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Wing Design for Light Transport Aircraft with Improved Fuel Economy

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Wing Design for Light Transport Aircraft with Improved Fuel Economy[1]

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Summary:

An advanced technology wing (Tragflügel neuer Technologie TNT) has been designed for a light utility and commuter service aircraft with the requirements for economy, safety and flexibility. Trade-off studies give optimum area and aspect-ratio of the wing. A new airfoil and a new flap were developed to fulfill the performance requirements. Wing planform and twist were chosen to give high maximum lift, low drag and good stall characteristics. Presen ailerons were optimized for wheel forces and lateral control. The applied aerodynamic methods including two- and three-dimensional wind tunnel tests are shown. Various structural configurations of the wing and various flap systems are evaluated. The cantilever tapered wing with integrally machined structure and a Fowler-flap with a two-lever mechanism were found to be the most economic ones. The wing was constructed and flight-tested with a modified Dornier Do 28 SKYSERVANT as a test bed. The new wing is applied to a family of light transport aircraft. To demonstrate the effect of advanced wing technology, the new model is compared with existing aircraft.

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1) The work described in this paper was sponsored by the Federal Ministry of Research and Technology (BMFT)

Introduction

Since 1975 Dornier has developed a new wing for a light transport aircraft which has been in flight testing since June 1979. The basic requirements for the design were safety, flexibility and improved economy compared with existing aircraft. The main sources of expected performance improvements were a new wing section and wing-tip shape. Trade-off studies to optimize wing area and wing aspect ratio for the specified performance and mission requirements were carried out using conventional methods. The new wing section with higher maximum lift and higher lift-drag-ratio was derived from the supercritical airfoils developed in [1] and [2].

The wing design was aimed to give high performance and good stall characteristics. A further improvement in performance comes from the wing tip configuration as investigated in [3]. Wing-fuselage structural attachment, Fowler flap extension mechanism and wing-box structure were chosen on the basis of facilities available at Dornier to give lowest DOC. Flight testing of the new wing with the Do 28-D2 as test bed started in June 1979.

1. WING PARAMETER DESIGN STUDY

This study is based on the specification (from realistic market analyses) for a utility and commuter aircraft in the weight category of up to 5.7 tons, with a range of 1000 - 1500 km and more than 400 km/hr cruise speed, Fig. 1. This specification includes particularly strict demands to be met by single-engine climb performance.

The wing parameter design studies followed the flow-chart shown in Fig. 2. Starting with a basic design, the parameters, wing area A, aspect ratio Λ, and take-off weight W were varied.

These variations result, on the one hand, in an incremental change in the structural weight. Together with the constant weight portion, there is a certain available fuel weight for the triple of values (W, A, Λ). On the other hand, varied drag polars are connected with an incremental change of the wing parameters A and Λ. These drag polars result in certain flight performances in connection with known thrust data. The required fuel weight is calculated from the range requirement. When the required fuel weight corresponds with the available fuel weight for the range requirement, the required take-off weight is established as a function of the wing parameters A and Λ. Based on this weight, the A and Λ combinations are determined with which required flight performance is obtained.

The required flight performance is calculated on the basis of the preliminary specifications. Fig. 3 shows the results of the parameter investigations. The three long dash curves each show for a constant take-off weight the combination of wing area and aspect ratio, which combinations can meet the range requirement of 1200 km. These curves form the basic net to which the families of limiting curves were superimposed for maximum cruise speed (solid lines), take-off distances (short dash lines), and single-engine climb gradients (dash-dot lines). Fig. 3 shows that the cruise speed requirement of 400 km/hr is fulfilled over the complete represented range. The admissible wing areas are limited downward by the required single-engine climb gradients (2.-segment climb) according to FAR 25. The curves for a flap deflection of 0° and 10° are drawn. The curve for a flap deflection of 10° was selected for further consideration. Hence it follows that shorter take-off distances on airports with high ambient temperature situated at a high altitude will result. The take-off distances are within the range given by the preliminary specification. The take-off weight changes very little. This means that wing weight, increasing with aspect ratio, is again balanced by lower fuel consumption.

These results do not yet allow a valid determination of aspect ratio and associated wing area. Therefore, cost was considered as an additional and finally determining criterion. The aircraft cost and thus the fixed charges (depreciation, insurance, interest) are proportional to the increasing aspect ratio on account of the higher wing weight, while the fuel cost is reduced. Fig. 4 shows overall yearly costs (fixed and variable) for a realistic lifetime (20 years) and for three different yearly revenues of the aircraft. With increasing revenue, the influence of the fuel cost is more and more important and shifts the optimum towards higher aspect ratios. Based on these results, the aspect ratio was chosen at nine with the respective wing area of 32 m^2. The respective take-off weight then is 5,550 kg.

2. AIRFOIL DESIGN

With the above chosen wing loading and aspect ratio and with the given performance requirements which are: range, cruise speed, climb angle and take-off distance, the aerodynamic design requirements for the airfoil are the following as shown on Fig. 5. In cruise, at a lift coefficient of $C_L \geq 0,4$, the airfoil should have minimum drag. By reason of low trim drag the zero lift pitching moment of the airfoil should be less nose down than $C_{M\,o} = -0,07$. Maximum lift, flaps up, should be $C_{L\,max} = 1,85$ while the minimum glide angle should be at 1,2 times stalling speed. For take-off under hot and high altitude conditions the flaps are to be set at 20^o and should give $C_{L\,max} = 2,9$ and minimum glide angle should be at 1,2 times stalling speed. Experience with the excellent short landing performance of the Do 28 indicates that the maximum lift of the airfoil with flaps at 40^o should be $C_{L\,max} = 3,4$.

A new airfoil, designated Do A-5, was developed [4], which is tailored to the requirements mentioned above. In its design, a turbulent boundary layer was assumed. Therefore, no better surface quality is required than for conventional airfoils. With a thickness of 16 % the wing has minimum weight with only an insignificant penalty in minimum drag compared to a 12 % thick airfoil.

Compared to the conventional NACA 23018 airfoil, the Do A-5 has more nose droop, a flat upper surface and a cambered rear part. This shape gives rise to a pressure distribution for the high lift with a low suction peak at the nose, favorable adverse pressure gradient on the upper surface behind the nose and a fairly high load on the aft portion of the airfoil. The high load on the front part of the lower surface reduces the zero lift nose-down moment.

Two-dimensional wind tunnel tests in the Laminar Wind Tunnel of the University of Stuttgart show that the design requirements are met very well as shown in Fig. 6. It is seen, that minimum drag and minimum glide angle are improved compared to the NACA 23018 airfoil, which was also measured in the same wind tunnel.

3. FLAP DESIGN

Because of cost it was necessary to fulfill the requirements without using a leading-edge device. Because of the camber of the rear part of the Do A-5 and due to a new developed two lever flap mechanism, which is described later, a single slotted flap is as efficient as a double slotted flap on a conventional airfolil. The 30 % chord flap is extended nearly 14 % chord behind the airfoil trailing edge in the take-off and landing position. Wind tunnel tests gave, for flaps at 20°, a maximum lift of $C_{L\ max} = 3{,}0$ and the best glide angle at a lift of $C_L = 2{,}0$. Both values show that the requirements are exactly met as shown in Fig. 7. Compared to the conventional airfoil with the same flap deflection the maximum lift and the lift-drag-ratio are considerably higher. Similar results are true for the landing configuration with flaps deflected 30° and 40°.

4. WING DESIGN

The planform of the wing consists of a rectangular inner part extending out to the engine nacelle and a tapered outer part. This planform results from a trade-off study of different wing configurations which is discussed later.

Initially, the necessary spanwise extend of the aileron was calculated to achieve a roll angle of 30° after not more than 1,8 seconds in accordance with MIL-8785 B for the landing condition. The remaining spanwise extend of the flaps was 65 %.

A trade-off study was made for the choice of optimum taper ratio and twist. The spanwise lift distribution of the wing was calculated by a nonlinear lifting line theory including viscous effects. The following criteria were used:

a) The aileron effectiveness, expressed by the local lift margin ΔC_L at 0,7 span at the moment when the maximum lift $C_{L\ max}$ at any inboard wing station is reached, should give a ΔC_L of not less than 0.13.

b) The aileron effectiveness beyond wing stall, expressed by the margin in angle of attack $\Delta\alpha$ between the angle of attack for maximum lift of the wing and the local angle of attack for local maximum lift at the 0,7 span station, should give a $\Delta\alpha$ of 3 degrees.

c) Maximum lift of the wing should be 80 percent of the 2-D $C_{L\ max}$ as shown in Figure 5.

d) Induced drag should be a minimum.

e) Weight of the wing should be a minimum.

Fig. 8 shows some results of this study for the take-off configuration. To fulfill the stall criteria a more tapered wing needs more twist which results in higher induced drag. A rectangular wing needs no twist but has less maximum lift. The same calculations were done for cruise and landing configurations. The structural weight of the wing decreases with more taper. A taper ratio of 0,7 and a twist of -3° was choosen as the optimum configuration of the outer wing panel.

