Walt Lounsbery

AIRCRAFT CONFIGURATION SYNTHESIS

1

by

John H. McMasters

Revised: December 1983

COURSE OBJECTIVES

Upon successful completion of this course the attendee:

WILL BE ABLE TO:

1. Appreciate the complexity of the process by which an aircraft (typical of Boeing's product line) configuration is derived.
2. Perform a first order airplane preliminary design study.
3. Understand the nature of the compromises and trade-offs involved in synthesizing an overall aircraft configuration.

WILL KNOW:

1. The fundamental elements of the process by which an aircraft configuration is synthesized.
2. The present state-of-the-art in aircraft design methodology.
3. A set of simple procedures for preparing an initial first order aircraft preliminary design study.

COURSE EVALUATION:

The course will be evaluated by conference with attendee's and line organization management after course completion.

Aircraft Configuration Synthesis I

Winter 1983

J. H. McMasters & J. C. Baer

Aircraft Configuration Synthesis - Bibliography (Pt. I)

Design Books & Reference Material.

1. Torenbeek, E., Synthesis of Subsonic Airplane Design, Delft Univ. Press, 1975.
2. Nicolai, L.M., Fundamentals of Aircraft Design, U. of Dayton, 1975
3. Stinton, D., The Anatomy of an Aeroplane, American Elsevier Pub. Co., NY, 1966.
4. Corning, G., Supersonic and Subsonic Airplane Design, 1953 (later ed. avail.)
5. Wood, K.D., "Aircraft Design", Johnson Publishing Co., 1966.
6. Teichman, F.K., "Airplane Design Manual, Pitman, 1950
7. Dommasch, D.O., et.al, Airplane Aerodynamics, Pitman, 1951, later eds also.
8. Miller, R. & Sawers, D., The Technical Development of Modern Aviation", NY Praeger 1968.
9. Swihart, J.M. (ed) Jet Transport Design, AIAA Reprint Series #8, 1969

Jane's All the Worlds Aircraft (annual)

Aviation Week & Space Technology.

Flight International

Interavia

Aircraft Engineering

Journal of Aircraft

Soaring

Sport Aviation

I. INTRODUCTION (3 hours)

A. Course outline & objectives

B. The Design Process

C. The General Design Problem

1. Philosophical - "The Design Onion"

2. Technical

D. Types of Design - Historical & State-of-the Art

1. Cut & Try - Empirical

2. Design by Analysis

3. Computer Aided Design - Synthesis

E. Levels of Design

1. Conceptual Design

2. Preliminary Design

3. Detailed Design

4. Product Improvement & Support

F. A Design Vocabulary

1. Trade-offs - compromises

2. Iteration

3. Analysis Synthesis

4. Optimization

G. Toward A General Theory of Locomotion

H. Outline of the Feasible Flight Spectrum

I. General Discussion of Flight Vehicle Types

The Book of Genesis

~from~

The Aerospace System Designers Bible

By W. Gillette & J.H. McMasters

And on the first day there was gravity and the spirit of Newton said:

$$F = K \frac{m_1 m_2}{r^2}$$

and Matters became weighty.

And then there was boundless energy and it was consolidated and Einstein quoth:

$$E = m c^2$$

and there was Motion, but it was merely transverse.

And on the third day, from the heavens, a voice cried out:

$$C_L = \int_x \int_y (C_{p_L} - C_{p_U}) \, dx \, dy$$

and there was Lift.

But on the fourth day, the Devil said:

$$C_D = C_{D_{p_{min}}} + C_L^2 / \pi AR e_w + \Delta C_{D_p} + \Delta C_{D_M} + \Delta C_{D_{p_{bugs}}} +$$

$$+ (\Delta C_{D_{bouyancy}}) + C_L \sin \alpha_{upflow} + Q_3 \int (erf)^{nerf} dz +$$

$$+ \Delta C_{D_{trim}} - \tfrac{1}{2}(\text{management requirement}) + 2\sigma + \text{H.O.T.} + C$$

and there was Drag.

On the fifth day a tiny voice from the wilderness cried out:

"...don't forget stability and control."

And this was echoed by various multitudes crying:

"...environmental control systems, ground support equiptment, and etc."

far into the night of the sixth day.

And on the last day, the sprit of Maynard Keynes proclaimed:

"He who controls the purse strings control the policy."

and there was Economic Reality.

Caveat Emptor. Amen

Some Thoughts on Design Education

by

John H. McMasters

I have been a student, and occasional participant, in the great debate regarding "design education" for almost twenty years. In this time period I have been involved in design, first as an undergraduate, and then as a design engineer in industry, as a teacher of design at two universities, and most recently as the teacher of an airplane design course offered by The Boeing Company. Despite my present formal position as an engineer in the Aerodynamics Research Unit at Boeing, I consider myself primarily a teacher by temperament, interest and experience. In fact much of my activity at Boeing involves teaching under the guise of other activites.

The intent of design education in a university, its goals, and its limitations have been subjects of sometimes heated debate in every university I have had dealings with for the entire duration of my working life, and apparently we are little closer to resolution of the central practical questions involved than ever. Despite years of discussion, we still have few satisfactory practical answers to the questions:

1. What is engineering design and how should it be taught?
2. Who should/can teach design?

I maintain that these two questions have some fairly obvious answers when viewed from the perspective of how something is actually "designed" in a commercially viable sense, who actually does the "design" work, and what a potential designer needs to know. I now believe that much of the length of the academic debate regarding design is a consequence of a misunderstanding of the design process and the full range of real world (i.e., commercial) design activities, by individuals with too little actual experience as designers. We too often attempt to discuss design and instead wind up talking at cross purposes because we have never actually designed anything and thus trip over definitions. Endless debate about "what design is" also seems to serve the purpose of evading the necessity of confronting some disagreeable realities regarding who should be hired to "teach" design.

What Is Design

Design in the material world is the process by which ideas, tastes, prejudices, basic scientific principles and available resources are weighed and combined into a well defined plan for the eventual construction of an object or system. This object or system may be a rug, an airplane or a computer program. As a process there is no difference in the way the final plan for the construction of any of these items is derived. There are only differences in magnitude, detail and orderliness of the final description of the object which has been designed, the tools used in the process and the effort which went into and supported the basic process.

It is the basic design process and the techniques available for its conduct within a given discipline (i.e., aeronautics) which is to be taught in the university. It is at this point that we run into trouble. How do we provide design education of lasting value, rather than fall into the trap of providing merely training in techniques which are currently fashionable, but may become rapidly obsolete? Who is to do this educating? Can we actually teach students to become good designers?

This last question is fundamental. Anyone can be taught to play the violin, but relatively few have the basic talent to play one well. My experience has been that given any class of university engineering students, that class can be broken down in very nearly the following way: 25% should not be in the university (they would be happier over their lifetimes training for and working in an honest blue collar trade), 65% will become competent to excellent analysts or experimentalist, and 10% are misfits with a real talent for design. This last 10% cannot be made into designers, they already are. They require only encouragement and the same sort of cultivation (in design) as one might devote to the cultivation of roses (which already know how to grow, but can be easily killed if neglected). What these 10% of my students require is: education in physics, mathematics, economics, philosophy; and experience. They already possess ingenuity, creativity, curiosity; and tolerance to imprecision and "open-endedness" - attributes inherent in a good designer, and characteristics which cannot be "taught" anyway.

It is my group of 65% student analysts/expermentalists who are the real constituents for courses in (about) design. Not because they can somehow be magically transformed into competent designers (they likely cannot be) but rather because they are the ones who will make up the great supporting army of soldiers who must work with and support those few who actually design something. Whether these people ever actually design anything or not is irrelevant. They serve a vital function in the design process and must have a thorough understanding of what design is, how many disciplines must interact effectively to produce a viable design even if it is "merely" a door knob; and will eventually supervise, assess and manage the work of subsequent designers.

The Design Education Debate

The above view of my typical class of engineering students may appear cynical, but I have elicited grudging verification of it from a number of faculty members at various universities, many of whom have far more experience than I have. Individual disagreements with portions of my assessment should not, however, alter the validity of my subsequent comments.

The majority of students who graduate with a degree in engineering will: go to work in industry, work for the government/military, or drop out of engineering altogether. It is that group which will either work as engineers in industry or will use or manage the work of engineers (e.g. the military pilot, the FAA employee, the government contract monitor) whom I am primarily interested in and qualified to discuss.

As a design educator, there are several messages I wish to deliver to these individuals. The first of these is contained in what I describe as the "Design Onion" (Figure 1). It is square in this diagram, and sometimes obscure in practice. Assuming a student intends to stay in engineering and work in the aerospace industry, his initial job will likely be in some small sector of one of the broad technical areas at the core of the onion. His job, however, is not independent of a very wide range of influences, over many of which he may have little or no control. He <u>must</u>, however, have a good appreciation of what all these influences (spread in wider and more general layers of the onion) are and how they affect his work. It comes as an often rude shock to the neophyte designer of the perfect wing (in an aerodynamic sense) that this wing can only be constructed of unobtainium, can only be manufactured on a machine that does not exist, or looks so "strange" to the customer that it is unacceptable. His work becomes a great exercise in futility, he is a failure, and I would fault not him, but rather his education. The content of the whole design onion must be understood, and preferably at the freshman or sophomore, rather than the graduate level.

The second message of importance is that within the context of design in industry there are at least four distinct levels of design effort and activity. Herein lies much of the difficulty in discussing design <u>in general</u>.

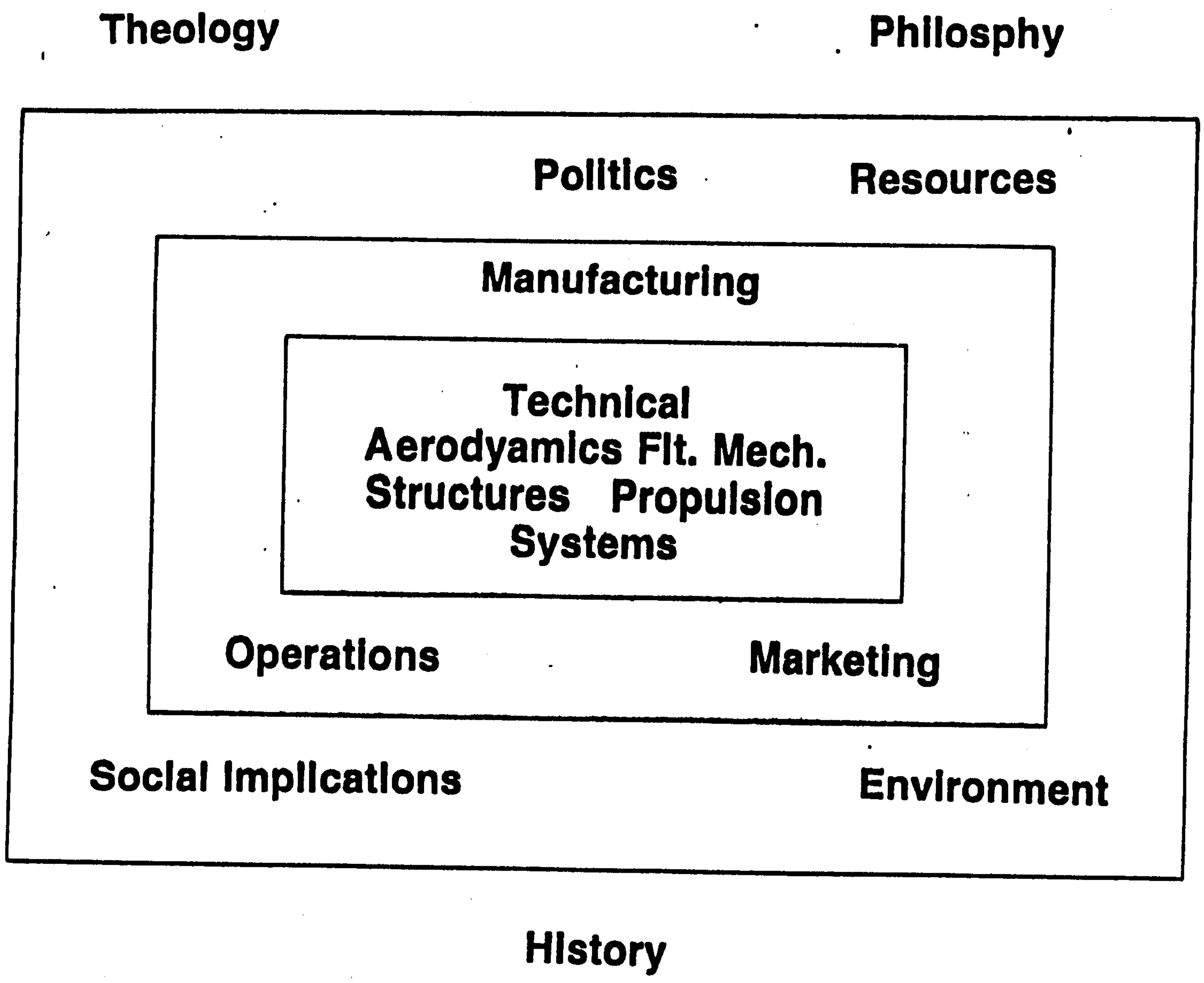

Figure 1 The Design Onion

Each of these levels of design activity (Figure 2) require different levels of effort, uses different tools; and within each, different levels of precision are acceptable. A goal of design education should be to make these different levels clear - all of them, not simply the normal "preliminary design" level taught in most university curricula in aeronautical engineering. It should be noted that for the near- and mid-future, that tool termed "computer aided design" (wherein an entire airplane system is to be modeled, optimized and graphically displayed) is applicable mainly to the preliminary design level. Conceptual design is best done with a slide rule. Detailed (project level) design takes its analytic methods and tools ultimately as extensions of these fundamentals presented in books like Schlichting and Kuethe & Chow and the basic courses in which such material is taught. At this level of detail design, large systems programs cannot be successful without many decades of further development - which may ultimately prove to be a good thing.

Again, the student (preferably freshmen and sophomores) need to see this whole picture. Very few of them will ever work in preliminary design, but they must know what it is and what one does when doing it. Conceptual design is done by very few - professionally. Project level (detail) design is "where it's at" for most "design" engineers, either as doers or, later, as supervisors. Computer aided design has a major place in this picture, but as a practical matter, it is less important to me as a designer of real airplanes than would be a good three-dimensional viscous flow wing analysis scheme, <u>coupled with</u> the basic education in fluid physics necessary to use it intelligently.

With so little time available in the engineering curriculum to teach courses specifically labeled "design," it is essential that priorities be set on what to teach in these courses. I believe it is wrong to spend much design course time having the students develop even simple airplane <u>system</u> design

Figure 2. Levels of Design

Design Level	Objective	Tools Used	Analysis Accuracy Required
I. Conceptual Design	To explore tentative, general possibilities.	Slide rule, envelope backs.	± 10 - 15%
II. Preliminary Design	To define and assess the main parameters of a new system.	Computer based parametric analysis. Computer graphics. First order theory.	± 10%
III. Detailed (Project) Design	To produce a documented set of performance predictions and detailed working drawing for the construction of a prototype	Experimental apparatus (wind tunnel, etc.). Detailed analysis computer programs.	± 2 - 5%
IV. Product Development & Support	To correct defects which came to light during operation. Explore "growth versions."	Whatever works from the above list.	± 2 - 5%

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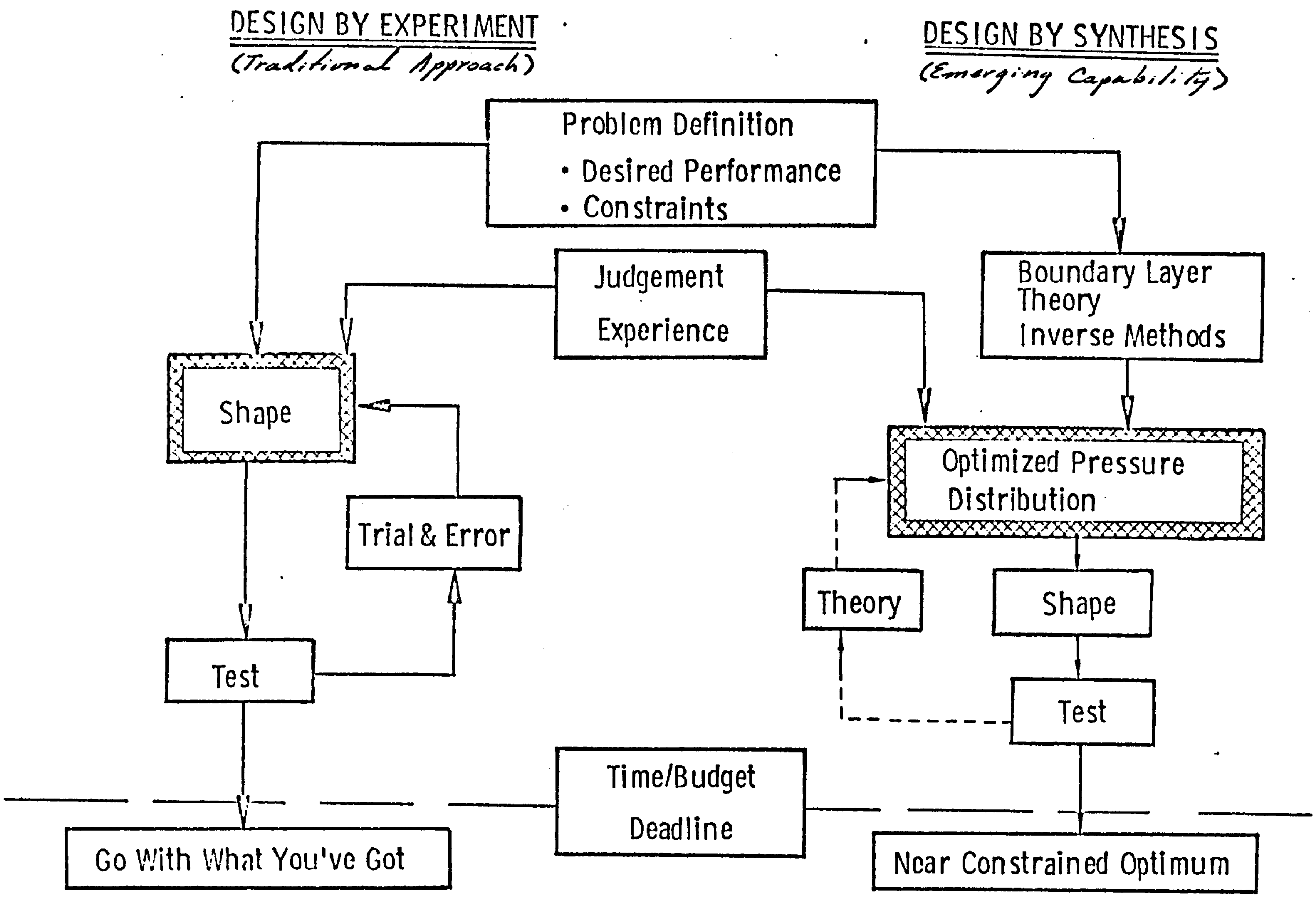
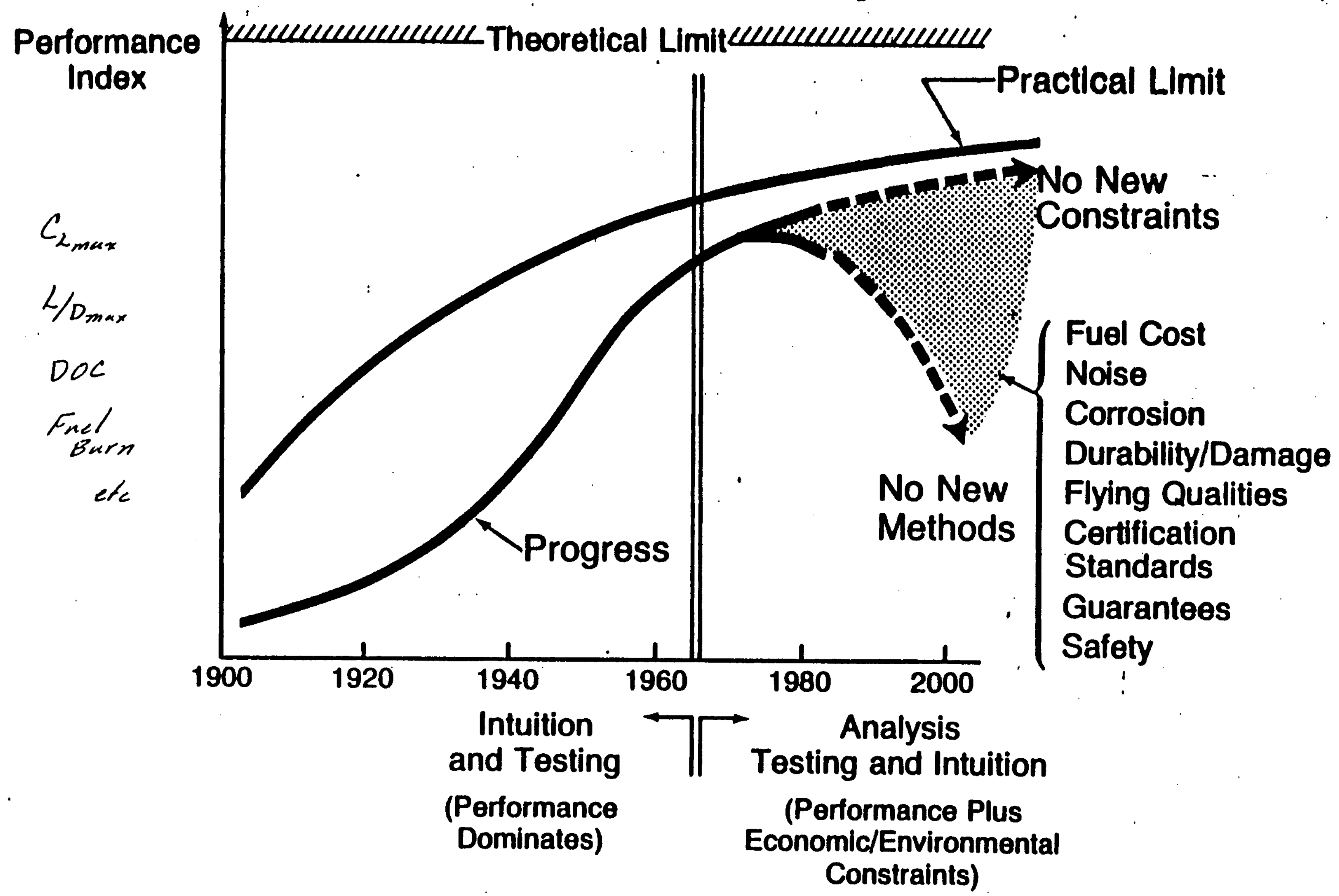
computer programs when many lack a basic appreciation for the design process in the first place. "Black box" programs of this sort merely handed to the student are also pedagogically worthless. I think the correct approach is to have the student stumble through the design process manually the first time. If he has the wits, talent and the necessary computer background, he can mechanize the process later. But not the first time (or likely the second). By extension, I am easily persuaded (in fantasy) that no one should be sold a pocket calculator prior to showing certified proof of proficiency with a slide rule. Once the university provides the education necessary to allow a student to make an intelligent decision, it is a relatively simple matter to train the student in the use of the tools to accomplish the task - and that training can be done in industry. An ignorant student armed with too powerful a tool before he knows what to do with it and why is a potential vandal.

The final message has to do with the state-of-the-art with respect to both the airplane design problem and the industry approach to the solution of the problem. As shown on the left side of Figure 3, past aerodynamic design has been conducted very largely by cut-and-try testing. (Whether this testing is done in the wind tunnel or on the computer, the process is the same.) As computational methods improve, and particularly as true "inverse" methods are developed, we can hope to see more design conducted by the so called "synthesis" process. At least we may pray for this possibility, because if such new methods do not materialize we will be confronted with the situation shown in Figure 4. As aerospace technology continues to mature, design in the future must be conducted by the best combinations of analytic methods (direct and inverse), experimentation, intuition and experience.

Fig. 3 AERODYNAMIC DESIGN PROCEDURES



Technology Development



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The Undergraduate Design Curriculum

The preceding comments can be distilled into a recipe for engineering education which contains a strong component of design. What is basically wanted is a student with a strong background in engineering fundamentals including physics, mathematics, computer science and graphics, English composition, and oral presentation. In addition, he should have some exposure to the elements (economics, history, etc.) which are involved in the whole design onion shown in Figure 1.

With this basic background, we have the raw material necessary to educate a student in design. To do this we likely need three design courses:

1. During the sophomore year a student should be given a survey course covering all the principle aspects of aeronautical engineering including design. He should be told how, in a design sense, all his subsequent courses interlock to provide him a basic engineering education. This course should also emphasize hardware - what is an airplane, what kinds of airplanes are there, what is the state-of-the-art in aeronautical technology. This course would thus provide him with motivation and give him a map or context within which to place the courses he will subsequently take.

2. During the senior year a classic course in aircraft preliminary design should be given - philosophically connected to the course above which he took when he started as a sophomore. Thus we provide a summary of this undergraduate education at the beginning and again at the end.

3. A course in "theoretical design methods" including optimization theory, computer based design methods, system analysis methods and so on, should be offered. These are the new and emerging tools of the future designer.

Who Should Teach Design?

Some one with professional experience as a designer should teach at least the first two courses listed. He/she should understand design, know airplanes and enjoy teaching undergraduates. He/she should have both experience as a designer, and an appreciation of emerging design methods and trends. A mere "history" teacher is not sufficient.

These people are rare, and even rarer are those who are willing to give up an industrial career for one in teaching. A good candidate will likely not have the full range of academic credentials normally associated with a university faculty candidate. He may have little interest in publishing or grant chasing. However, if a university is seriously interested in quality design education, as I believe all should be, there seems to be little choice. Relaxing employment standards in a single case, in exchange for attracting a talented designer who can teach a very difficult but important topic with enthusiasm and expertise seems a not unreasonable proposition.

Conclusions

From the preceding discussion it is possible to draw some general conclusions regarding design education within the overall engineering curriculum. Among these conclusions are:

1. Technical hardware design is not an end in itself. A new "thing" has meaning and utility only within the broader context of an operational, manufacturing, marketing, economic, social and political environment. Engineering students should be made fully aware of this and educated in the elements of all of these disciplines.

2. Intuition, curiosity and experience cannot be taught. These are, however, the basic attributes of a "designer." To be able to work effectively he must be armed with the best tools available and these can be taught. The student must know what these tools are and be given some instruction in how to use them correctly.

3. Relatively few engineers will become really innovative designers. However, all engineers must deal with design and work with those few who are innovators. Thus design education is an essential part of any engineering cirriculum.

4. There are several levels of design activity (c.f. Figure 2). Those levels which are normally "taught" in a "design" course (i.e., conceptual and preliminary design) are not those in which most engineers will work. Further, the tools used at each level are different in kind and application. At the project (detailed) design level where most engineers will work, analysis methods and experimentation will be the primary tools for at least the remainder of this century.

5. The value of courses in "preliminary design" are:

    a. They can be highly motivational.

b. They can be used to display the basic design process at a level an inexperienced student can grasp.

c. These courses are usually the only place in the engineering cirriculum where an attempt is made to show the interrelation of the many disciplines which must be combined to produce a viable piece of technological hardware.

6. The basic design process (with all its non-linearity and irrationality), together with the necessary synthesis of many disciplines to create a viable hardware item, are the two things that must be taught to all engineering students. Education in these topics is, in my view, the basic goal of design education. Everything else is technique.

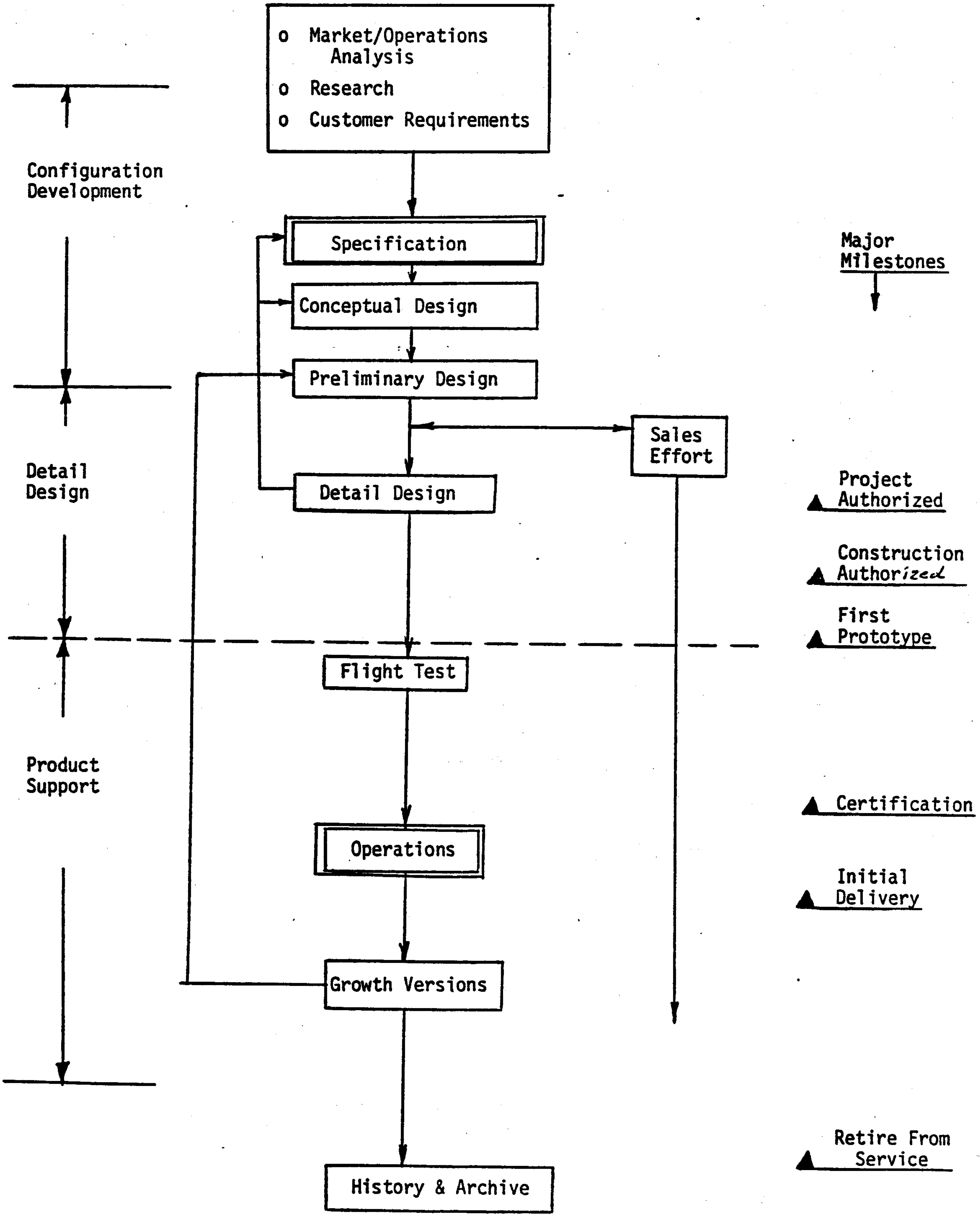
7. Because of its display of the interrelation of individual disciplines, individually taught in detail in the remainder of the engineering curriculum, and because of its motivational value (it gives a student on overall rationale for his engineering studies), design should be taught both at the beginning (freshmen/sophomore) and at the end of the undergraduate curriculum.

8. Offering design courses alone is not sufficient. Courses in engineering fundamentals (fluid mechanics, structures, propulsion, etc.) should also include a "design prespective." This is rather simply done by assigning several problems for which no closed form (text book) solutions exist. Thus the student learns that the final boundary condition in the formulation of his problem is judgement, and the world is not as neat and orderly as a mathematician might prefer.

9. Design courses should be taught by individuals with design experience, and who have a thorough understanding of the global view of the technological development process which is design. Candidates for these positions are regretably rare, and a different set of academic standards necessary for their employment may be required.

