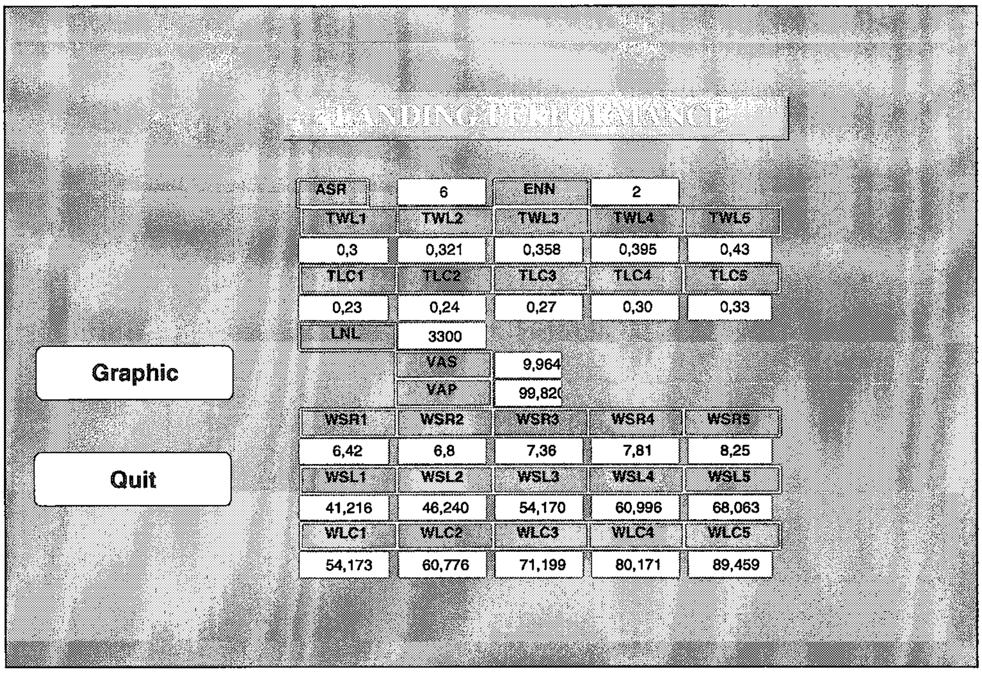


Figure 3.3 Landing Performance 1 Input Screen

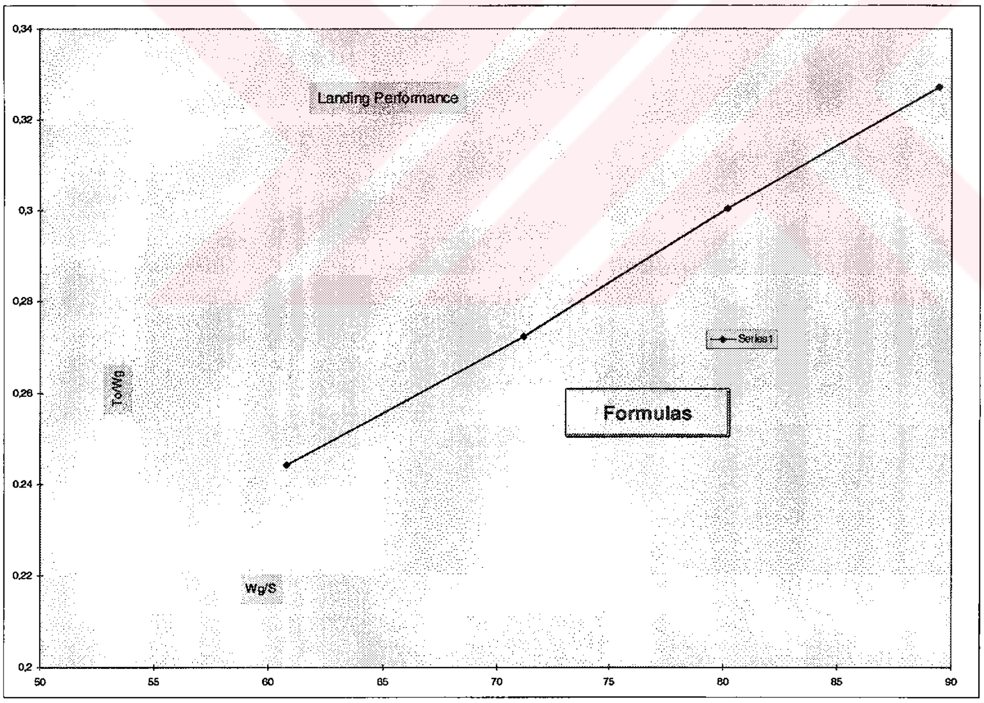


Figure 3.4 Landing Performance 2 Graphical Display

3.2.1.1 Take-Off and Landing Weight Relationships

In order to relate the thrust requirements for these two manuveaur, the maximum landing weight must be known in terms of the maximum take-off weight. Maximum landing weight varies in accordance with the range of the aircraft. The relation between the landing weight and range is determined based on the data of the current aircraft. There are 3 range categories:

Short Range : 1000-1500 nm = 1850-2780 km

Mid Range : 2000-3000 nm = 3700-5550 km

Long Range : 3000-4000 nm = 5550-7400 km

The approximate value of the ratio of W_{mxi} / W_g for short, mid and long-range aircraft is 0.91, 0.82, 0.73. The thrust loading for these situations is:

$$(T_0 / W_i)(T_0 / W_g) = 1/(W_i / W_g) = 1/0.91 = 1.099 \text{ (Short Range)}$$

$$(T_0 / W_i)(T_0 / W_g) = 1/(W_i / W_g) = 1/0.82 = 1.22 \text{ (Mid Range)}$$

$$(T_0 / W_i)(T_0 / W_g) = 1/(W_i / W_g) = 1/0.78 = 1.37 \text{ (Long Range)}$$

The ratio of landing weight take-off weight is shown in Fig. 3.4

3.2.2. Take-Off Field Length

During the take-off, the speed of the aircraft starts from zero and reaches the take-off speed V_{yk}. The equation for this state is:

$$L = W_k = C_{lyk} \sigma V_{yk}^2 S / 294 \tag{3.8}$$

The FAR take-off field length contains certain inherent safety features to account for engine failure situations. If an engine should fail during the take-off roll at a critical speed, called the decision speed, the pilot is offered two action options. He may elect to continue the take-off on the remaining engines. In this case the take-off distance is defined as the distance from the point where the take-off run is initiated to the point where the aircraft has reached an altitude of 35 ft. In the second alternative, the pilot may elect to shut down all engines and apply full braking. The decision speed is chosen in such a way that the sum of the distance required to accelerate to decision speed and then decelerate to a stop is the same as the total distance for the case in which the take-off is continued following engine failure. If we get the same take-off field length for the both situations, it is called FAR balanced field length.