In spite of the lack of a theory which takes into account the propeller slipstream, the margins of a) and b) were choosen cautiously, so that the stall behaviour with propeller slipstream will be good. The choice was confirmed by wind tunnel and flight tests.

Fig. 9 shows some tuft studies in the wind tunnel for the landing configuration. During approach with idle power setting the angle of attack of maximum lift is $\alpha\ (C_{L\ max}) = 15^{\circ}$. The ailerons are efficient far beyond this angle of attack. In the case of a go-around with full thrust, the $\alpha\ (C_{L\ max})$ is increased by $2{,}5^{\circ}$ up to $17{,}5^{\circ}$ by the propeller slipstream. The tuft picture shows attached flow on the whole aileron at this angle of attack. Flow separation spreads more rapid but the effectiveness of the ailerons is high enough to avoid untolerable bank angles. The prediction of good stall behaviour of the aircraft was fully confirmed by the flight tests.

Wind tunnel tests with a 1:8 scale model in different German low speed wind tunnels confirmed the theoretical results. Induced drag of the aircraft with horizontal tail fixed is shown in Fig. 10. It is compared with test results of the Do 28 which has the same horizontal tail and nearly the same fuselage but a conventional wing with the NACA 23018 airfoil. The theoretical induced drag compared to the measured one shows, that the improvement comes not only from the higher aspect ratio of the new wing, but must also come form the less lift dependent drag of the new airfoil.

A triangular-like shape was choosen for the wing tip as shown in Fig. 11. Compared to conventional wing tip shapes with the same area it has less induced drag for the same wing root bending moment. To avoid high local peaks in the lift distribution, the triangular wing tip is twisted and cambered.

5. AILERON

It was decided to use slotted ailerons because of their greater potential in maximum lift, aileron effectiveness and stall behaviour. The drag of the open slot in cruise was tolerated. For this reason it was possible to use symmetrical deflection of the ailerons to increase the maximum lift by $\Delta C_{L\ max} = 0{,}15$ and the maximum lift-drag-ratio by 13 %, Fig. 12. The latter improvement comes from the more favourable (near to elliptical) lift distribution.

The optimization of the aileron was done by two-dimensional wind tunnel tests at a full scale Reynoldsnumber and by three-dimensional wind tunnel tests. Many criteria have to be fulfilled, Fig. 13. The effectiveness $C_{l\ \xi}$ has to be linear, only with small changes with α and with not too much decrease beyond. The hinge moment $C_{r\ \xi}$ has to be linear with only small changes with α. The aileron yawing moment $C_{n\ \xi}$ should be proverse. The hinge moment due to yaw should be small. The rolling moment due to side slip in the case of aileron preset should be small and proverse. Aileron buffet should be small. Maximum lift in the case of aileron preset should be as high as possible. The minimum drag in the case of aileron preset should be small.

The variable parameters for the optimization were: shape of the aileron nose, shape of the shroud, slot width, pivot point in x- and z-direction and aileron preset. The wind tunnel tests showed, that one has to live with a compromise:

Aileron preset of 10° is favourable for maximum lift, lift to drag ratio, stall behavior and hinge moment but there is an unfavorable adverse yawing moment due to aileron deflection.

6. FLAP SYSTEM

For the aircraft specifications as defined in the first section a trade-off study was made to optimize costs for two different flap systems, Fig. 14:

- Single slotted Fowler-flap with a two-lever mechanism
- Single slotted flap with a fixed hinge point.

As shown in the aerodynamic part of this paper the Fowler-flap has high maximum lift and a high lift-drag-ratio for take-off and landing configurations. It can be shown, that the loads on the lever due to aerodynamic lift and drag are relatively low, so that the cross section of the lever is very small and there is no drag penalty in the take-off, landing and cruise configuration compared to the fixed hinge point system. On the other hand, the single slotted flap has less maximum lift so that the wing area must be increased.

It was found, that the Fowler-flap, with two lever mechanism, is considerably cheaper than the system with a fixed hinge point.

7. WING STRUCTURAL DESIGN

Comparison studies were aimed at determining a wing shape that is optimal with respect to weight and production cost. There were four promising solutions to choose from:

- Braced Rectangular Wing RA
- Cantilever Rectangular Wing RF
- Rectangular-Tapered Wing RT
- Tapered Wing T

These possibilities, constant design parameters and the results of the evaluation are shown in Figure 15. The tapered wing is the lightest, as was to be expected, but its first cost is very high. One obtains a completely different impression with respect to the operating cost. The DOC comparison took into account that the different structural weight of the wings influences the payload capacity with a quality factor and that additional drag (struts) increase the fuel consumption.

It could be verified that the following DOC-related priority appeared from a utilization rate of approximately 500 flight-hours/year:

1. Rectangular Tapered Wing
2. Tapered Wing
3. Cantilever Rectangular Wing
4. Braced Rectangular Wing

The usual revenues of utility and commuter aircraft of > 1,500 flight hours/year allowed only the selection of a rectangular tapered wing even with the low fuel cost recorded in 1975. Latest developments in fuel costs have shown that, compared to the first cost, the operating cost is of high importance in view of the revenues and life of the aircraft.

8. WING-BOX STRUCTURE

The design studies concentrated on the optimum structure of the spar box with respect to weight and production cost.

Four designs were compared, Fig. 16.

1. Design: Integrally machined structure; whole wing.
2. Design: Integrally milled inner wing, conventional (skin-stringer) outer wing.
3. Design: Pure differential wing, i.e. conventional skin-stringer-rivet design; fuel cell compartment is integrally sealed in the inner wing.
4. Design: Like design 3, however with rubber bladder tanks.

Fig. 16 shows the weight and production cost comparison results on the basis of a series production of 300 aircraft.
The table shows, that the integrally machined structure has the lowest weight and the lowest production cost and hence, this one was chosen.

9. ZKP-TNT-PROGRAMME AND FLIGHT TEST RESULTS

The design of an advanced wing (Tragflügel neuer Technologie TNT) was sponsered by the Federal Ministry of Research and Technology (BMFT) within the ZKP-Programm (Ziviles Komponenten Programm). The TNT programm started in July 1975 with design studies and the aerodynamic and structural design. The structural calculations, construction and assembly of the aircraft took from the end of 1976 until the beginning of 1979. The TNT experimental aircraft made its first flight on June 14th 1979. Until now more than 140 flights have been made.

For data acquisition a programmable PCM-timemultiplexsystem is used. It allows recording of analog as well as digital signals. There were 120 probes and sensors on the aircraft installed to measure:

- conditions of the atmosphere and of the flight,

- configurations of the aircraft and of the engines, (flag-deflection, throttle-setting etc.)
- structural and load data.

The flight test data were used to calculate the performance of the new wing. The lift and drag polars were calculated from thrust- and inertia-forces. Thrust was calculated by comparison of fuel flow, revolution and torque with the performance chart of the uninstalled propeller. A propeller installation efficiency of 0,88 was used. This number was found from static tests with the experimental aircraft and from wind tunnel tests with a powered model of the aircraft and with an uninstalled model propeller. Inertia forces are calculated from take-off weight, fuel flow, flight speed and accelerations normal and tangential to the flight path.

Fig. 17 shows the maximum lift versus thrust coefficient. The thrust coefficient is defined as the thrust divided by the dynamic pressure and the wing area. The thick lines come from flight tests with a Reynolds number between 3,8 and 6 million. The thin lines were measured in the wind tunnel with a Reynolds number between 1 and 0,3 millions. The correction of $C_{L\ max}$ due to Reynoldsnumber is approximately 0,3 for zero thrust. The maximum trimmed lift without thrust is 1,75 with flaps retracted and 2,65 with flaps 40^{o} deflected. These numbers are in good agreement with the requirements for the airfoil shown in Fig. 4.

The aircraft has gentle stall behaviour for every flap deflection and thrust setting. There is a low to moderate buffet before maximum lift. Beyond maximum lift the aircraft dips nose down without lateral roll-off.

Fig. 18 shows the drag polars of the experimental aircraft and compares them with the prediction from wind tunnel tests published in [5]. The wind tunnel data have been corrected by the influence of Reynolds number on friction drag. The drag of the landing gear has also to be added since the wind tunnel model had none. The experimental aircraft has a landing gear from the Alpha-Jet with a drag coefficient of 0.012, so that the minimum drag of the aircraft without landing gear is 0.03. The 20° flap setting, which is used for take-off has a low drag, so that even under hot and high altitude conditions the requirements for the balanced field length and for the climb gradient shown in Fig. 3 are well fulfilled. The landing configuration with 40° flaps has a landing distance which is approximately the same as the balanced field length for take-off, which is a well balanced design.

The characteristics of the ailerons were similar to that predicted in the wind tunnel tests as mentioned in chapter 5. The aileron preset of 10° in the take-off and landing configuration increases maximum lift and lift to drag ratio, gives very gentle stall behaviour and a linear and well balanced stick force. There is however an unfavourable adverse yawing moment due to aileron deflection. A 4° rudder deflection is necessary to balance the maximum yawing moment in the worst case. For ILS-approaches this is an uncomfortable behaviour for the pilot. One simple possibility to counter the aileron-yawing moment is to couple the rudder with the aileron.