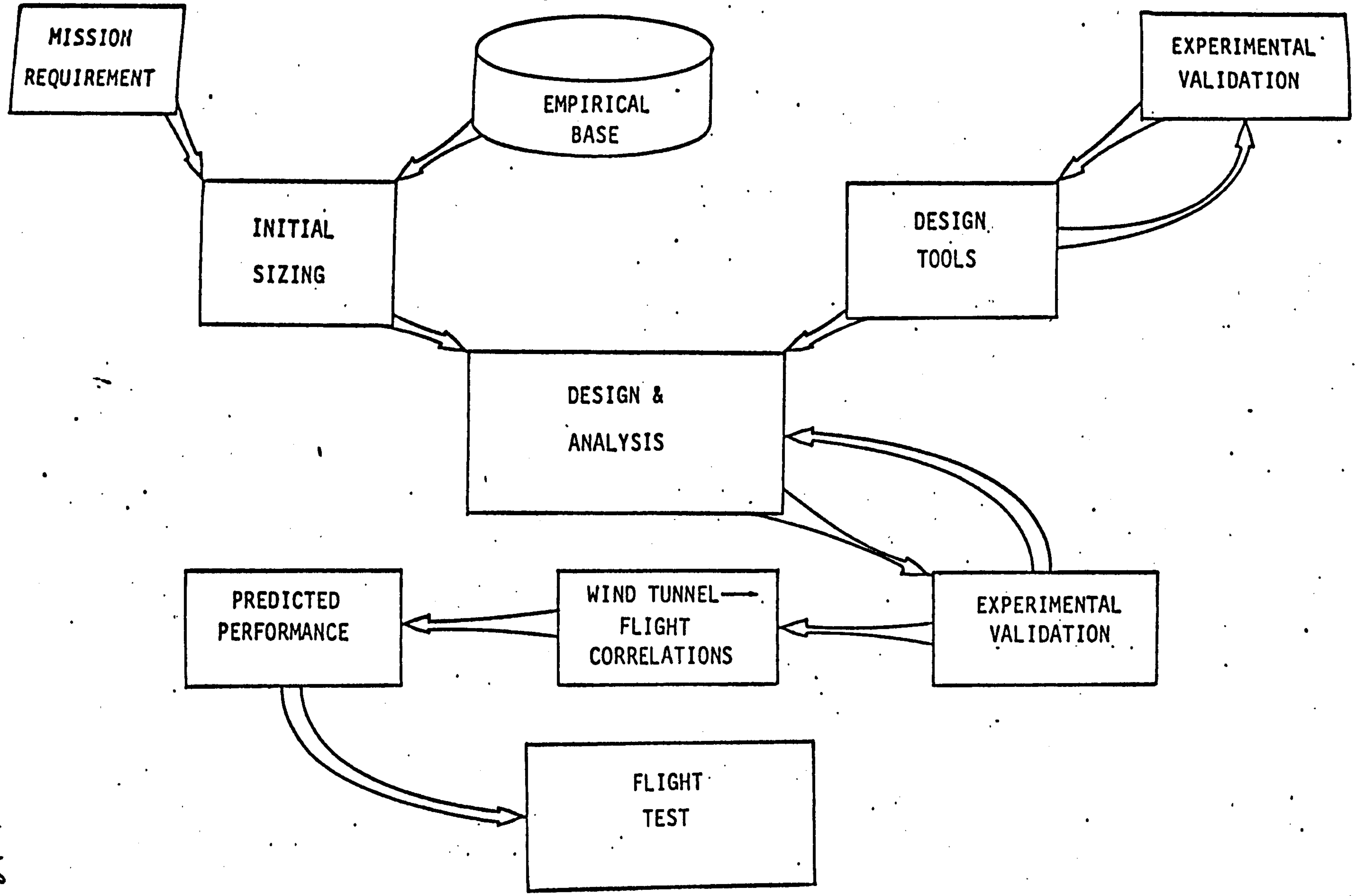
All of these conclusions are rather general, but can easily be particularized to the education of aeronautical engineers. I cannot understand why anyone would become an aeronautical engineer unless he was fundamentally interested in the airplane as an item of hardware reality. To design (or analyze) a modern airplane requires a profound knowledge of engineering fundamentals including mathematics, physics and computer science. However, without grounding these sometimes esoteric fundamentals in the hardware reality of the airplane iteslf, we seem too often to lose the student whom we are trying to educate. Design courses, taught by an individual who knows and is fascinated by airplanes, is the single opportunity most students have for making the connection between science and the art of aeronautics.

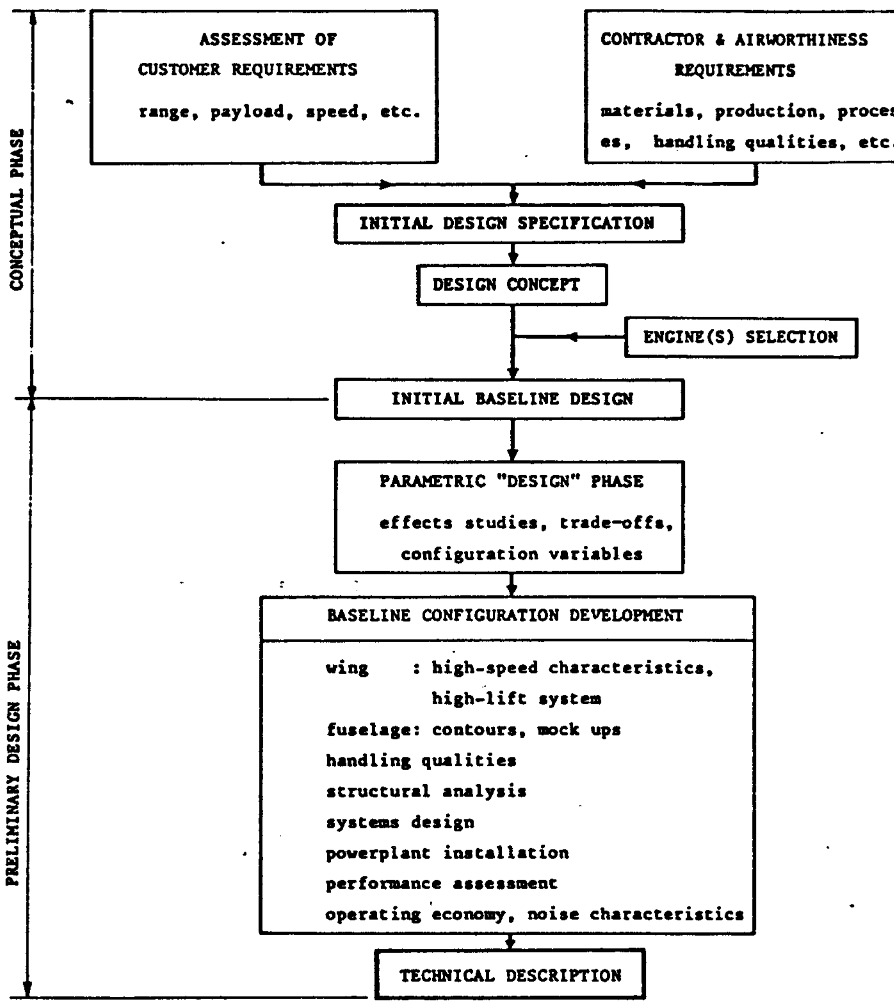
COMMERCIAL AIRPLANE DEVELOPMENT



1.18

OUTLINE OF AIRPLANE DESIGN PROCESS





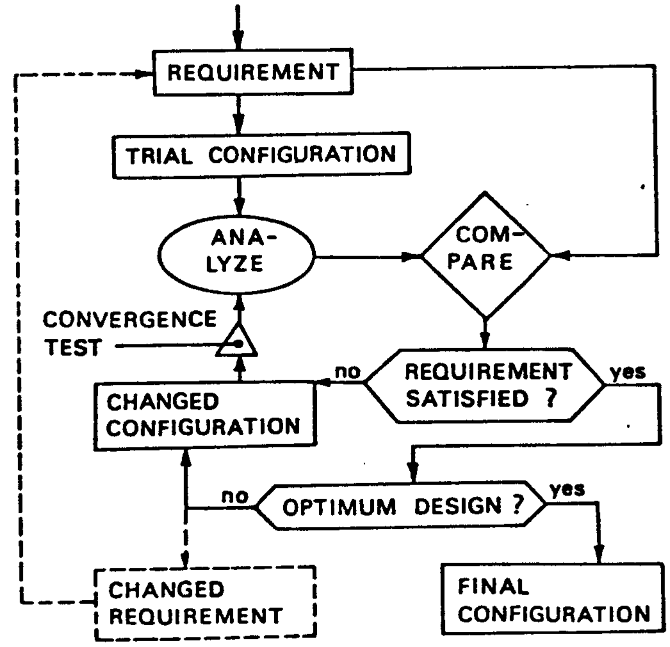


Fig. 1-7. General design procedure

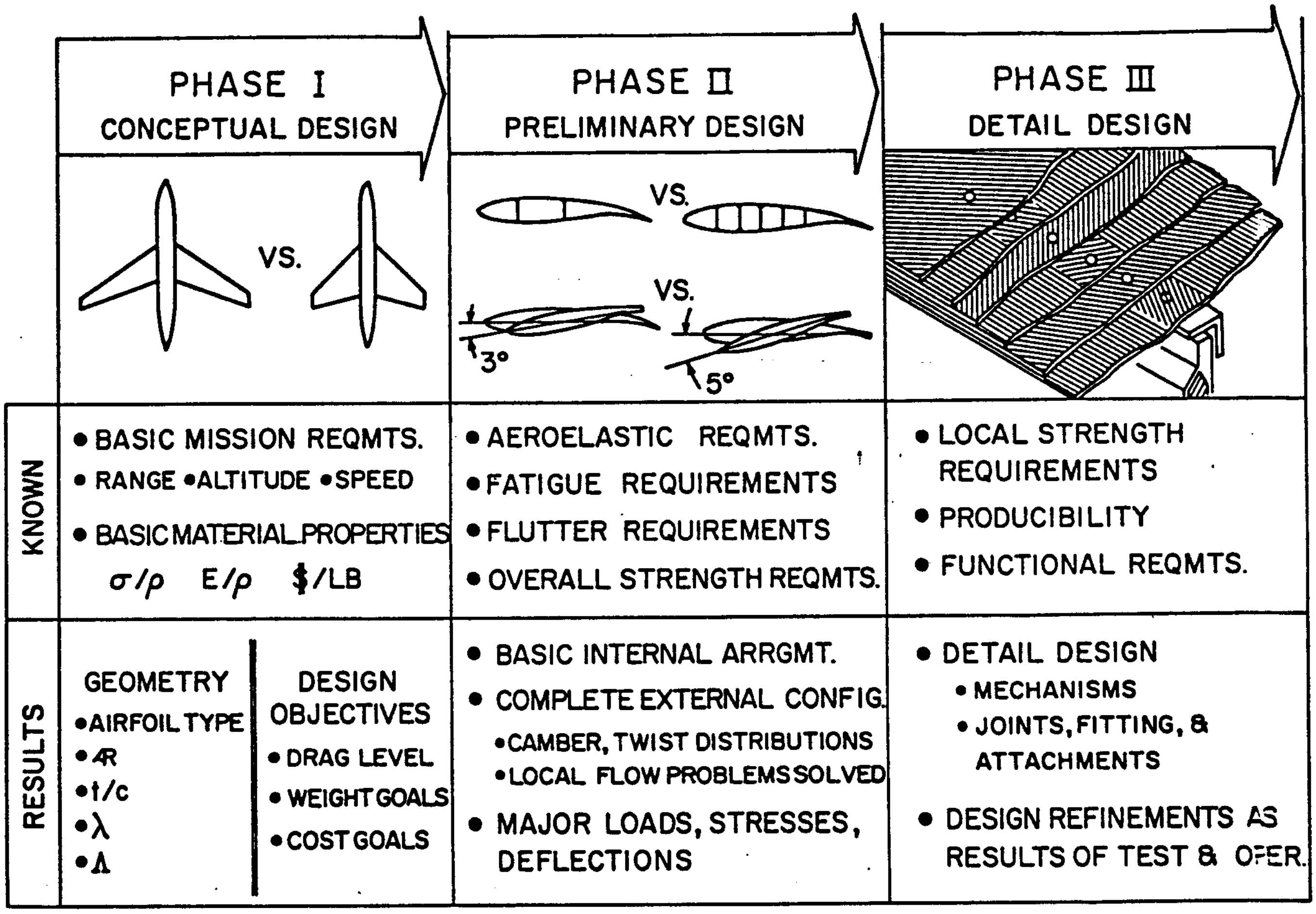
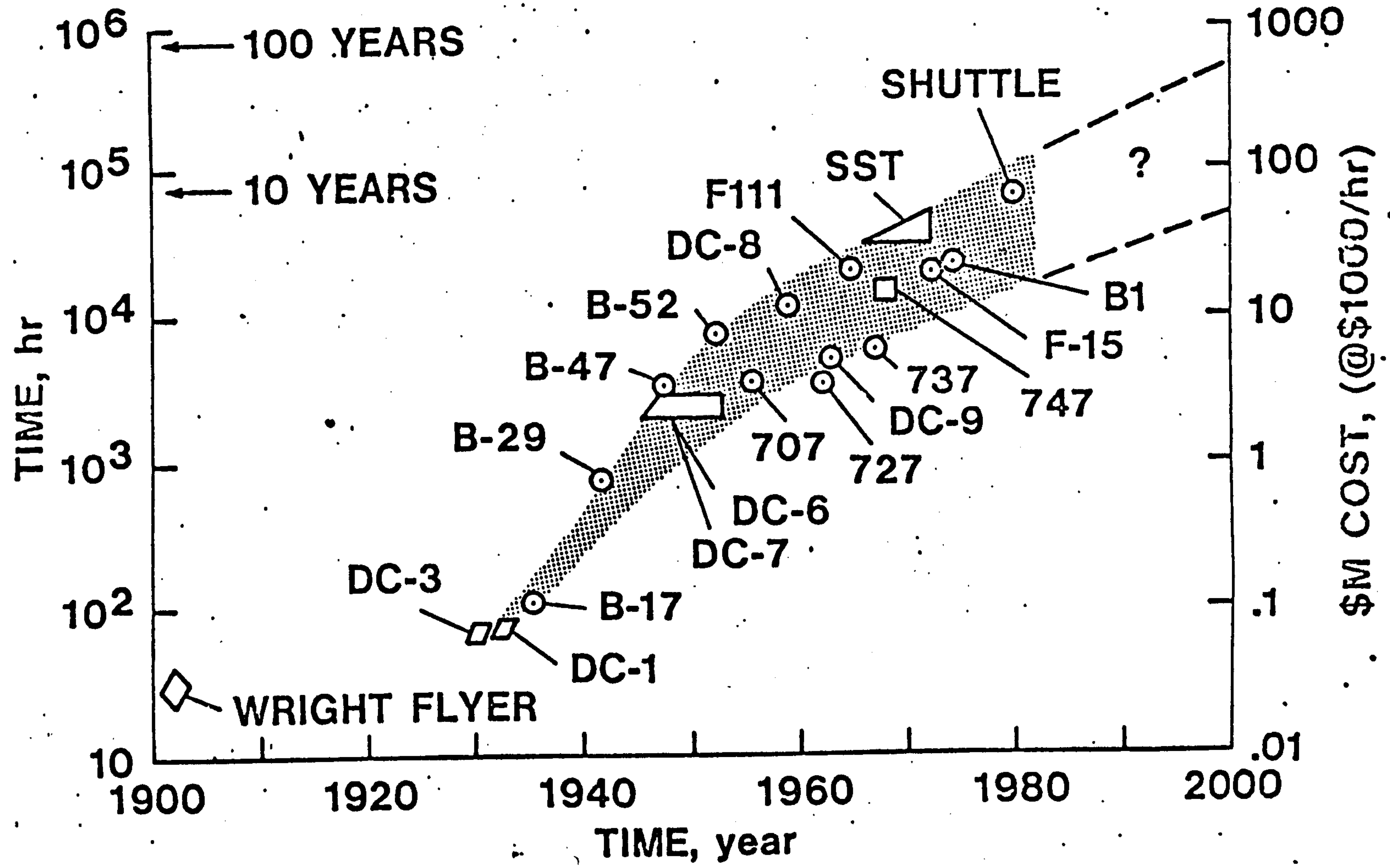


Fig. 1.4 The Three Phases or Levels of Aircraft Design (Reference 4)

TOTAL WIND TUNNEL TEST HOURS FOR DEVELOPMENT OF VARIOUS AIRCRAFT



Stimulating Change

1.22

WIND TUNNEL PROGRAMS THRU FIRST FLIGHT

WIND TUNNEL OCCUPANCY HOURS	727	737	747
HIGH SPEED	1803	3818	7070
LOW SPEED	2500	2563	7279
	(2.2) MY	(3.2) MY	(7.2) MY
W.T. ENGINEERING MAN HOURS	90,363 (49) MY	91,090 (49) MY	229,639 (124) MY
W.T. SHOP MAN HOURS	240,968 (130) MY	197,885 (107) MY	529,039 (286) MY

Excludes Engineering Design of Configurations to be Tested.

FIGURE 1

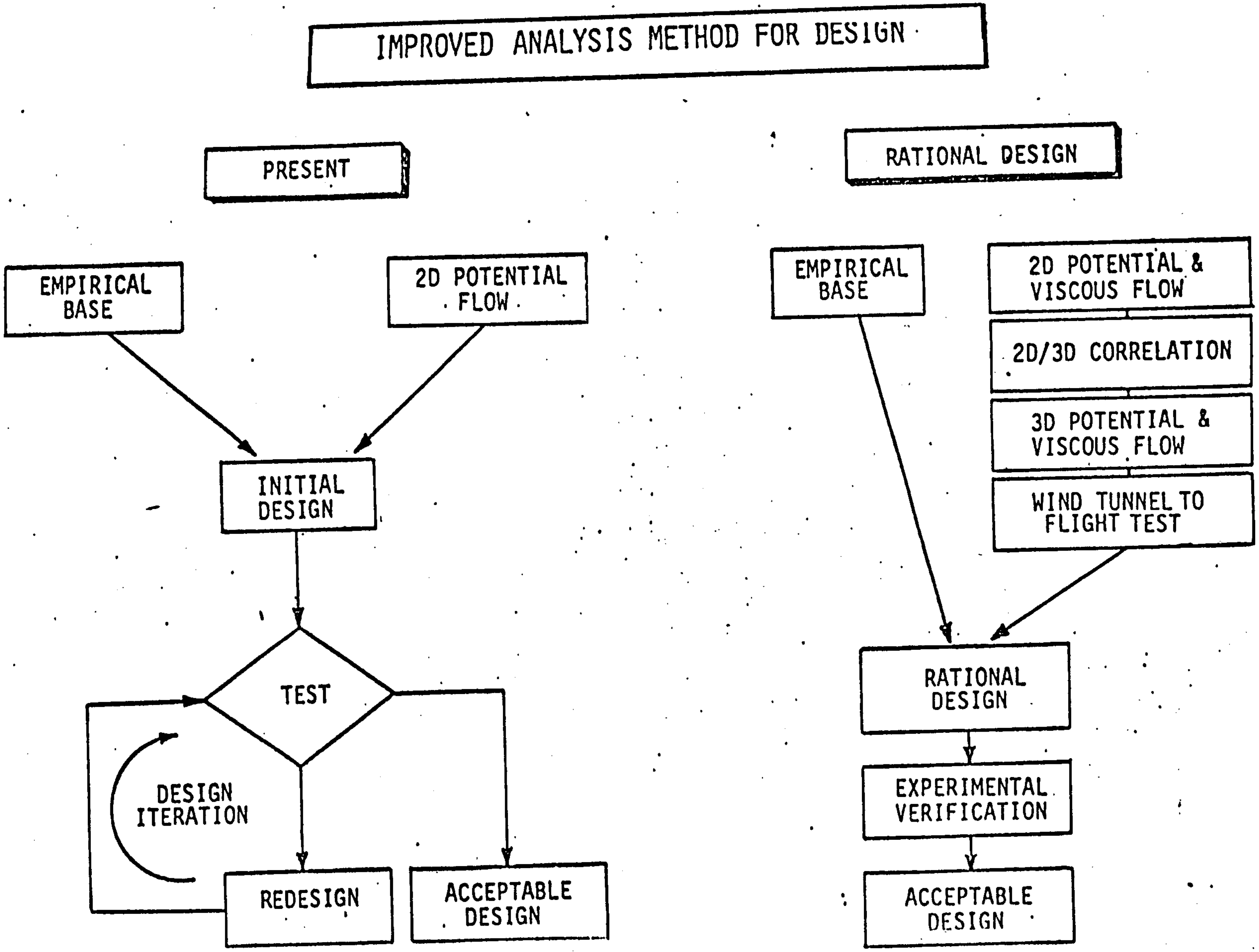
PG. 14

GOALS OF COMPUTATIONAL AERODYNAMICS

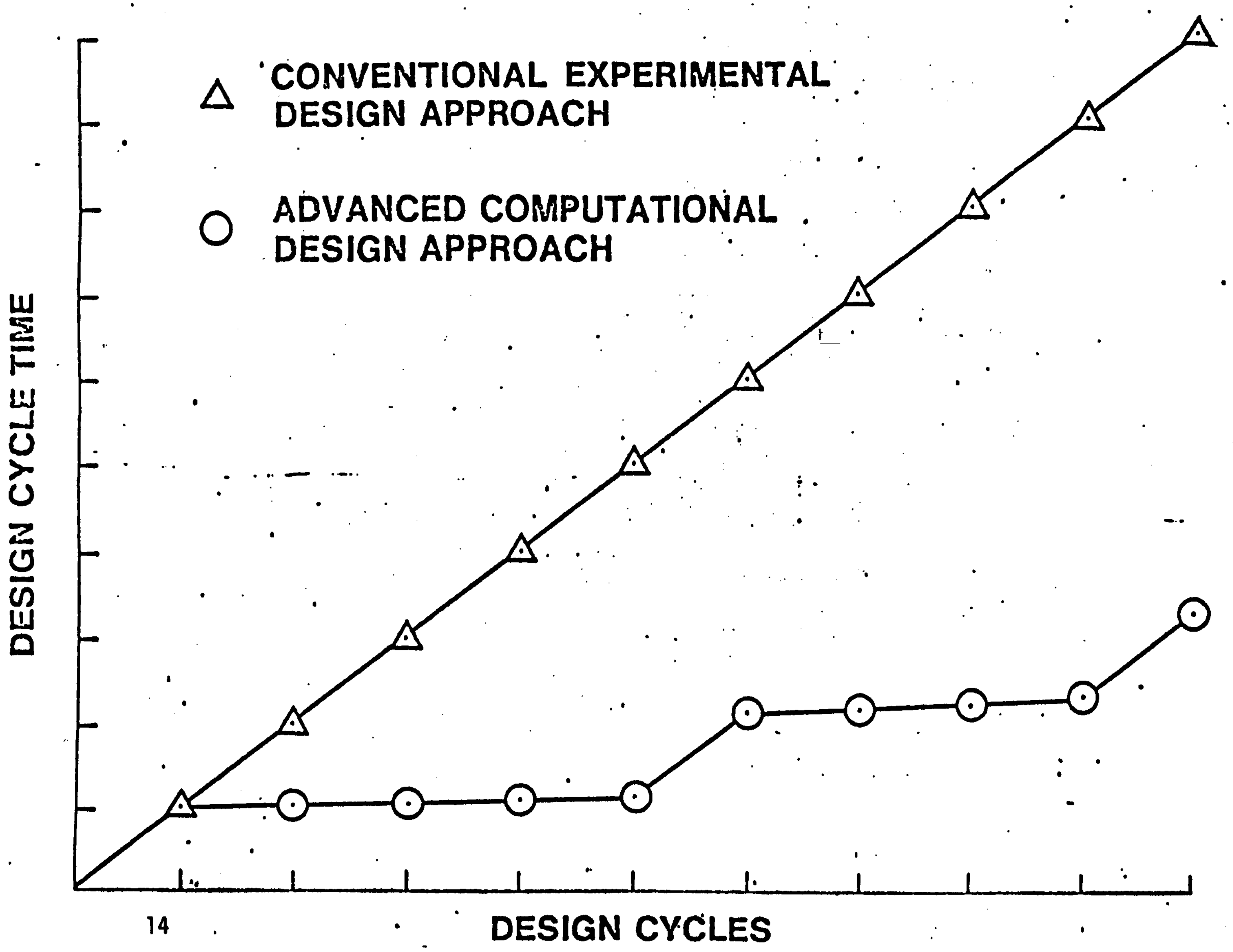
- RAPID AND INEXPENSIVE SIMULATION OF FLOW PHENOMENON THAT HAVE A LARGE EFFECT ON DESIGN
- DETAILED FLOW FIELD DESCRIPTIONS
- SIMULATION OF FLOWS THAT ARE EXPERIMENTALLY IMPRACTICAL OR IMPOSSIBLE
- RAPID CONFIGURATION OPTIMIZATION

⇒

- REDUCED DESIGN COST
- REDUCED DESIGN TIME
- REDUCED DESIGN RISK
- IMPROVED DESIGN


IMPROVED ANALYSIS METHOD FOR DESIGN
PRESENT
EMPIRICAL BASE
2D POTENTIAL FLOW
INITIAL DESIGN
TEST
DESIGN ITERATION
REDESIGN
ACCEPTABLE DESIGN
RATIONAL DESIGN
EMPIRICAL BASE
2D POTENTIAL & VISCOUS FLOW
2D/3D CORRELATION
3D POTENTIAL & VISCOUS FLOW
WIND TUNNEL TO FLIGHT TEST
RATIONAL DESIGN
EXPERIMENTAL VERIFICATION
ACCEPTABLE DESIGN

AERODYNAMIC DESIGN CYCLE APPROACHES



AIAA

PACIFIC NORTHWEST SECTION
AMERICAN INSTITUTE OF AERONAUTICS AND ASTRONAUTICS

NOONTIME TECHNICAL LECTURES

OFF-THE-WALL AERODYNAMICS

BY

JOHN MCMASTERS

CAN THE BUMBLE BEE FLY? HOW BIG CAN A FLYING ANIMAL BE? WHY? WHAT ARE THE TRUE ORGINS OF THE HANG GLIDER? WHAT AERODYNAMIC FEATURES MIGHT A MODERN WINDMILL ROTOR SHARE WITH A BIRD'S WING? THESE AND OTHER TOPICS IN NONCOMMERCIAL AVIATION THAT YOU'LL NEVER READ ABOUT IN THE BOEING NEWS WILL BE THE SUBJECT OF THIS LIGHTLY TECHNICAL PRESENTATION.

RENTON 10.80 THEATER	TUESDAY, JANUARY 11
D.C. 9.101 R-7 THEATER	WEDNESDAY, JANUARY 12
KENT 18.05 IUS MIC ROOM	THURSDAY, JANUARY 13
PLANT 2 2.25 THEATER	THURSDAY, JANUARY 20
EVERETT 40.86 ENG. CONTROL ROOM, 11D11, BAY D11	TUESDAY, JANUARY 18

BEHOLD:
REVISED DATE

TIME 12:00 TO 12:40

BRING YOUR LUNCH

EVERYONE WELCOME

1.27

OFF-THE-WALL AERODYNAMICS

"TO PROVE THAT PIGS CANNOT FLY

IS NOT TO DEVISE A MACHINE THAT WILL"

D. KÜCHEMANN

THE AVOIDANCE OF STRICT ACCURACY IS

A GENERAL SIGN OF GOOD BREEDING

SOCRATES (470-399 BC)

OFF-THE-WALL AERONAUTICS
(A Work in Progress)

by

John H. McMasters
Aerodynamics Staff
Boeing Commercial Airplane Company
Seattle, Washington

April 1983

"To prove that a pig cannot fly is not to devise a machine that does so."

D. Küchemann (1911-1976)

Introduction

This paper is the outgrowth of a talk given at various locations in the Seattle area during the winter and spring of 1983 as one of the Noontime Technical Lectures sponsored by the Pacific Northwest Section of the American Institute of Aeronautics and Astronautics. Eight presentations of the lecture drew a total audience of 500 and has encouraged the author to prepare an expanded written text of the lecture material.

The lecture presented reflects the author's long time professional and avocational interests in "everything that flies," and draws upon information collected and lessons learned in the course of a number of diverse activities. These include experience both as a university student and professor, as an industrial designer (of light weight expedition tents and backpacking equipment), as a research aerodynamicist, as an enthusiast of soaring and human powered flight; and finally as the instructor of a course in "aircraft configuration synthesis" offered for the past five years by the Training Unit of the Boeing Commercial Airplane Company.

There were several motivations for preparing the original lecture which this paper documents. Having served for several years as a program coordinator and participant in the Noon Lecture Series, the author is a firm advocate of the value of the program. When the present program chairman, Howard Rush (a former student of the author's at Purdue University) asked for yet another presentation - offering carte blanche on the topic - the request was not to be refused.

The topic for this lecture occurred to the author because of his belief that periodically it is valuable to stand very far back from a topic in the details of which one is deeply immersed on a daily professional basis and attempt to see a whole picture - to see ones work in full perspective. The effort can be immensely refreshing - and humbling.

Closely connected to this first thought, is the author's conviction, based on years of experience in engineering education, that most university aeronautical engineering deparments miss the boat very badly in laying out the entirety of the rich tapestry of all the things that fly - whether of direct current commercial value or not. In the author's rather curmudgeonly opinion too much time is spent in training student in the details and techniques of engineering science - before a properly broad framework, and education in first principles, has been provided. All too frequently such a broad framework (which allows a student to see clearly the connection between required courses in differential equations, fluid mechanics, structural analysis, and what you do with them) is never adequately presented. If one considers a plot, for example Figure 1, which shows a standard index like Reynolds number versus flight speed for a wide variety of both natural and man-made flying devices, and then compares the contents of any standard university text book on aerodynamics, one sees immediately that such books and the curricula upon which they are based, hardly begin to address the full topic of flight. What configuration is appropriate to which region of the total flight spectrum Why don't dragonflies look like sea gulls, and why do neither look like the range of modern transport aircraft? Most universities don't teach that sort of thing.

There is, of course, little obvious commercial potential in designing better bumble bees and hence no prospect in making a living by doing so. It is the author's contention, however, that understanding the underlying principles of the flight of everything that flies, the similarities and contrasts, and the order and pattern in the full spectrum can be of inestimable value in making one a better designer of these vehicles which do have commercial value.

From a modern bionic point of view, one does not (in general) extrapolate the hardware solution nature has derived, but rather one seeks the underlying connections between the operational problem which the natural device solves and the physical basis for that solution. With these first principles understood, one can then seek to extrapolate the ideas involved to a human scale device - and seldom will an exact mechanical replica of the natural solution be the proper solution to the human problem. Thus winglets, derived from the pinion feathers on a bird's wing, are not necessarily appropriate on the wing of a business jet. Regardless of this, the study of natural models of flying devices continues to offer a very rich source of inspiration to the would-be aircraft designer. While a paper such as this one can hardly present a comprehensive overview of the topic, it is hoped that it can bring to the attention of some a very fruitful area of further inquiry.

A final motivation for this presentation has been the steady encounter in the semi-literate popular press of mindlessly stupid jibes at technology typified by the giggled observation "Didn't the Aerodynamicists (whoever he was) prove that bumble bees can't fly?" As will be described later, bumble bees fly very well, thank you, and, no, they do not violate the laws of themodynamics in doing so. Likewise, one would like once and for all to lay to rest the tired old anti-bureacratic saw: "A camel is a horse designed by committee." All airplanes beyond the complexity of a hang glider are of necessity designed "by a committee," and we can only damn this process by ignoring the obvious unique and marvelous advantages of a camel over a horse under certain operational conditions.

With this accumulation of thoughts, and burrs under his saddle, the author set about putting together the following pastiche of "first principle" considerations and specific examples in an attempt to both amuse and amaze, and ultimately stimulate some thought among, those seeking some light entertainment during lunch on a cold winter day.

Towards a General Theory of Optimal Locomotion

Once upon a time, with the inspiration and enthusiasm which seems to come only to neophyte graduate students, the author began the study of the general topic of locomotion (and its optimization) covering the entire range of both natural and man-made devices traveling through the air, on land, and in or on water. It was hoped that eventually certain semiempirical relations could be derived between speed, size, shape and energy consumption of a variety of locomotion schemes based on the laws of resistance to motion, sustaining and propulsive force generation, and structural mechanics and material properties.

It rapidly became apparent, however, that such a comprehensive study could be a ridiculously complex task, requiring a thorough knowledge of a wide range of engineering disciplines, the biological sciences, and the state-of-the-art in the vehicular technologies involved. The topic remained fascinating however, and the thought occurred that by considering the problem in the simplest possible fundamental terms, it might still be possible in a reasonable time to find means of rationally comparing the relative "effectiveness" of "transportation" devices as dissimilar as horses, submarines, sea gulls and airplanes. Having thus established some sort of index or indicies for comparing such devices*, it might then be possible to use them to study the size, weight and shape parameters which might lead to optimal values of the indices for a given class of transportation device (e.g. low speed flying machines).