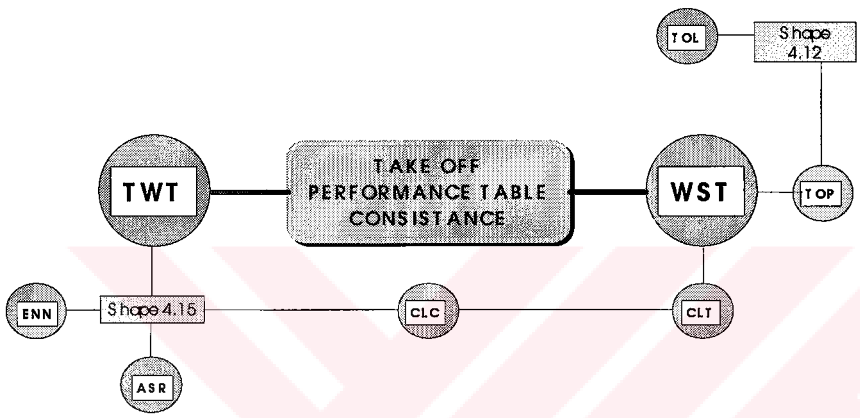


Figure 3.5. Take-off Performance Table

$$l_{yk} = \frac{V_{yk}^2}{2a} \tag{3.9}$$

$$T_0 / W_g = a / g \tag{3.10}$$

$$V_{yk}^2 = \frac{2W_g}{\rho C_{lyk} S} \tag{3.11}$$

In these equations; l_{yk} shows the distance required to accelerate to the lift-off speed, a shows average acceleration of aircraft along ground, r shows atmospheric density, g shows acceleration due to gravity, T_0 shows sea level static thrust, W_g shows maximum take-off weight.

$$K = \frac{1}{\rho_0 g} \tag{3.12}$$

$$\sigma = \frac{\rho}{\rho_0} \tag{3.13}$$

$$l_{yk} = \frac{K}{C_{lyk}\sigma} \frac{W_g / S}{(T_0 / W_g)} \tag{3.14}$$

$$l_{yk} = K.TOP \tag{3.15}$$

In these equations; ρ_0 is the atmospheric density for standard sea level conditions, K is a constant value, TOP is the take-off parameter. There is a linear relation between the TOP and take-off field length. The relation is given with fig. 3.6.

$$T_0 / W_g = \frac{N}{N-1}(\frac{1}{L/D} + \gamma) \tag{3.16}$$

equation will be used. This equation is the same as landing field length equation, but here take-off values are used. The variation of thrust loading in accordance with aspect ratio and lift coefficient is shown in fig. 3.7.

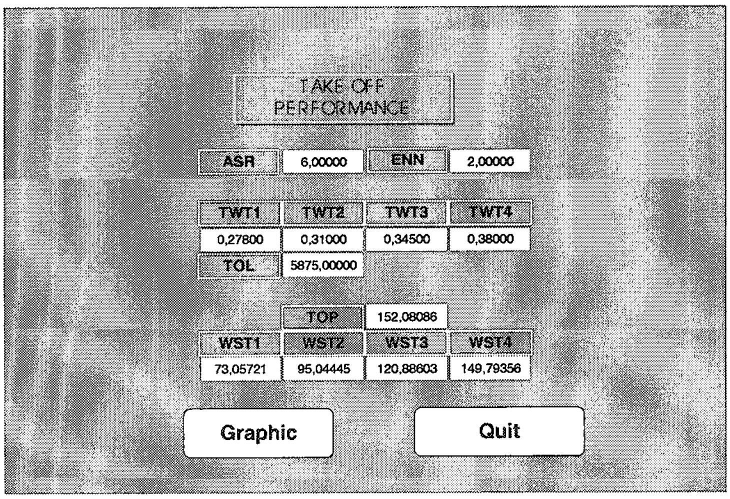


Figure 3.6 Take-off Performance 1 Input Screen

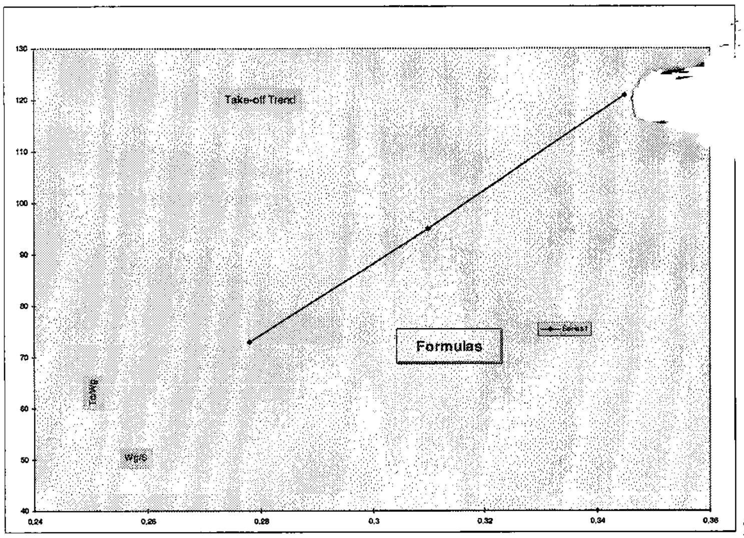


Figure 3.7 Take-off Performance 2 Graphical Display

3.2.3 Cruising Analysis

A cruise analysis will be done considering the gi[illegible]onditions for cruising performance. The cruising criterion to be used here is[illegible]t t[illegible]e pertinent engine and airframe characteristics will be matched in such a way [illegible] to [illegible]ermit achievement of a specified design range at a given cruising Mach numb[illegible] for a minimum amount of fuel. The quantitative relationship between the range [illegible]he significant aircraft and engine characteristics, and the fuel used during cruising [illegible]ght is given by the Breguet range equation:

$$B = \frac{V(L/D)}{SFC} \qquad (3.17)$$

$$W_y / W_g = 1 - \frac{1}{e^{R/B}} \qquad (3.18)$$

where B is the Breguet factor, R is range (nm), V is spe[illegible] (knots), W_y is aircraft fuel weight (lb), SFC is engine specific fuel consumption (lil[illegible] of fuel per liber of thrust per hour).