10. APPLICATION OF THE NEW TECHNOLOGY WING

The new wing, as successfully flight tested on the TNT-Experimental aircraft, has been applied to a family of light transport aircraft designs, including the Do 228-100 for 15 and the -200 for 19 passengers. These configurations are like the TNT but with two different fuselage lengths and a rectractable landing gear.

These new aircraft were designed to demonstrate the effect of advanced wing technology, incorporating features such as

- a new airfoil with high lift, high lift/drag ratio, moderate zero lift pitch moment and
- fully, integrally machined load carrying wing box.

The two new models are compared with existing aircraft. Performance is compared at ISA-conditions, a range of 500 nm and 13 passengers payload,

The equation for climb speed is shown in Fig. 19. Power, propeller efficiency weight and wing loading was held constant whereas the optimum climb angle is better with the new section. This is the reason that the climb performance is considerably improved. A further advantage is the longer range compared to today's aircraft. Due to the higher lift to drag ratio, a smaller wing area is allowed and less installed power is required for the same performance. Both effects decrease the aircraft/size and empty weight and therefore fuel consumption and airframe cost. At equal maximum take-off weight, longer ranges can be flown. The integrally machined wing box with integral fuel tanks provides an additional saving in weight.

The most important gain of the advanced technology wing becomes evident in fuel economy. Fuel cost per ton-kilometer is primarily a function of payload to aircraft empty weight, fuel consumption of the engine in climb, cruise and descent condition. In Figure 20 cost of fuel burned per ton kilometer is shown versus range. Fuel price is assumed at 1,50 $/US-gallon. Over the entire range the new aircraft consumes only 75 % fuel of contemporary aircraft.

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Wing Design for Light Transport Aircraft with Improved Fuel Economy

Purpose of Aircraft

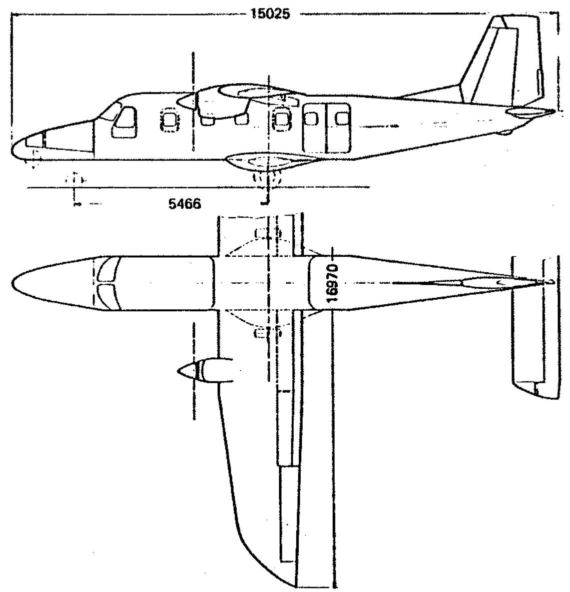
- utility
- commuter

Basic Data

– take-off weight	5700 kg
– range	1000 - 1500 km
– cruise speed	> 400 km/h, H = 1000 ft

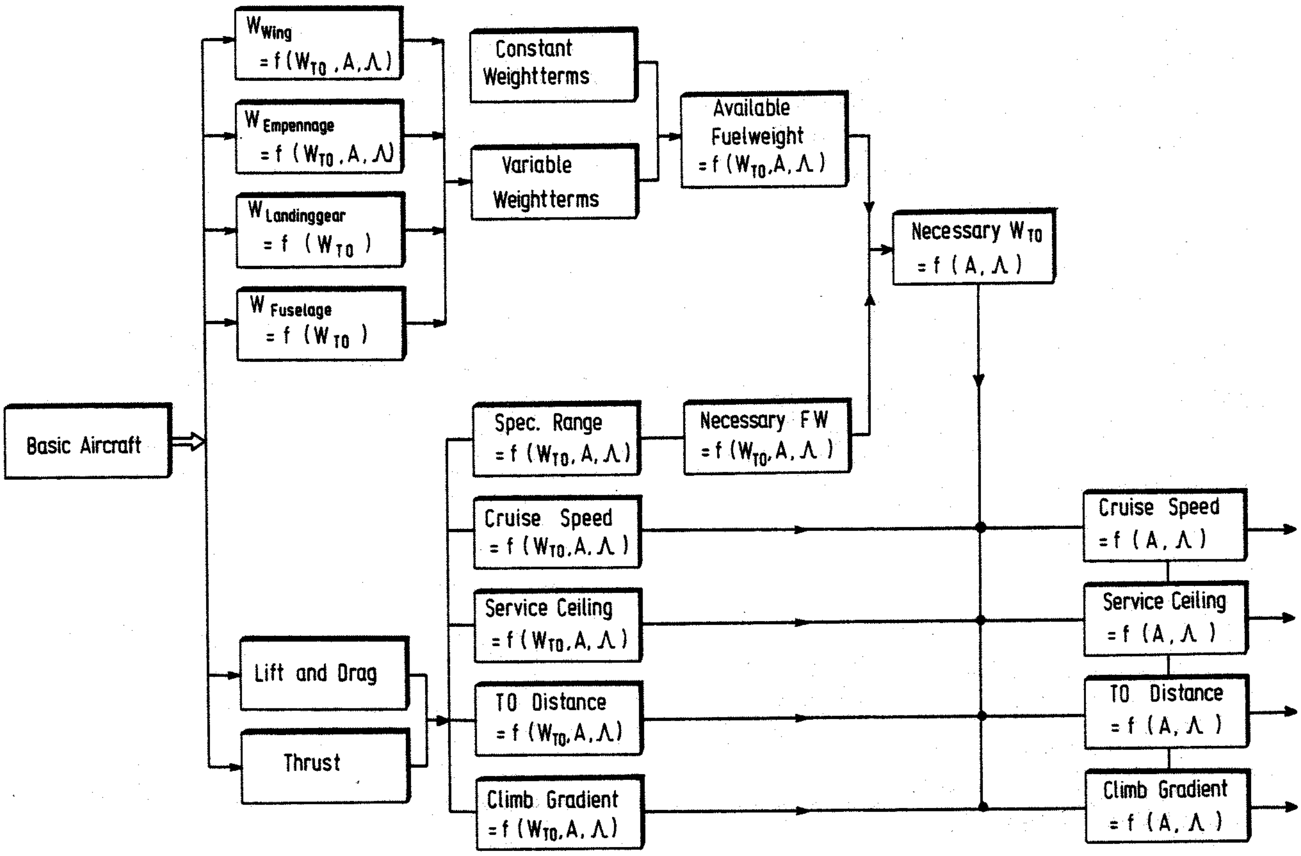
Design Requirements

- Economy
- Safety
- Flexibility



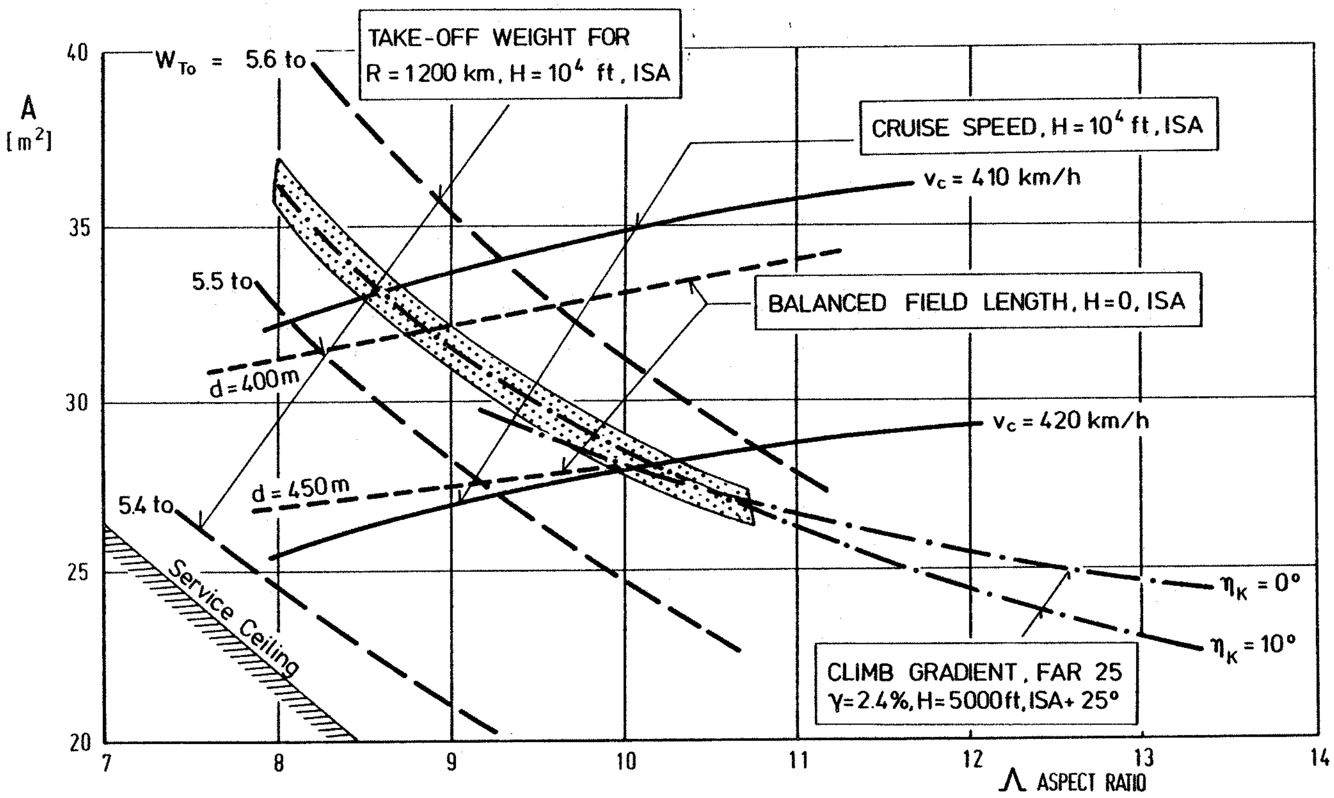
BASIC OBJECTIVES OF THE ZKP/TNT PROGRAMME