Let us proceed with such an inquiry by defining the <u>purpose</u> of any system capable of locomotion to be to transport some "load" from one point in space to another. Such a "load" may be only the device itself (e.g. an animal in search of food) or the device plus its "payload" in the case of a commercial vehicle such as a truck. One may further hypothesize that to be viable, the system must operate "economically" whatever that may mean. The object now is to derive quantitative indicies which describe this transportation process, and a very simple way to do this is to use dimensional analysis.

Assume that at least the technological portion of the transportation problem can be described by some combination(s) of the following quantities (expressible in turn by the fundamental units of mass[m], length[l] and time[t]):

* It should be noted here that the author takes a mechanistic view of the biological systems involved in such a study. This is done without apology to biologists who sometimes find such notions offensive. From an engineering rather than a theological or metaphysical point of view, it need only be observed that at the level of the present discussion, all things that move appear to obey the same laws of physics and chemistry and are thus bound by the same constraints. Bones, steel, fiberglass and feathers all have corresponding measureable physical properties. Questions of free will and the process of deciding <u>which</u> direction to move do not arise. Further, the approach in no way denies the possibility of the existence of God.

W	-	The total operating weight of the system. This is also one measure of "size."	$[W] = ml/t^2$
U	-	The "useful" load to be transported	$[U] = ml/t^2$
P	-	The power required to move the system	$[P] = ml^2/t^3$
V	-	The speed at which the system travels	$[V] = l/t$
R	-	The distance to be traveled	$[R] = l$

So far each of these quantities has a clear intuitive meaning in the context of the transportation problem. The set is incomplete, however, because it does not contain a quantity explictly describing the economics of the transportation process. One can easily argue that economics are included implicitly in these basic parameters. For example, it is known that there is a strong correlation between a vehicle's cost and its weight. As the author's strength of materials professor used to say (at least once a day): "You buy steel by the pound." Whatever you do to the steel is making the product, the cost is still proportional to the weight of the basic raw material. Likewise, "time is money" goes the cliche and if one is transporting perishable goods the combination of speed and distance traveled yields time which may be convertable to profit or loss in dollars. And for a given propulsion scheme the power required is proportional to the fuel consumed and the fuel cost can be computed directly in dollars. All of this is far too circuitous however, and to complete the set, let us select an easily quantifiable economic parameter - the energy consumption (E), where $[E] = ml^2/t^2$.

We now have a set of six parameters (W,U,P,R,V,E) in three basic dimensions (m,l,t). According to dimensional analysis (Buckingham's Pi-theorem) it should be possible to form three non-dimensional groups of the six parameters - yielding hopefully the equivalent of the Mach or Reynolds numbers of the transportation process. By inspection, a possible set is:

(1)
$$\pi_1 = P/WV$$
$$\pi_2 = E/W\,R$$
$$\pi_3 = U/W$$

Having performed this little exercise, one asks: So what? A little thought and a review of the literature shows that several investigators have had a good time with these parameter groups, they have interesting physical meanings, and one even has a name. The first group (P/WV) is usually referred to as the "transport economy index." To clarify its meaning, consider the following further manipulations:

Assume that in the transportation process the motion is steady, i.e., that V = constant. In this case the "tractive force" (T) producing the motion is equal to the sum of the forces (F) resisting the motion. For flying devices the tractive force, T, would be the thrust (T) and the resisting force, F, would be the drag (D). Now we observe that:

(2)
$$R = V\,t \text{ and } P = E/t = E\,V/R = TV = FV$$

thus we find:

(3)
$$\pi_1 = P/WV = EV/WRV = E/WR = \pi_2$$

1.32

and the transport economy index (P/WV) is the energy consumed per unit weight per unit distance traveled in the transportation process. Further, if one confines one's attention to flying devices (and assumes either that the "power required" is that delivered directly to the air or that there is no aero/mechanical loss in the system), then:

(4) $$\pi_1 = P/WV = TV/WV = T/W = 1/(L/D) \text{ where } L = \text{lift} = W$$

and we find that the transport economy index is merely the reciprocal of the flight vehicle's lift-to-drag ratio. For subsonic flying devices it can further be shown (using relations from the next section on variations in vehicle size with weight) that:

(5) $$\pi_1 = P/WV \propto V^2/W^{1/3}$$ since $P = TV = DV$

$D \propto V^2 S$

$S \propto W^{2/3}$

(π_1 decreasing $\Rightarrow$ good)

But again, so what? What new clarifications to the overall transportation problem, and its "optimization," does this bit of arithmetic provide?

Two examples, one due to engineers, and the other due to a biologist are of interest. The first example is from a classic paper entitled: "What Price Speed?" written in 1950 by Gabrielli and Von Karman. Having discovered the transport economy index, the authors set about collecting power, weight and speed data for a wide variety of transportation devices from standard references. An immediate difficulty in such a search is that standard references seldom list the "optimum values" of such variables - rather they generally list merely extremes. Thus the authors based their analysis on values of maximum speed, maximum uninstalled power and maximum gross weight. Thus, in this work $P/WV = P_{max}/W_{max} V_{max} \neq 1/(L/D)_{max}$.

With large samples of data for each type of vehicles considered, Gabrielli and von Karman plotted the value of $P_{max}/W_{max} V_{max} = T_{max}/W_{max}$ (calling T/W the "specific tractive force") versus speed (V). Next they observed that if only data for "single unit" vehicles (i.e., excluding multi-unit trains, tractor-trailer trucks, etc.), these appeared to be a limiting line relation in speed beyond which the values of "maximum" transport economy or specific tractive force resulted in "uneconomical" vehicles. Stated another way, if one wished to produce a specific type of vehicle (an automobile, transport airplane, etc.) capable of traveling at a certain speed, one has to pay an appropriate "minimum" price in terms of energy consumed per unit weight per unit distance traveled. The basic Gabrielli-von Karman plot demonstrating this result is shown in Figure 2. Periodically, subsequent investigators have updated the analysis and made attempts to derive this limiting line relationship theoretically. While attempts at theoretical prediction have been unsuccessful, later statistical analyses indicate that the limit line relation shown by the original authors still exists - but that it is "technology dependent," moving parallel to itself to the right a small increment each decade.

The second study of interest employing the transport economy index was done by the biologist Vance Tucker at Duke University. In this case Tucker was interested in the relative energetic requirements of various biological systems and had data from experiments on the "optimum" energy consumption of such systems. Dividing his (limited) data into those for swimming, flying and walking/running systems, he plotted $(P/WV)_{min} = (T/W)_{min} = 1/(L/D)_{max}$ (for fliers) versus weight to produce Figure 3.

What one observes from Figure 3 is that if one wants to transport something in the most "economical" way possible, making it "swim" or float would be the choice. Flying is the next best way to go and walking is a poor third choice - for devices of equal weight. The second trend shown in Figure 3 is that for the biological devices considered, there is a dramatic improvement in economy with increasing size. For example, considering the category of walker/runners one sees that mice at the upper end of the line should not attempt to migrate and are nearly two orders of magnitude less economical than horses and elephants at the other end of the line.

Finally we observe the apparently poor relative energy economy of man-made devices compared to their biological counterparts. Ignoring the fact that the man-made vehicles are "commercial" (i.e., they carry some discrete pay load) versus the fact that the biological devices merely travel professionally (they move about to find food and thus "make a living"), the discrepancy between the two is due to the trade which has been made between absolute "economy" in the biological systems and speed (which determine productivity) for the man-made commercial vehicles.

Before moving on to the question of productivity versus energy economy, it is interesting to examine aerodynamic efficiency (maximum lift-to-drag ratio) of a variety of low-speed flying devices. From the information in Figures 3 and 4 one might conclude that if one was prepared to reduce the cruise speed of a Boeing 747 to under M = 0.1, a machine of unprecedented economy (compared to any natural counterpart) might be produced. Likewise, no natural device is capable of flying at the Mach numbers of the actual 747. Lest one become arrogant about man's ability to excell nature, however, it should be observed that the 747 remains unable to lay eggs and reproduce itself.

To complete the present (admittedly simplistic) outline story of transporation, it should be noted that the notion of transport economy indices have played little if any part in aviation over the first seventy years of its development. If one doubts this, one need only examine Figure 5 which shows the trend in absolute world speed record since 1906 when such records were first recognized. The auxillary plot in Figure 5 shows the corresponding trend in the measure which has driven commercial aviation, the so-called productivity index.

(6) $\phi = \frac{U}{W} V$ - productivity index (a dimensional quantity) Note: ϕ increasing $\Longrightarrow$ good

Here we see dramatic progress and it will remain of interest to see how the inherent conflict between transport economy (eqn. 5) and productivity (eqn. 6) is reconciled as fuel costs rise and fossile fuel resources decline in the decades to come. The flattening in the trend in productivity index in commercial aviation development during the last decade is already clearly reflective of a realization that the world in which we live is no longer infinite nor is the supply of natural resource readily available to us. Thus the productivity imperative which leads naturally and inexorably from DC-3 to Boeing 707 to SST (based on 10¢ a gallon jet fuel) has been dramatically arrested by the rise of OPEC and a far-reaching questioning of many of the assumptions on which past technological progress had been based.

On Being the Correct Size

Having examined in broad outline the energy consumption versus productivity relations in a generalized transportation or locomotion process, it is now of interest to examine in the same spirit questions such as: "How big (or small) can an animal be?" and "Are there general relations between size and weight for all the things that fly?"

To investigate the first of these questions, a useful organizing principle is provided by the simple square-cube law (cf. Figure 6)*. The square-cube law says that the dimensions (size) of homogeneous, geometrically similar objects can be described by some characteristic length (L). This done, all surface or cross-sectional areas are proportional to the characteristic length squared (L^2) and the weight is proportional to the volume or the characteristic dimension cubed (L^3).

With these notions in hand let us begin by applying them to the geometry of a class of grazing animals - specifically, consider a spherical cow. Assume that the basic materials from which we wish to construct a family of such cows have common properties of yield or ultimate stress, and so on. The first of our cows (A) will not be ambitious in scale and its conservative body and head are easily supported by the nimbly articulatable "slender column" legs of an antelope.

Our second attempt at cow design (B) is bolder, involving doubling the size (L) of the first prototype. If we attempt to maintain strict geometric similarity between our first and second designs, however, we run into trouble. Cow B, according to the square-cube law, must weigh eight times as much as cow A, but the cross-sectional area of the supporting legs has only increased four-fold. Assuming the structure of cow A was properly optimized and the stress levels in the legs were established such that no excess bone material beyond that necessary to support the weight was used, then the stress level in cow B's legs would exceed the stress level in the legs of cow A by a factor of two, and cow B would collapse. To avoid this difficulty, we must abandon strict geometric similarity (at least in the leg structure) and thicken the legs of the B-model cow, with a consequent loss in agility.

Playing this game on to absurdity, we eventually arrive at the absolute cow (C) wherein the legs necessary to support the weight become equal to the dimensions of the brute's body, locomotion is no longer possible and unless grass grows very rapidly under its mouth, it will die. Somewhere between B and C lie the elephant and the now extinct Brontosaurus, representing practical extremes in giantism in "spherical" grazing animals.

Smallness may also be investigated using square-cube law arguments. In this case consider the giant spherical mouse. Our family of mice are warm blooded and thus must maintain some level of internal temperature. To do so the job of these mice is to eat foods of high caloric content. In this process about 20 percent of the energy content of the food consumed can be

* The size-weight augments to be advanced on the basis of the simple square-cube law must be considered first order preliminary only. An excellent critique of the square-cube law applied to biological systems, together with a more sophisticated analysis is presented by McMahon (1973).

converted by the mouse's muscles into useful work (e.g. moving about in search of more food, finding a mate, fleeing from predators). The remaining 80 percent goes into heat which maintains internal temperature, and any excess is transferred from the body surface (which is proportional to the square of the characteristic dimension of the mouse).

In this case, as our mouse shrinks, its heat conserving mass decreases more rapidly than its surface area and at some point the poor creature must become truly voracious, eating continuously twenty-four hours a day, merely to balance its caloric intake with its heat loss - regardless of how much insulating fur it grows. In practice the smallest warm blooded animal is a species of pygmy shrew about the size of one's thumb nail.

Our interest in this dissertation, however, is neither in cows nor mice but in things that fly. It is thus of interest to evaluate the consequences of the square-cube law vis-a-vis birds, insects and airplanes, the geometric similarity of which is tenuous at best. Undaunted by this detail, the task has been made much easier by the massive compilations by Greenewalt (1962) and Hartman (1961) of data relating geometric size (wing span and area, aspect ratio, tail area, etc.) and weights (total wing, muscle and internal organ) for nearly the entire range of animal and insect fliers. From these sources supplemented by data from standard aeronautical references (e.g. Jame's All the World's Aircraft), it is possible to construct Figures 7 and 8, relating respectively wing span and wing area, to vehicle weight, covering twelve orders of magnitude in the latter variable. Also shown are the author's square-cube law fits to the entire ranges of data.

A casual perusal of the data presented in both Figures 7 and 8 shows that while there is substantial scatter about the global square-cube law means, particularly in the insect range, basic square-cube law lines do seem to capture the trends over the full range (again twelve orders of magnitude in mass variation) of all the different types of flying device - with a few noteworthy exceptions. Principal among these exceptions are hummingbirds (Greenewalt, 1962), human powered airplanes* (McMasters, 1977) and model airplanes (no demonstrated here).

The data in Figure 7 shows far less deviation from the global mean than that in Figure 8, however, Figure 8 is by far the more technically interesting. One can go through a lengthy discussion of the detailed trends in weight versus wing area in Figure 8 for the variety of fliers shown, but for purposes of the present discussion it is only necessary to note the following points:

- o Again, while a single square-cube law line ($M = 15\ S^{3/2}$) captures the trend in most conventional fliers, the deviations from this mean even within a given category (e.g. insects) may be large.

* It can be argued that devices like human powered and model airplanes are anomalous because they are "recreational" rather than professional or commercial fliers. Thus they are driven by no economic imperative, with size established merely by the basic laws of aerophysics and structural mechanics.

- If one takes a closer (but not too close) look at the data, now basing the assessment on the way different devices fly, one sees interesting further support for a simple square-cube law variation of mass with wing area. Specificially, if one considers "light gliders" (e.g. the Zinnonia seed, butterflies, pterosaurs and hang gliders), all of these types display the trend $M = S^{3/2}$. Likewise beetles, turkeys and Cessna's can all be connected by another square-cube law trend line, and soaring birds and sailplane are seen to be roughly equivalent in a square-cube law sense.

Despite the lack of strict geometric similarity of the devices considered in Figures 7 and 8, there does appear to be a strong pattern in size-weight relations between them, particularly, when viewed from the proper perspective of their diverse modes of flight. Further discussions of these matters have been published by Cleveland (1970), Greenewalt (1962, 1976), McMasters (1974, 1976), Brower and Veinus (1981), and Pedley (1977).

We now come to the rather hotly debated topic of how big can a flying animal get. At present this question relates to both flying birds and prehistoric flying reptiles (Pterosaurs). The square-cube law says the following things on this matter:

Wing area varies as	$W^{2/3}$
Wing loading varies as	$W^{1/3}$
Flight speed varies as	$W^{1/6}$
Power required varies as	$W^{7/6}$

The literature on the subject shows that for birds, flight muscle weight (and hence power available) tends to be a constant fraction of total weight - that is: Power available varies directly as W. The British ornithologist C. J. Pennycuick (1966) thus showed that square-cube law extrapolations of his own measurements of power required to fly for pigeons predicted the heaviest bird capable of flight (i.e. power available just equals power required at one "design point") would have a mass of approximately 20 kg. This he found consistent with the weight of a wild South African turkey - the Kori Bustard - which indeed is just barely able to fly.

While the Bustard may be the heaviest flying bird extant, it is not (dimensionally) the largest. Depending on ones point of view the largest living flying birds are the condors (California and Andean) and/or the Wandering Albatross (Diomedae exulans) shown in comparison in Figure 9. While enormous, the condor is further dwarfed by the 4 meter span Ice Age vulture Teratornis merriami, now extinct and known only from complete fossile remains from the La Brea tarpit in California. Recent reports of a fossile teratorn found in Argentina presently credit it (based on scaling the length of a single inner wing bone to deduce total span) with a wing span of 7 meters. Taking the existing Condor as a model for these larger teratorns, the square-cube law would predict weights of 23.7 kg (52 lbs) for the La Brea tarpit species and 127 kg (280 lbs) for the Argentine monster, both beyond Pennycuick's limits, the latter by a huge margin.

A tentative partial explanation for these apparent anomalies lies in the fact that a major flaw in classic square-cube law predictions of the sort made so far is that they neglect the effects of fluid dynamic scale factors - specifically Reynolds and Mach numbers. The author (McMasters, 1974) has been able to show that it is a relatively easy matter to accommodate first order approximate Reynolds number scale effects in the context of square-cube law.* *Appendix A* Thus it can be shown that if the vultures under discussion all had completely turbulent boundary layer flow on their surfaces, the power required should vary not as $W^{7/6}$ but as $W^{65/57}$, an apparently small but significant difference. In fact, playing with Pennycuick's data, it can be shown the maximum weight of a "giant pigeon" accounting for Reynolds number scale effects is on the order of 40 kg (compared to 20 kg when Reynolds numbers are ignored). Based on this new value, the biggest feasible pigeon would have a wing span of about 4.75 m, a little smaller than the monster Ice Age Teratornis incredibilis known from a few wing bones discovered in a cave in Nevada.

What all of this says is that one probably can't manipulate a pigeon into a turkey (or a sea gull) and that birds larger than those presently in existence did exist during the last Ice Age - and they didn't get that big by sitting on their tails because they were too big to fly. As the American ornithologist Carl Welty (1955) has observed: "Birds simply dare not deviate widely from sound aerodynamic design. Nature liquidates deviationists much more consistently and drastically than does any totalitarian dictator."

One might further observe that giantism is something of a luxury in nature and generally represents an extreme in specialized development. The ability of a biological device to specialize to such a degree - to expand to the limit of some particular ecological niche and continue to survive - must depend on the stability or permanence of the environment which permitted such development. Climatic changes between the Ice Age 10-20,000 years ago and today doomed the giant teratorns to extinction. Regrettably, the modern condor seems equally close to extinction due to the encroachment of civilization on its habitat - a change in environment no less profound than a change in climate and food supply. As will be discussed later, the same sort of fate befell the dinosaur and its fantastic airborne counterpart the Pterosaur.

From Generalities to Specifics

An attempt has been made to lay out the broad outlines of some first order relations between size, weight and energy consumption of devices which fly. As was to be hoped, these considerations demonstrate that rather than being separate disconnected topics, natural and human flight represent the two ends of a continuous, very broad and fascinating spectrum. A huge range of diverse devices and configurations are all tied together by the underlying requirement that each individual element in the full spectrum must obey the same basic laws of physics. Further, the range of viable flying devices is restricted by the requirements of the environment (physical and/or economic) in which they must operate. Within these restrictions, however, the variety of devices which do manage to fly and remain viable is nearly mind boggling.

The previous general trends in size, weight and power requirements displayed do very little, however, to clarify how devices as obviously dissimilar in form and function as dragonflies, bats and jet transports have been designed to fit so neatly into their individual, complimentary, ecological niches - let alone demonstrate what lessons the study of the detailed design of natural flying devices might teach the designer of a modern airplane. It is not the purpose of this presentation to address this latter issue directly, but rather to stimulate the reader to devote some thought to the matter, perhaps study the topic farther, and draw his or her own conclusions. The subsequent sections will merely describe some examples of good design in specific models of natural flying devices. The intent is to demonstrate how general rules have been put to practice or how misunderstanding or ignorance of basic principles has led to erroneous and sometimes absurd conclusions. No particular criteria have been used in selecting the examples to be discussed other than that they interested or amused the author.

The Design of Insects

The author is a strong advocate of the disireability, at the "conceptual design" level, of constructing a configuration matrix. The intent in doing so is to explore in the sense of brain storming, all the possible ways of solving a problem regardless of the probable practically of any particular solution. Among the author's favorite examples of such a configuration matrix is that shown in Figure 11, displaying (not to scale) the various forms of flying insects - all of which exist and appear to be highly viable despite the best efforts of the pesticide industry.

At present there are about 800,000 named species of insect and it is estimated that there are that many more which have not yet been formally identified. A very large percentage of these insects fly, ranging in size from the nearly microscopic thrip (lower right hand corner in Figure 9) to large tropical butterflies with wing spans of 10-12 centimeters. Far more dramatic are examples of 280 million year old dragonflies embedded in amber with wings spanning up to 70 cm (30 inches!). These monsters presumably became rapidly extinct in the face of competition from later flying reptiles and birds. Current examples of the form with wing spans of up to 10 cm remain highly viable, however. The basic design has been successful enough to have survived for 300 million years, and insects in general exhibit a marvelous array of ingeneous solutions to very difficult design and manufacturing problems. Despite this, the author's various attempts to interest the students in his airplane design classes in bug design usually elicites the response: "Well, that's all very interesting we suppose ... but they're not in Boeing's product line and we'd really rather not hear about that." So much for creativity and expanded horizons, and the mystery remains as to why so many turned up to find out why bumble bees can't fly at the AIAA noon lecture.

Before discussing why bumble bees <u>can</u> fly, several points regarding the general design of the flying insects <u>need</u> to be reviewed. Several excellent modern references on the subject exist (e.g. Nachtigall, 1974; Dalton, 1977; Hertel, 1966, Weis-Fogh, 1956, Pringle, 1957, Lighthill, 1975).

In addition to a diversity of size and shape of the configurations shown in Figure 11, the figure also displays the general trend in insect development from the earliest forms with four discrete, non-foldable wings to the most recent (but still ancient) models such as the fly and bee in which the hind wings have degenerated to a pair of small nubs thought to serve the function of flight orientation sensors, while the fore wings have the ability to fold along the sides of the body when not in use.

The architecture of the wings of the various insects is of very considerable interest both technically and aesthetically, and to fully understand them one must refer back to Figure 1 wherein the Reynolds number range of insects versus other types of flying device is shown. One sees that no insect flies in the Reynolds number range beyond about 10^4 and it turns out that, in consequence, the airfoil sections employed by the insects are dramatically different than those used on any bird or airplane. For reference, the performance of airfoils over the Reynolds number of interest is shown in Figure 12.

A very interesting study of insect wings has been conducted by Rees (1975). What emerges from Rees' work is a remarkable example of a wing designed to provide good aerodynamic characteristics (at the Reynolds numbers is question) coupled with desireable structural and mechanical properties, which at the same time is readily manufactureable from the materials at hand. The shape of a typical insect wing and its airfoil sections are shown in Figure 13. Rees writes:

> "[Natural] Selection is likely to have resulted in the evolution of insect wings which combine aerodynamic efficiency with a rotational moment of inertia about the wing base small enough to reduce as far as possible the energy expenditure in their repeated accelerations. Their construction has to leave them stiff enough to remain aerodynamically efficient when under inertial or aerodynamic load, and free from buckling, however light they become. Insect wings are very light structures - 11 g/m^2 in the dragon fly Aeschna cyanea, 16.7 g/m^2 in Locusta migratoria and 7.4 g/m^2 in Tipula sp."

Based on Rees' work and the cited references on insects, the story then goes like this: The insect to be starts life in a larval state and when its time comes to metamorphose into its final form it emerges as a legged body to which are attached what resemble limp wet sacks where the wings should be. This characature of an insect then positions itself, perhaps on the convenient branch of a tree, and begins to pump fluid through the veins in the wing sacks, expanding them to their full extent. Under the action of sun and breeze the fluid then evaporates leaving a series of properly located hard hollow tubes of devious cross section, arrayed over the wing surface and connected by truss-like cross members, while the intervening spaces are covered by the thin membrane film of the dried and collapsed sack material. The film which thus makes up the lifting surface is on the order of a mere 2-6 micrometers in thickness. As seen in Figure 13 this final structure is far from flat, but neither does it have the smooth continuous cambered form of a conventional airfoil. Rather, it is a ragged, rough surface in cross-section, and it turns out that this has major aerodynamic and structural advantages.

While the details of the low Reynolds number flow over a wing of such contour remains to be understood, the advantage is undemiable as demonstrated by experiments conducted by Rees, the results of which are shown in Figure 13. Here Rees first encased an insect wing in epoxy and examined various cross-sections under an electron microscope to determine their shapes. He then constructed greatly enlarged stainless steel replicas of a typical section which he then tested in a water tunnel to achieve the correct viscous scale conditions. He then built a plastic model of the conventional shaped airfoil which would form the envelope curve around the actual insect airfoil and tested this in the same tunnel at the same scale conditions. It is readily seen from the results in Figure 13 that the "conventional" smooth airfoil is inferior to the actual section in both maximum lift coefficient and in drag coefficient - due primarily to laminar separation on the smooth contour. Thus the contour the insect can manufacture, if not optimum aerodynamically, certainly represents a good shot at optimality in a very difficult flow regime.

Having explored the aerodynamics of the insect wing, Rees then takes up the question of structural optimization [following Hertel (1966)]. Here Rees examines the strength and deflection characteristics of various flat and corrugated plates both with and without longitudinal hollow tube stiffeners scaled and located to approximate those in the insect wing. Here he observes: "We thus see that the introduction of corrugation is associated with scarsely any weight penalty ... although these are immense reductions both in deflection and in the maximum stress experienced for a given [bending] load. The tubes when added to the beam of corrugated cross section increase its mass by a factor of 1.66, but result in a decrease in maximum stress by a factor of 2.87. Inclusion of tubes in a plane section [no corrugation] results in almost as great a mass as in the corrugated, tubed section, but [the uncorrugated, tubed beam] still deflects 337 times as far ... for a given end load, is 38 times as highly maximally stressed, and yet its rotational moment of inertia is only 2 percent less than that of [the corrugated, tubed beam]."

Rees further notes: "In a real insect wing these tubes (veins or nervures) often carry haemolymph or serve as conduits for nerves or tracheae. In a corrugated beam with a very thin web there is in bending a tendency to collapse suddenly and catastrophically by buckling at the folds. The inclusion of tubes at these positions reduces the likelihood of this form of failure. Insect wings show irregular depth of corrugation, both chordwise and spanwise. Corrugation tends to be deeper in the interior half of the wing and probably reflects the chordwise distribution of aerodynamic forces."

Altogether, the picture Rees (and Hertel) draws for us shows a remarkable piece of systems engineering in its best sense. The neophyte designer of an airplane wing who approaches the problem merely from the perspective of its aerodynamic optimization might well profit from the study of examples such as this one - provided by a mere entomologist.

The Flight of the Bumblebee

Having examined the broad outlines of flight and the detailed architecture of insect wings we come to one of the central questions in this dissertation: "Can the bumblebee fly?" More to the point, "Who was the Aerodynamicist who 'proved' they can't?"

The author has devoted considerable effort to a search for the identity of the knave who is purported to have consciously denied the contradicting evidence before his very nose. In the course of this investigation, a most remarkable recent (1979) book entitled Bumblebee Economics written by an entomologist, Bernd Heinrich, has been discovered which, while not divulging the identity of the notorious Aerodynamicist, has made the search altogether worthwhile. Questions of flight aside, the bumblebee is worthy of serious study for at least two major reasons. First it represents a remarkable example of superb thermodynamic system engineering and, second, it provides a vivid example of a process of coevolution between plants and animals which mutually benefits both of them and mankind as well.

Before reviewing these more interesting aspects of the bumblebee, let us dispose of the mythological aerodynamicist. The search for the villain leads to the conclusion that he apparently does not exist as a single individual, but rather he is archetypical of several investigators (usually entomologists) who have attempted to work with concepts and tools outside their own field of specialization and - for a variety of reasons - blew it. The mischief is then compounded when a journalist identifies the investigator by the field he happens to be working in at a particular moment (aerodynamics) rather as a member of a discipline in which he has legitimate training and expertise (entomology). Thus folklore is born and, once published, perpetuated. Aerodynamics (as interpreted during the 1920's and 30's by entomologists) rather than the Aerodynamicist appears to be the culprit. Pringle (1957) provides a brief review of good and bad analyses of insect flight and names a number of individuals who apparently contributed to the general bumblebee myth - although none apparently cited the bumblebee explicitly as being incapable of flight. That part of the story seems to be a purely journalistic construct.

All this is not to say that an entomologist cannot be a more than competent aerodynamicist or computational fluid dynamicist. It turns out, however, that such combinations are even rarer than the few recorded examples of the sort of serendipitous collaboration between biologists and aerodynamicists as reflected in the work of von Holst and Küchemann (1941), Weis-Fogh and Jensen (1956) and Weis-Fogh and Lighthill (1975).

So much for the myth. The reality of the details of insect flight has greatly advanced due to improvements in experimental techniques (particularly high-speed photography), computational capabilities and a general improvement in both aerodynamic and biological knowledge. As Pringle (1957) wrote:

> "Forward motion of the [insect] body as a whole may be necessary for flight, but in the most advanced fliers, which can hover or fly backwards and sideways, the analogy of a helicopter is closer than that of a conventional aeroplane. Unlike a helicopter the movement is oscillatory rather than rotary and the axis of rotation is horizontal rather than vertical, but in both types of flying machine (sic) changes in the angle of attack of the wing in different parts of the stroke are necessary to produce lift, propulsion and control."

Pringle then notes:

> "The movement of the wings during flapping flight is an extremely complicated action involving, as well as elevation and depression, promotion and remotion (fore and aft movement), pronation and supination (twisting) and changes in shape by folding and buckling. The kinematics have [as of 1957] been fully described in only one example, the locust <u>Shistocerca gregaria</u> (Jensen, 1956)."

Indeed the monumental work of Torkel Weis-Fogh and Martin Jensen (funded in part by the United Nation's World Health Organization in an attempt to discover the inherent migratory capability of one of the greatest of the Biblical plagues), has finally shed a great deal of light on the complex problem of insect flight and, more importantly perhaps, served as the model for properly conducted investigations using a truly interdisciplinary approach to the problems of animal flight in general. Some results of Weis-Fogh's and Jensen's experiments on locust wing motion are reproduced in Figures 15 and 16.

Simplified analyses of insect lift and thrust production using a model of the wings executing simple harmonic motion give at best a "zeroth" order estimate and frequently lead to the conclusion that our poor bumblebee's wing must generate lift coefficients well in excess of two at Reynolds numbers in the order of 10^3. Even under unsteady conditions such values are not very likely.

Bumble Bee Thermodynamics

While there are plenty of aerodynamic problems with the flight of the bumblebee, there are a number of biological problems as well. In discussing these problems, it is necessary to take a closer look at the bumblebee itself, observing that it has to be one of the cleverest pieces of fine mechanism design extant. As shown in Figure 14 the bee's body is divided into three discrete parts: the head which contains the brain, eyes and mouth; the thorax, a nearly spherical shell containing the motor muscles and to which are attached the wings and legs; and the abdomen which serves as the payload container and houses the digestive and reproductive organs.

Heinrich (1979) observes:

"Bumblebees are able to inhabit cool temperate and arctic regions with short growing seasons in part because they are able to be active at low temperatures, and that ability is a consequence of their remarkable thermo regulatory physiology. The bees can maintain high body and nest temperatures, but they can do so only by expending prodigious amounts of energy derived from [the sugar in nectar in] flowers. Since reproductive success is based largely on the amount of energy available for rearing offspring, it is important for the bee to optimize the energy return per unit of time and the energy spent in foraging."

The magnitude of the problem of high energy expenditure for insects in general is cited by Pringle who notes:

The very high total metabolic rate of intact flight muscles [in insects] is stressed. The indirect muscles of some Diptera [flies] and Hymenoptera [bees, wasps] reach values of 2000 kcal./kg/hr, nearly twice that calculated for hummingbirds and ten times the maximum for human heart muscle."

Thus one sees that if one bases estimates (as was generally the case until forty years ago) on data only for the muscles of vertebrates, the flight of bumblebees becomes impossible - aerodynamic issues completely aside.

The bumblebee is thus constructed to conserve as much as possible the precious energy on which it relies. Heinrich describes some of these adaptations as follows:

"Anatomically, bumblebees are constructed so that heat loss is minimized, as is the energy expended in shivering necessary to oppose it. Only the thorax, housing the flight muscles, needs to be maintained at a high temperature in order for flight to be possible. Heat loss from the thorax to the surrounding air is minimized by the pile [fuzz] that covers the body and acts as insulation. Conductive heat loss to the abdomen is reduced by the narrow petiole that connects the two body parts and by a large air sac that insulates the anterior portion of the abdomen from the thorax. In addition, a counter current heat exchanger retards heat loss to the abdomen while allowing blood to circulate from the thorax."

To complete this part of the story on the short, sweet life of the bumblebee it is of interest to compare the fuel used and the power-to-weight ratio of the flight muscles of various insects compared to other animals and a reciprocating airplane engine. Typical values, taken from Nachtigall (1974) are listed in Table 2.