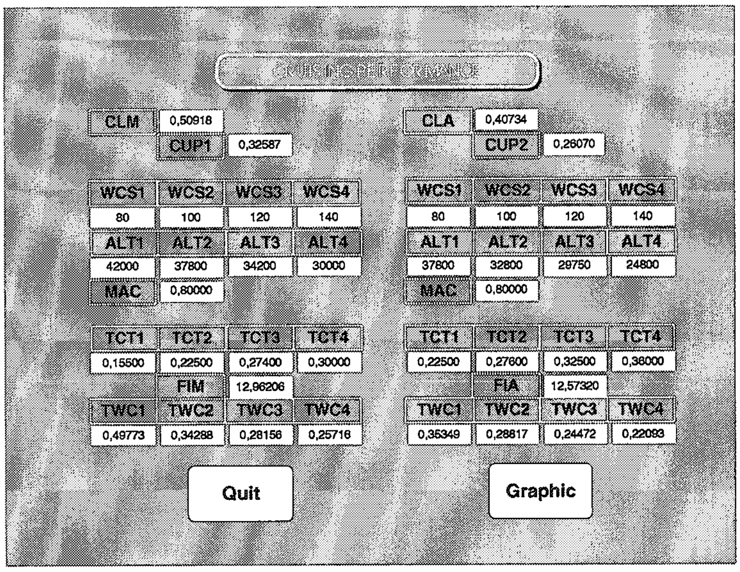


Figure 3.8 Cruise Performance 1 Input Screen

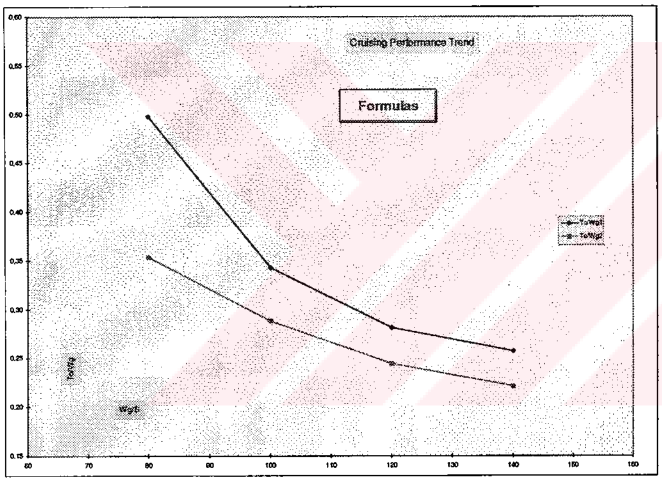


Figure 3.9 Cruise Performance 2 Graphical Display

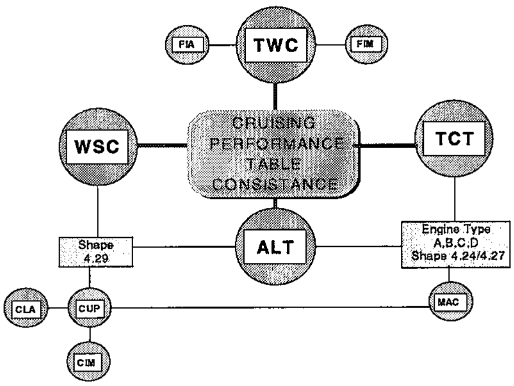


Figure 3.10 Cruise Performance Table

3.2.3.1 Aircraft L/D Characteristics

Highly sophisticated and accurate methods are available for estimating the lift and drag characteristics of aircraft. There are several methods for estimating drag polar. One of the methods is Roskam Method that permits a rapid estimation of the aircraft lift-drag characteristics.

If the drag polar is assumed to be symmetrical about zero lift, the drag coefficient at any lift coefficient below the stall may be may be expressed as follows:

$$C_d = C_{d0} + \frac{C_l^2}{\pi A e} \tag{3.19}$$

$$L / D = C_l / C_d = \frac{C_l}{C_{d0} + C_{di}} = \frac{C_l}{C_{d0} + \frac{C_l^2}{\pi A e}} \tag{3.20}$$

where

C_{di} induced drag coefficient

C_d total drag coefficient

A wing aspect ratio

e Oswald Factor

C_l lift coefficient

The value of the lift coefficient for the maximum value of L/D, and the value of $(L / D)_{\max}$ may be deduced by taking the derivative of equation (3.16) with respect to C_l and equating this derivative to zero. Thus,

$$C_{di} = C_{d0} = \frac{C_l^2}{\pi A e} \tag{3.21}$$

Equation 3.21 indicates the very important result that the induced drag coefficient is equal to the zero-lift drag coefficient at the maximum value of the lift-drag ratio. The total drag coefficient at $(L/D)_{\max}$ is then equal to twice the zero-lift drag coefficient or twice the induced drag coefficient.

$$C_d = 2C_{do} = \frac{2C_l^2}{\pi Ae} \tag{3.22}$$

From equation 3.22 the lift coefficient corresponding to $(L/D)_{\max}$ is given by

$$C_{lm} = \sqrt{\pi AeC_{do}} \tag{3.23}$$

Since the total drag coefficient at $(L/D)_{\max}$ is twice the zero-lift drag coefficient

$$(L/D)_{mx} = C_{lm}/C_d = \frac{\sqrt{\pi AeC_{do}}}{2C_{do}} \tag{3.24}$$

or, more conveniently,

$$(L/D)_{mx} = 0.5\sqrt{\frac{\pi Ae}{C_{do}}} \tag{3.25}$$

Equations 3.23 and 3.25 are extremely useful and will be utilized in subsequent analysis.

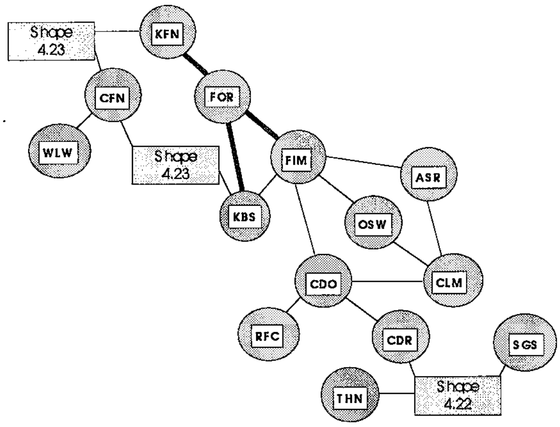


Figure 3.11 Aerodynamic Variables Table

3.2.3.2 Estimation of Maximum Lift Drag Ratio

$(L/D)_{\max}$ can be calculated in two ways: Utilizing wetted area of the aircraft and using Reynolds number. Both methods need Reynolds ratio (L). Regardless of which method you use, the value of the zero-lift drag coefficient is estimated by extrapolating a known value for a current, reference aircraft. The methods give following results:

$$L/D = 0.5\sqrt{\frac{\pi Ae}{C_{do}}} \tag{3.27}$$

$$C_{do}/C_{dor} = \frac{S_{ts}/S}{(S_{ts}/S)_r} \tag{3.28}$$

The subscript r refers to reference aircraft which is a modern, four engine, long-range transport aircraft with 0.0131 zero-lift drag coefficient, with S_{ts}/S ratio of 5.0 (wetted area/wing area), with an optimum cruising altitude of 35.000 ft, and with the multiplication of the velocity in knot and root chord in ft is 16.300.

$$S_{ts}/S = \frac{\pi d^2}{S}\left(\frac{l}{d} - 1\right) + 2(1 + S_{ky}/S) + S_{mb}/S \tag{3.29}$$

where

d fuselage diameter

l fuselage length

S_{ky} sum of vertical and horizontal tail area

S_{mb} nacelle wetted area

An examination of the characteristics of current aircraft yielded average values of 0.44 and 0.47 for S_{mb}/S and S_{ky}/S respectively.

$$S_{is} / S = 3.38 - \frac{4\pi d^2}{4S}(1 - \frac{l}{d}) \qquad (3.30)$$

$$C_{do} / C_{dor} = (\frac{R_r}{R})^{1/6} \qquad (3.31)$$

If cruising at the same altitude is considered, the equation takes the form

$$C_{do} / C_{dor} = (\frac{(Vc)_r}{Vc})^{1/6} \qquad (3.32)$$

where v cruising speed, c root chord.

$\underline{L}$ ratio is given below:

$$\underline{L} = \frac{Vc}{16300} \qquad (3.33)$$

3.2.3.3 Variation of Lift Drag Ratio with Lift Coefficient

The value of the lift coefficient at $(L/D)_{max}$ was calculated in previous sections. The value of the lift-drag ratio at lift coefficients different from that for $(L/D)_{max}$ is frequently needed.

$$C_d / C_{dm} = \frac{C_{do} + \dfrac{C_l^2}{\pi A e}}{C_{do} + \dfrac{C_{lm}{}^2}{\pi A e}} \tag{3.34}$$

$$C_{lm}{}^2 = \pi A e C_{do} \tag{3.35}$$

$$\underline{C_l} = C_l / C_{lm} \tag{3.36}$$

$$C_d / C_{do} = \frac{(1 + \underline{C_l^2})}{2} \tag{3.37}$$

$$\frac{L/D}{(L/D)_{mx}} = \frac{2\underline{C_l}}{1 + \underline{C_l^2}} \tag{3.38}$$

where $\underline{C_l}$ the ratio of the lift coefficients, C_{dm} drag coefficient corresponding to maximum lift.

The last equation is shown in fig. 3.12.