FIGURE 1



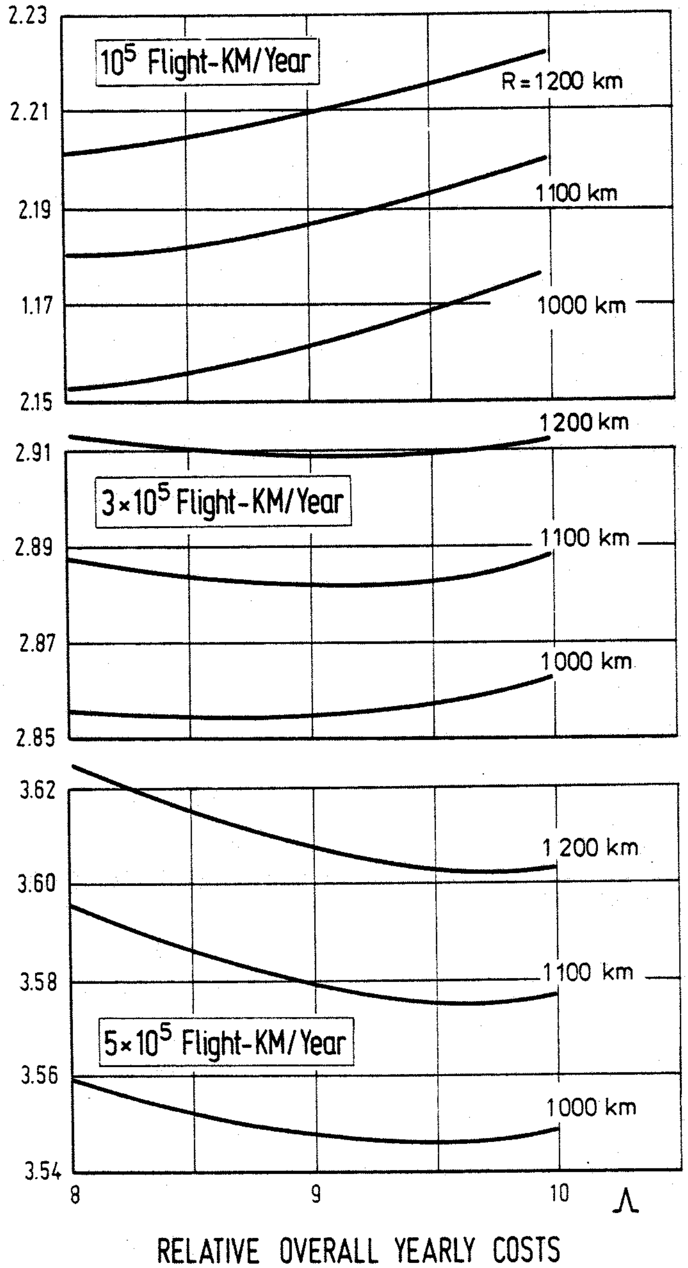
FLOWCHART FOR WINGDESIGN TRADE-OFFS

FIGURE 2



WING DESIGN CHART

FIGURE 3



2.23
2.21
2.19
1.17
2.15
10⁵ Flight-KM/Year
R = 1200 km
1100 km
1000 km
2.91
2.89
2.87
2.85
3×10⁵ Flight-KM/Year
1200 km
1100 km
1000 km
3.62
3.60
3.58
3.56
3.54
5×10⁵ Flight-KM/Year
1 200 km
1100 km
1000 km
8
9
10
Λ
RELATIVE OVERALL YEARLY COSTS

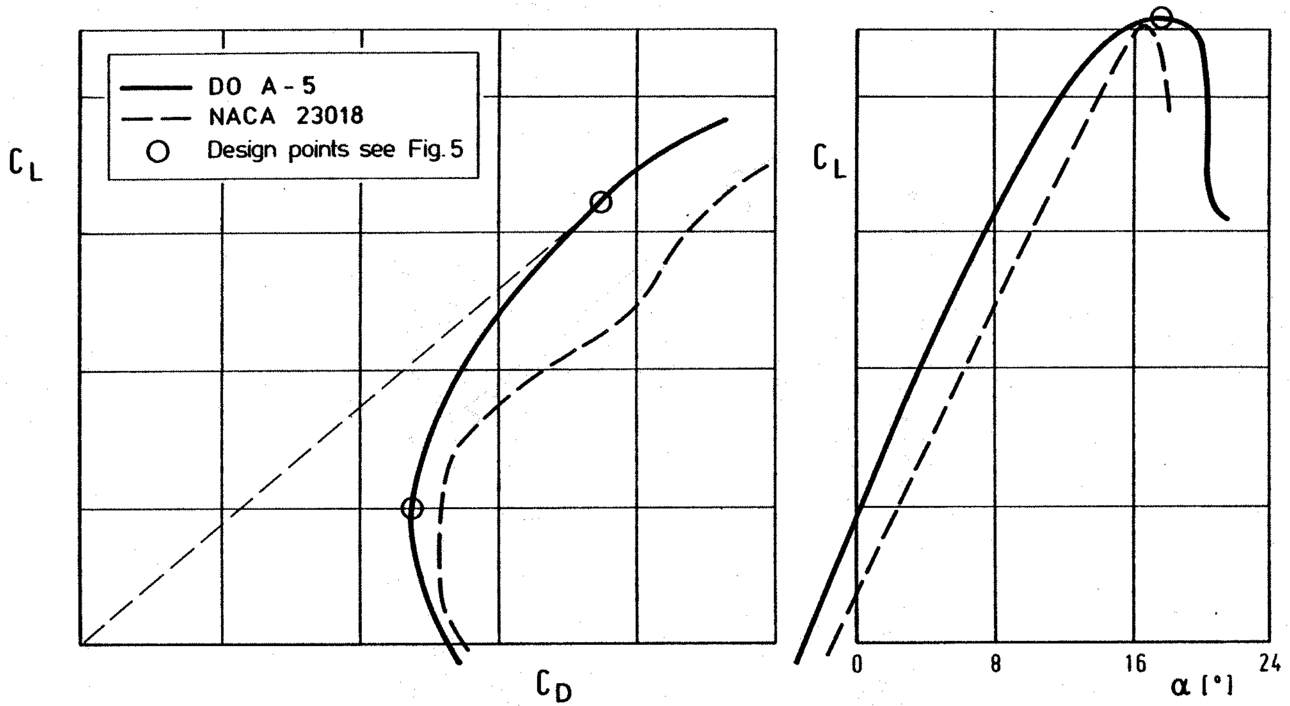
FIGURE 4

FIGURE 5

PERFORMANCE REQUIREMENTS ⟹		REQUIREMENTS FOR THE AIRFOIL		
Range	$R = \frac{\eta}{b} \cdot \ln \frac{W_o}{W} \cdot \left(\frac{C_L}{C_D} \right)$	$C_{D_{min}}$ at $C_L = 0.4$		–
Cruise Speed	$V = \sqrt{\frac{T}{C_D \frac{\rho}{2}}}$	$C_{M_o} > -0.07$		–
Climb Angle	$\gamma = \frac{T}{W} - \frac{C_D}{C_L}$	$\left(\frac{C_D}{C_L} \right)_{min\,1.2 V_s}$ $C_{L_{max}} = 1.85$	Flaps 0°	ISA, H=0
Take-off Distance	$s \sim \frac{1}{C_{L_{max}}}$	$\left(\frac{C_D}{C_L} \right)_{min\,1.2 V_s}$ $C_{L_{max}} = 2.9$	Flaps 20°	Hot and High
Short Landing as Do 28		$C_{L_{max}} = 3.4$	Flaps 40°	–