The Case of the Supersonic Deer Fly

While physics, and the aerodynamicist in particular, remain the losers in the exchange with the popular press over the flight of bumble bees, Nachtigall (1967) relates an amusing example of a modern myth demolished by a few simple calculations, thought and a trivial experiment. It remains a popular past-time to over estimate the speed of moving objects casually observed, but perhaps the most amazing example of this was the claim by a biologist that a deer botfly could reach supersonic speeds. During the 1930's an article appeared in the Proceedings of the New York Entomological Society which contained the following statement:

> "... on 12,000 ft. summits in New Mexico, I have seen pass me at an incredible velocity what were quite certainly the males of Cephenomyia pratti. I could barely distinguish that something had passed - only a brownish blur in the air of about the right size for these flies, and without sense of form. As closely as I can estimate, their speed must have approximated 400 yards per second."

The story was widely reprinted, appearing in various books of "world records" for several years. It also annoyed the Nobel Prize winning physicist Irving Langmuir exceedingly, and he set about deflating the claim in 1938.

The first problem with the biologist's estimate was that 400 yards per second at 12,000 ft. happens to be about Mach 1.1 and no sonic boom was reported. Langmuir calculated that the most optimistic estimate of drag due the botfly's body (ignoring the wings), required the botfly to generate nearly half a horsepower. In addition, the dynamic pressure generated during flight at $M = 1.1$ should have been sufficient to crush its head.

Botflies tend not to be the most elegant of fliers and on occasion while zipping around, run into things - deer and people for example. Langmuir calculated that the impact of a 0.2 gram botfly traveling at $M = 1.1$ would produce an impact equivalent to a large caliber pistol bullet - making hiking on the "summits in New Mexico" a somewhat risky business.

Finally Langmuir checked the circumstances under which the observations were made by attaching a small weight the size of a botfly on the end of a thread and by whirling it round found the upper and lower bounds of speed at which the object appeared as "a brownish blur in the air." The mean value turned out to be about 25 mph (1 yard/second) which turns out to be consistent with energy consumption requirements of the actual botfly. Thus the botfly was quickly struck from the list of the fastest flying machines, although the biologist's estimates continued to appear in popular reference books for several years thereafter.

Flapping Wings - A Clever Insect Adaptation

Studies based on vertebrate flight muscles directly attached to a beating wing indicate that wing beat frequencies greater than about 80 H_z should be impossible. Very small insects have measured beat frequencies of up to 1000 H_z however (Figure 17). The manner in which the insect circumvents the classical limit is extremely clever, involving use of materials which have so far defied synthesis by modern chemistry. Figure 18 shows the relevant mechanical arrangements and the mechanisms are described by Dalton (1977) and Nachtigall (1974) among others.

The Evolution of Vertebrate Wings

The evolution of various natural flying devices and their direct inspiration of human flying machines is shown in Figure 19. Utilizing the same basic skeletal structures, the various ways to construct a wing are shown in comparison in Figure 20. In the bird this leads to the dramatic variable geometry capability shown in Figure 21. While the bird can dramatically alter its wing span and area, its ability to control twist and chamber are limited. In contrast, the bat has a very limited capability for planform alteration in flight. However, by retaining the capability to articulate its "fingers" it has a high degree of control over its camber distribution making them highly maneuverable. The topic of the relative efficiencies of birds, bats and pterosaurs is of interest.

Pterosaurs and Hang Gliders

One generally credits the experiments in the late 1800s by Lilienthal, etc. as providing the inspiration for modern hang gliding. This is short sighted. For a period in excess of 100 million years (until their extinction about 65 million years ago) the Pterosaurs were the dominant form of vertebrate flying device. These creatures are of great interest for a number of reasons not least of which are that they appear to be not only the direct aerodynamic counterpart of modern Rogallo wing hang gliders, but show remarkable structural parallels as well.

Pterosaurs covered a wide range of size (Figures 22 and 23), the largest of which appear to be the largest flying animal ever evolved. A study of their skulls (Figure 24) also indicates that they adapted to a wide variety of diets and hence flight modes to search for their prey (Bramwell and Whitfield, 1974; Langston, 1981; McMasters, 1976).

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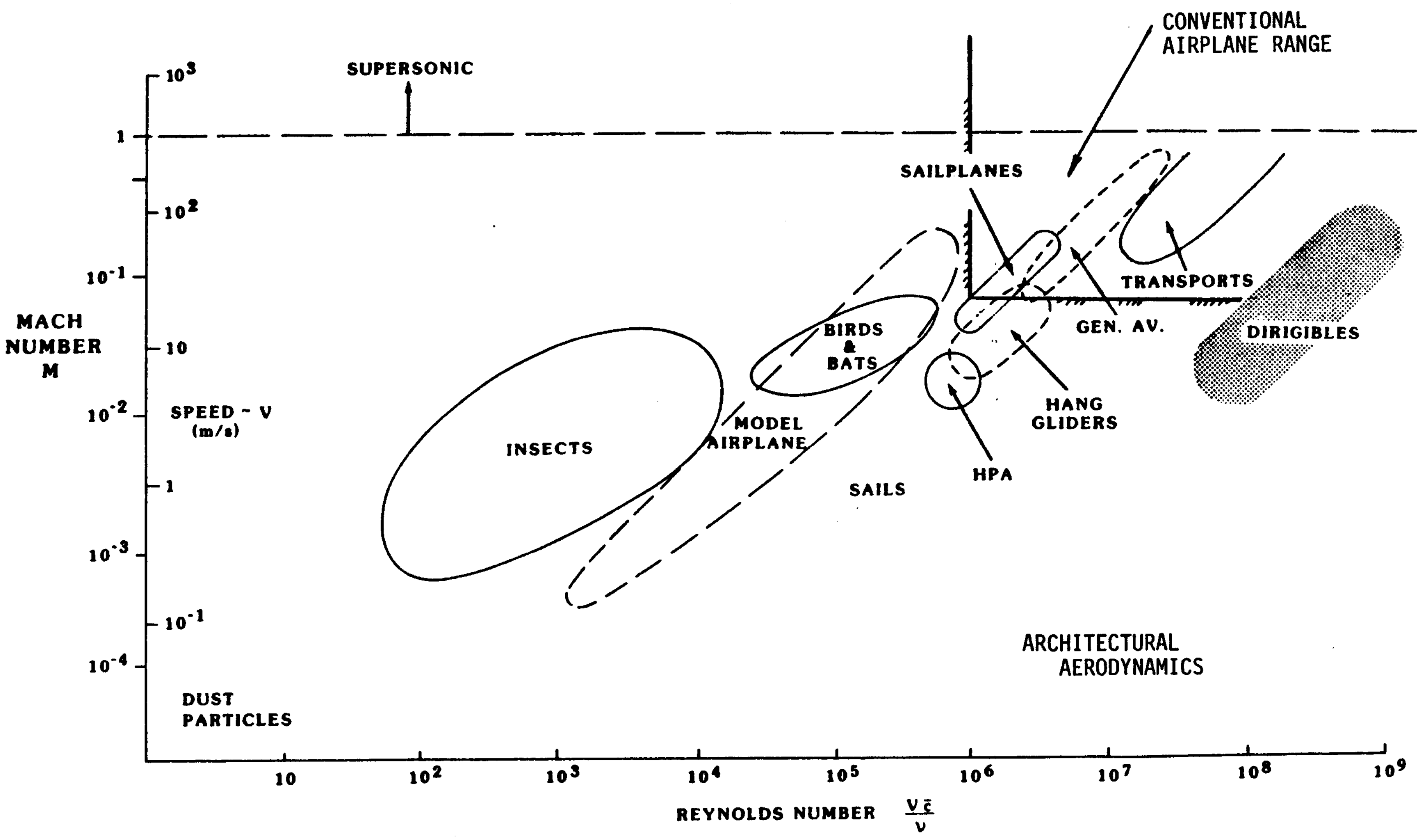


Figure 1. The General Range of Reynolds Number Versus Flight Speed for a Variety of Flying Devices.

INDICIES OF PERFORMANCE

- DIRECT OPERATING COST $= \dfrac{\text{TRIP COST}}{\text{PASSENGERS x RANGE}} = \$ / \text{SEAT - MILE}$

- PRODUCTIVITY INDEX $= \dfrac{\text{USEFUL LOAD}}{\text{GROSS WEIGHT}} \times \text{SPEED} = \dfrac{U}{W} \times V$

- TRANSPORT ECONOMY INDEX $= \dfrac{\text{POWER}}{\text{WEIGHT x SPEED}} = \dfrac{P}{W\,V} = \dfrac{\text{ENERGY CONSUMED}}{\text{W x DISTANCE TRAVELED}}$

P = maximum uninstalled power (lb-ft/sec)

W = gross weight (lb)

V = speed (ft/sec)

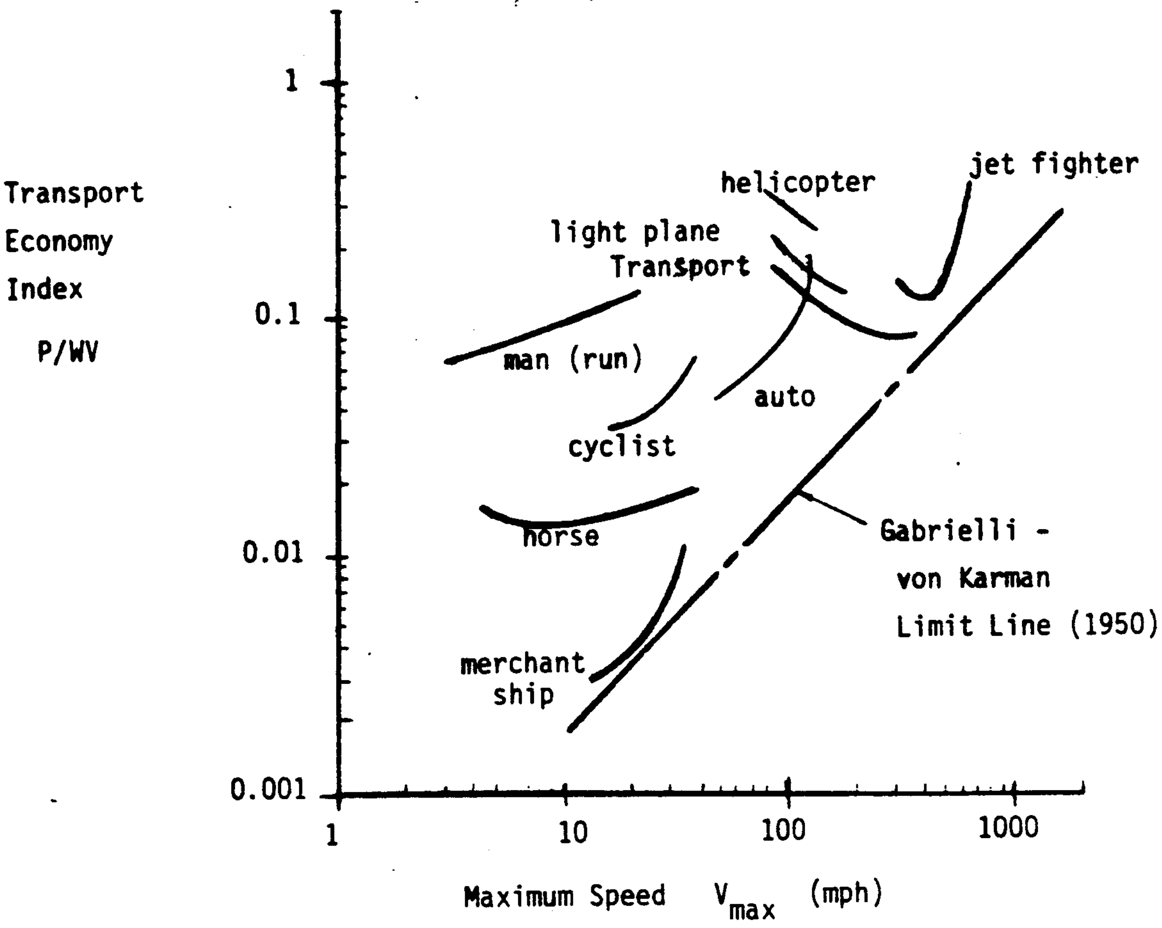


Figure 2. Variation in "Maximum" Transport Economy Index With Maximum Speed (Single Unit Vehicles)

non dimensional Φ = 0.427 kcal/ kg-km

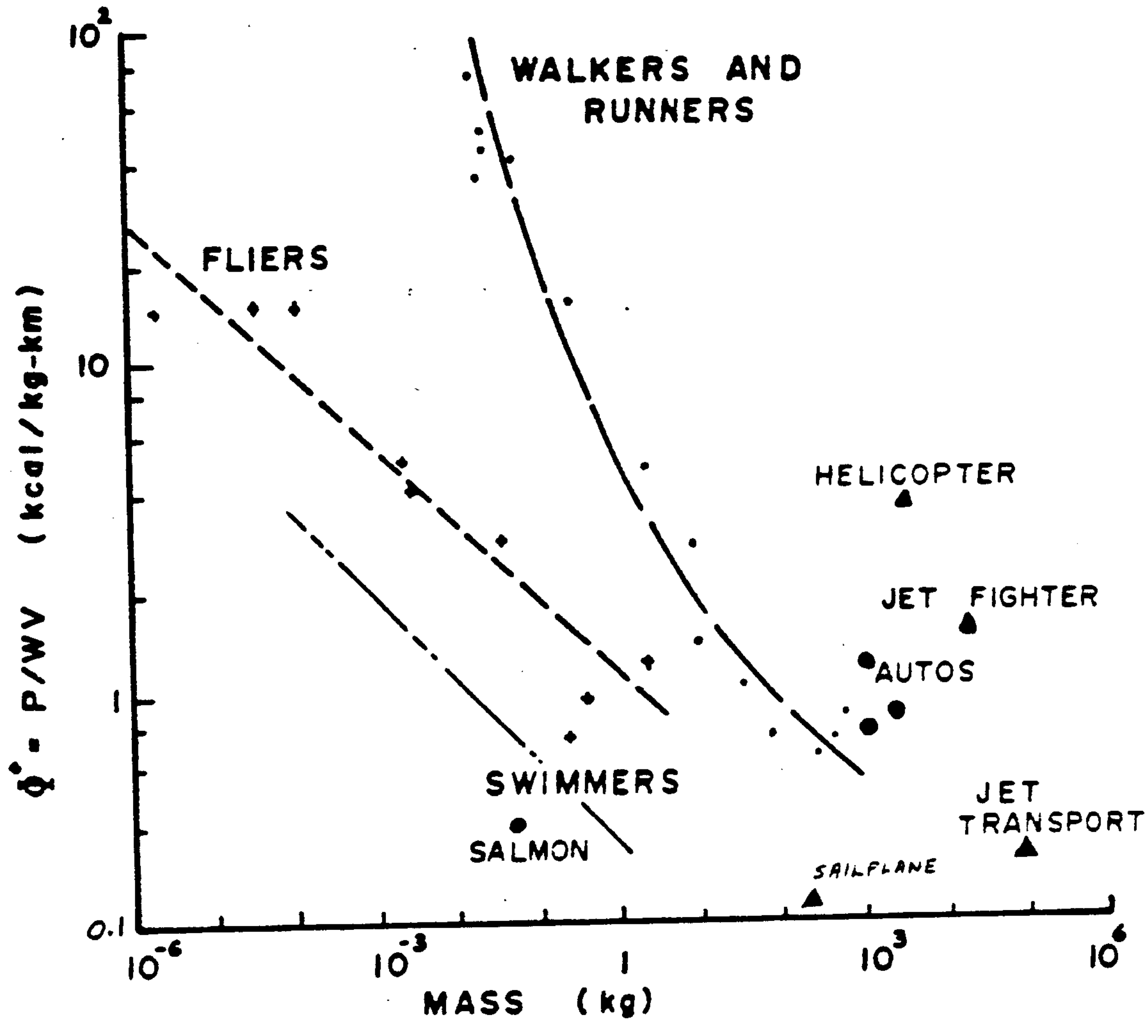


FIGURE 3. VARIATION IN MINIMUM TRANSPORT ECONOMY ($\Phi^* = P/WV$) WITH LOADED MASS

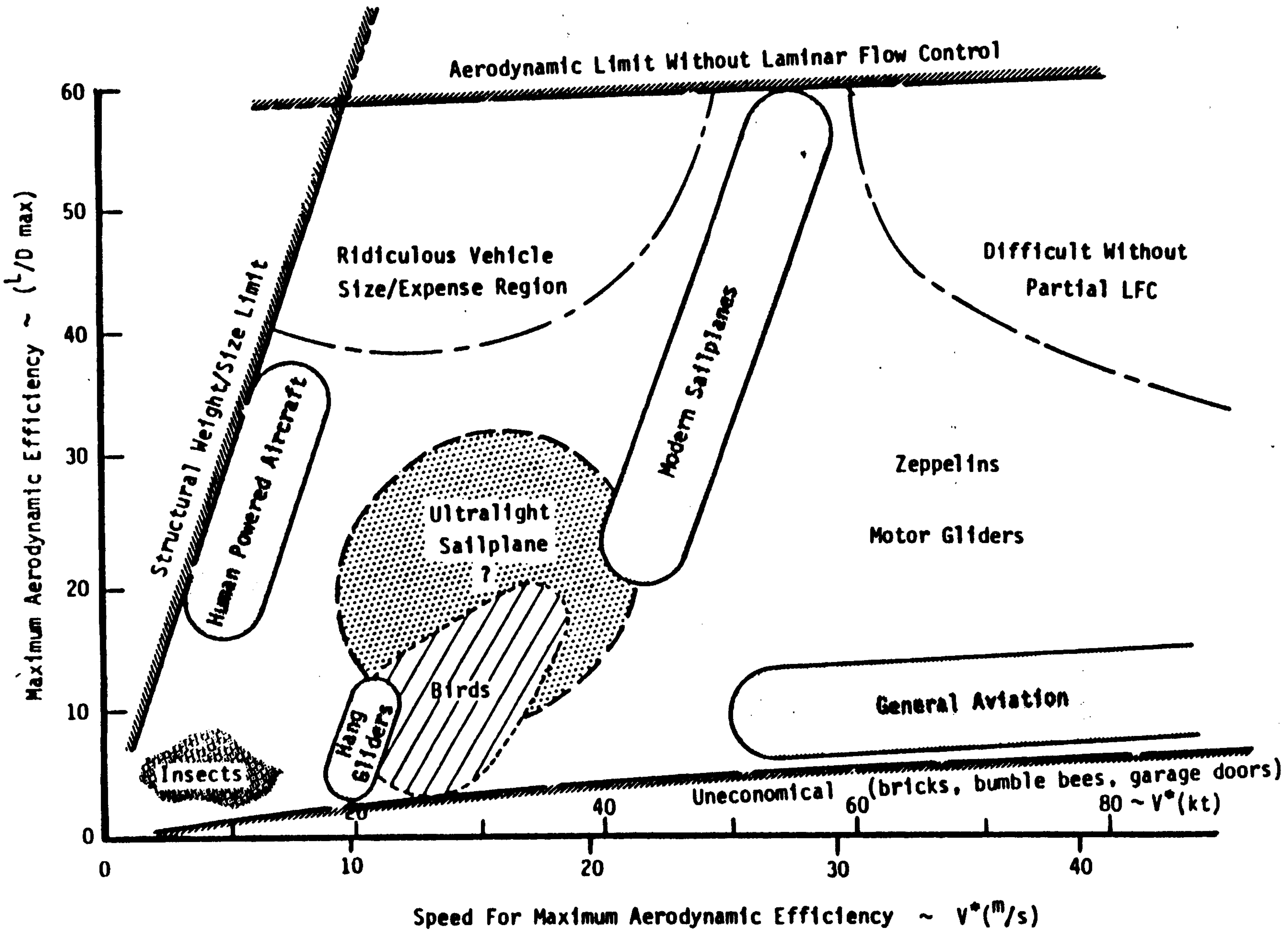


FIGURE 4. APPROXIMATE BOUNDARIES OF THE FEASIBLE/ECONOMICAL LOW-SPEED FLIGHT SPECTRUM

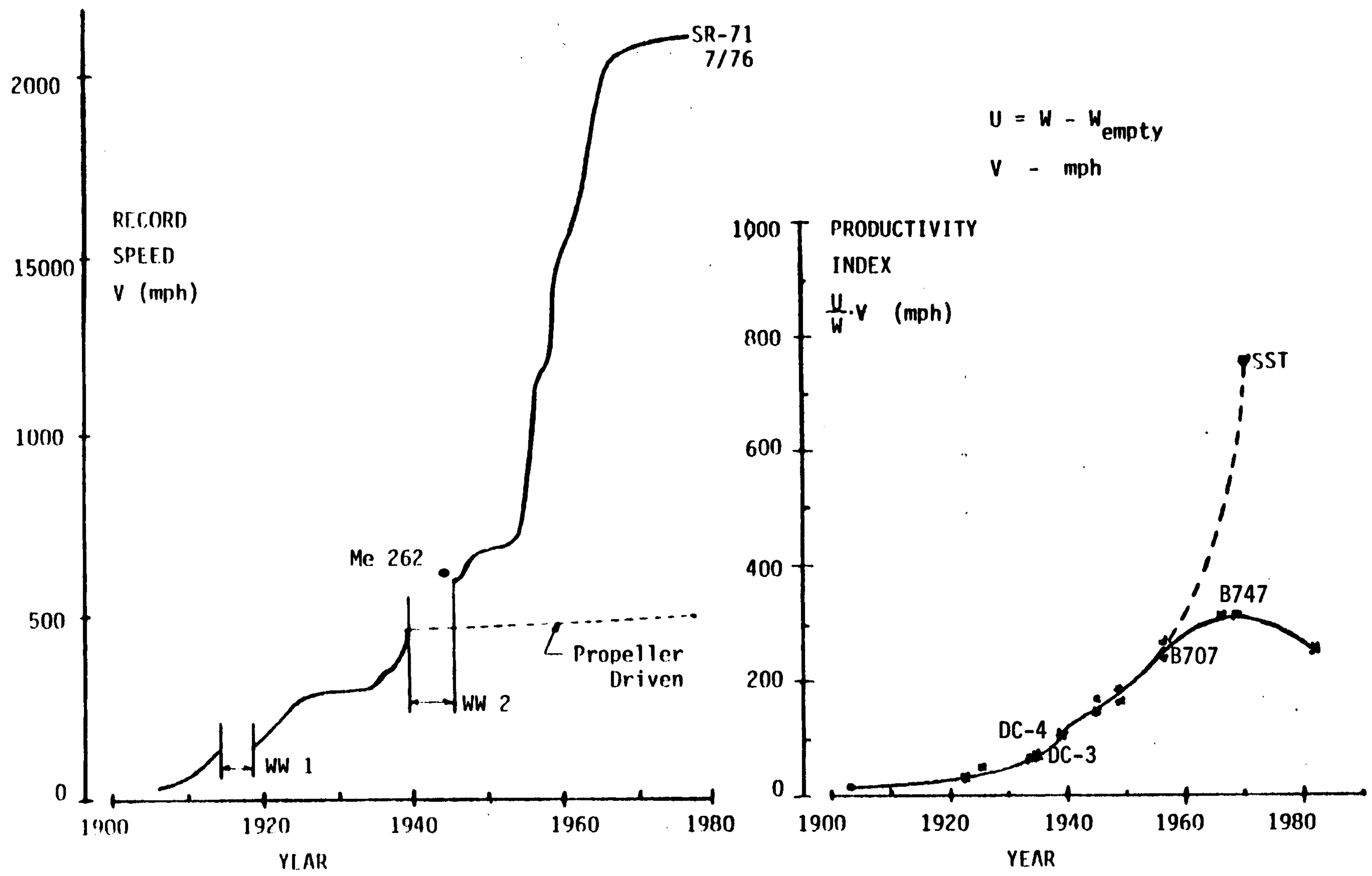


Figure 5. Trends in World Speed Record and Commercial Airplane Productivity Index with Historical Time

Figure 6. THE SQUARE - CUBE LAW

LENGTH ~ CHARACTERISTIC DIMENSION	L
AREA ~ STRESS, SURFACE AREA	L^2
VOLUME ~ WEIGHT	L^3

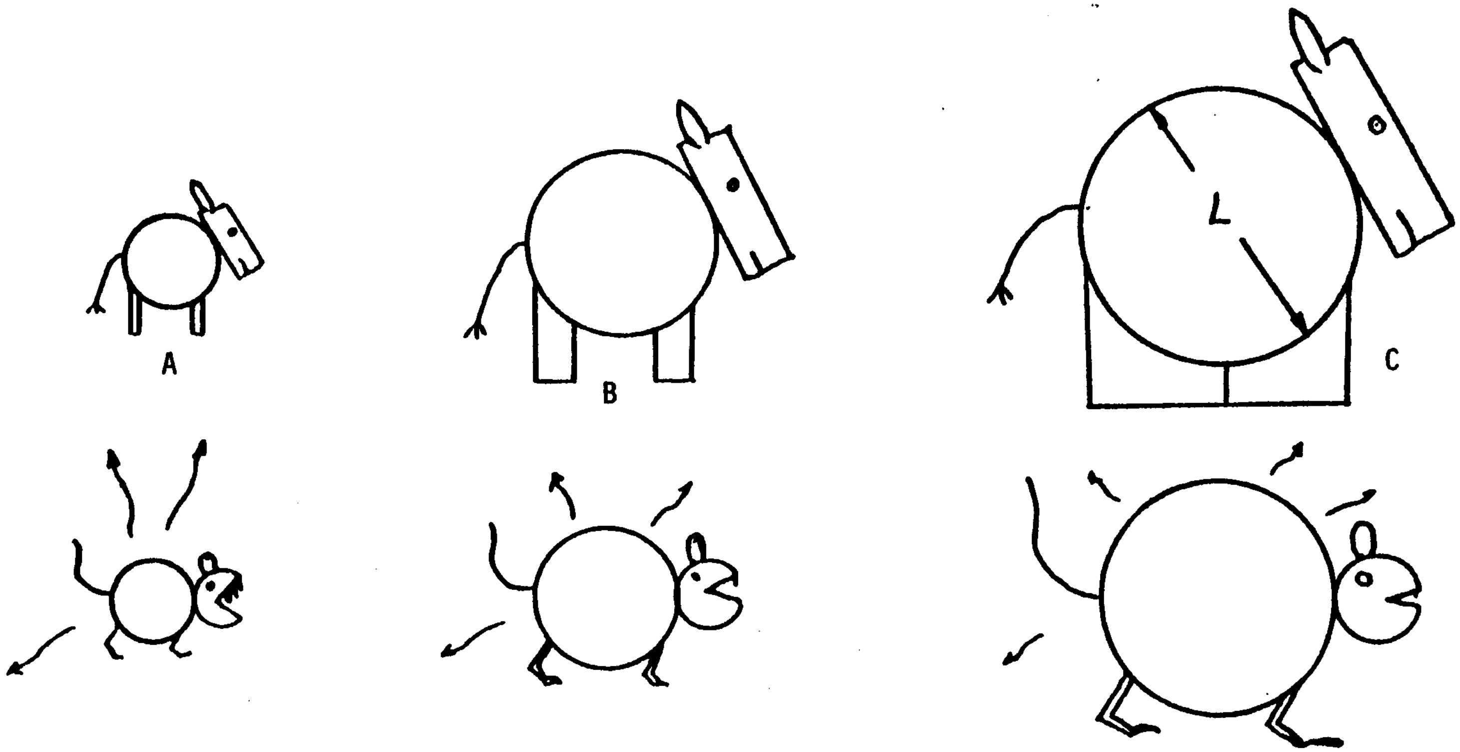


Table 1. Representative Flying Device Characteristics

Type	Wing Span b (m)	Wing Area $S(m^2)$	Aspect Ratio Æ	Loaded Mass M (kg)	Optimum Speed V (m/s)	Max. Lift-Drag Ratio L/D_{max}	Wing Loading M/S (kg/m^2)
1. Housefly (<u>Musca</u>)	0.013	$3x10^{-5}$	5.6	$1.2x10^{-5}$	2		0.4
2. Butterfly (<u>Papillio</u>)	0.082	$3.6x10^{-3}$	1.87	$3x10^{-4}$	3.5		0.83
3. Locust (<u>Schistocerca</u>)	0.10	$2x10^{-3}$	5.0	$2x10^{-3}$	4.15	2.2	1.0
4. Dragonfly (<u>Aeschna</u>)	0.10	$1.85x10^{-3}$	5.4	$1.5x10^{-3}$	10		0.85
5. Pigeon (<u>Columba</u>)	0.65	0.063	6.7	0.4	12.4	5.4*	6.35
6. Fulmar Petrel (<u>Fulmaris</u>)	1.09	0.102	11.7	0.725	12.2	8.3*	7.11
7. Black Vulture (<u>Coragyps</u>)	1.32	0.323	5.4	1.79	12.5	~12*	5.54
	1.44	0.364	5.7	2.3	15	~12*	6.32
8. White Backed Vulture (<u>Gyps</u>)	2.2	0.69	7.0	5.4	13	~14*	7.83
9. Rüppel's Griffon Vulture (<u>Gyps</u>)	2.5	0.83	7.55	7.5	14.5	~15*	9.04
10. Wandering Albatross (<u>Diomedae</u>)	3.5	0.60	20.4	9.2	20	~20*	15.3
	3.45	0.725	16.5	9.8	16	~19*	13.5
11. Dog Faced Bat (<u>Rousettus</u>)	0.554	0.057	5.42	0.119	8	6.4*	2.09
12. Pterosaur (<u>Pteranodon</u>)	7.0	4.2	11.5	16	9	~14*	3.81
	7.6	4.6	12.5	11.3	8	~14*	2.46
13. Flying Seed (<u>Zinnonia</u>)	0.15	$6.2x10^{-3}$	3.63	$3x10^{-4}$			0.05
	0.115	$5x10^{-3}$	2.65	$1.75x10^{-4}$			0.04
14. Rogallo Hang Glider	6.58	18.4	2.36	81.5	10	4	4.4
15. "Quicksilver" Hang Glider	9.15	10.75	7.8	118	9.8	7	11.0
16. "Icarus V" Hang Glider	9.75	14.9	6.4	100	10	8	6.7
17. VJ-24 Hang Glider	11.1	15.15	8.15	141	9	9	9.3
18. "Puffin II" HPA	28.4	36.3	22.2	132	8	36**	3.64
19. "Toucan" HPA	37.5	55.8	25.2	240	8.25	40**	4.3
20. "Dumbo" HPA	36.7	44.6	30.2	127	7.45	43.5**	2.85
21. SF-27M Motor Glider	15.0	12.1	18.6	370	34.2	31.5	30.6
22. Schweizer 1-26 Sailplane	12.2	14.9	10.0	270	21.6	21.6	18.1
23. "Std. Cirrus" Sailplane	15.0	10.0	22.5	333	26.2	37.9	26.2
24. ASW-12 Sailplane	18.3	13.0	25.8	412	24.6	43.3	31.7
25. "Nimbus 3" Sailplane	24.5	16.8	35.7	703	22.2	~ 60	41.8
26. Piper PA-18 "Super Cub"	10.76	16.58	7.0	794	32	10	47.9
27. Beech "Bonanza"	10.2	16.8	6.2	1417	48	12	84.4
28. Cessna 310F	10.9	16.3	7.3	2190	56	12	134.4
29. DC-9-30 Jet Transport	28.4	93.0	8.7	$4.9x10^{4}$	190	17	479.5+
30. Boeing 707-320	44.4	283.4	7.2	$1.51x10^{5}$	210	18	444.6+
31. Boeing 747-200	59.6	511	6.95	$3.56x10^{5}$	240	18	584.7+

Note: * Power-off glide ** Flight in ground effect (3m height) + Half fuel weight