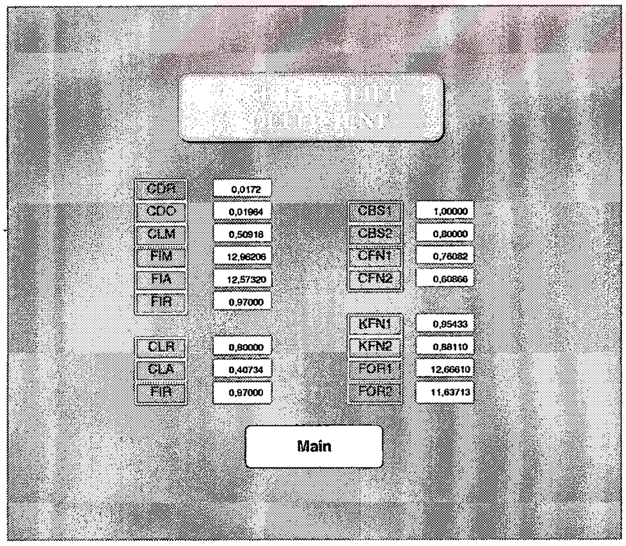


Figure 3.12 Aerodynamic Coefficients Input Screen

3.2.3.4 Airplane Efficiency Factor: Oswald Factor

Induced drag coefficient is directly proportional with the square of lift coefficient and inversely proportional with aspect ratio and efficiency factor. The efficiency factor can be for the entire airplane or for the wing alone. Induced drag coefficient is generated in two ways: First, a wing operating in an inviscid flow generates a drag due to lift. Second the drag due to viscosity. The expression for Oswald Factor;

$$e = \frac{u}{1 + K\pi A u} \tag{3.26}$$

where u span loading parameter.

According to the equation; the airplane efficiency factor increases with decreasing aspect ratio for a given value of K. The variation in viscous drag coefficient with lift coefficient becomes a larger proportion of the total drag coefficient increment due to lift as the aspect ratio increases, and, hence, is responsible for the effect of aspect ratio on the airplane efficiency factor indicated by equation.

The value of the airplane efficiency factor for clean aircraft (flaps and landing gear retracted) is usually between 0.70 and 0.85. In our calculations, considering safety it will be taken as 0.70.

3.2.3.5 Engine Characteristics

The engines must provide sufficient thrust to balance the aircraft drag in cruising flight at the desired combination of altitude and Mach number at which the aircraft is to be operated. The thrust that the engine is capable of producing is primarily a function of the altitude and Mach number.

Two different thrust ratings are given for a particular engine. These ratings are defined as the maximum take-off thrust, which can be used for only short periods of time, and the maximum continuous thrust. The maximum continuous thrust is almost always less than the maximum take-off thrust. A relationship between the maximum continuous thrust and maximum take-off thrust is needed for use in matching the engine to the aircraft so that sufficient thrust is available for cruising flight at the desired conditions.

The ratio of maximum continuous thrust to maximum sea level thrust is presented as a function of Mach number for different altitudes for four different turbofan engines. Engines A and B are modern high compression ratio engines with bypass ratios of 4.5 and 6.0, respectively, whereas engines C and D are older engines with bypass ratios of 1.4 and 1.1, respectively. The amount of fuel that the engine consumes in order to produce a given amount of thrust is another important performance parameter. The specific fuel consumption, expressed in pounds of fuel per pound of thrust per hour, is a function of Mach number and altitudes.

3.2.3.6 Cruise Matching

The relationship between the wing loading and thrust loading for the cruise condition must be chosen in such a way that the aircraft can fly with sufficient thrust to balance the drag at the desired Mach number and at an altitude which permits operation at some specified design lift coefficient. Typically, for the purpose of matching the engine to airframe, the specified lift coefficient is chosen to be that for maximum lift-drag ratio, and the wing loading and thrust loading are based on maximum take-off gross weight.

$$T_c = D = \frac{W_g}{(L/D)_{mx}} \tag{3.39}$$

$$T_c / T_o = \frac{W_g / T_o}{(L/D)_{mx}} \tag{3.40}$$

$$T_o / W_g = \frac{1}{(T_c / T_o)(L/D)_{mx}} \tag{3.41}$$

where T_c is the cruise thrust.

In order to obtain the maximum range, lift-drag ratio must be the maximum. T_0 is the thrust of the installed engine. The installed engine thrust is somewhat less than the uninstalled value quoted for the engine by the manufacturer. Thrust ratio varies with altitude and Mach number. The variation of wing loading as a function of altitude and cruising parameters is shown in fig. 3.13.

$$W / S = C_l q = C_l M^2 \frac{q}{M^2} \tag{3.42}$$

$$\frac{q}{M^2} = \frac{\rho \frac{V^2}{2}}{\frac{V^2}{a^2}} \tag{3.43}$$

$$a^2 = p\gamma\rho \text{ and } \frac{q}{M^2} = \rho\frac{a^2}{2} \tag{3.44}$$

$$\frac{q}{M^2} = \gamma\frac{p}{2} \tag{3.45}$$

where p is the static pressure, g is the ratio of specific heats, a is the speed of sound and q is the dynamic pressure.

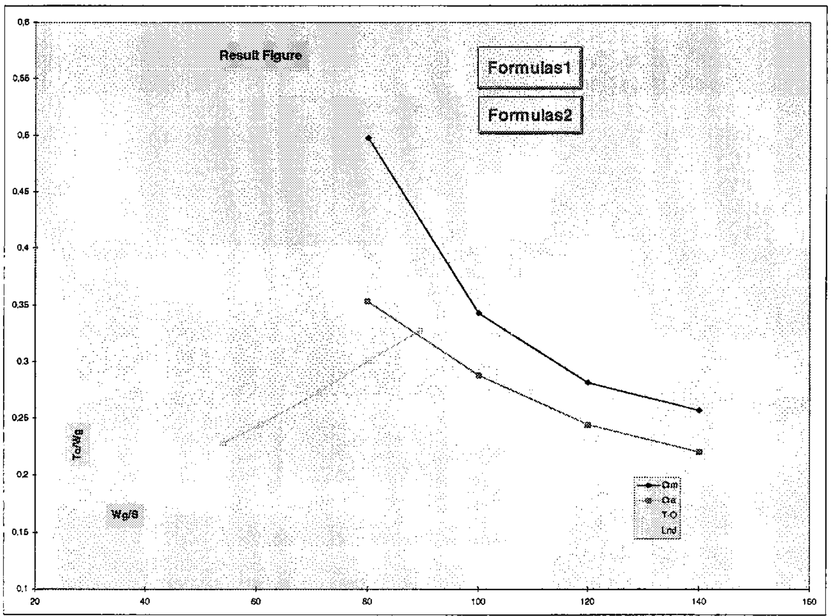


Figure 3.13 Performance Evaluation Graphical Display

3.2.4 Aircraft Sizing

The matching procedure gives values of the wing loading and thrust loading needed to meet cruise and airport performance objectives. In this section aircraft gross weight will be calculated.