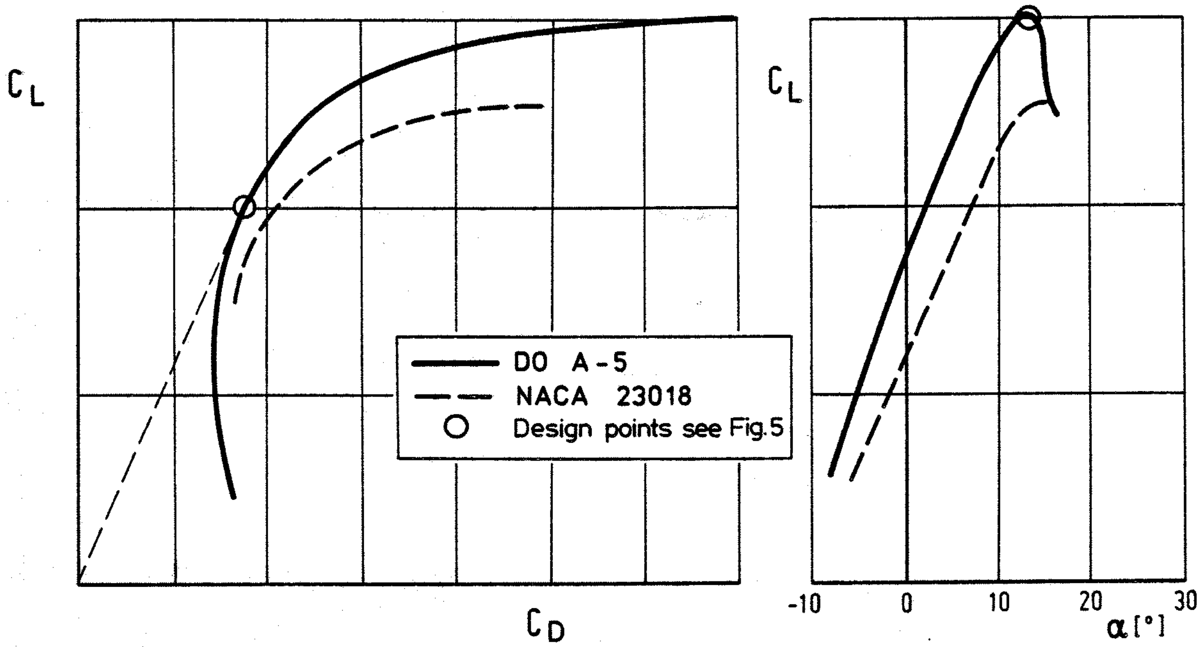
DESIGN REQUIREMENTS FOR THE AIRFOIL

FIGURE 6



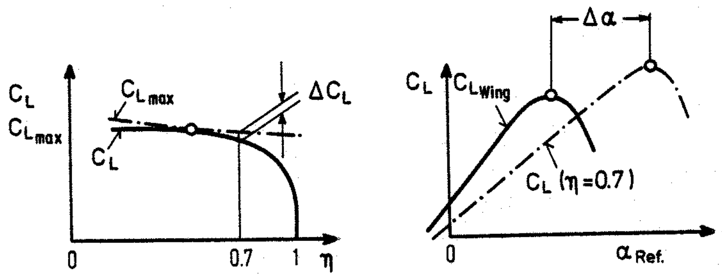
COMPARISON OF SECTION CHARACTERISTICS OF Do A-5 AND NACA 23018 AIRFOILS.

M=0,15; Re=5 x 10^6 ; Transition at x/c=17 % on Upper and Lower Surface



COMPARISON OF SECTION CHARACTERISTICS OF Do A-5 AND NACA 23018 AIRFOILS WITH FLAPS IN TAKE-OFF POSITION.
$\eta_F = 20°$; M=0,15; Re=5 x 10^6

FIGURE 7

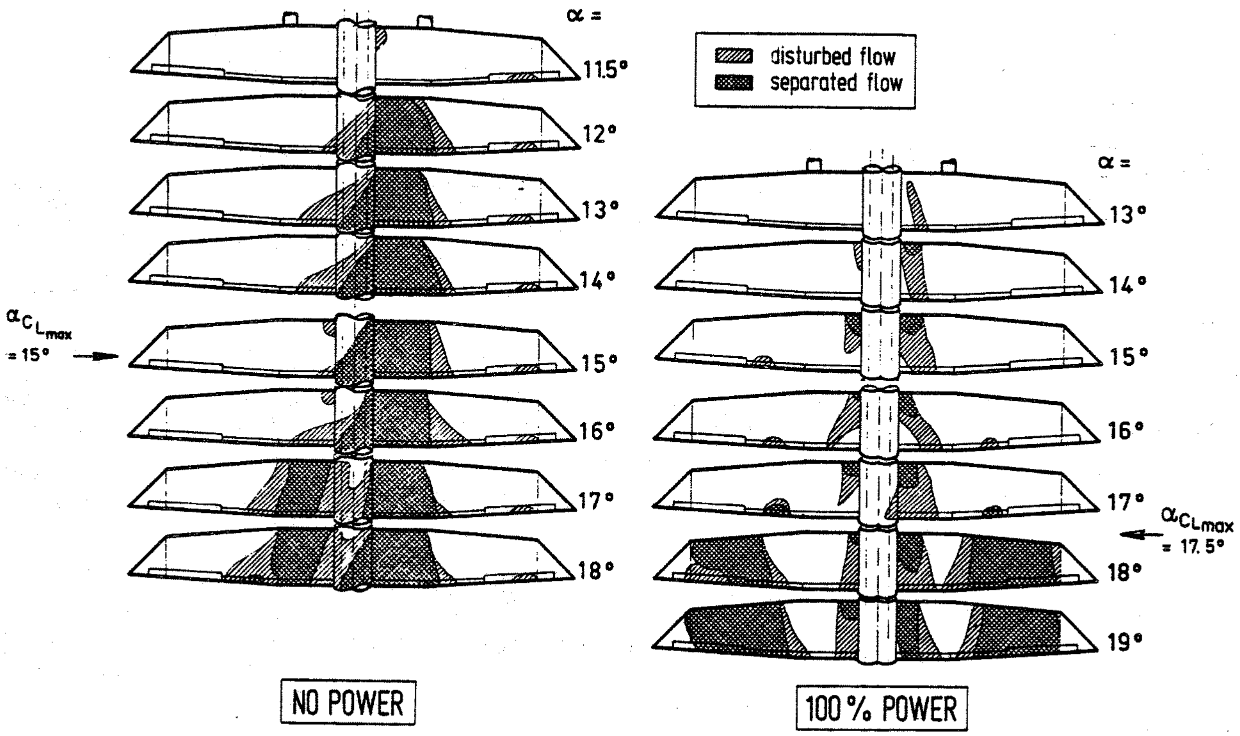


a) $\Delta C_L \geq 0.1$ b) $\Delta \alpha \geq 3°$

taper ratio twist	0.65 0	0.65 -3.5°	0.8 0	0.8 -2°	1 0	0.7 -3°
ΔC_L	0.02	0.15	0.12	0.18	0.27	0.13
$\Delta \alpha$	0	2°	2°	4.2°	5°	3°
$C_{L max}$	2.36	2.35	2.33	2.31	2.29	2.35
C_{Di} (C_L=1)	0.0400	0.0440	0.0400	0.0404	0.0382	0.0419