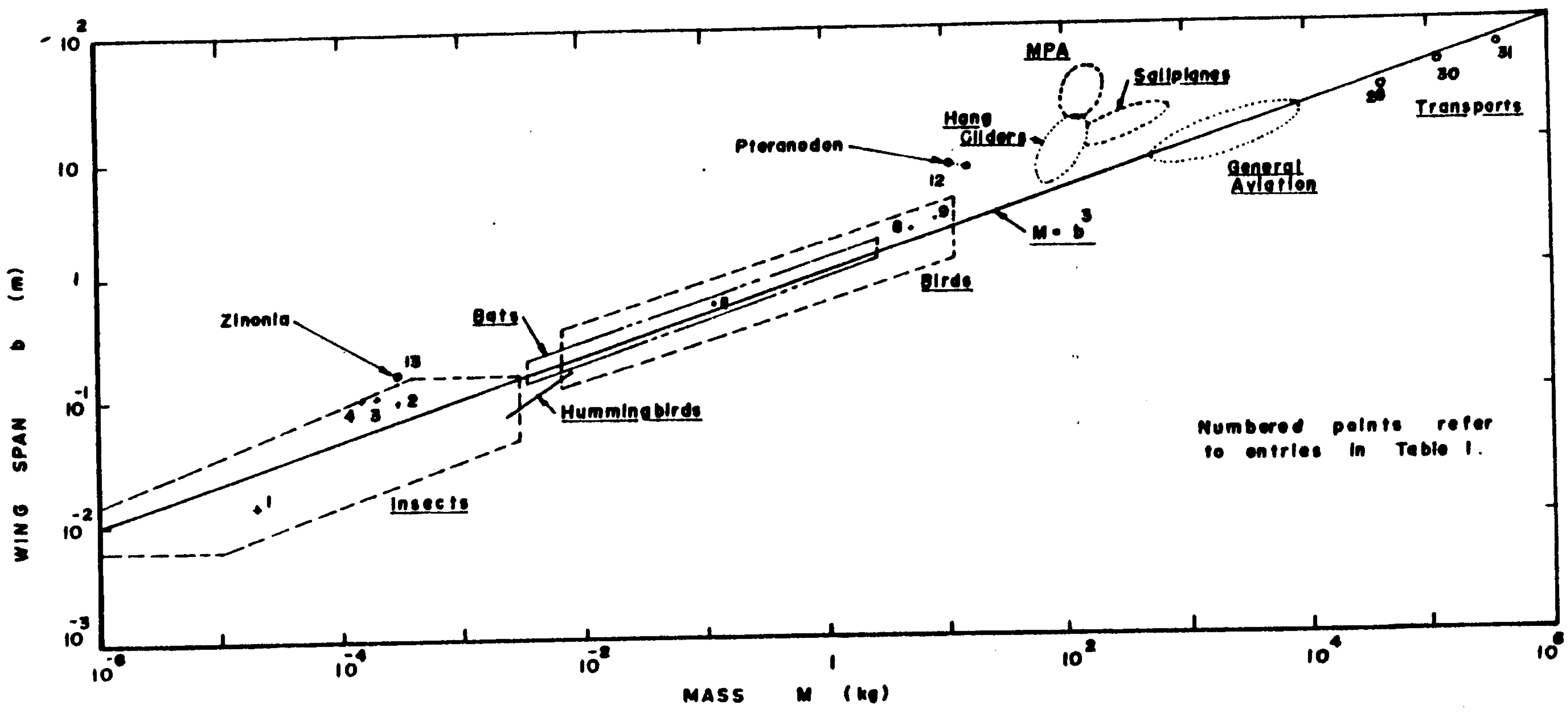


FIGURE 7. VARIATION IN WING SPAN WITH LOADED MASS

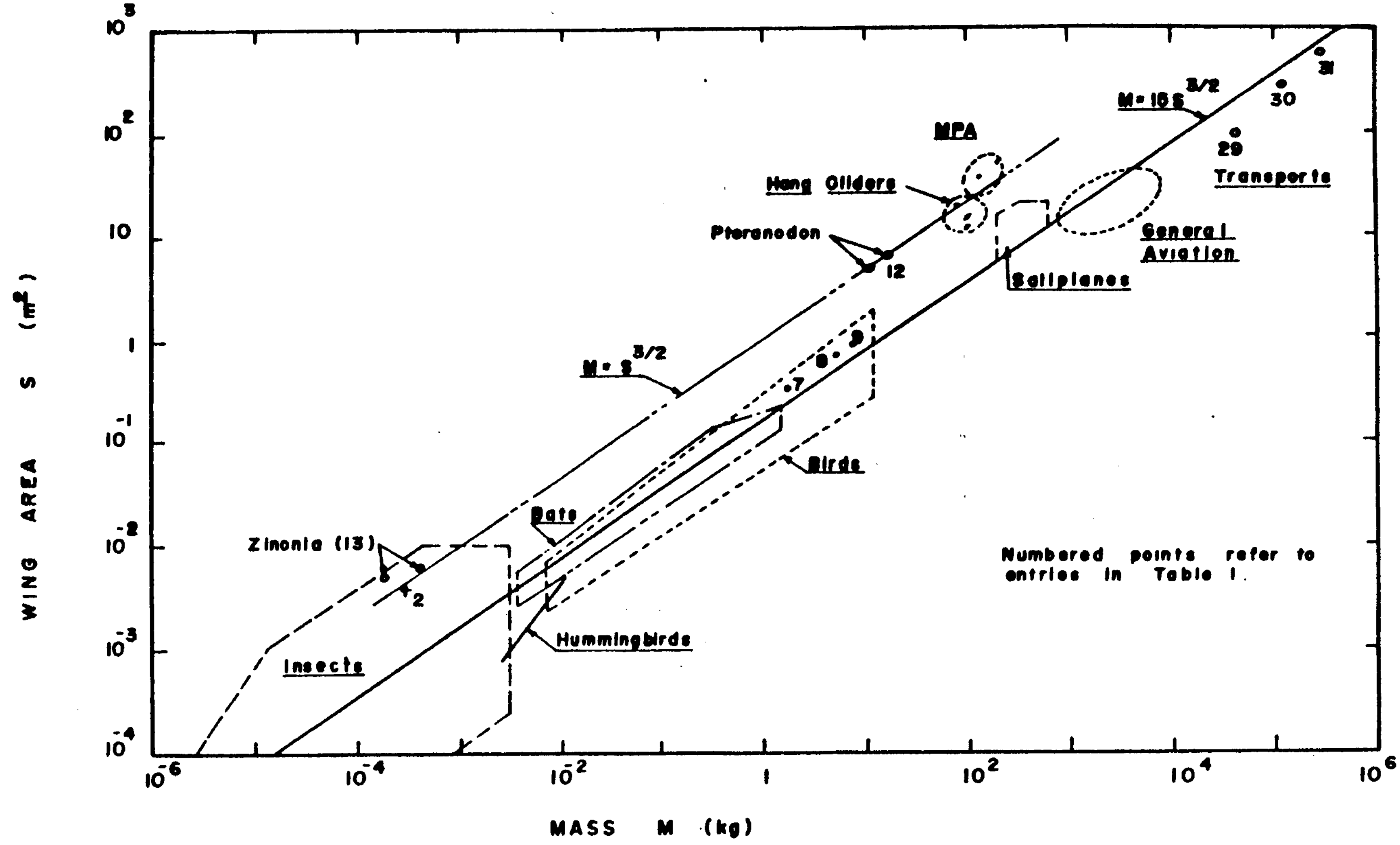
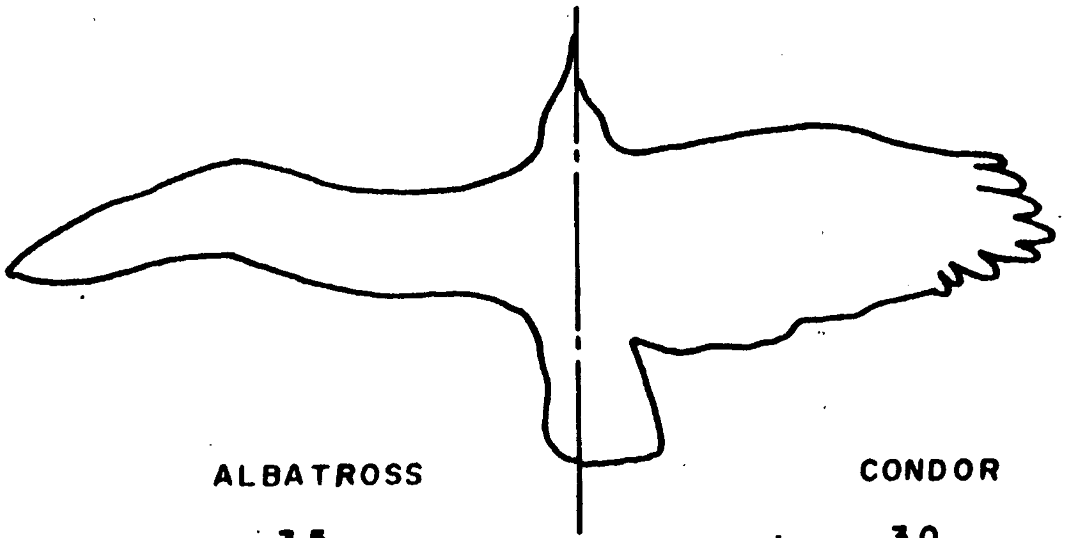


FIGURE 8. VARIATION IN WING AREA WITH LOADED MASS

1.57

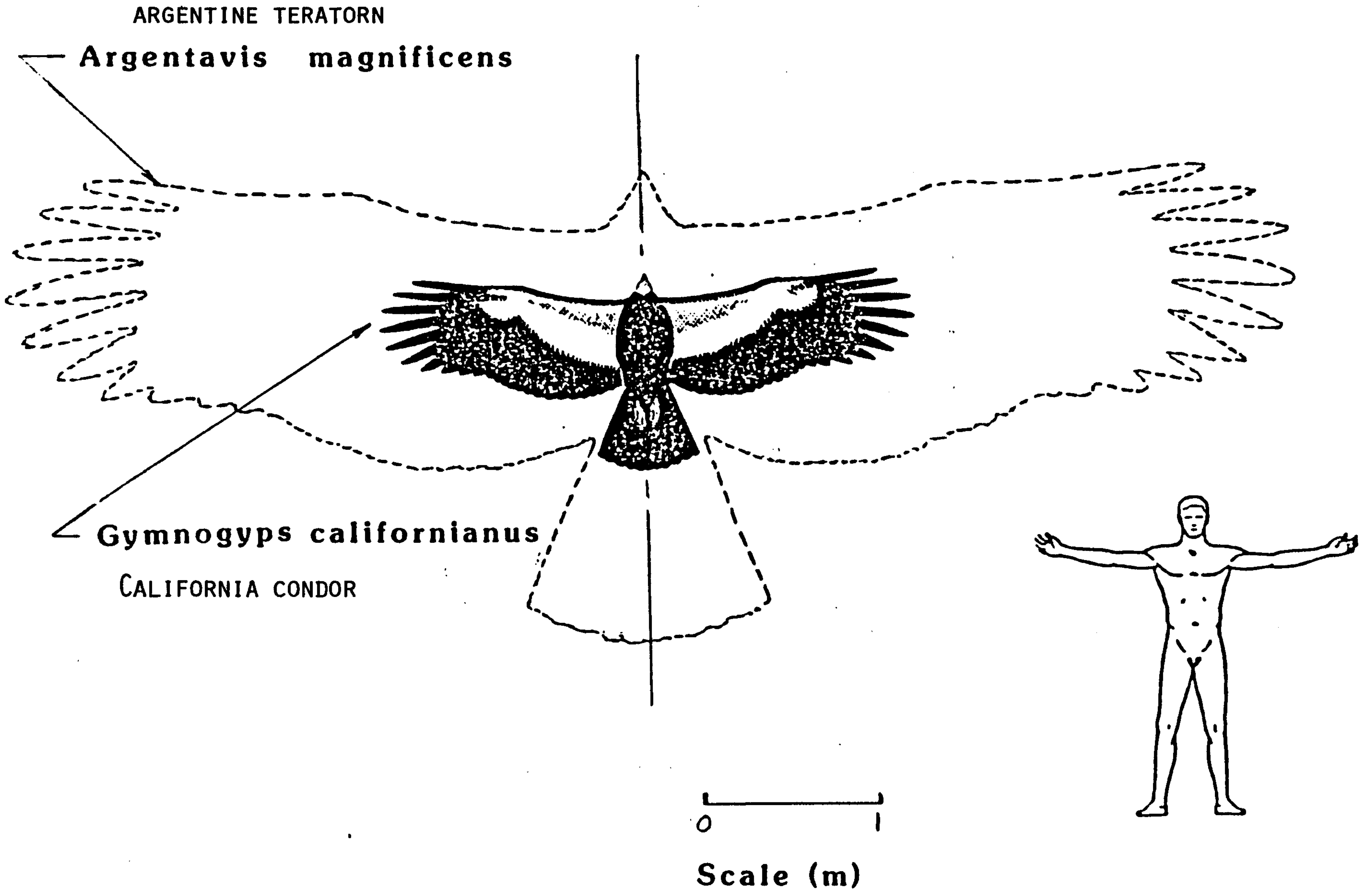
WANDERING ALBATROSS
(Diomedae exulans)

CALIFORNIA CONDOR
(Gymnogyps californius)



	ALBATROSS	CONDOR
WING SPAN (m)	3.5	3.0
WING AREA (m^2)	0.72	1.5
ASPECT RATIO	17	6
MASS (kg)	9.8	10
WING LOADING (N/m^2)	133	65

FIGURE 9. PLANFORM COMPARISON OF LARGE LAND AND SEA SOARING BIRDS


ARGENTINE TERATORN
Argentavis magnificens
Gymnogyps californianus
CALIFORNIA CONDOR
0
1
Scale (m)

1.59

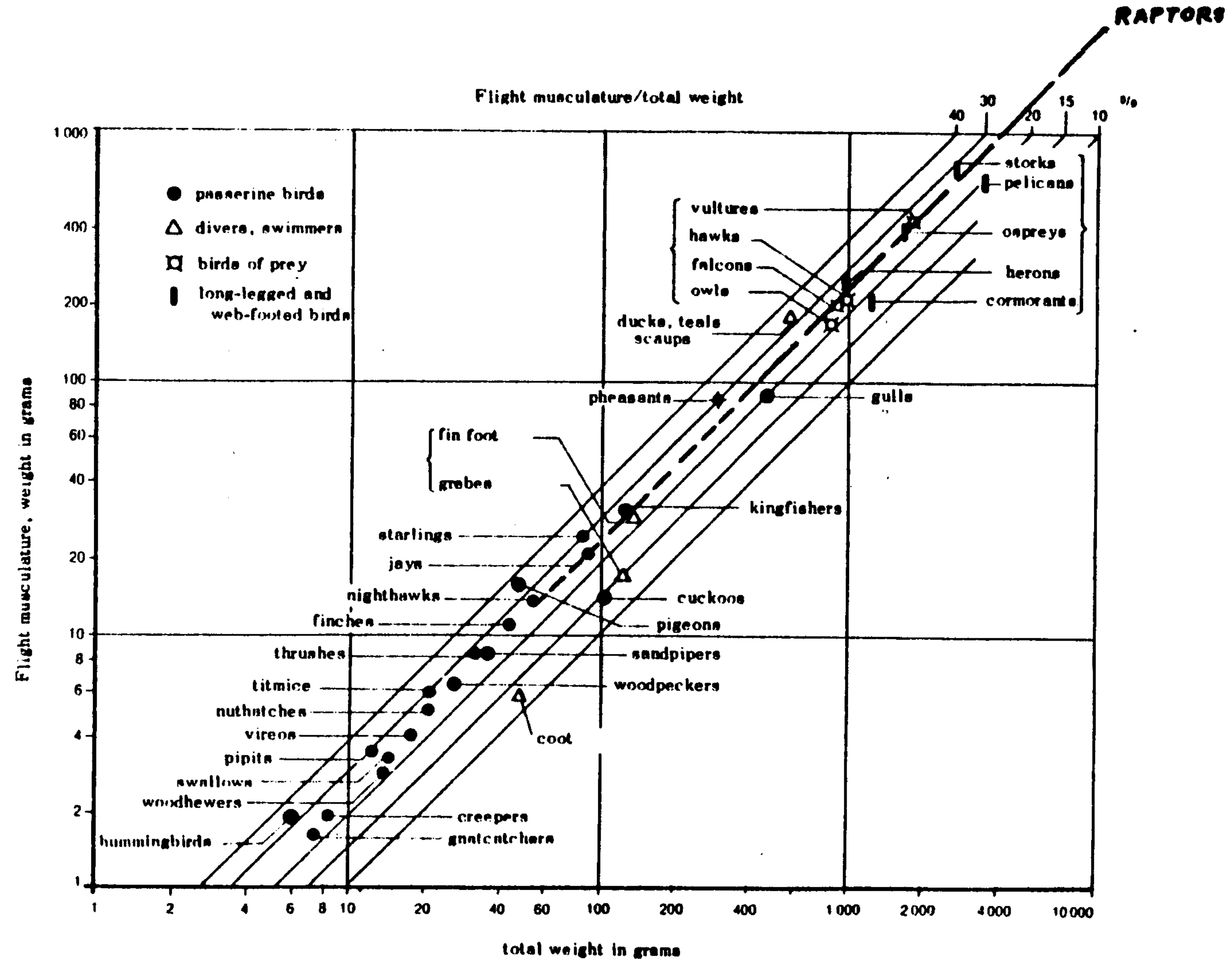


Figure 10. Relationship Between Flight Muscle Weight and Total Weight for a Variety of Birds Including Birds of Prey (Raptors)

1.60

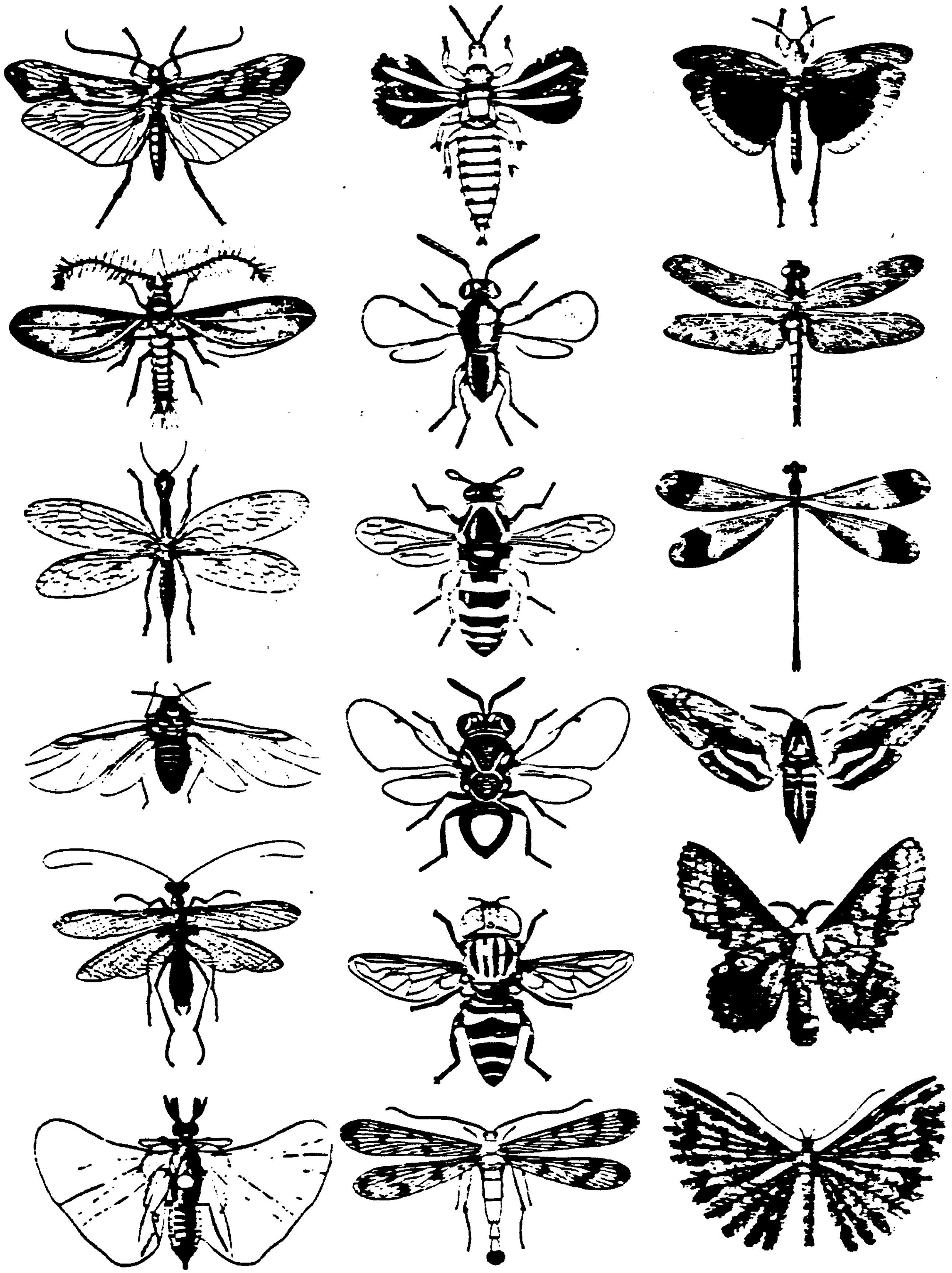


Figure 11. Insect Configuration Matrix

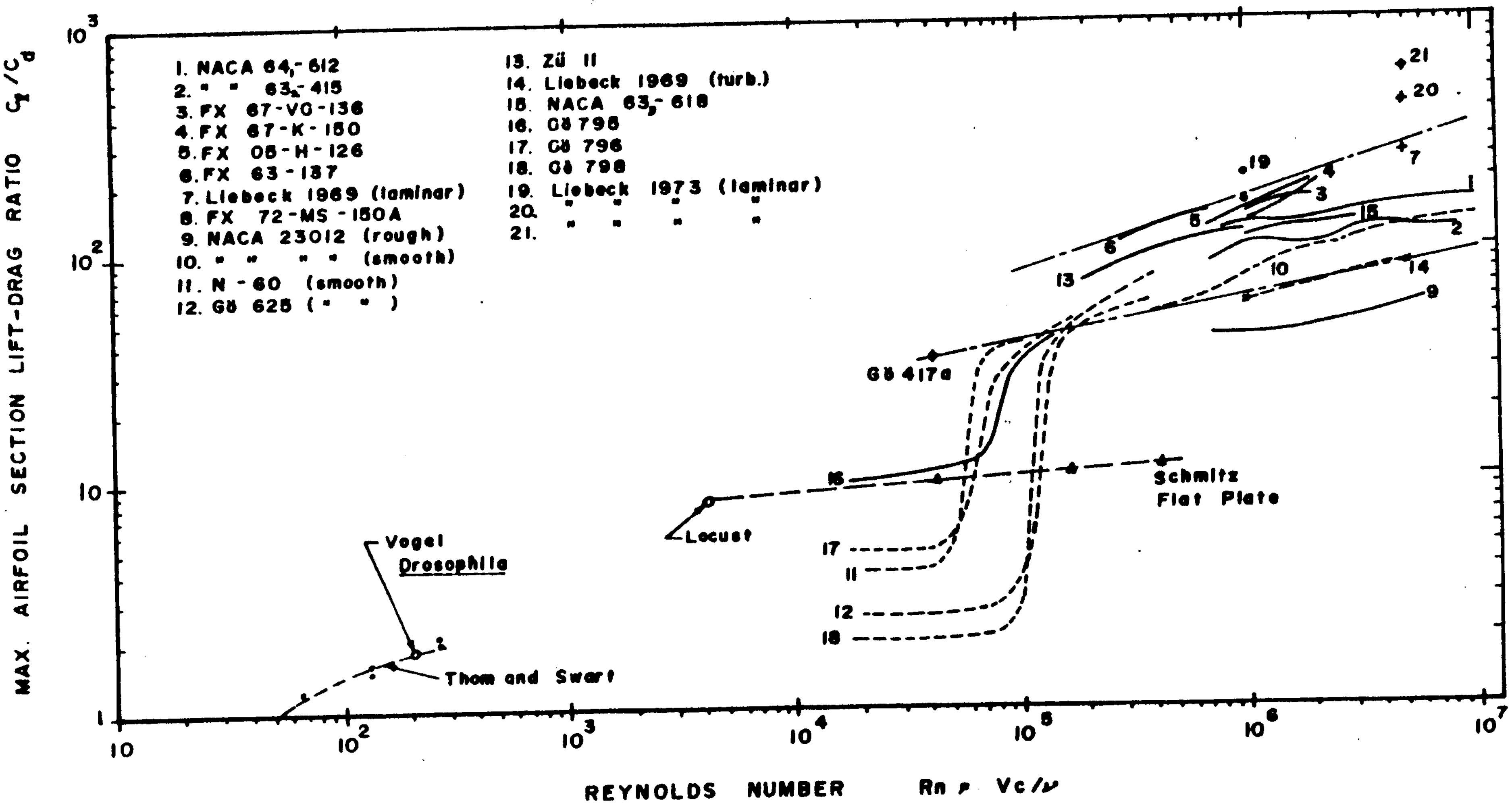


FIGURE 12 VARIATION IN MAXIMUM AIRFOIL SECTION LIFT-DRAG RATIO WITH REYNOLDS NUMBER

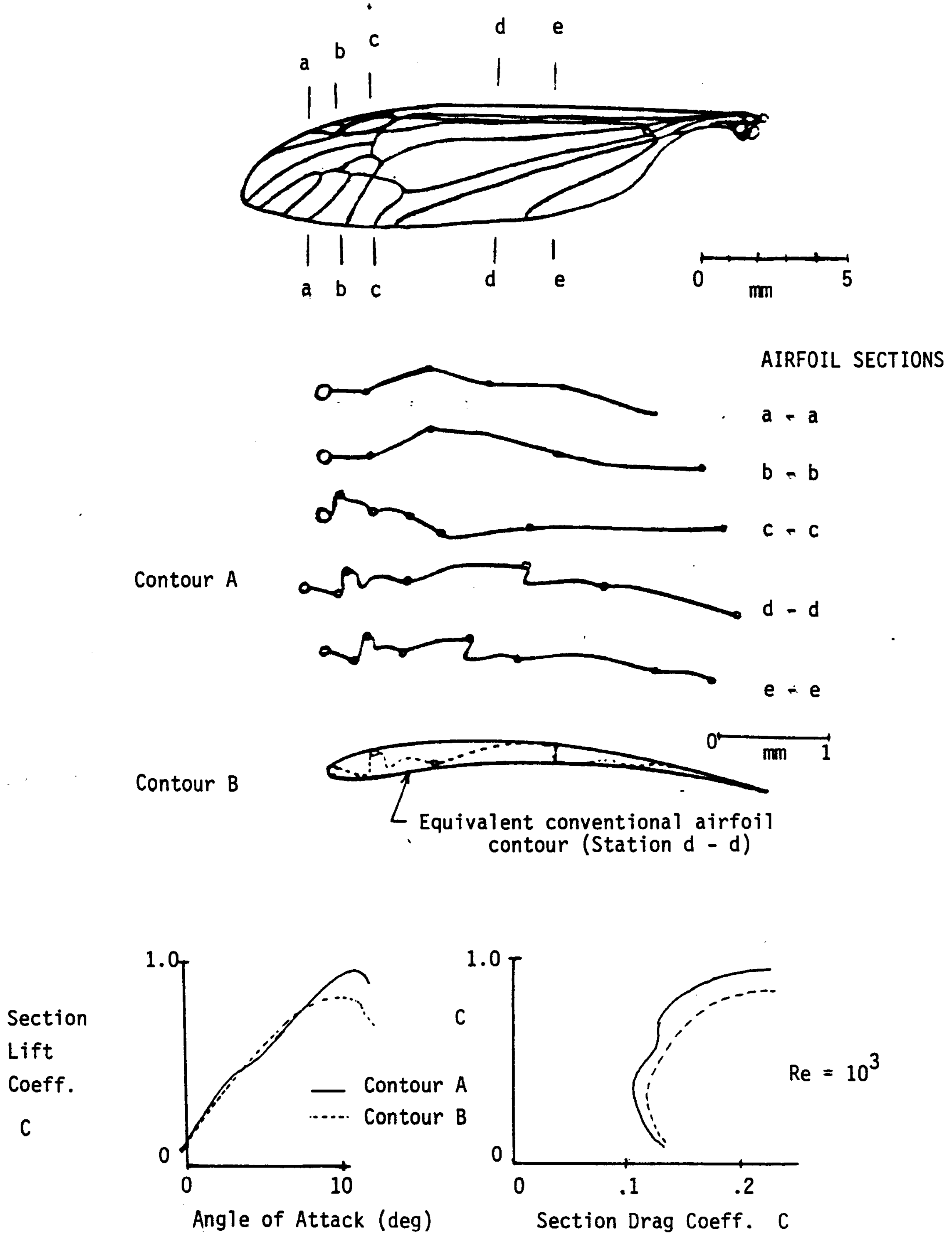


Figure 13. Architecture and Performance of the Wing and Airfoil of a Crane Fly (T. oleracea)

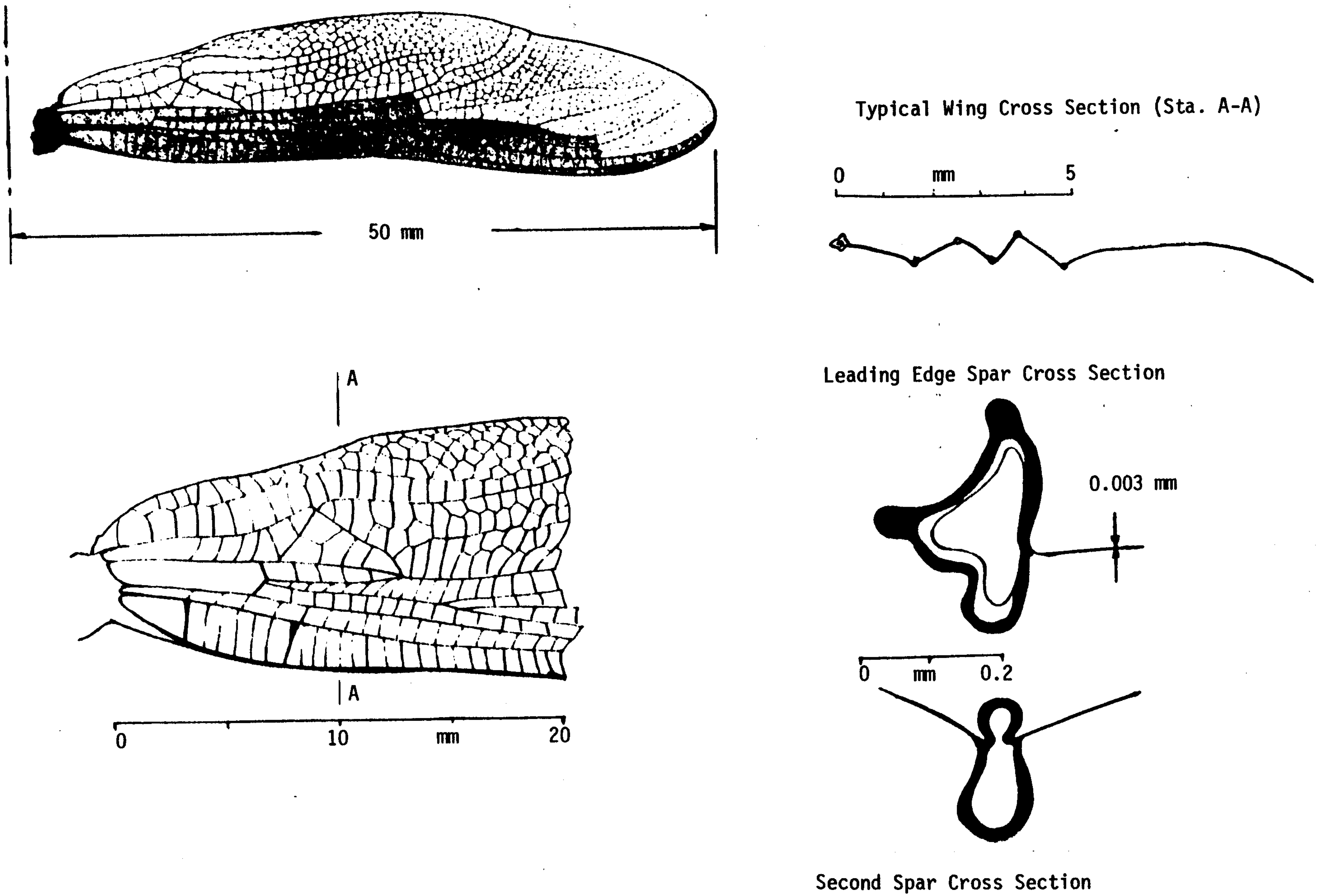


Figure 13 (b). The Wing of a Blue Dragonfly (Aeschna cyanea)