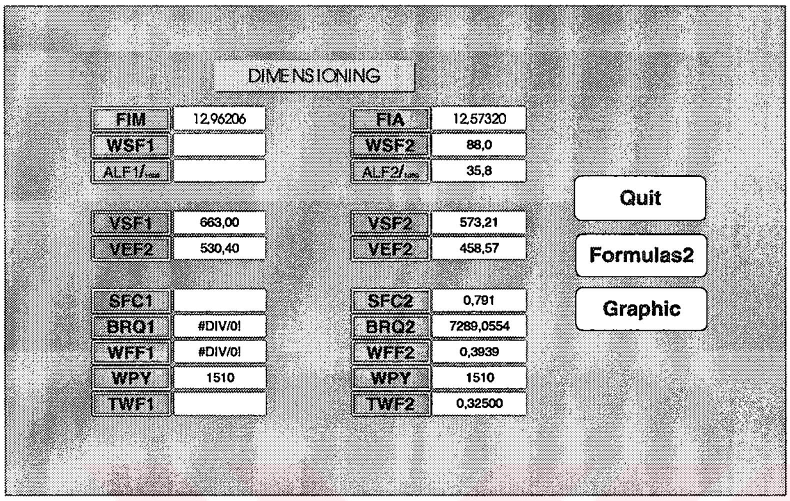


Figure 3.14 Sizing Input 1 Screen

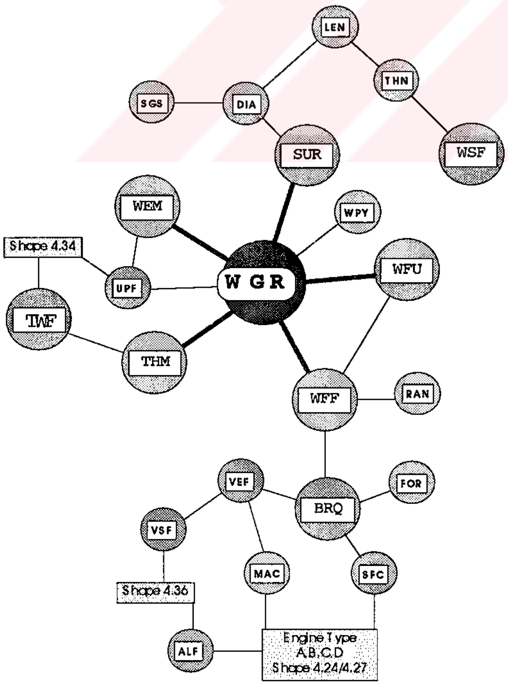


Figure 3.15 Parameters Table

3.2.4.1 Fuel Fraction

The Breguet equation for calculating the fuel fraction is;

$$W_y / W_g = 1 - \frac{1}{e^{R/B}} \tag{3.47}$$

where B is Breguet range factor, R is range (nm), V is speed (knots), W_y is aircraft fuel weight (lb), c is specific fuel consumption (liber of fuel per liber of thrust per hour), L/D is average cruise lift-drag ratio and Wg is aircraft gross weight (lb). Breguet factor is given by the expression;

$$B = \frac{V}{SFC}\frac{L}{D} \tag{3.48}$$

In this equation L/D must be an average value during the flight. In order to calculate the average value of (L/D), we should define new parameters:

$$F_{or} = \frac{F_{b½} + F_{bt}}{2} \tag{3.49}$$

$$K_{b½} = \frac{F_{b½}}{F_{mx}} \tag{3.50}$$

$$K_{bt} = \frac{F_{bt}}{F_{mx}} \tag{3.51}$$

$$F_{or} = F_{mx}\frac{K_{b½} + K_{bt}}{2} \tag{3.52}$$

$$\underline{C_{lb½}} = \frac{C_l}{C_{lmx}} \tag{3.53}$$

$$\underline{C_{lbt}} = \underline{C_{lb^-}}\frac{W_i}{W_g} \tag{3.54}$$

F_{or} is average value of (L/D), F_{bs} is the value of (L/D) at the beginning, F_{bt} is the value of (L/D) at the cruise end, K_{bs} is the ratio of F_{bs} to the maximum value of (L/D), K_{bt} is the ratio of F_{bt} to maximum (L/D), $\underline{C_{lbs}}$ is the ratio of beginning lift coefficient to maximum lift coefficient, $\underline{C_{lbt}}$ is the ratio of ending lift coefficient to maximum lift coefficient, W_i / W_g is the ratio of landing weight to take-off weight. Using the value of $\underline{C_{lbs}}$ corresponding to K_{bs}, then $\underline{C_{lbt}}$ is calculated by multiplying $\underline{C_{lbs}}$ by W_i / W_g. Using $\underline{C_{lbt}}$ again, K_{bt} is found. Using the value in equation 3.52, F_{or} is calculated.

3.2.4.2 The Calculation of Other Characteristics

In order to calculate the gross weight, we need W_p, W_y / W_g, $\underline{U}$ useful load fractions. $\underline{U}$ useful load fractions can be found as a function of thrust loading.

$$W_g = \frac{W_p}{\underline{U} - \dfrac{W_y}{W_g}} \tag{3.55}$$

$$\underline{U} = \frac{W_p + W_y}{W_g} \tag{3.56}$$

For the empty weight W_{bv}, fuel weight W_y, wing area S, engine thrust T_0, fuselage diameter d and fuselage length l; the following expressions will be used:

$$W_b = W_g(1 - \underline{U}) \tag{3.57}$$

$$W_y = W_g \frac{W_y}{W_g} \tag{3.58}$$

$$S = \frac{W_g}{W_g / S} \tag{3.59}$$

$$T_o = W_g \frac{T_o}{W_g} \tag{3.60}$$

$$d = \sqrt{4 \frac{S}{\pi} (\frac{S_g}{S})} \tag{3.61}$$

$$l = d(l/d) \tag{3.62}$$

The altitude of the aircraft will be calculated by interpolating the wing loading. By multiplying the sound velocity by given Mach number for the mission, cruising speed is determined.

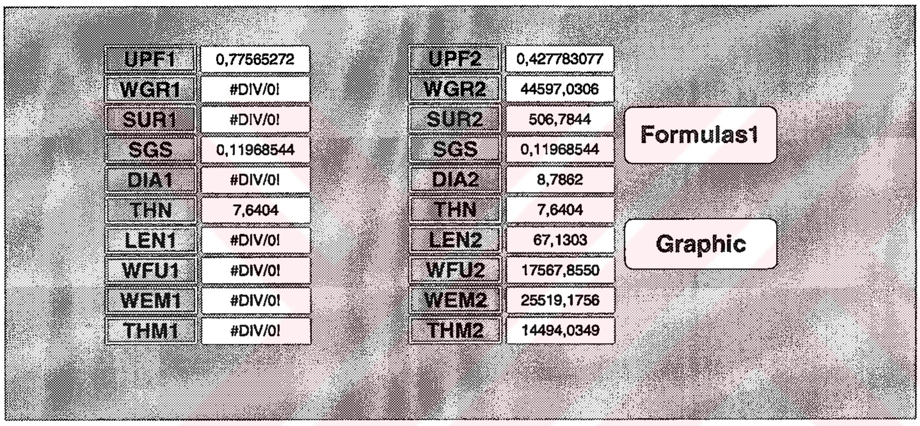


Figure 3.16 Sizing Input 2 Screen

3.2.5 Findings

In this section, the variations of fundamental parameters according to each other will be summarized.

The increase in Mach number means increased altitude and wing loading. Since the velocity of the sound decreases with the increase in altitude, Mach number must be greater at high altitudes than at lower ones.

With the increase of useful load fraction gross weight decreases. Gross weight again decreases if Fuel/Weight ratio decreases. Engine thrust should be proportional with gross weight.

If the range of the aircraft increases, fuel/weight ratio decreases. With the increase of the Brequet Factor, W_y / W_g increases that means that more fuel is needed per unit airplane weight. Consequently, less payload can be considered. In order to decrease the Brequet Factor, either specific fuel consumption can be increased (ie. cruising altitude can be decreased) or aircraft speed can be decreased (ie. cruising altitude can be increased). To find an optimization, a comparison must be done between the variation gradient of specific fuel consumption as a function of altitude for the chosen engine type and the variation gradient of sound velocity at design altitudes. Thrust-to-weight ratio has no effect on the range. Useful load fraction varies with the range. Brequet factor is also independent from the range. W_y / W_g depends on the range. Since the payload does not vary with the range, gross weight varies with the range. The most important parameter on gross weight is payload. This is a very important result.