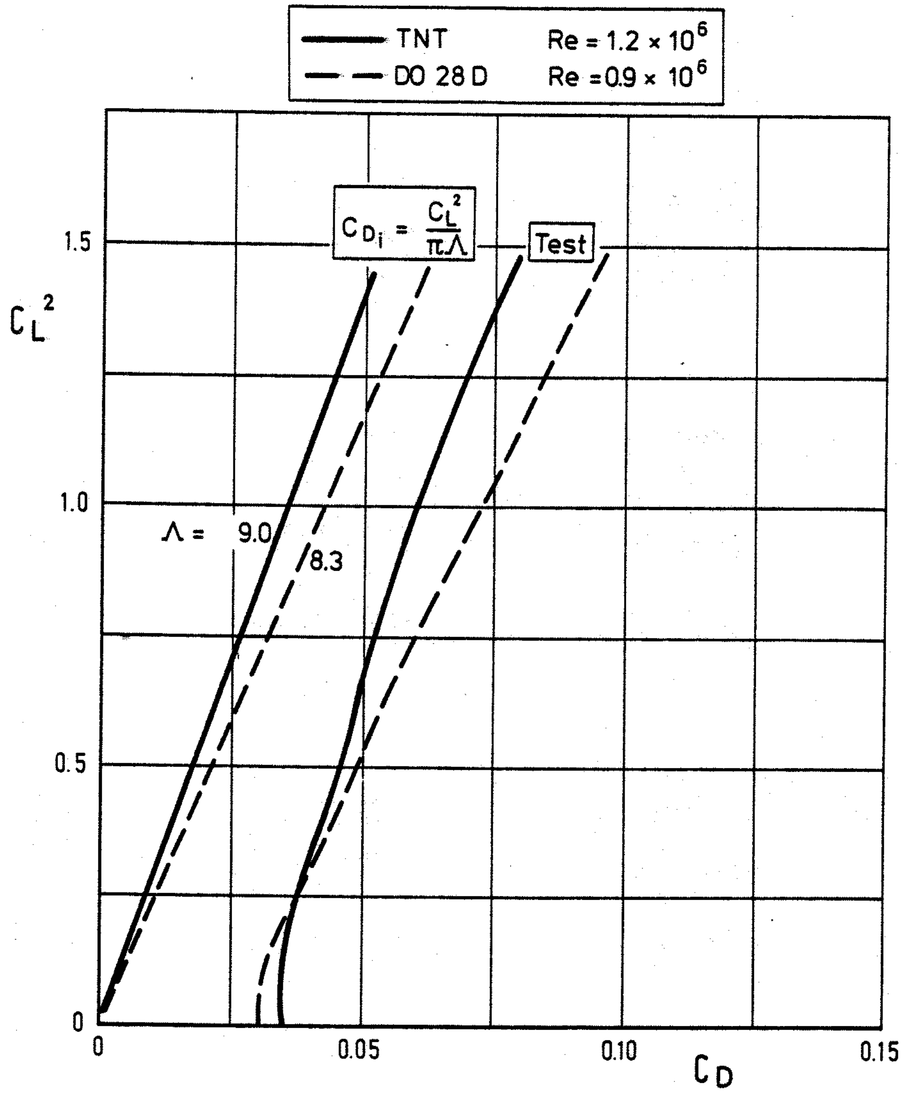
STALL CHARACTERISTICS OF THE WING WITH FLAPS IN TAKE-OFF POSITION

FIGURE 8



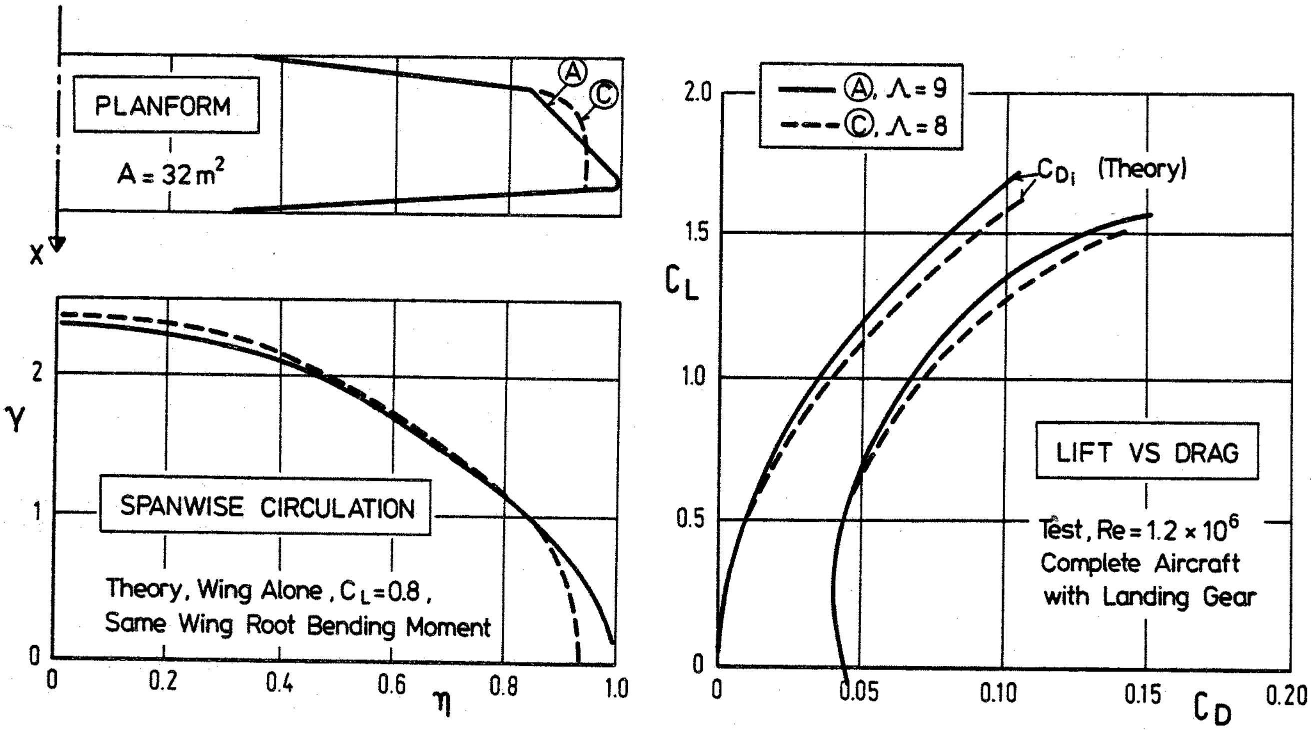
TUFT STUDIES IN THE WIND-TUNNEL; FLAPS η_F =40°; INFLUENCE OF PROPELLER SLIPSTREAM ON STALL

FIGURE 9



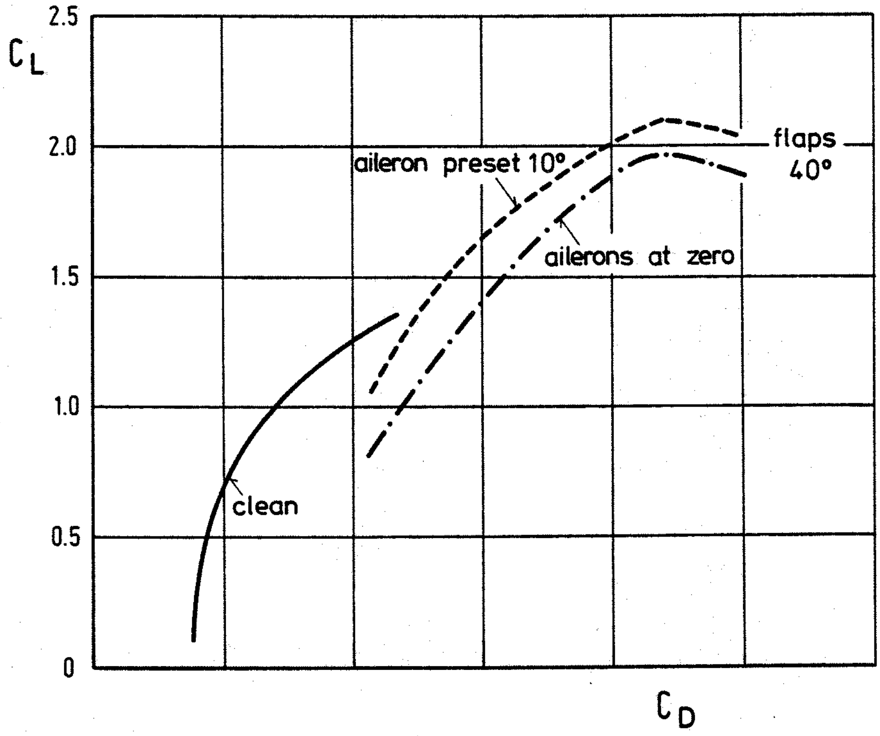
DRAG POLAR OF UNTRIMMED AIRCRAFT

FIGURE 10



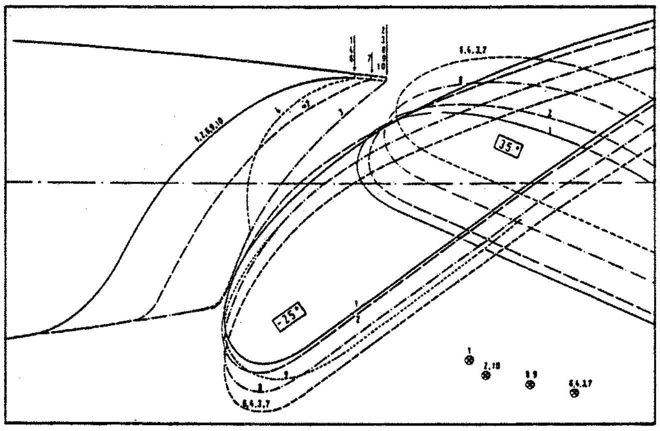
INFLUENCE OF WING TIP PLANFORM ON DRAG

FIGURE 11



INFLUENCE OF AILERON PRESET ON DRAG AND LIFT; Re=5,2 x 10^5

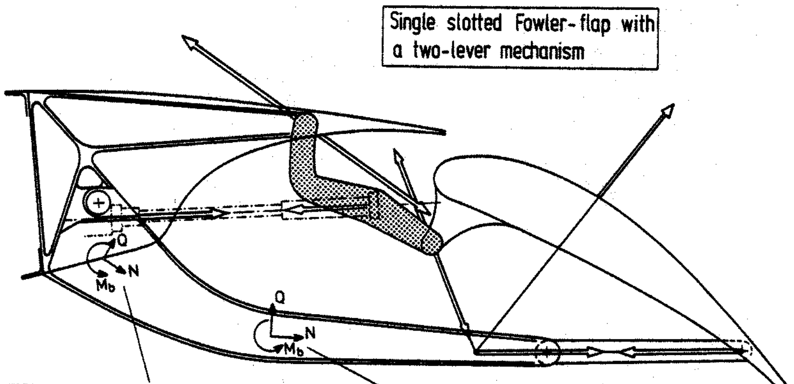
FIGURE 12



Criteria for Aileron Evaluation	Wind Tunnel Test 2-D	Wind Tunnel Test 3-D	Flight Test
Effectiveness	x	x	x
Hinge Moment/Stick Force	x	-	x
$C_{n\xi}$	-	x	x
$C_{r\beta}$	-	-	x
$C_{l\beta}$	-	x	x
Aileron Buffet	x	-	x
Stall	x	x	x
$C_{L_{max}}$	x	x	x
$C_{D_{min}}$	x	x	x