1.64

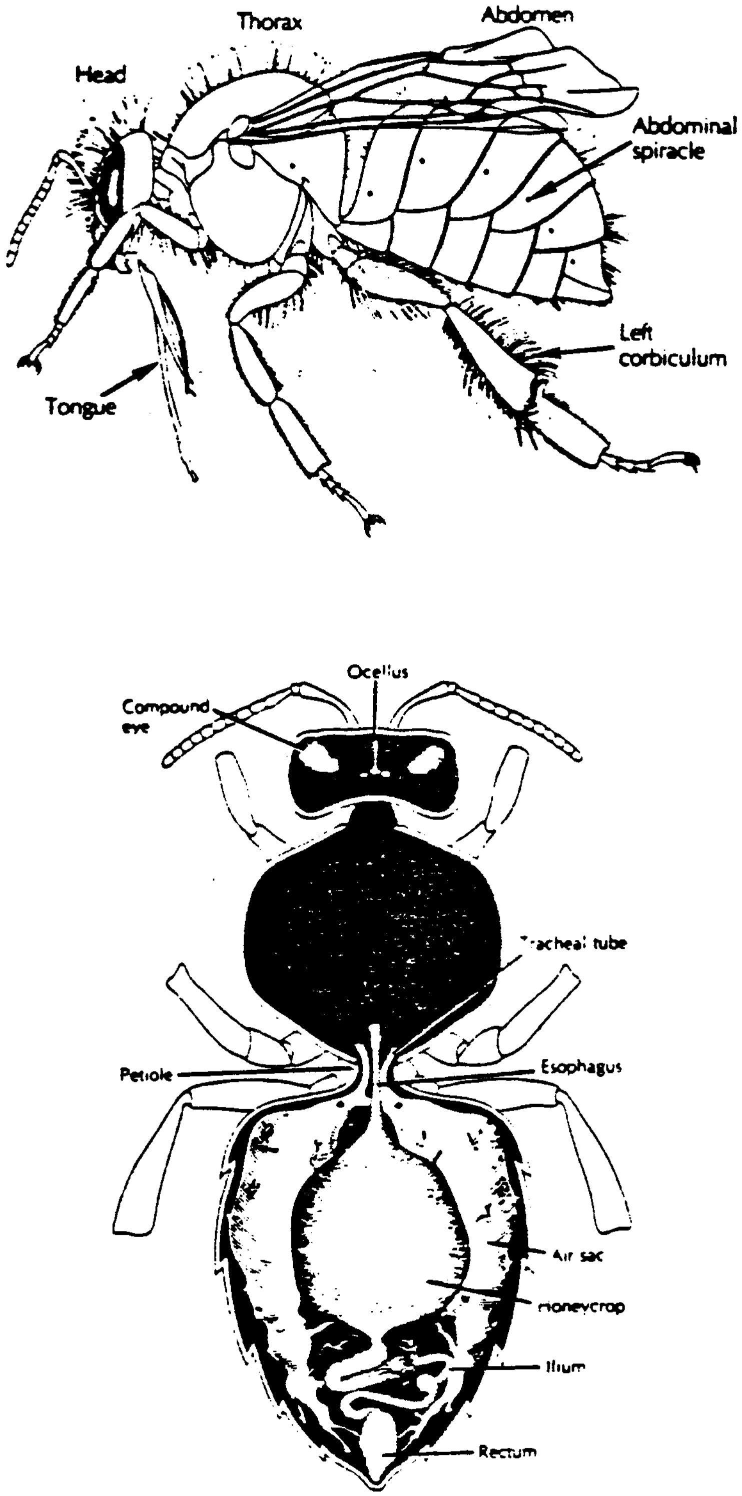


Figure 14. The Architecture of a Bumble Bee (<u>Bombus</u>)

Table 2. FUEL EFFICIENCY

	FUEL	MUSCLE PERFORMANCE (HP/KG)
HUMAN LEG MUSCLE	FAT + SUGAR	0.08 - 0.1
HUMMINGBIRD FLIGHT MUSCLE	SUGAR	1.12 - 1.6
DESERT LOCUST	FAT	0.64 - 1.28
FRUIT FLY	FAT	1.04
BLUE BOTTLE FLY	SUGAR	2.72
HONEY BEE	SUGAR	3.84
AIRPLANE PISTON ENGINE	GASOLINE	3.2 - 6.4

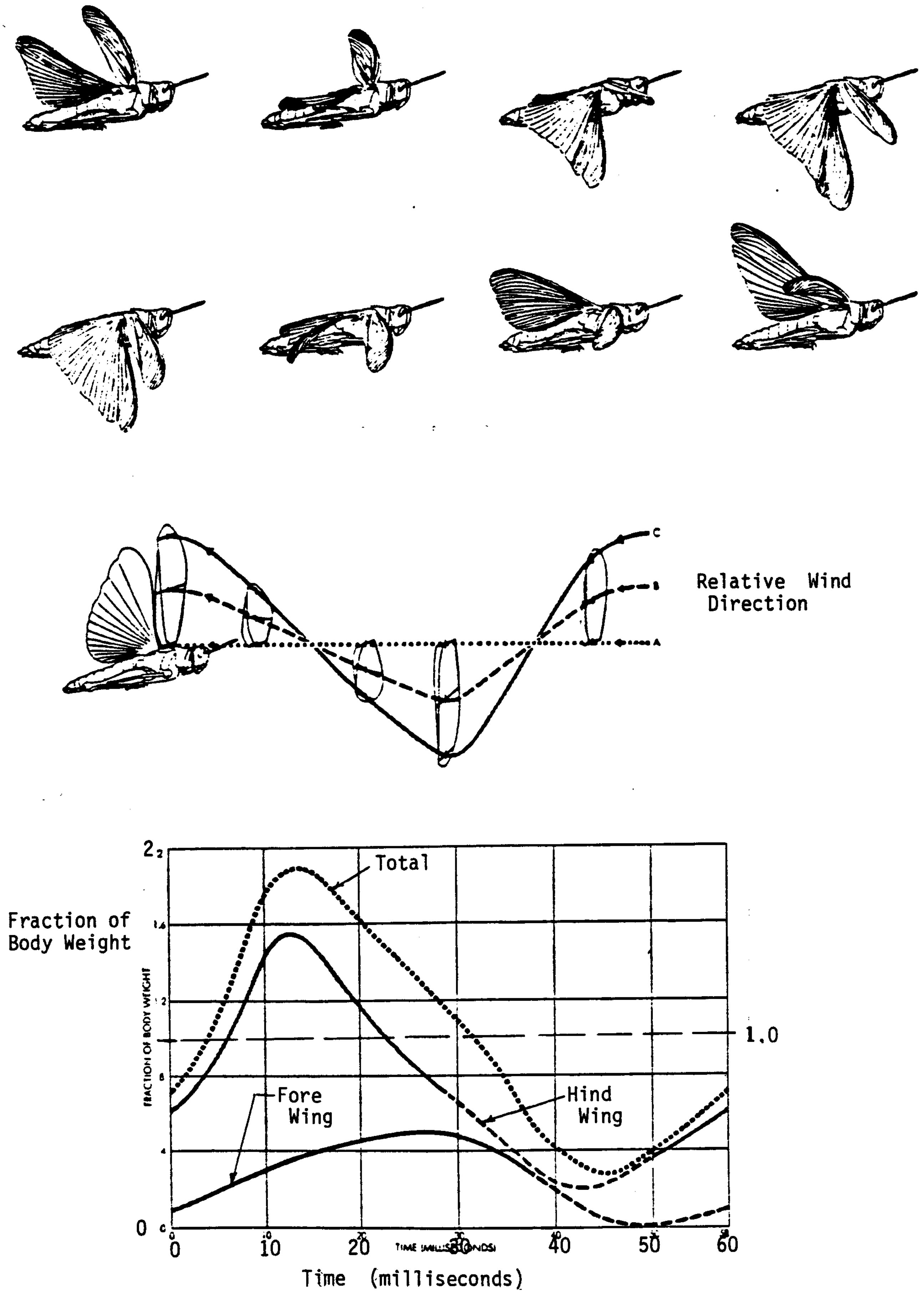
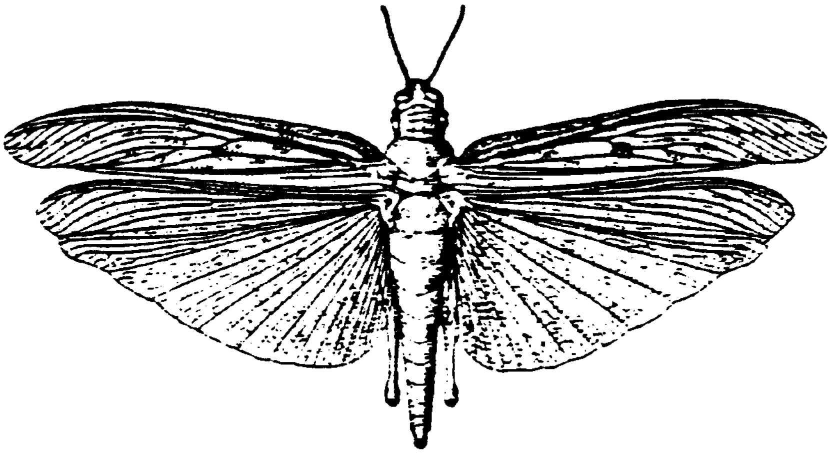
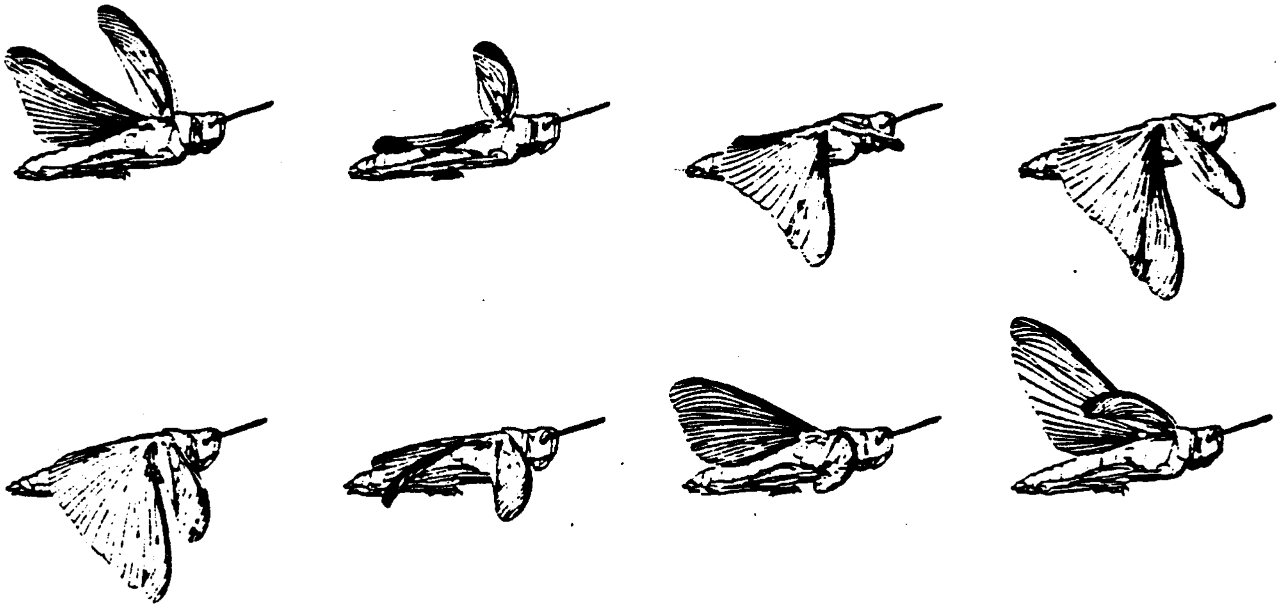


Figure 16. Forces Generated By the Flapping Wings of a Desert Locust.



Desert Locust (Schistocerca gregaria)



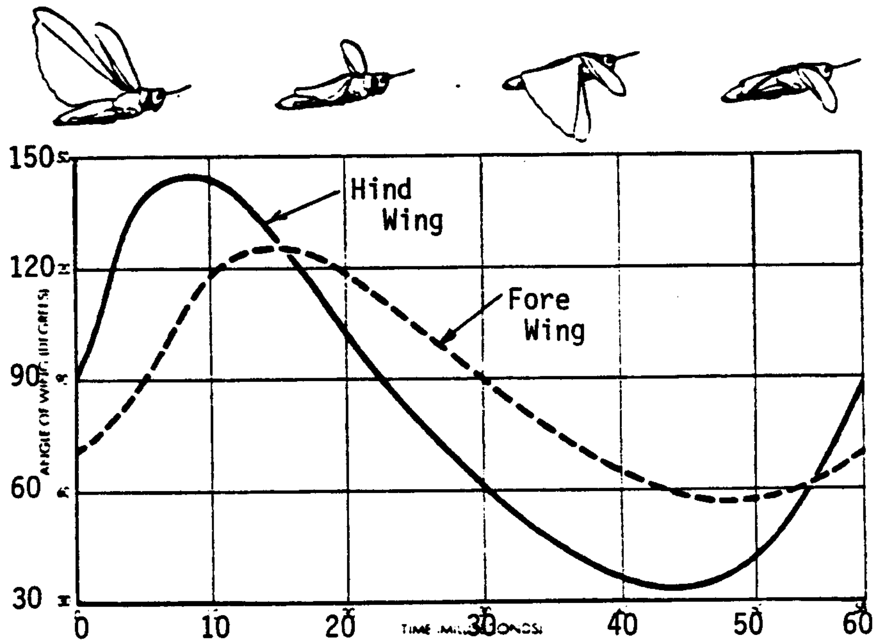


Figure 15. The Wing Flapping Cycle of The Desert Locust

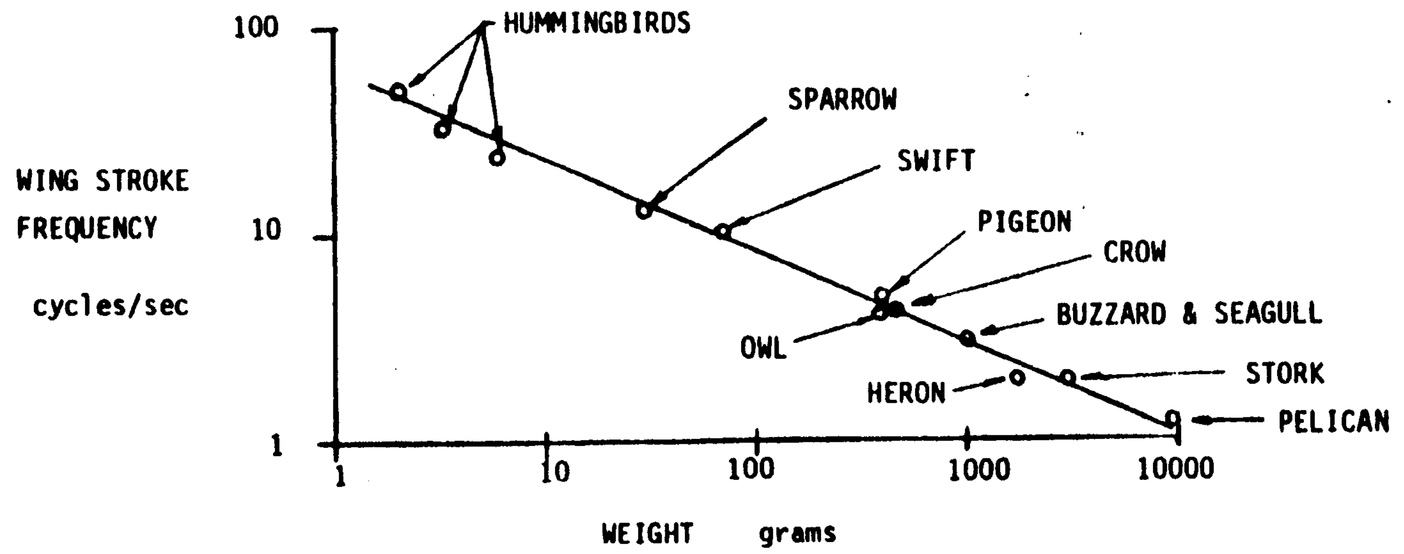


Figure 17. Correlation of Wing Beat Frequency with Weight for Birds.

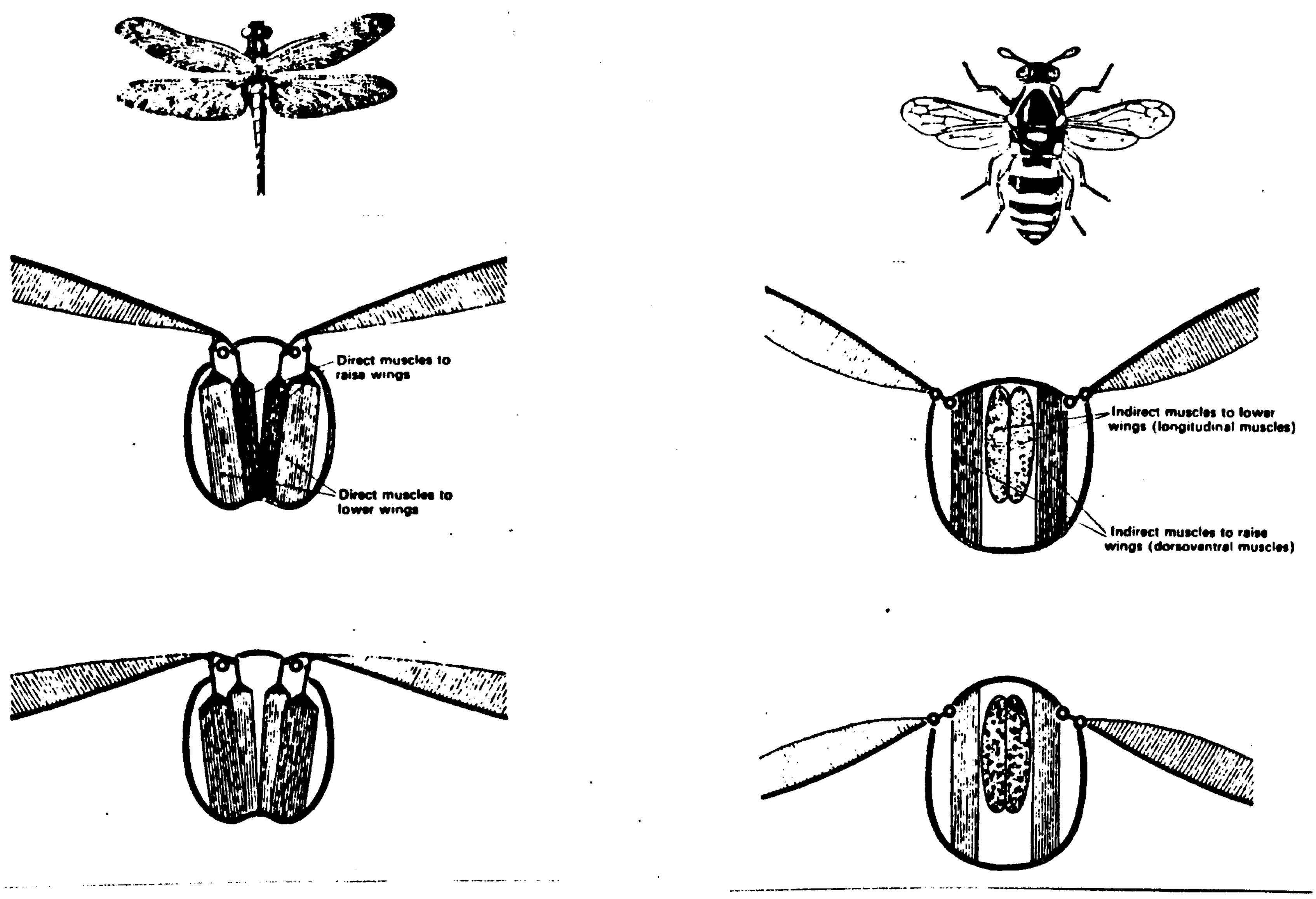


Figure 18. Alternative Flight Muscle Arrangements in the Thorax of Insects

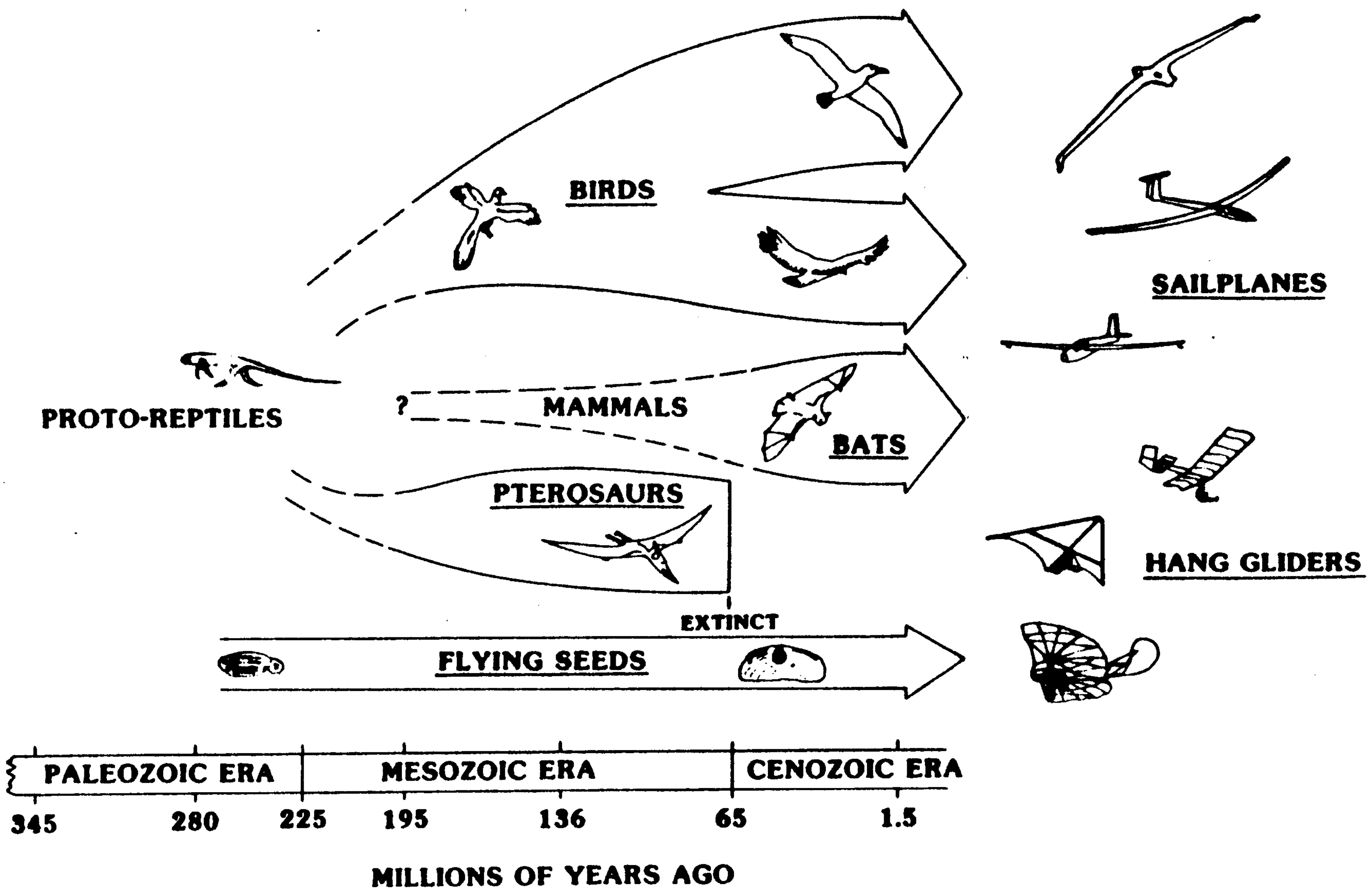


Figure 19. The Evolution of Low-Speed Flying Devices

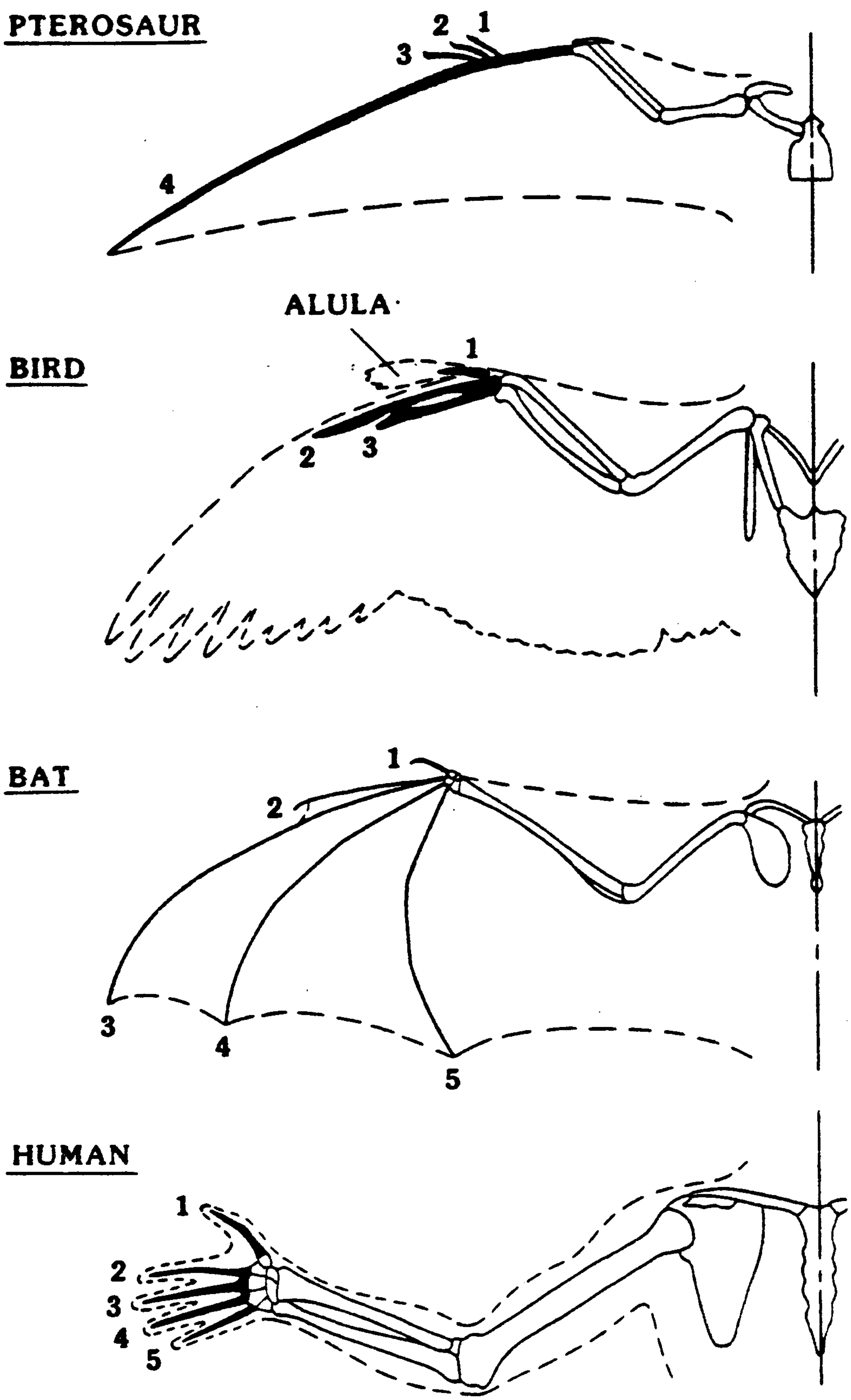
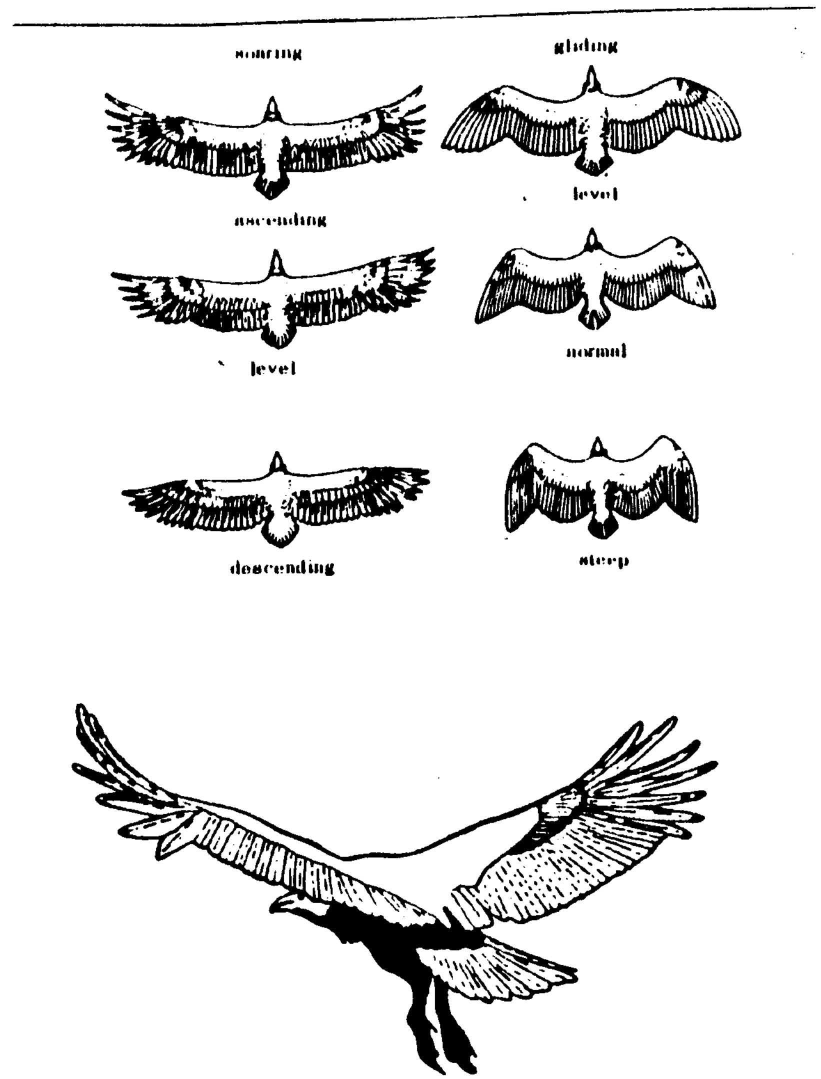
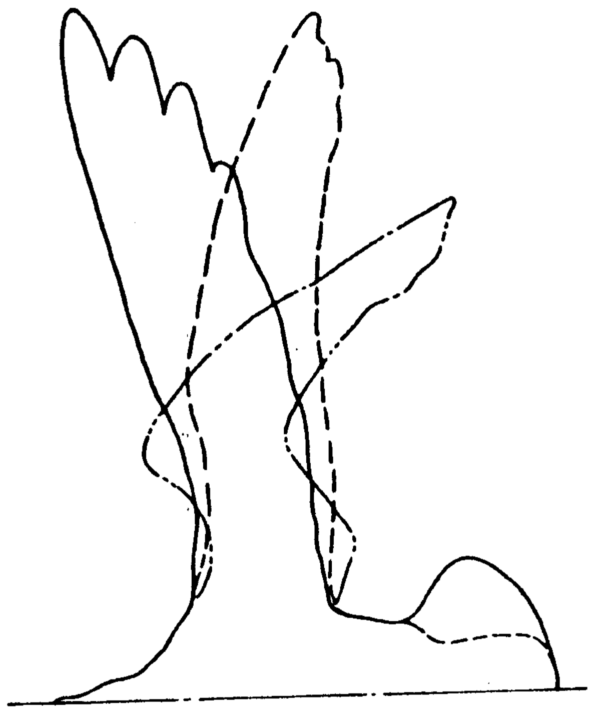


Figure 20. Various Vertibrate Wing Structures Compared to the Human Arm



CONDOR (Gymnogyps californius) IN A LOW SPEED GLIDE WITH GLIDE PATH CONTROL DEVICES (FEET) EXTENDED



VARIABLE GEOMETRY CAPABILITY OF THE FULMAR PETREL (Fulmoris)

Figure 21. The Variable Geometry Capability of Soaring Birds.