Thrust-to-weight ratio is inversely proportional with (L/D) ratio. But wing loading can be proportional with (L/D). (L/D) ratio varies with zero-lift drag coefficient and aspect ratio. If zero-lift drag coefficient remains constant while aspect ratio is increasing, the cruising parameter increases and so wing loading increases. The most important parameters effecting the wing loading are Mach number, take-off and landing field lengths. If the cruise performance curve does not match the take-off and landing performance curves, either Mach number can be decreased or take-off and landing field lengths can be increased. Thrust-to-weight ratio has no effect on the range. But generally, if requirements for the take-off are met, requirements for the landing are also satisfied. The range has got an effect on thrust-to-weight ratio during landing performance. The effect of range decreases for mid-range and long-range.

Without increasing take-off and landing field lengths, take-off and landing curves can not be matched cruising curves (only speed can be decreased but this way is not generally preferred). For the values of the lift coefficient until 2.0 extrapolation methods can be used and in this way corresponding values of T_0 / W_g and W_g / S can be calculated. The aircraft having the same aspect ratio will have the same take-off and landing curves. Independent from the mission of the aircraft, different wing loading will be obtained for constant thrust. The thrust loading is independent from the mission of the aircraft, different wing loading will be obtained for constant thrust. The thrust loading for an aircraft is generally about 0.2. With the increase of aspect ratio, a decrease in thrust loading is obtained for the same wing loading. This is an expected situation because the greater the aspect ratio is, the less the additional drag due to three-dimensional body and the higher the efficiency is. Zero-lift drag coefficient, maximum lift coefficient, Mach number, and maximum lift/drag ratio are not considered in take-off and landing performance calculations. These effect only cruising performance. It means that, surface area, aspect ratio, (L/D) ratio effect only cruising conditions as well.

4. SAMPLE DESIGN

The range of the aircraft will be 3,150 nm. The distance for flight to an alternative airport due to an unplanned situation will be 500 nm. Total range is 3,650 nm. There will be 7 people totally on the aircraft and their baggage will be totally 1,510 lb. Take-off field length is 5,875 ft and landing field length is 3,300 ft. The cruising speed will be 0.8 Mach and the aircraft will have two C type turbofan engines with low bypass ratio. The fuselage length will be 68 ft, fuselage diameter will be 8.9 ft So l/d ratio is 7.64, the ratio of fuselage section to wing area (SGS) is 0.119, reference wing area is 520 square ft The aspect ratio will be taken as 6. The targeted airplane has got an aspect ratio of 8.5. In order to match the cruising curve with ground conditions curve, the aspect ratio is assumed as 6. The target aircraft is actually an existing one. At the end of this method, the characteristics and major dimensions of the final aircraft will be determined.

4.1 The Steps of the Design

1. VAP: LNL= The square root of square of approach speed at 3,300 ft.

2. CLC Approach lift coefficient. It is a value between 1.2-2.2 for current aircraft.

3. WSR: Square root of wing loading parameter. It is found considering VAP and CLC.

4. WSL: Landing wing loading parameter. It is calculated by taking the square power of WSR for s=1.

5. TWL: Landing thrust-weight parameter. It is found according to CLC lift coefficients for ASR= 6 and ENN= 2.

6. WLW: Landing and take-off weight ratio. It is found for RAN= 3,650 nm. as 0.760.

7. WLC: Landing wing loading. (WSL/WLW)

8. TLC: Landing thrust-to-weight ratio. (TWLWLW)

9. TOL: Take-off field length according to FAR. (given as a design condition).

10. CLC: Second segment climb gradient lift coefficient. It is between 1.2-2.0 for current aircraft.

11. CLT: Aircraft lift-off lift coefficient. It is a value between 1.73 and 2.88. (1.44CLC)

12. TWT: The thrust loading needed to satisfy minimum climb gradient requirement during second segment climbing. It is found for CLC and ASR=6.

13. TOP: Take-off parameter. It is found according to TOL.

14. WST: Take-off wing loading. It is calculated by the expression CLTTWTTOP assuming s=1 (ie. at sea lev:el and ISA conditions).

15. ENT: Engine type. (A, B, C or D)

16. MAC: Cruise Mach number. It is a given value as 0.8.

17. FIM: Cruise maximum (L/D) ratio. It was previously calculated.

18. CLM: Cruise maximum lift coefficient according to FIM.

19. CUP: Cruise parameter. It is calculated by the expression CLM MAC^2

20. WCS: Cruise wing loading. It is obtained from current aircraft data.

21. ALT: Altitude. It is found according to CUP and WCS. (It is not the altitude given in the beginning.)

22. TCT: The ratio of THC (engine altitude thrust) required for steady state cruise at cruise altitude to THR (thrust at ALT=0 and ISA conditions). It is found for engine type C according to MAC and ALT (found from CUP and WCS).

23. TWC: Cruise thrust to weight ratio. It is the ratio of THR to WGR. It is calculated by the expression $1/(FIMTCT)$.

24. FIB: Initial cruise (L/D) ratio and taken as 1 and 0,97.

25. FOR: Average cruise (L/D) ratio. It was calculated by using the values of ASR= 6 and FIM= 12,96.

26. ENT: Engine type C.

27. WSF: Matching wing loading values.

28. TWF: Matching thrust loading values.

29. ALF: Matching altitude values. It is calculated by interpolation using WCS and ALT (cruise) values used during the cruise performance.