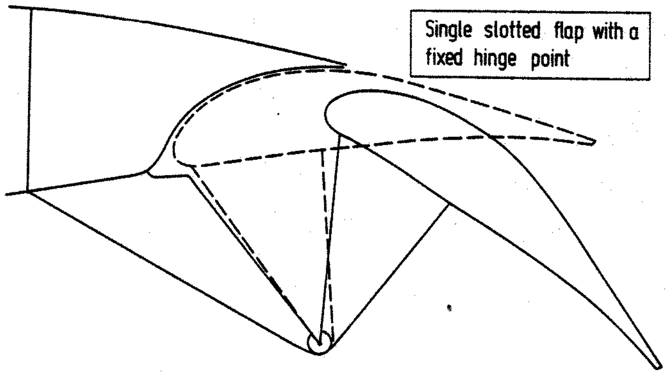
PROCEDURE FOR EVALUATING AILERON MODIFICATIONS

FIGURE 13



η	5°	18°	40°
N [kp]	0	88	440
Q [kp]	54	190	330
M_b [kp]	3600	10 300	8900

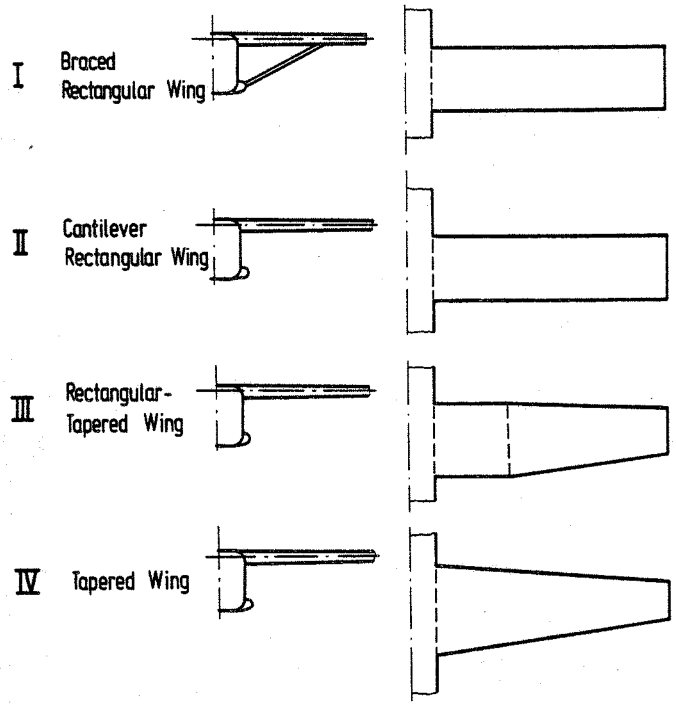
η	5°	18°	40°
N [kp]	38	176	338
Q [kp]	45	110	35
M_b [kp]	2000	4 900	1560



VARIOUS FLAP CONFIGURATIONS

FIGURE 14

CONFIGURATIONS FOR EVALUATION:



DESIGNPARAMETERS WHICH ARE HELD CONSTANT :

- -Cruise Speed v_c = 410 km/h
- -Load Factor n = 5
- -Wing Area A = 31 m^2
- -Aspect Ratio Λ = 9
- -Position of Frontspar x/c = 17%
- -Position of Rearspar x/c = 60%
- -Airfoil Section DO A-5

RESULT OF EVALUATION:

Configuration	Weight	First Costs	DOC (500 FLH / YR)
I	4	2	4
II	2	1	3
III	3	3	1
IV	1	4	2

VARIOUS WING CONFIGURATIONS, DESIGNPARAMETERS AND RESULT OF COSTEVALUATION

FIGURE 15

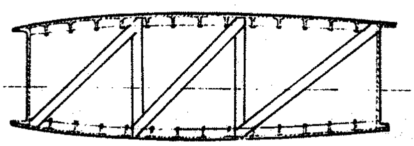
RESULT OF EVALUATION

CONFIGURATION	WEIGHT kg	ASSEMBLY COST %	MATERIAL %	TOOLING COST %	PRODUCTION COST
I	326	66	30	4	100
II	351	64	32	4	107.6
III	410	65	31	4	119
IV	430	63	33	4	114

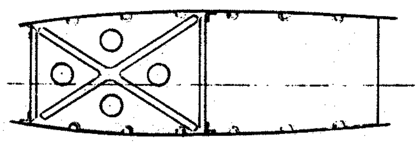
Configurations for Evaluation

I Integrally machined structure, whole wing
II Integrally machined structure, inner panel
Rivetted structure, outer panel
III Rivetted structure,wet inner panel
IV Rivetted structure, rubber fuel cells.