Note: All scale bars represent 20 cm

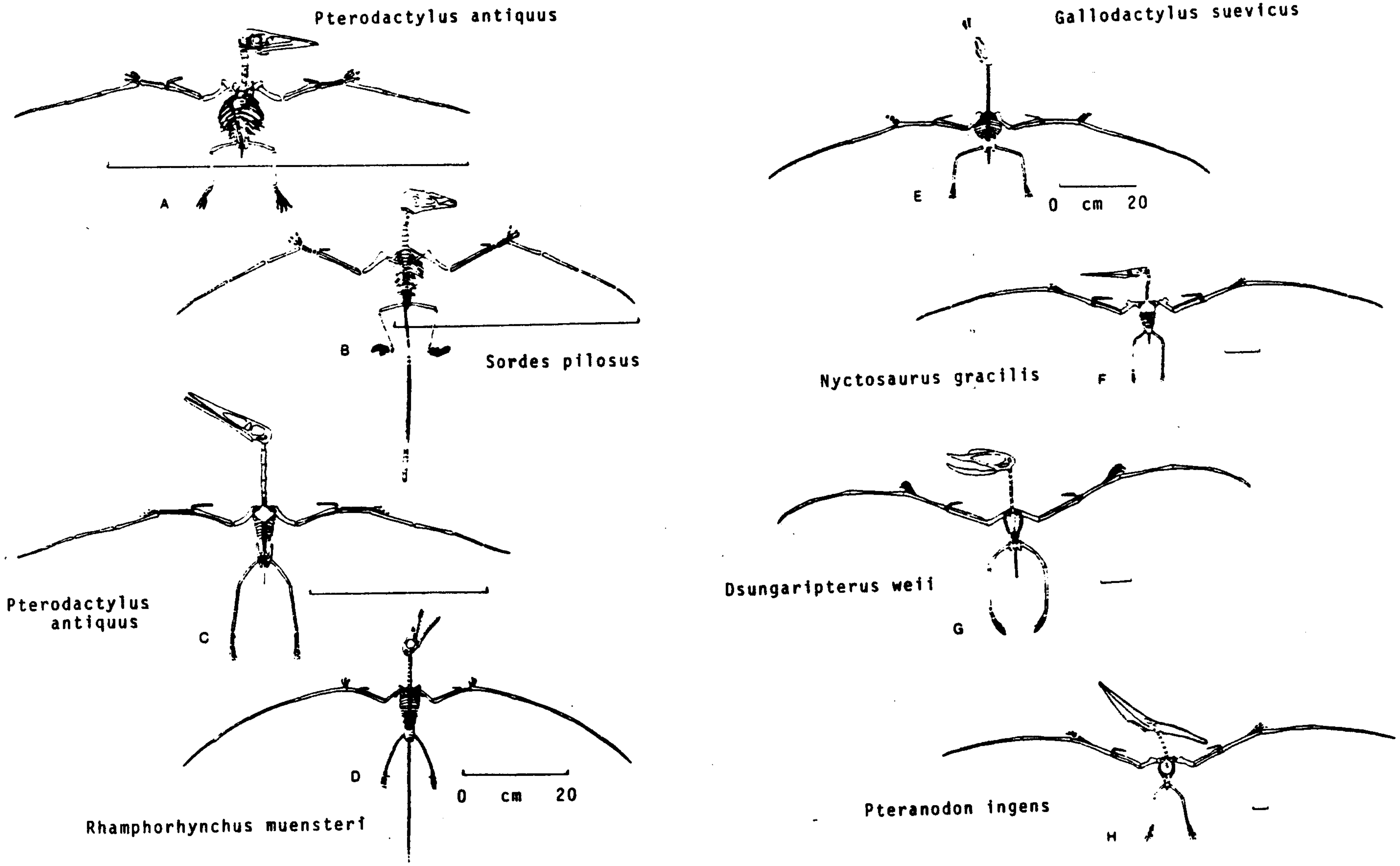


Figure 22. A Representative Sample of Prehistoric Flying Reptiles (Pterosaurs)

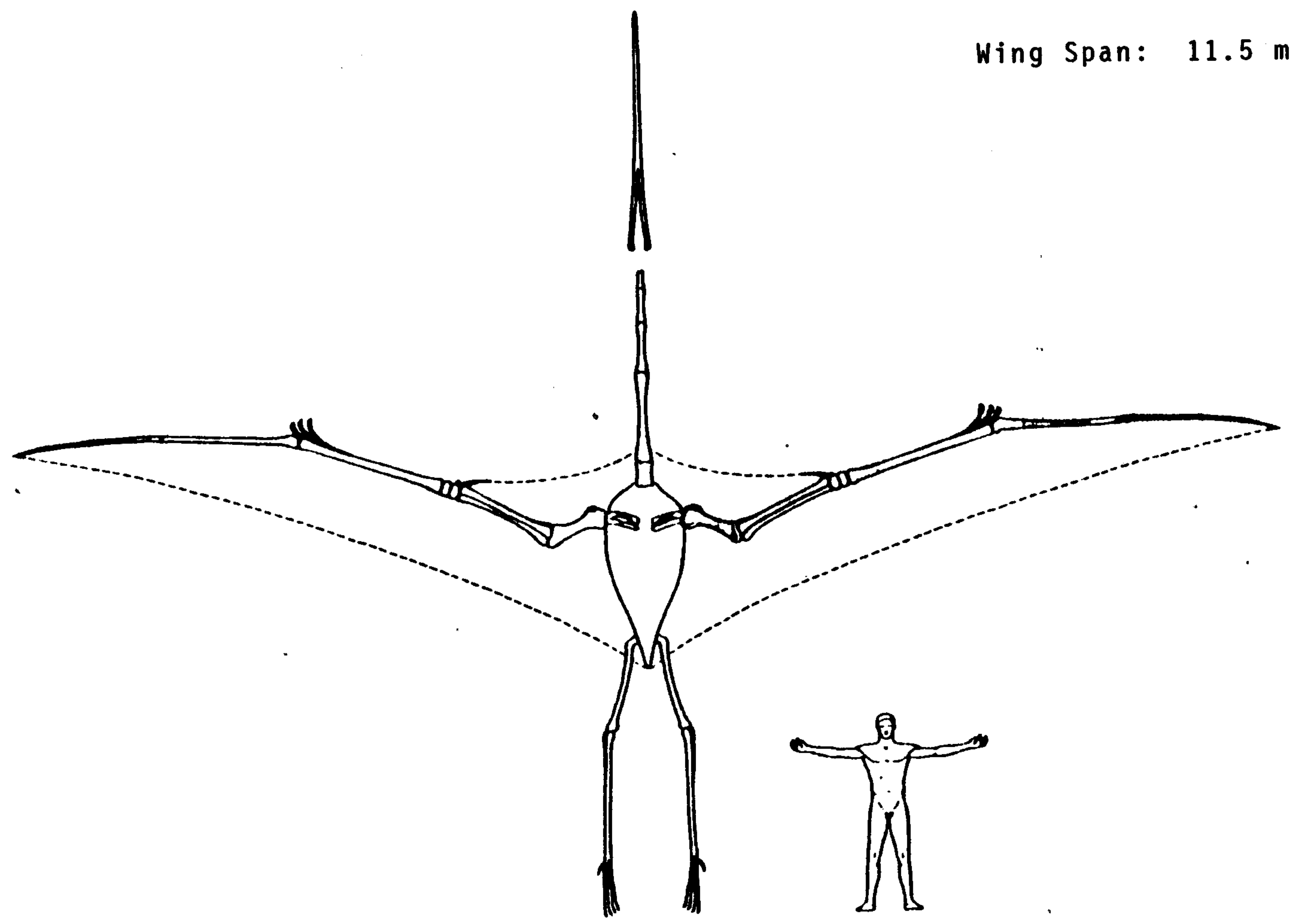


Figure 23. The "Texas Pterosaur" (<u>Quetzalcoatlus</u> <u>northropi</u>)

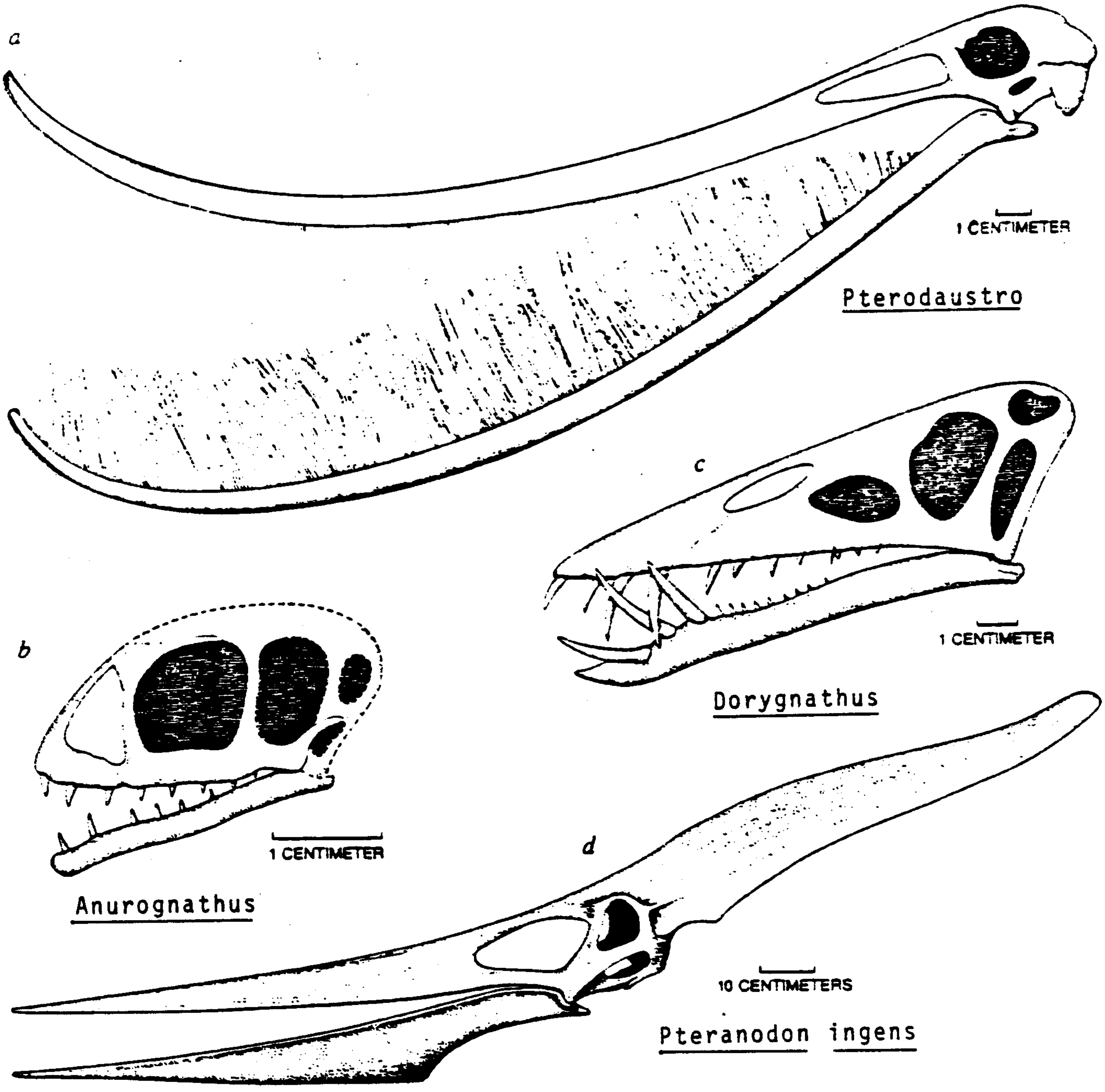
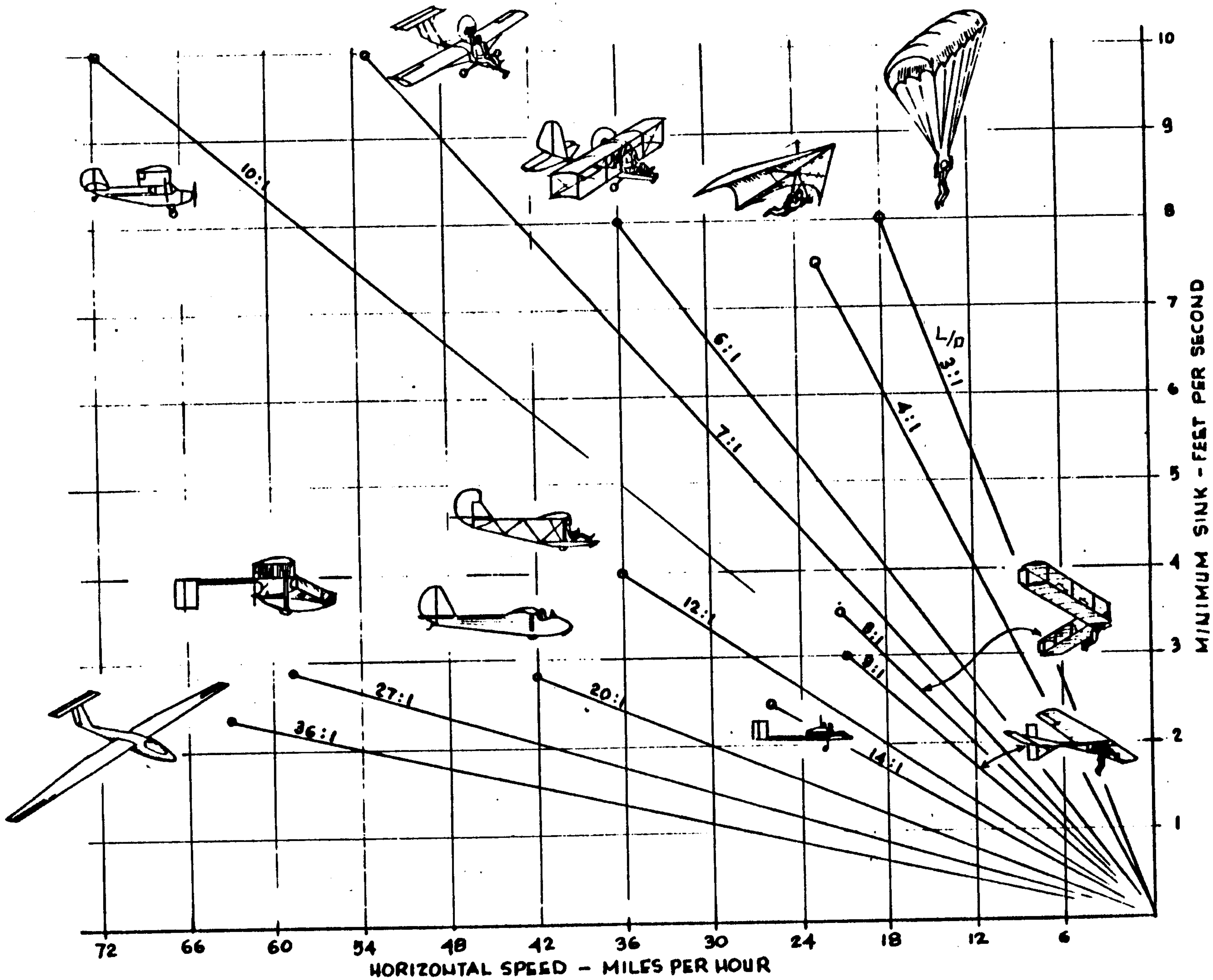


Figure 24. Variations in the Skulls of Several Pterosaurs

②



10:1
6:1
7:1
L/D 3:1
4:1
12:1
8:1
9:1
27:1
20:1
36:1
14:1
10
9
8
7
6
5
4
3
2
1
MINIMUM SINK - FEET PER SECOND
72
66
60
54
48
42
36
30
24
18
12
6
HORIZONTAL SPEED - MILES PER HOUR

1.77

APPENDIX A

Geometric Programming and its Application to Aerodynamic Optimization Problems

Geometric programming is a non-linear optimization technique which, even in the restricted form to be discussed here, offers a number of advantages over more conventional techniques based on classic calculus. Important features of the method are:

- The method is often computationally convenient. A problem may reduced from one of solving m non-linear equations in m unknowns (the optimum values of the design variables), to one of solving m+1 linear equations in m+1 new variables which may have important physical significance in their own right. Once the optimum values of these new (dual) variables are determined, the optimum (primal) variables can be easily determined by routine calculations.

- In a problem with constraints (which may be non-linear), the constraints are reduced to linear form and inequality constraints are handled quite naturally.

- A major restriction of the method is that the primal objective function and any constraints must be specified in the form of polynomials, the terms of which are made up of products of the design variables raised to arbitrary real exponents. In addition, the total number of terms in the primal objective function and the constraints must exceed the number of primal design variables by at least one.

- It is in the nature of the method that a minimum value of the primal objective function is sought. In the restricted case (to be discussed here), where all the terms in the objective function and constraints are positive, and the constraint inequalities are of the correct sense (i.e., ≤ 1), the method guarantees the establishment of a global minimum (that is, in a function which may have several local minima, the extreme minimum will be established).

The simplest way to present the method and to illustrate the features described above, is to describe the history of its development together with appropriate examples. This done, the application of the method to examples important to the discussion in this paper will be demonstrated.

The Historical Development of Geometric Programming

Geometric programming was discovered by Claurence Zener (inventor of the zener diode) of Westinghouse in 1961. Zener (1961) noted that the sum of the component "costs" of a process sometimes can be minimized almost by inspection, provided these "costs" are functions of the product of the variables involved in each cost term, each variable raised to arbitrary (but known) real exponents. The exponents are not restricted to positive integers, but may be any real number, either positive or negative. At this early stage, it was further necessary that the coefficients of each term in the cost function summation be positive. Zener and his colleagues (Duffin, Peterson and Zener, 1967) coined the word "posynomial" to describe these positive polynomial functions. For example:

(A-1) $\Phi(\underline{x}) = x_1^2 x_x^{-1} - 28 x_1^{-1} + 6 x_2^{-3}$ is <u>not</u> a posynomial

while:

(A-2) $\Phi(\underline{x}) = 2 x_1^2 x_2^{-1} + 4 x_2^3 x_3^{1/2} + x_1^{-2} x_2 x_3^{-1/2} + 2 x_2^{-3}$

is a posynomial.

One's immediate instinct when confronted with an objective function like equation (A-2) is to regard the x_i as the independent variables, the optimum values of which may be established by the set of (in this case) N non-linear equations $\partial \Phi / \partial x_i = 0$, i = 1, 2 .. N. Zener pointed out, however, that one could with equal justification regard each term in the objective function $\Phi(x_i)$ as an independent variable and seek to establish the optimal contribution of each term to the total minimum sum. Once the minimum value of the objective function was thus established, and that value was of sufficient interest, then the additional work necessary to determine the optimum values of the x_i could be performed.

Zener's procedure involved forming a dual function of the original primal function. This dual function was constructed by dividing each term in the primal function by a weight factor (the dual variables) and then raising each of these modified terms to the power given by its weight. Each of these new terms, when multiplied together, formed the dual objective function. [Note: In the subsequent terminology of the method, this new formulation which retains the primal design variables, the x_i explicitly, is referred to as the pre-dual function of the primal objective function.] As an example, the dual (or pre-dual) function of eqn. (A-2) is:

$$\text{(A-3)}\quad d(w_j, x_i) = \left[\frac{2 x_1^2 x_2^{-1}}{w_1}\right]^{w_1} \cdot \left[\frac{4 x_2^3 x_3^{1/2}}{w_2}\right]^{w_2} \cdot \left[\frac{x_1^{-2} x_2 x_3^{-1/2}}{w_3}\right]^{w_3} \cdot \left[\frac{2 x_2^{-3}}{w_4}\right]^{w_4}$$

$w_j = w_1, w_2, w_3, w_4$ = dual variables (weights)

The values of the dual variables (weights) are to be chosen such that: (1) the dual function is non-dimensionalized with respect to the x_i (the primal design variables) and (2) the sum of the w_j must equal unity. The first requirement yields a set of orthogonality conditions and the second yields a normality condition. For eqn. (A-3), these conditions would be:

(A-4) Normality: $w_1 + w_2 + w_3 + w_4 = 1$

Orthogonality: $2 w_1 - 2 w_3 = 0$ (x_1)

$- w_1 + 3 w_2 + w_3 - 3 w_4 = 0$ (x_2)

$1/2\, w_2 - 1/2\, w_3 = 0$ (x_3)

When the values of the weights which satisfy the conditions (e.g. eqn. A-4) are substituted into the dual function (e.g. A-3), a pure number results (because of the non-dimensionalizing process). Zener then noted the apparently remarkable fact that this number is the minimum value of the primal function. To demonstrate this, consider the solution (by inspection) of the simple set of linear algebraic equations (A-4):

$$w_1^* = w_2^* = w_3^* = w_4^* = 1/4 \qquad \text{(A-5)}$$

where $(\ )^*$ represents the optimum value.

Then, substituting these values into eqn. A-3 one obtains:

$$d(w_j^*) = \Phi^*(x_i) = \left[\frac{2}{1/4}\right]^{1/4} \cdot \left[\frac{4}{1/4}\right]^{1/4} \cdot \left[\frac{1}{1/4}\right]^{1/4} \cdot \left[\frac{2}{1/4}\right]^{1/4} = 8 \qquad \text{(A-6)}$$

Now, if the value of $d(w_j^*) = \Phi^*$ is of interest, the optimum values of the primal variables can be extracted from the following equations (which can be made linear by taking appropriate logarithms):

(A-7)

$$2\, x_1^2\, x_2^{-1} = w_1^*\, \Phi^* = 2$$

$$4\, x_2^3\, x_3^{1/2} = w_2^*\, \Phi^* = 2$$

$$x_1^{-2}\, x_2\, x_3^{-1/2} = w_3^*\, \Phi^* = 2$$

$$2\, x_2^{-3} = w_4^*\, \Phi^* = 2$$

$$x_1^* = 1 \qquad x_2^* = 1 \qquad x_3^* = 1/4$$

Note: These results can be verified simply by considering the equivalent process using eqn. (A-2) and $\partial\phi/\partial x_i = 0$. Thus:

(A-8)

$$\partial\phi/\partial x_1 = 0 = 4\, x_1\, x_2^{-1} - 2\, x_1^{-3}\, x_2\, x_3^{-1/2}$$

$$\partial\phi/\partial x_2 = 0 = -2\, x_1^2\, x_2^{-2} + 12\, x_2^2\, x_3^{1/2} + x_1^{-2}\, x_3^{-1/2} - 6\, x_2^{-4}$$

$$\partial\phi/\partial x_3 = 0 = 2\, x_2^2\, x_3^{1/2} - \frac{1}{2}\, x_1^{-2}\, x_2\, x_3^{-3/2}$$

It then can be verified that $x_1 = x_2 = 1$, $x_3 = 1/4$ satisfies these equations and that $\phi^* = 8$ in consequence. Higher order derivative tests must then be performed to assure that the solution is in fact the desired minimum (rather than a maximum or a saddle point).

Two factors in the geometric programming procedure demonstrated are important:

- The dual variables w_j^*, in addition to being computationally convenient to extract, have physical significance in their own right. The values of w_j^* obtained from eqn. (A-4) measure the relative importance (or contribution to the total) of each term in the optimal value of the primal objective function. Thus, in the example shown, at the optimum (minimum) value of Φ, each term in eqn. (A-2) is exactly one quarter of the total. In addition, regardless of the values of the numerical coefficients in the terms in eqn. (A-2), if the values of the exponents on the x_i do not change, the optimum weight of each term is invariant and only the numerical value of Φ^* would be altered according to eqn. (A-6).

- In the initial work by Zener, only objective functions with exactly one more term (T) than the number (N) of primal design variables could be considered (i.e., $T \equiv N + 1$). Under these circumstances, there are always N + 1 dual variable w_j, and exactly N orthogonality conditions plus the normality condition which results in N + 1 linear algebraic equations for the N + 1 unknowns (w_j^*). Thus, the w_j^* (if they exist) are always uniquely determined. Such a geometric program is said to be of <u>zero degree of difficulty</u>. The case where $T > N + 1$ (where <u>T - N - 1 = degree of difficulty</u>) is of great practical importance and will be discussed later. It should be noted, however, that a zero degree of difficulty geometric program in which N = 50 (requiring T = 51) is, in principle, no more difficult to solve than one in which N = 1 (and T = 2).

The Theoretical Foundations of Geometric Programming

As described so far, geometric programming seems no more than a parlor trick. It remained for Richard Duffin of Carnegie Tech. to put the method on a firm theoretical foundation. Duffin had been developing a duality theory with application to non-linear programming and when he learned of Zener's work, he was immediately able to make two important contributions.

Duffin observed that if the primal objective function (in posynomial form) was considered a <u>weighted arithmetic mean</u>, and the dual function a corresponding <u>weighted geometric mean</u>, then Cauchy's inequality (Hardy, et. al, 1959) states that the geometric mean is always less than or equal to the arithmetic mean. The simplest example of this relation is:

$$\text{(A-9)} \qquad \overline{U}_{arith} = \frac{1}{2} U_1 + \frac{1}{2} U_2 \geq U_1^{1/2} \cdot U_2^{1/2} = \overline{U}_{geometric}$$

where U_1 and U_2 are positive numbers or functions, and the equality holds if and only if $U_1 = U_2$.

It can be shown that eqn. (A-9) can be generalized such that: Given the set of positive numbers or functions U_n and a set of positive weights w_n, then

$$\text{(A-10)} \qquad \overline{U}_{arith} = w_1 U_1 + .. + w_n U_n \geq U^{w_1} \ldots U_n^{w_n} = \overline{U}_{geometric}$$

with $\overline{U}_{arith} = \overline{U}_{geometric}$ if all U_n are equal.

Now, defining a set of new quantities $V_n = w_n U_n$, eqn. (A-10) becomes:

$$\text{(A-11)} \qquad V_1 + V_2 + \ldots + V_n \geq \left[\frac{V_1}{w_1}\right]^{w_1} \cdot \left[\frac{V_2}{w_2}\right]^{w_2} \cdots \left[\frac{V_n}{w_n}\right]^{w_n}$$

It is readily seen that eqns. (A-2) and (A-3) are in exactly the same form as the left and right hand sides of the inequality eqn. (A-11). Duffin, et. al., (1967) call the inequalities (A-10) and (A-11) geometric inequalities for brevity. The use of the geometric inequality and vector concepts of orthogonality and normality led Duffin to name the overall procedure <u>geometric programming</u>.

Duffins' recognition of the connection between Zener's procedure and the geometric inequality allowed him to show formally that the problem of minimizing a posynomial primal objective function $\Phi(x_i)$ could be transformed into one of maximizing its dual function $d(w_j)$ with respect to its weights (the dual variables, w_j). Duffin could further show that, in fact, the maximum of the dual function was exactly equal to the minimun of the primal function (if a minimum existed), and that the normality and orthogonality conditions imposed on the w_j were necessary and sufficient conditions to establish the optimum w_j^* (Duffin, et. al., 1967).

Duffins' second contribution to Zener's basic procedure was to show how to extend the method to objective functions in which the number of terms in the primal functions exceeds the number of primal design variable by more than one (i.e., problems with $T > N + 1$, or one or more degrees of difficulty). This extension will be demonstrated later by example.

Although, as a consequence of Duffins' initial work, the basic method rested on firm theoretical ground, it remained for Charnes and Cooper (1962) and Duffin and his student Peterson (1964) to raise the method to the status of a fully viable optimization technique by showing how to handle inequality constraints in the form of posynomials less than or equal to unity. All this work, with rigorous proofs, is summarized together with many useful examples and transformations for expressing practical optimization problems in proper posynomial form in the book by Duffin, Peterson and Zener (1967). An even further extension, which permits application of the procedure to general polynomials (at the expense of being unable to guarantee more than the establishment of a stationary point for the primal function) is described by Passy and Wilde (1967, 1968) and in the book by Wilde and Beightler (1969).

Geometric programming, when it works, can yield elegant and often insightful solutions to optimization problems which would be difficult and/or laborious to solve by classical techniques. This is especially true when the problem can be formulated as one of zero degree of difficulty. It is frequently possible by judicious choice of design variables and simplifying assumptions in the objective function formulation to express complex physical problems in a model with no more than one or two degrees of difficulty. As a general rule of thumb, one may say that if the number of degrees of difficulty is equal to or exceeds the number of primal design variables, it is advisable to resort to some other method (e.g., gradient search techniques). With this final limitation in mind it remains to demonstrate the application of the technique to a few simple, but interesting, problems in aeronautics.

<u>Summary of Restricted Geometric Programming</u>

Given a set of positive polynomials of the form:

(A-12) $$\Phi_m (x_n) = \sum_{t=1}^{T_m} C_{mt} \prod_{n=1}^{N} x_n^{a_{mtn}}$$

where: $m = 0, 1, \ldots, M$ (eqn. number)

$t = 1, 2, \ldots, T_m$ (term number)

$C_{mt} > 0$ (coefficients)

$x_n > 0$ (primal design variables)

$n = 1, \ldots, N$

The problem is to <u>minimize</u> the <u>primal</u> objective function $\Phi_o (x_n)$ subject to the M constraints:

(A-13) $$\Phi_m (x_n) \leq 1 \qquad m = 1, \ldots, M$$

The corresponding dual problem is to <u>maximize</u> the dual objective function $d(\underline{w})$ in terms of the T dual variables $\underline{w}$. The dual objective function is:

(A-12) $$d(\underline{w}) = \prod_{m=0}^{M} \prod_{t=1}^{T_m} (c_{mt}\, w_{mo} \quad w_{mt})^{w_{mt}}$$

where: $\max d(\underline{w}^*) = \Phi_o (\underline{x}^*)$ = a minimum if it exists.

The dual variables, $\underline{w}^*$ must satisfy the following relations:

(A-15) <u>Normality Condition</u>: $$\sum_{t=1}^{T_o} w^*_{ot} = 1$$

(A-16) <u>Orthogonality Conditions</u>: $$\sum_{m=0}^{M} \sum_{t=1}^{T_m} a_{mtn}\, w^*_{mt} = 0$$

1.83

(A-17) **Non-Negativity Condition:** $w^*_{mt} \geq 0$

(A-18) **Inequality Constraint:** $w^*_{mo} = \sum_{t=1}^{T_m} w^*_{mt}$

$$m = 1, \ldots, M$$

By convention, define:

(A-19) $w_{oo} = 1$

(A-20)
$$\lim_{w_{mt} \to 0} \left[\frac{C_{mt} W_{om}}{w_{mt}}\right]^{w_{mt}} = 0$$

When the optimum dual variables $\underline{w}^* = w_{mt}$ are found from eqns. (A-15)-(A-20), and the value of $d(\underline{w}^*) =$ $(\underline{x}^*)$ is established from eqn. (A-14), the optimum values of the primal variables $(\underline{x}^*)$ can be determined from the relations:

(A-21)
$$C_{ot} \prod_{n=1}^{N} x_n^{a_{mtn}} = w^*_{ot} \; \Phi_o \, (x^*_n)$$

$$m = 0 \;\; ; \;\; t = 1, \ldots, T_o$$

(A-22)
$$C_{mt} \prod_{n=1}^{N} x_n^{a_{mtn}} = w_{mt}/w_{mo}$$

$$t = 1, \ldots, T_m \;\; ; \;\; m = 1, \ldots, M$$

__Note:__ The value of the quantity $T - (N+1)$ is the number of __degrees of difficulty__ of the geometric program. If the problem has zero degrees of difficulty (i.e., $T = N + 1$) then eqns. (A-15)-(A-20) are necessary and sufficient to uniquely define the required w^*, and the relation $d(\underline{w}^*) = \Phi_o^*$ guarantees that Φ_o^* is __global__ minimum (if one exists). If $T > N + 1$, the additional conditions to establish the $\underline{w}^*$ values must be generated from the condition that: $\max d(\underline{w}) = \Phi_o^* = \min \Phi_o$ (see example 2 to follow).

Some Examples of Restricted Geometric Programming

Example 1. Optimum Speed to Fly for a Glider

Given a gliding device of known geometry and weight, a common question is: At what speed should the device fly to minimize its glide angle. As shown in Figure A-1, if the device has a sufficiently high lift-drag ratio and viscous scale effects are ignored, the problem becomes:

Minimize $\Phi_0 (V) = \gamma \approx \frac{D}{L} = C_{01} V^2 + C_{02} V^{-2}$

where: $D = \frac{1}{2} \rho V^2 C_D S \qquad L = \frac{1}{2} \rho V^2 C_L S \approx W$

$$C_D = C_{D_0} + \frac{k C_L^2}{\pi AR} \qquad AR = b^2/S$$

$$C_{01} = \rho C_{D_0} S/2W$$

$$C_{02} = 2 k W/\rho \pi b^2$$

If no constraints are placed on the problem, the result is an objective function $\Phi_0(V)$ in the form of a simple posynomial with two terms and one design variable and the geometric programming problem has zero degrees of difficulty. This simple problem is easily solved by inspection using Zener's original technique (i.e., forming the pre-dual function of the primal function $\Phi_0(V)$).