30. VES: Sound velocity in knot at altitude ALF. It is found from standard atmosphere tables.

31. MAC: Design Mach number. It is given in the beginnig.

32. VEF: Cruise speed. It is calculated by the expression VESMAC.

33. SFC: It is found for engine type C for ALF and MAC=0,8.

34. BRQ: Brequet range factor. It is calculated by the expression VEFFOR/SFC.

35. RAN: Design range. İt is given as 3650 nm.

36. WFF: Fuel gross weight ratio. It is calculated by the expression $1-1/\left(e^{R/B}\right)$.

37. WPY: Payload weight (staff, passengers and baggage). It is given in the beginning as a design condition.

38. UPF: Useful load fraction. It is found for different values of TWF.

39. WGR: Gross weight. It is calculated by the formula WPY/(UPF-WFF).

40. WEM: Empty weight. It is calculated by the formula WGR (1-UPF).

41. WFU: Fuel weight. It is calculated by the formula WFFWGR.

42. SUR: Wing area. It is calculated by the formula WGR/WSF

43. THR: Take-off maximum static engine thrust. It is calculated by the formula TWFWGR.

44. SGS: The ratio of fuselage cross section to wing area. It is chosen according to the reference aircraft.

45. DIA: Fuselage diameter. It is calculated by the formula 4SURSGS/3,14.

46. LEN: Fuselage length. It is calculated by the formula THNDIA.

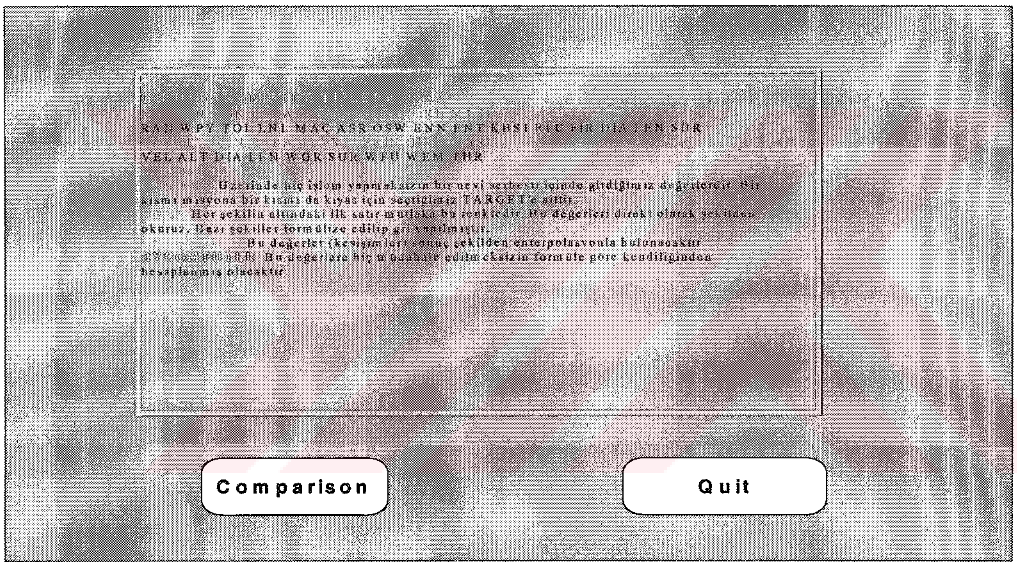


Figure 4.1 Help Screen 1

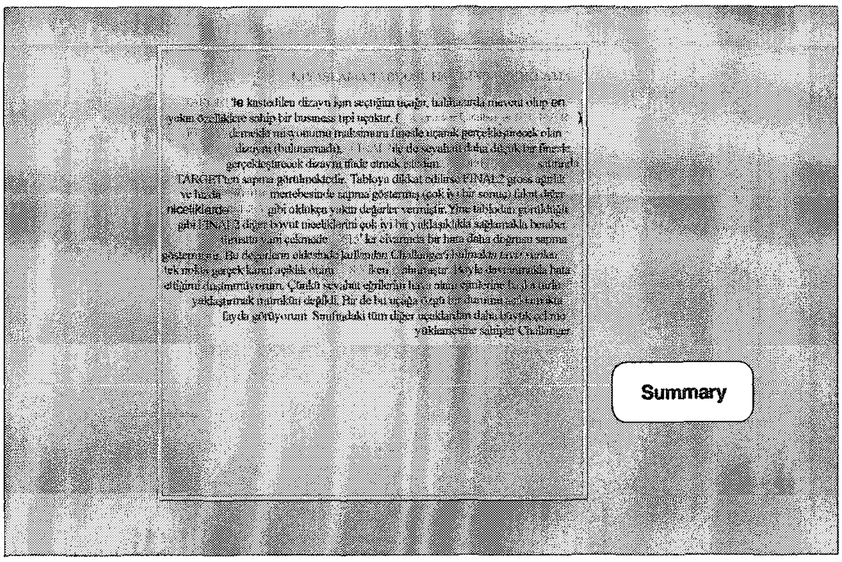


Figure 4.2 Help Screen 2

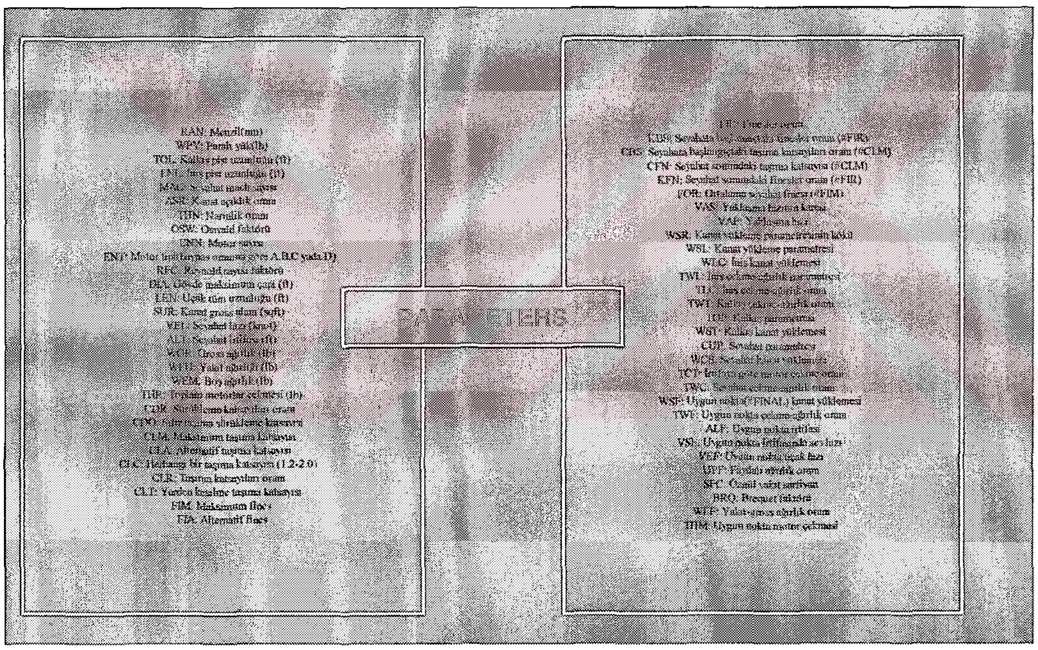


Figure 4.3 Parameters Screen

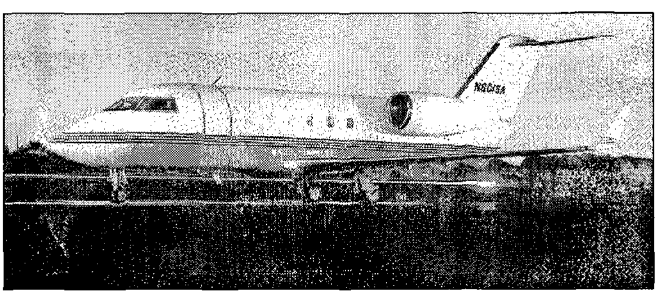


Figure 4.4 Canadian Challenger Business Jet Aircraft

4.1.1 Mission Characteristics

A/C COMMERCIAL NAME	Canadian Challenger
A/C MODEL	601-3A/ER
ENGINE TYPE	GENERAL ELECTRIC CF 34-3A
A/C TYPE	Business A/C with two turbofan engines.
MAXIMUM TAKE-OFF WEIGHT	20230 Kg (44600 lb)
MAXIMUM FUEL WEIGHT	8119 Kg (17900 lb)
EMPTY WEIGHT	11310 Kg (24935 lb)
ENGINE THRUST	EA 40.66 kN (9140 lbst)
WING LOADING	418.8 Kg/(m^2) (87.7 lb/ft^2)
THRUST LOADING	262.86 Kg/kN (2.58 lb/lbst)
TOTAL WING AREA	48.31 (m^2) (520 ft^2)
WING ASPECT RATIO	8.5
TOTAL LENGTH	20.85 m (68 ft)
FUSELAGE DIAMETER	2.69 m (8.9 ft)
CRUISE SPEED	851 Km/h (459 Knt)
MAXIMUM CRUISE ALTITUDE	12500 m (41000 ft)
TAKE-OFF FIELD LENGTH	1791 M (875 ft)
LANDING FIELD LENGTH	1006 m (3300 ft)
RANGE	6764 Km (3650 nm) (2 crews and 5 passengers)

4.2 Program Parameters

RAN: Range (nm)

WPY: Payload (lb)

TOL: Take-off field length. (ft)

LNL: Landing field length (ft)