Integrally machined structure

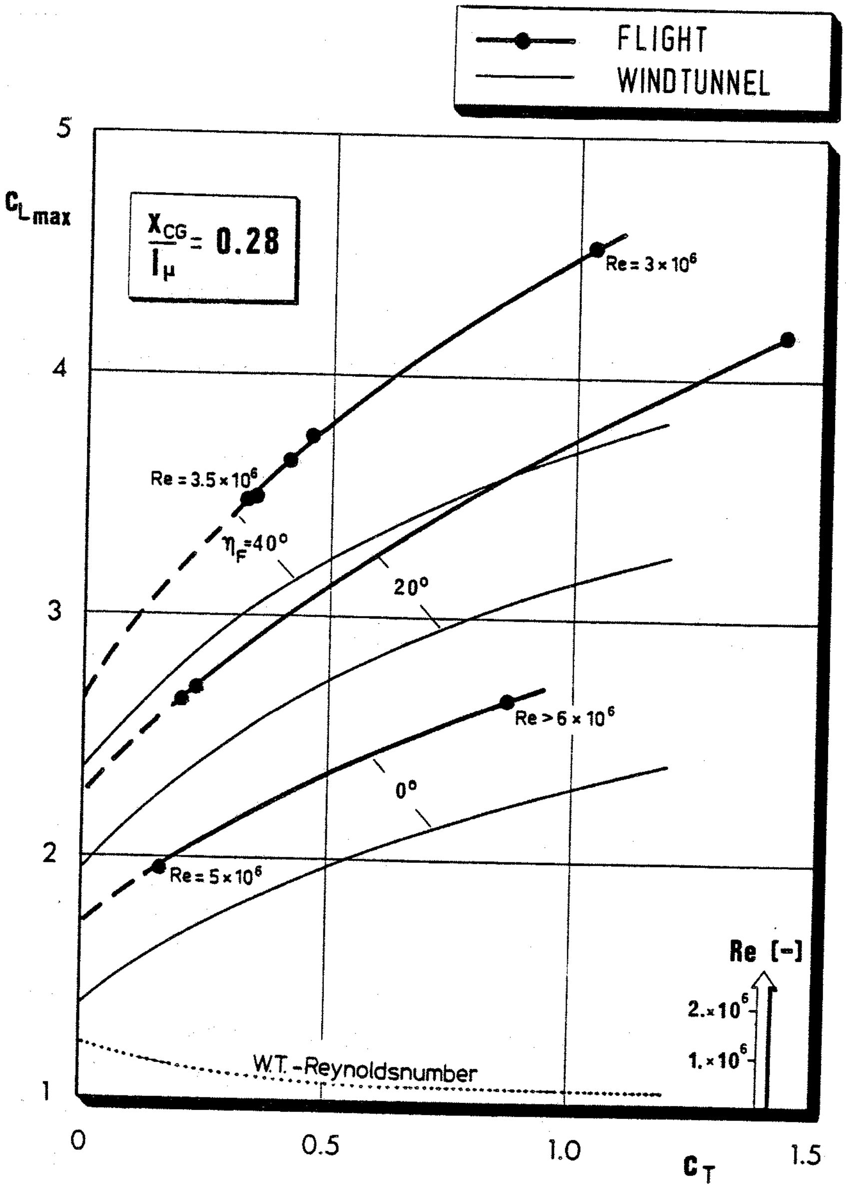


Rivetted structure



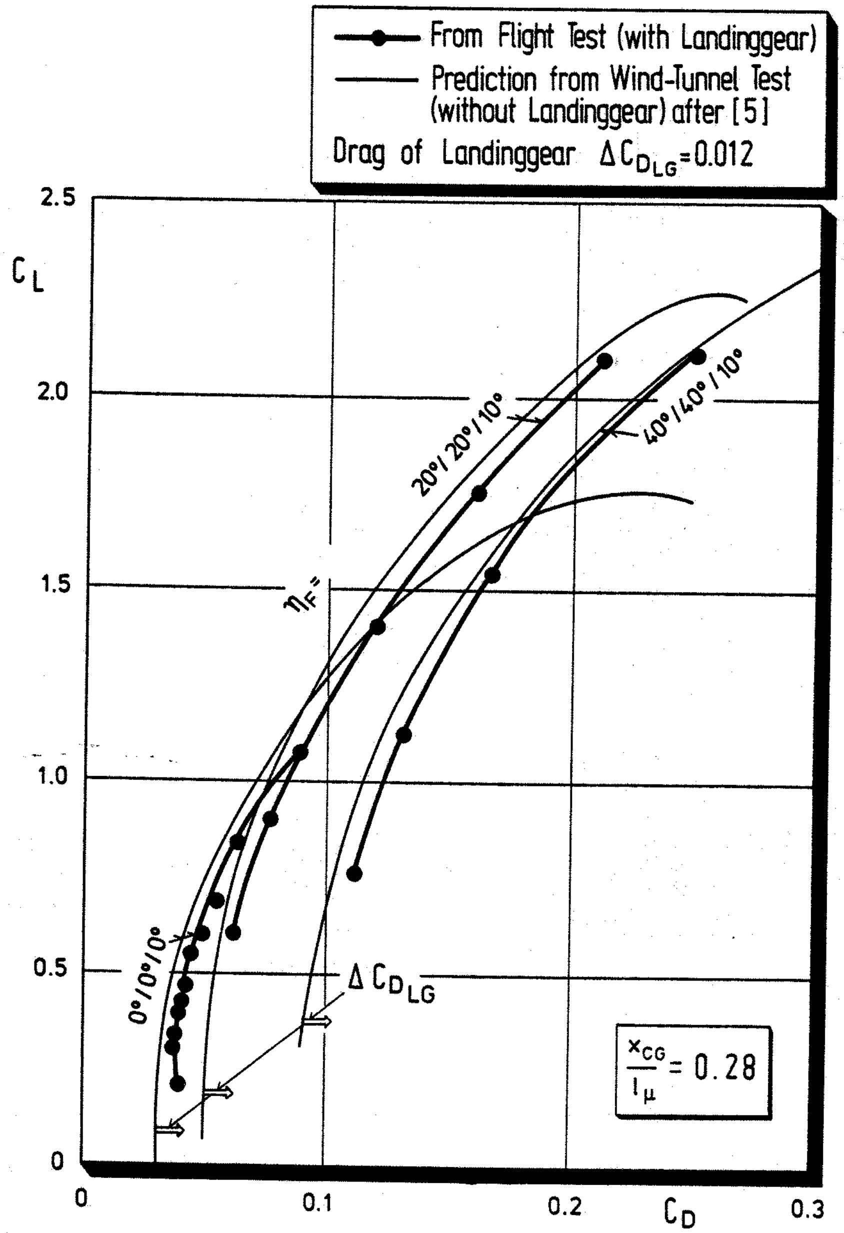
VARIOUS STRUCTURAL CONFIGURATIONS AND RESULT OF EVALUATION

FIGURE 16



INFLUENCE OF THRUST ON MAXIMUM LIFT

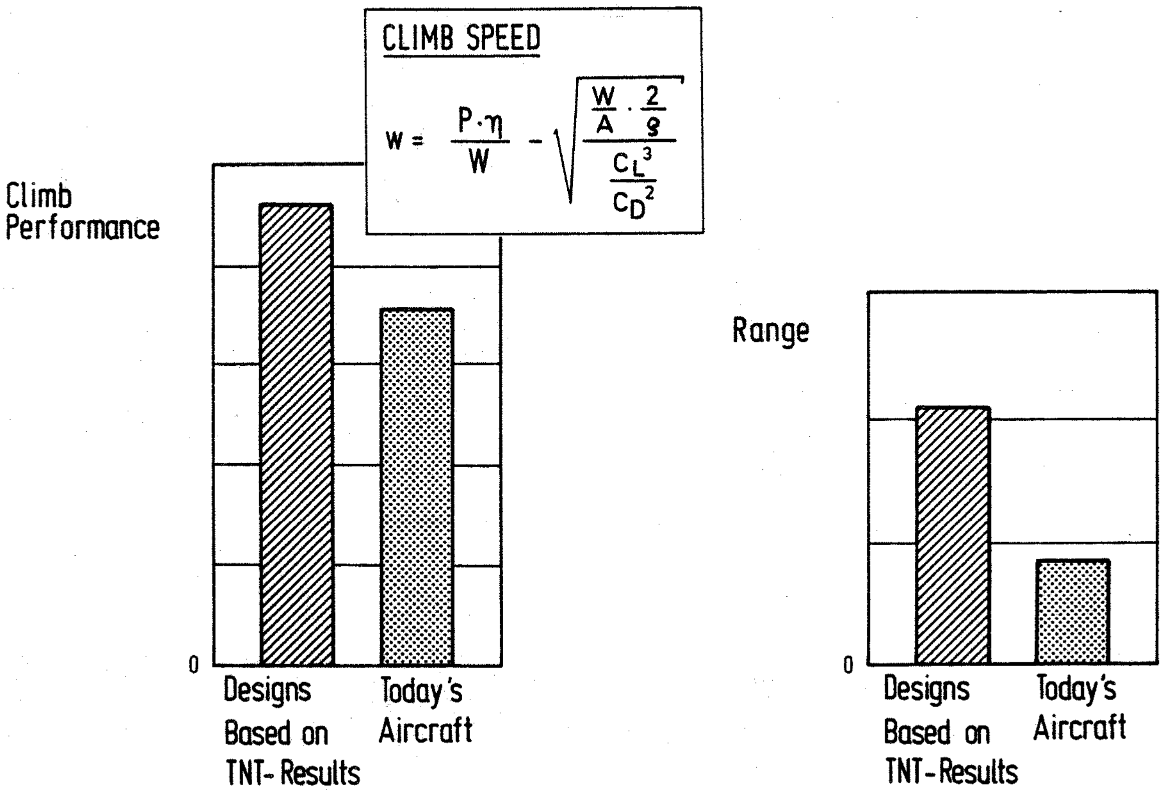
FIGURE 17



TNT DRAGPOLAR

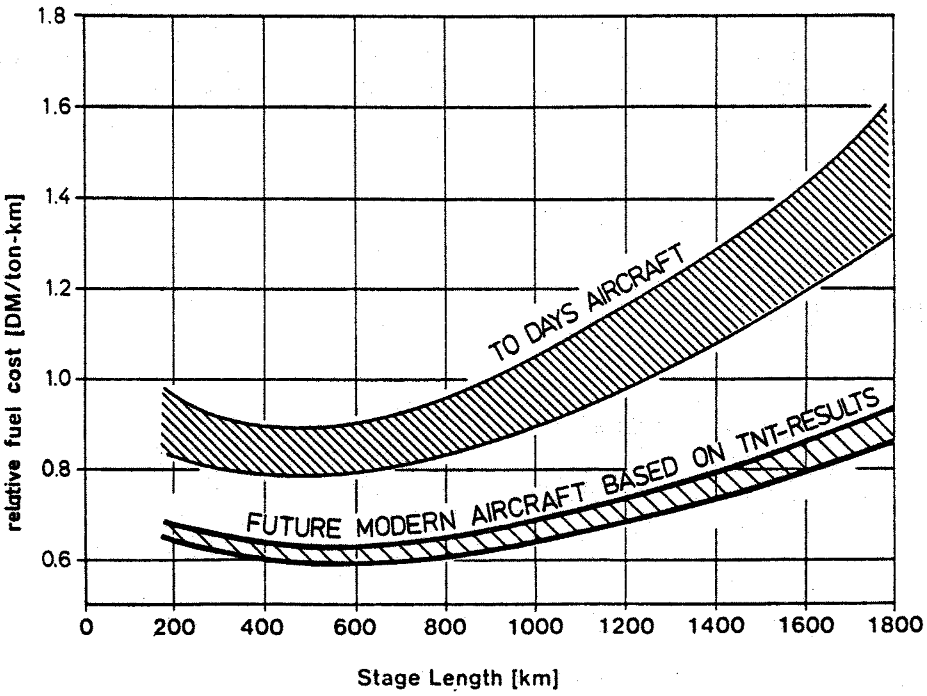
Comparison of Flight-Test with Prediction from Wind-Tunnel Test

FIGURE 18



IMPROVEMENT CLIMB/RANGE PERFORMANCE

FIGURE 19



IMPROVEMENT IN FUEL COST PER TON-KM

FIGURE 20