MAC: Cruise Mach number

ASR: Wing Aspect Ratio

THN: (L/D) ratio

OSW: Oswald factor

ENN: Number of engines

ENT: Engine type (A,B,C or D according to by-pass ratio)

RFC: Reynolds coefficient

DIA: Fuselage maximum diameter (ft)

LEN: Airplane length (ft)

SUR: Wing gross area (ft^2)

VEL: Cruise speed (knot)

ALT: Cruise altitude (ft)

WGR: Gross weight (lb)

WFU: Fuel weight (lb)

WEM: Empty weight (lb)

THR: Total engine thrust (lb)

CDR: Drag coefficients ratio

CDO: Drag coefficient at zero-lift

CLM: Maximum lift coefficient

CLA: Alternative lift coefficient

CLC: A lift coefficient between 1.2-2.0

CLR: Lift coefficients ratio

CLT: Lift-off lift ratio

FIM: Maximum (L/D) ratio

FIA: Alternative (L/D) ratio

FIR: The ratio of (L/D) ratios

KBS: The ratio of (L/D) ratios at the beginning of the cruise

CBS: Lift coefficients ratio at the beginning of the cruise

CFN: Lift coefficient at the end of the cruise

KFN: The ratio of (L/D) ratios at the end of the cruise

FOR: Average cruise (L/D) ratio (FIM)

VAS: Square of approach speed

VAP: Approach speed

WSR: Root of wing loading parameter

WSL: Wing loading parameter

WLC: Landing wing loading

TWL: Landing thrust to weight parameter

TLC: Landing thrust to weight ratio

TWT: Take-off thrust to weight ratio

TOP: Take-off parameter

WST: Take-off wing loading

CUP: Cruise parameter

WCS: Cruise wing loading

TCT: Engine thrust ratio according to altitude

TWC: Cruise thrust to weight ratio

WSF: Matching point wing loading

TWF: Matching point thrust to weight ratio

ALF: Matching point altitude

VSF: Sound speed at matching point altitude

VEF: Matching point airplane speed

UPF: Useful weight ratio

SFC: Specific fuel consumption

BRQ: The Brequet factor

WFF: Fuel gross weight ratio

THM: Matching point engine thrust

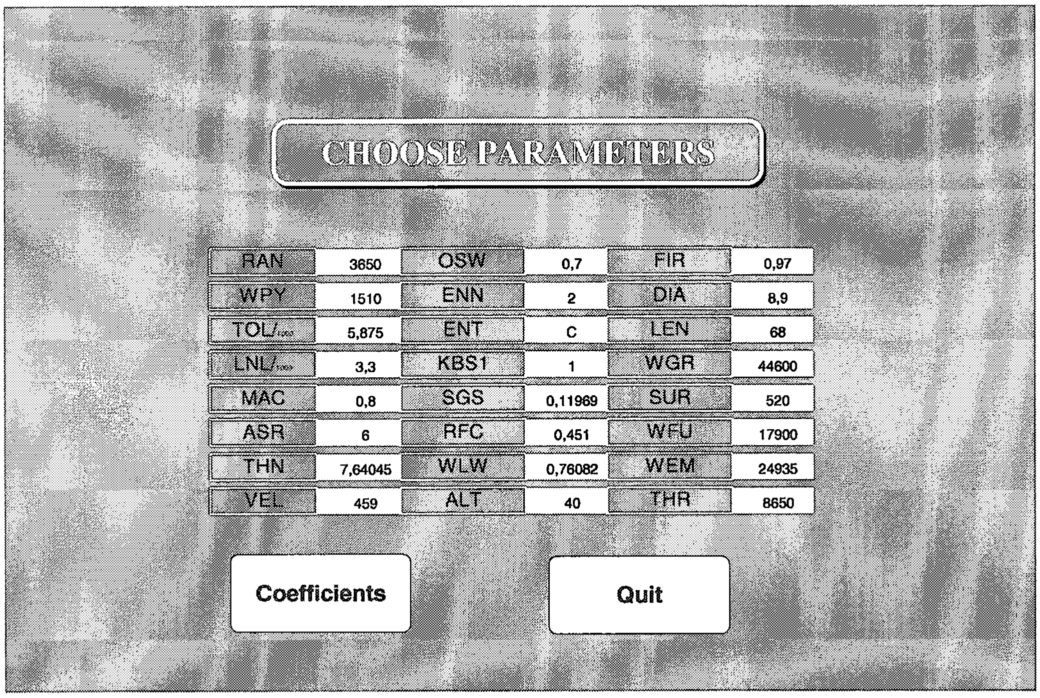
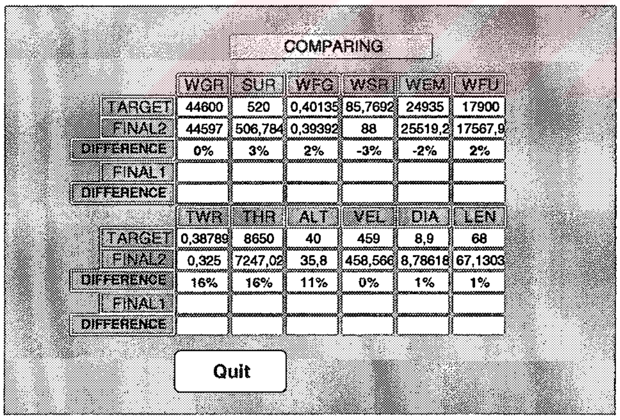


Figure 4.5 Definition Input Screen Display

5. CONCLUSION

Using the Loftin method, the effects of the variables to each other are determined precisely and tabulated. The relationships between these variables are formulated. Additional data can be collected for small-size aircraft and figures are modified. In this way, the variation of fundamental characteristics for small-size aircraft is found. In order to make the calculation, the output data of Loftin method for current aircraft models are used. The output data of Loftin method are gross weight, thrust and major dimensions. Using the known values of current aircraft flight characteristics can be calculated using the known formulas and relationships. In this way, a more realistic design procedure based on the database is obtained. But there may be difficulties in finding the data of current aircraft or the data we found can not be in the format we need. Some sources can contain different definitions such as the speed at average cruise weight etc.


COMPARING

	WGR	SUR	WFG	WSR	WEM	WFU
TARGET	44600	520	0,40135	85,7692	24935	17900
FINAL2	44597	506,784	0,39392	88	25519,2	17567,9
DIFFERENCE	0%	3%	2%	-3%	-2%	2%
FINAL1						
DIFFERENCE						

	TWR	THR	ALT	VEL	DIA	LEN
TARGET	0,38789	8650	40	459	8,9	68
FINAL2	0,325	7247,02	35,8	458,566	8,78618	67,1303
DIFFERENCE	16%	16%	11%	0%	1%	1%
FINAL1						
DIFFERENCE						

Figure 5.1 Size Comparison Output Screen

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ÖZGEÇMİŞ

1974 yılında İstanbul' da doğdum. 1991 yılında Vefa Lisesi'nden mezun oldum ve aynı yıl İstanbul Teknik Üniversitesi Uçak ve Uzay Bilimleri Fakültesi, Uçak Mühendisliği bölümünde öğrenime başladım. 1995 yılında Uçak Mühendisliği bölümünden mezun oldum ve aynı sene Uçak Mühendisliği anabilim dalında yüksek lisans programına kayıt oldum. 1996 senesinde Türk Hava Yoları'nda Yapısal Sistemler Mühendisi olarak işe başladım ve halen bu görev çerçevesinde uçak yapılarında bakım, onarım ve modifikasyon çalışmaları yapmaktayım.

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