Thus:

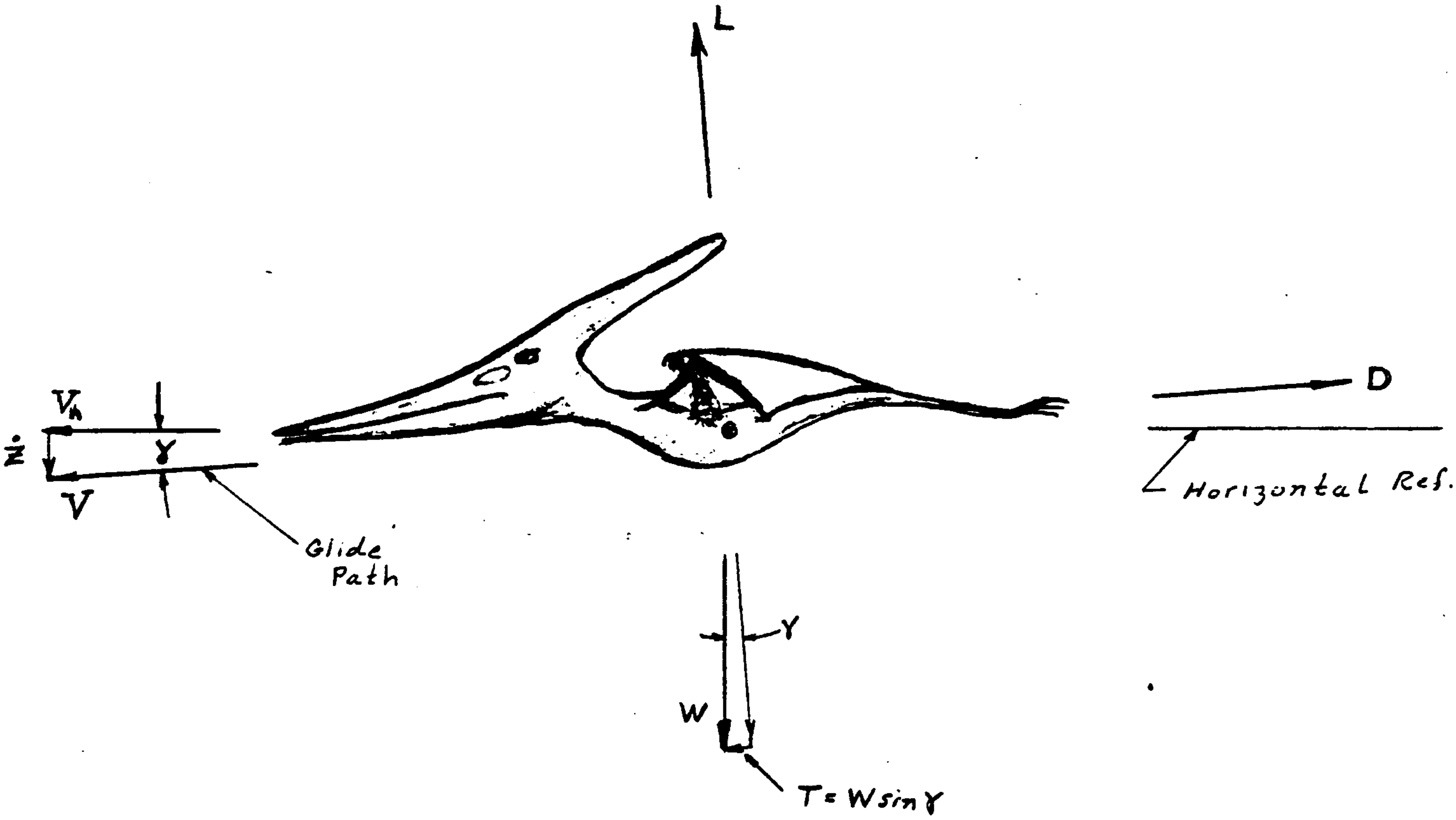
$$d(\underline{w},V) = \left[\frac{C_{01} V^2}{w_{01}}\right]^{w_{01}} \cdot \left[\frac{C_{02} V^{-2}}{w_{02}}\right]^{w_{02}}$$

Observing that the dual function is stationary with respect to $\underline{w}$ when it has been non-dimensionalized with respect to V, the resulting value of $d(w^*)$ will then be the minimum value of $\Phi_0 = \Phi_0 (V^*)$. It is readily seen that the unique values of w_{01}^* and w_{02}^* (subject to the normality condition $w_{01}^* + w_{02}^* = 1$) which remove V from the function $d(\underline{w},V)$ above are:

$$w_{01}^* = w_{02}^* = 1/2$$

It then follows that:

1.85



$$\Sigma \vec{F} = \vec{0} \qquad \Sigma F_n = 0 = L - W\cos\gamma$$

$$\Sigma F_t = 0 = W\sin\gamma - D = T - D$$

$$\therefore\ L/D = \cot\gamma = V_h/\dot{z}$$

If L/D is "large" (eg. $L/D > 6$):

$$L \cong W$$

$$V \cong V_h$$

$$L/D \cong V/\dot{z} \cong 1/\gamma$$

Figure A-1. Forces on a *Pteranodon ingens* in a Rectilinear Glide in Still Air

$$d(\underline{w}^*) = \Phi_o(V^*) = \gamma_{min.} = \left[\frac{C_{01}}{1/2}\right]^{1/2} \cdot \left[\frac{C_{02}}{1/2}\right]^{1/2}$$

or $$L/D_{max} = 1/\gamma_{min} = \frac{1}{2}\left[\frac{\pi\,AR}{k\,C_{D_o}}\right]^{1/2}$$

This is the classic result readily obtained by simple differentiation ($d\Phi_0/dV = 0$) which states that at the condition of maximum $L/D = (1/\gamma_{min})$, the "parasite" drag equals the "induced" drag (i.e., $w_{01}^* = w_{02}^*$) independent of the weight of the glider.

Unlike the classic calculus based solution, the initial result obtained from the geometric program is the optimum value of the objective function itself (i.e., $\Phi_o^* = \gamma_{min}$) rather than the speed at which the glider should fly to achieve it. To obtain the optimum value of the "design" variable (V^*) from the geometric program results one writes:

$$C_{ot}\sum_{n=1}^{N} x_n^{a_{tn}} = w_{ot}^*\,\Phi_o^* \qquad t = 1, 2 \quad ; \quad n = 1 = N$$

Thus: $$C_{01}\,V^2 = w_{01}^*\,\Phi_o^*$$

$$C_{02}\,V^{-2} = w_{02}^*\,\Phi_o^*$$

or $$V^* = \left[\frac{C_{02}}{C_{01}}\right]^{1/4}$$

which again yields the classic results:

$$V^* = V\Big|_{L/D_{max}} = \left[\frac{2\,W}{\rho\,S}\right]^{1/2} \cdot \left[\frac{k}{\pi\,AR\,C_{D_o}}\right]^{1/4}$$

$$C_L^* = \left[\frac{\pi\,AR\,C_{D_o}}{k}\right]^{1/2}$$

As a final result, if one assumes a family of geometrically similar gliders of varying weights (W) (again ignoring viscous scale effects), then according to the square-cube law:

$$S \sim W^{2/3}$$

$$AR = b^2/S = \text{constant}$$

$$b \sim W^{1/3}$$

1.87

Then:

$$V^* \sim \left[\frac{W^2}{b^2\ S}\right]^{1/4} \sim W^{1/6}$$

Example 2. Minimum Power Required to Fly

Given a flying machine of known geometry and weight, flying at constant speed at a given altitude. Referring to figure A-1 and assuming the drag of the device may be expressed by a parabolic variation with lift coefficient, the geometric programming problem of determining minimum power required may be formulated as follows:

$$\Phi_o\,(\underline{x}) = P_{req.} = TV = DV = \frac{1}{2}\,\rho\,V^3\,C_D\,S$$

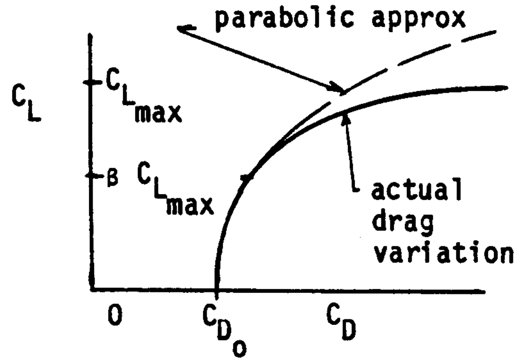
$$\text{with:}\quad W \cong L = \frac{1}{2}\,\rho\,V^2\,C_L\,S \qquad V = \left[\frac{2}{\rho}\,\frac{W}{C_L\,S}\right]^{1/2}$$

$$C_D = C_{D_o} + \frac{k\,C_L^{\,2}}{\pi\,AR} \qquad 0 \le C_L \le \beta\,C_{L_{max}}$$

$$AR = b^2/S \qquad 0 < \beta \le 1.0$$

which results in:

$$\Phi_o\,(C_L) = P_{req} = C_o\left[C_{01}\,C_L^{-3/2} + C_{02}\,C_L^{1/2}\right]$$



subject to the constraint:

$$\Phi_1\,(C_L) = C_{11}\,C_L \le 1$$

where:

$$C_o = \left[\frac{2\,W^3}{\rho\,S}\right]^{1/2}$$

$$C_{01} = C_{D_o}$$

$$C_{02} = k/\pi AR$$

$$C_{11} = 1/\beta\,C_{L_{max}}$$

1-88

In this problem as formulated there are three terms in the objective function/ constrain system, and one design variable (C_L). Thus we have a <u>one degree of difficulty</u> geometric program. The formal solution to the problem then proceeds as follows:

The dual function is:

$$d(\underline{w}^*) = \Phi_o\,(C_L^*) = C_o\cdot(C_{01}/w_{01}^*)^{w_{01}^*}\cdot(C_{02}/w_{02}^*)^{w_{02}^*}\cdot(C_{11}\,w_{10}/w_{11}^*)^{w_{11}^*}$$

where the $\underline{w}^*$ satisfy:

normality: $w_{01}^* + w_{02}^* = 1$

orthogonality: $-\frac{3}{2}\,w_{01}^* + \frac{1}{2}\,w_{02}^* + w_{11}^* = 0$

non-negativity: $w_{01}^*,\ w_{02}^*,\ w_{11}^* \geq 0$

constraint: $w_{10} = w_{11}^*$

In this latter system, the normality and orthogonality conditions yield only two equations for the three unknown dual variables (w_{01}^*, w_{02}^*, w_{11}^*). Thus one can merely write:

$$w_{01}^* = \frac{1}{4}\,(1 + 2\,w_{11}^*)$$

$$w_{02}^* = \frac{1}{4}\,(3 - 2\,w_{11}^*)$$

$$w_{10} = w_{11}^*$$

We note also that the non-negativity condition requires that:

$$w_{02}^* = \frac{1}{4}\,(3 - 2\,w_{11}^*) \geq 0 \qquad w_{11}^* \leq \frac{3}{2}$$

In order to find a third equation to uniquely determine the optimal dual variables, we can make use of the fact that the maximum of $d(\underline{w})$ is the minimum of Φ_o (C_L). Therefore we construct the <u>substituted dual function</u>

$$d(w_{11}^*) = C_o\left[4\,C_{01}/(1+2\,w_{11}^*)\right]^{(1+2\,w_{11}^*)/4}\cdot\left[4\,C_{02}/(3-2\,w_{11}^*)\right]^{(3-2\,w_{11}^*)/4}\cdot\left[C_{11}\,w_{11}^*/w_{11}^*\right]^{w_{11}^*}$$

1.89

Now taking the derivative of logarithm of this function with respect to the unknown weight (w_{11}^*) and setting the result equal to zero, we obtain the desired remaining condition. Thus:

$$w_{11}^* = \frac{1}{2}\left[\frac{3 - (C_{01}/C_{01}\ C_{11}^2)}{1 + (C_{02}/C_{10}C_{11}^2)}\right] = \frac{1}{2}\left[\frac{3-\varepsilon}{1+\varepsilon}\right]$$

where $$\varepsilon = \frac{C_{02}}{C_{01}\ C_{11}^2} = k\ \beta^2\ C_{L_{max}}^2 / \pi R\!\!\!/ \ C_{D_o}$$

Again we note that the non-negativity condition is violated when $\varepsilon > 3$, thus we specify that when $\varepsilon \geq 3$, $w_{11}^* \equiv 0$ which means that in these cases the constraint on C_L becomes inactive. Now, with the value of w_{11}^* established, the remainder of the solution becomes:

$\varepsilon < 3$:

$$w_{01}^* = 1/(1 + \varepsilon)$$

$$w_{02}^* = \varepsilon/(1 + \varepsilon)$$

$$w_{11}^* = \frac{1}{2}\ [(3 = \varepsilon)/(1 + \varepsilon)]$$

$$\varepsilon = \frac{k\ \beta^2\ C_{L_{max}}^2}{\pi R\!\!\!/ \ C_{D_o}}$$

$\varepsilon \geq 3$:

$$w_{01}^* = 1/4$$

$$w_{02}^* = 3/4$$

$$w_{11}^* = 0$$

For the case $\varepsilon \geq 3$, we have merely the classic result that without constraint on C_L, at the condition of minimum power, the "parasite" drag is one-quarter of the total drag and "induced" drag represents the remaining three-quarters of the total.

For the case $\varepsilon \geq 3$, we may then write:

$$\phi_o\ (C_L^*) = P_{min} = 2.556 \left[\frac{W^3}{\rho\ S}\right]^{1/2} \left[\frac{k^3\ C_{D_o}}{(\pi R\!\!\!/)^3}\right]^{1/4}$$

1.90

$$C_L^* = \left[\frac{3\, C_{D_o}\, \pi \mathcal{R}}{k}\right]^{1/2}$$

$$V^* = \left[\frac{4\, W^2\, k}{\rho^2\, C_{D_o}\, \pi \mathcal{R}\; S^2}\right]^{1/4}$$

And again, for geometrically similar devices:

$$S \sim W^{2/3} \qquad \text{with } \mathcal{R} = \text{constant}$$

and thus:

$$P_{min} \sim \left[\frac{W^3}{S}\right]^{1/2} \sim W^{7/6}$$

$$V^* \sim \left[\frac{W}{S}\right]^{1/2} \sim W^{1/6}$$

Example 3: Reynolds Number Scaling in the Context of the Square-Cube Law

Suppose:

$$C_D = C_{D_o} + \frac{k\, C_L{}^2}{\pi \mathcal{R}} \qquad \text{where} \qquad \mathcal{R} = \frac{b}{\bar{c}} = b^2/S$$

$$C_{D_o} = k_o\, R_n^{-n}\, (S_w/S)$$

$$R_n = \frac{V\bar{c}}{\nu}$$

$$\bar{c} = S/b$$

In steady, level flight:

$$\Phi = \frac{P}{WV} = \frac{TV}{WV} = \frac{DV}{WV} = \frac{T}{W} \approx \left[\frac{L}{D}\right]^{-1} \qquad \text{since} \qquad L \cong W$$

$$\text{and} \quad \frac{L}{D} = \frac{q\, C_L\, S}{q\, C_D\, S} = \frac{C_L}{C_D} \qquad \text{and} \qquad V = \left[\frac{2}{\rho}\frac{W}{C_L S}\right]^{1/2}$$

Therefore:

$$\Phi = \frac{C_D}{C_L} = \frac{C_{D_o}}{C_L} + \frac{k\, C_L}{\pi AR} = k_o \left[\frac{S_w}{S}\right] \left[\frac{2}{\rho \nu^2} \cdot \frac{W}{AR}\right]^{-\frac{n}{2}} \cdot C_L^{-(1-n/2)} + \frac{k\, C_L}{\pi AR}$$

Note: $R_n = \frac{V\,\bar{c}}{\nu} = \frac{VS}{\nu b} = \sqrt{\frac{2}{\rho \nu^2}} \cdot \sqrt{\frac{W\,S}{b^2\, C_L}} = \sqrt{\frac{2}{\rho \nu^2}}\ \sqrt{\frac{W}{AR\, C_L}}$

Thus:

$$\Phi(C_L) = C_1\, C_L^{-(1-\frac{n}{2})} + C_2\, C_L$$

where $C_1 = k_o \left[\frac{S_w}{S}\right] \left[\frac{2}{\rho \nu^2} \cdot \frac{W}{AR}\right]^{-n/2}$

$C_2 = \frac{k}{\pi AR}$

$$d(\underline{w}^*) = \left[\frac{C_1}{w_1^*}\right]^{w_1^*} \cdot \left[\frac{C_2}{w_2^*}\right]^{w_2^*}$$

$$w_1^* + w_2^* = 1$$

$$-(1 - \tfrac{n}{2})\, w_1^* + w_2^* = 0$$

$$w_1^* = \frac{2}{4 - n}$$

$$w_2^* = \frac{2-n}{4 - n}$$

Note: (1) No Rn scaling $n = 0$: $w_1^* = 1/2$, $w_2^* = 1/2$

(2) Laminar flat plate $n = \frac{1}{2}$: $w_1^* = 4/7$, $w_2^* = 3/7$

(3) Turbulent flat plate $n = \frac{1}{5}$: $w_1^* = 10/19$, $w_2^* = 9/19$

1.92

In general:

$$\phi_{min} = \phi^* = d(w^*) \sim \left[\frac{S_w}{S}\right]^{2/4-n} \cdot W^{-\left[\frac{n}{4-n}\right]} \cdot AR^{-2\left[\frac{1-n}{4-n}\right]}$$

$$\boxed{L/D_{max} \sim \left[\frac{W^n\, AR^{2(1-n)}}{(S_w/S)^2}\right]^{1/(4-n)}}$$

Then: 1. No Rn scaling $n = 0$: $L/D_{max} \sim AR^{1/2} = AR^{0.5}$

2. Laminar flat plate $n = 1/2$: $L/D_{max} \sim W^{1/7} AR^{2/7} \cong W^{0.14}\, AR^{0.29}$

3. Turbulent flat plate $n = 1/5$: $L/D_{max} \sim W^{1/9}\, AR^{8/19} \cong W^{0.05}\, AR^{0.42}$

$$C_L^* \sim W^{-\frac{n}{4-n}}$$

$$V^* \sim \left[\frac{W/S}{C_L^*}\right]^{1/2} \sim \left[\frac{W^{1/3}}{C_L^*}\right]^{1/2} \sim W^{(2+n)/3(4-n)}$$

$$P^*_{req.} = \frac{W\,V^*}{L/D_{max}} \sim [W]^{\frac{14-5n}{3(4-n)}}$$

(For constant AR)	No Scaling ($n = 0$)	Lam. Flat Plate ($n = 1/2$)	Turb. Plate ($n = 1/5$)
C_L^* ~	constant	$W^{-1/7}$	$W^{-1/19}$
V^* ~	$W^{1/6}$	$W^{1/7}$	$W^{11/57}$
P^*_{req} ~	$W^{7/6}$	$W^{23/21}$	$W^{65/57}$
	↓	↓	↓
	$W^{1.17}$	$W^{1.10}$	$W^{1.14}$

1.93

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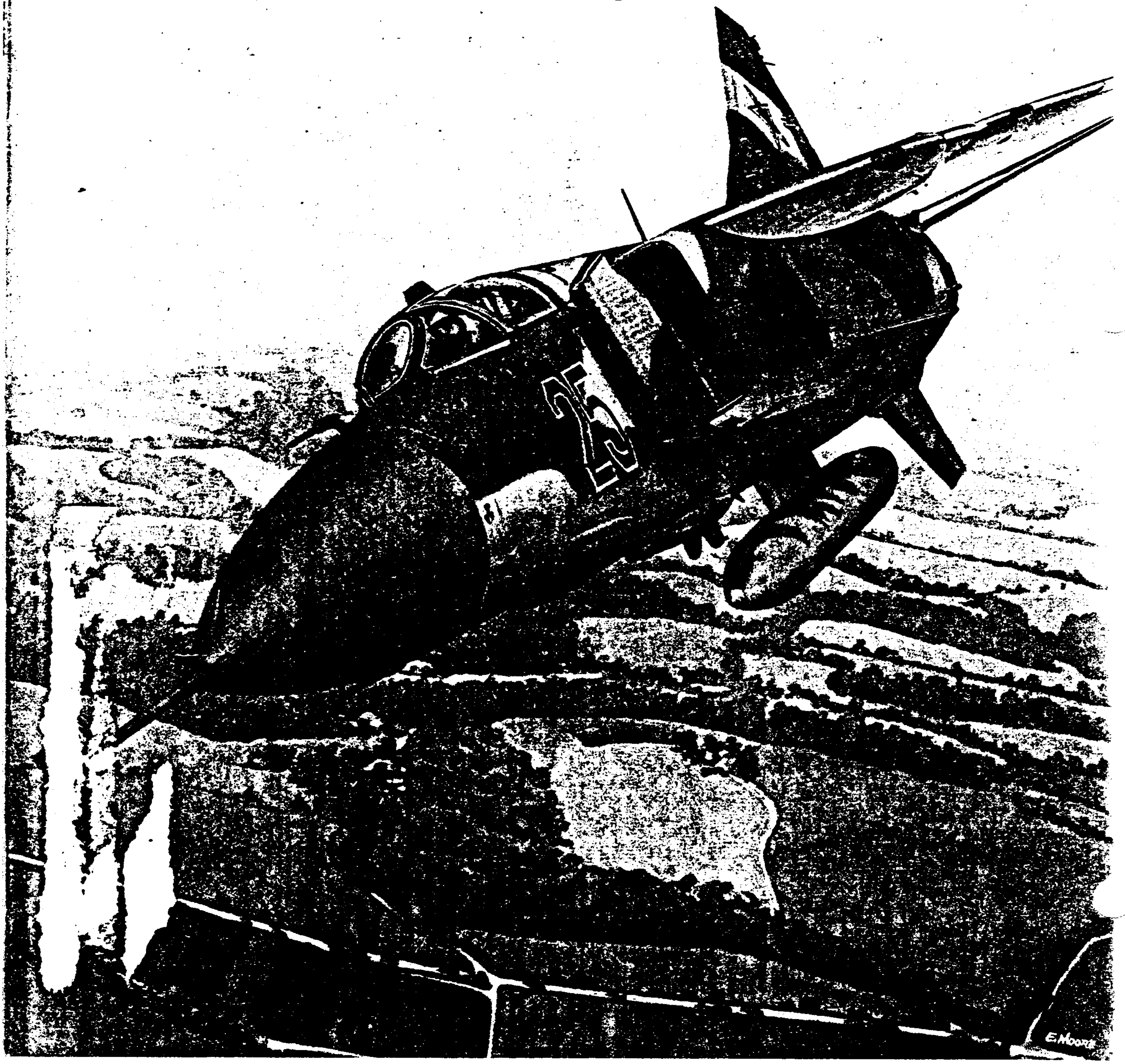
SOVIET MILITARY AIRCRAFT DESIGN



SOVIET PRACTICE IN DESIGNING & PROCURING MILITARY AIRCRAFT

By RICHARD D. WARD
General Dynamics/Ft. Worth Div.

Soviet aircraft procurement presents both an example and a contradiction of the Communist political system. The designs generated in the Ministry of Aircraft Industry are controlled by the bureacracy of a highly centralized government. However, the design bureaus, like U.S. aircraft firms, exist in a highly competitive environment offering financial rewards for success. Soviet designs are judged by their simplicity, maintainability, and ease of manufacture. The result of the incentives and requirements for success are that each generation of aircraft is derived from the previous, using only proven and approved technology. This article cites prominent examples of Soviet military aircraft procurement to show its evolution since World War II.

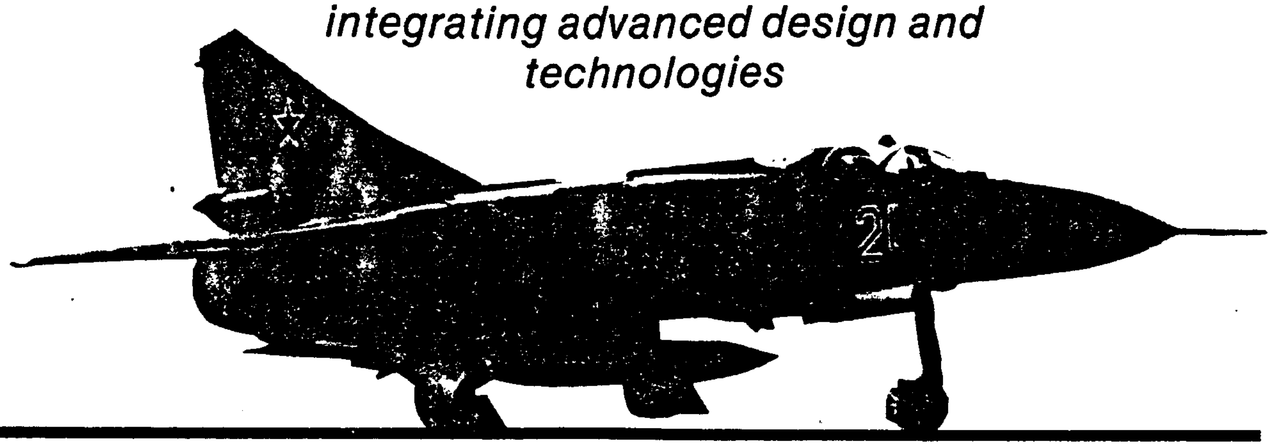
Military aviation enjoys a special position in the USSR's military/industrial sector. Understanding this position requires some knowledge of the Soviet governmental system, the organization of the aircraft industry, and the methods of aircraft procurement. The Soviet government is obviously more complex than can be thoroughly treated here. This discussion thus treats only areas pertinent to aviation, and, in the same vein, the involved post-war history of Soviet aviation can be given here only as an abridgement. But this limited treatment should serve to reveal the main lines of Soviet practice.

Aircraft procurement in the Soviet Union is a compromise between state-controlled and open-market economics. Under Communist doctrine, all aspects of manufacturing should be regulated through centralized economic planning. In the civilian sector, this regulation holds true. Industrial output is established by a form of centralized planning directing that production goals take precedence over customer needs, but in military procurement the customer predominates. This fundamental difference shows the top priority given to national defense by Soviet leadership. To ensure that the performance of weapons meets this defense policy, military customers are allowed to set standards of quality that the defense industry must satisfy. Through these standards the customer controls the quality of the product and can reject the inferior.

The government of the Soviet Union is an interlocking organization of the Communist Party, the Supreme Soviet, and the Council of Ministers (F-1). Ultimate power officially resides in the Communist Party, the pinnacle of this power being the office of General Secretary. The Party appoints the Politburo (political bureau), which elucidates national policy. The principal organ of state is the Supreme Soviet, or national legislature. The membership of the Supreme Soviet elects a presidium, or executive committee, whose chairman is the nation's chief of state. With the approval of the Politburo, the Supreme Soviet appoints the members of the Council of Ministers, which administers national policy. The Council elects a chairman who is the nation's chief executive officer. Within the Council are 62 ministries, five directly involved in the aviation industry (F-2).

Five Ministries Direct Most Aviation-Related Functions: The five aviation-related ministries buy, sell, or produce aircraft. Military aircraft are obtained through the Ministry of Defense (F-3), which has procurement priority over all other customers. The Defense Ministry directs all military forces and maintains extensive weapon research institutes. The

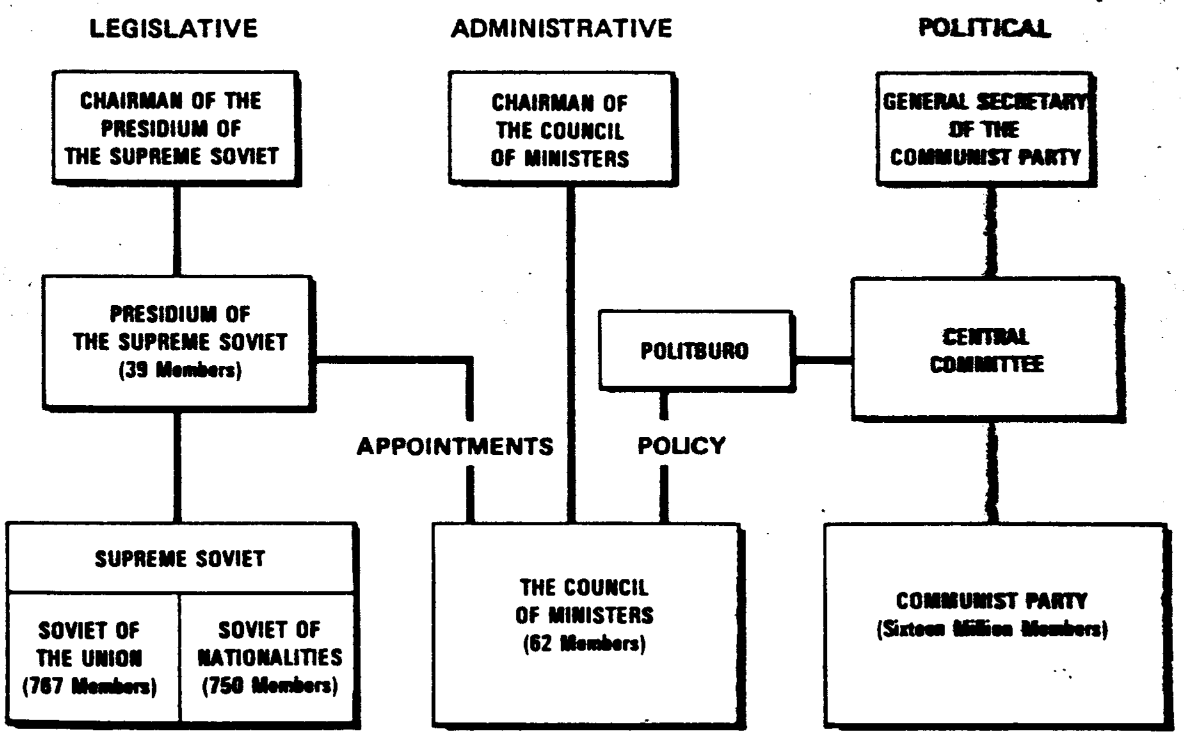
Prototypes and technology demonstrators have been giving way to preproduction aircraft integrating advanced design and technologies



Ministry of Civil Aviation is the predominant customer for nonmilitary aircraft; it operates the national airline Aeroflot. Foreign sales of aircraft are transacted by the Ministry of Foreign Trade through the aviation export office Aviaexport. The Ministry of Higher and Specialized Education directs the academic institutes, which supply most of the engineers and technicians for the aviation industry. Also, several professors and students in these institutes carry out aviation-related research with their own technical facilities. Finally, there is the aircraft-production organization highlighted in this article, the Ministry of Aircraft Production (MAP).

MAP comprises the institutes responsible for almost all research, design, and production of aircraft (F-4).

F-1 USSR GOVERNMENT



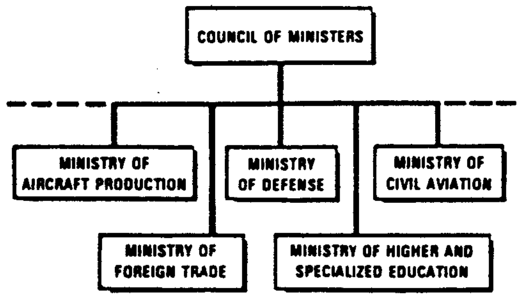
—The Central Aero-Hydrodynamics Institute (TsAGI) conducts basic aviation research in its laboratories and wind tunnels and uses this research to establish aerodynamic methods and forms.

—The Scientific Research Institute for Aircraft Equipment (NISO) sets standards for the instruments, avionics, and accessories required and tests this equipment before delivery.

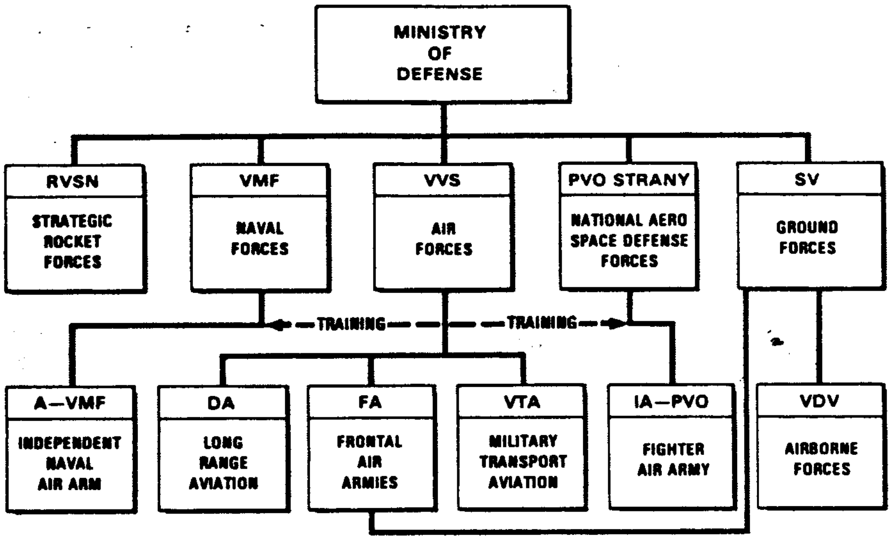
—The All Union Institute of Aviation Materials (VIAM) is responsible for and must approve the type, proportion, and usage of materials in aircraft.

—The Flight Research (Test) Institute provides

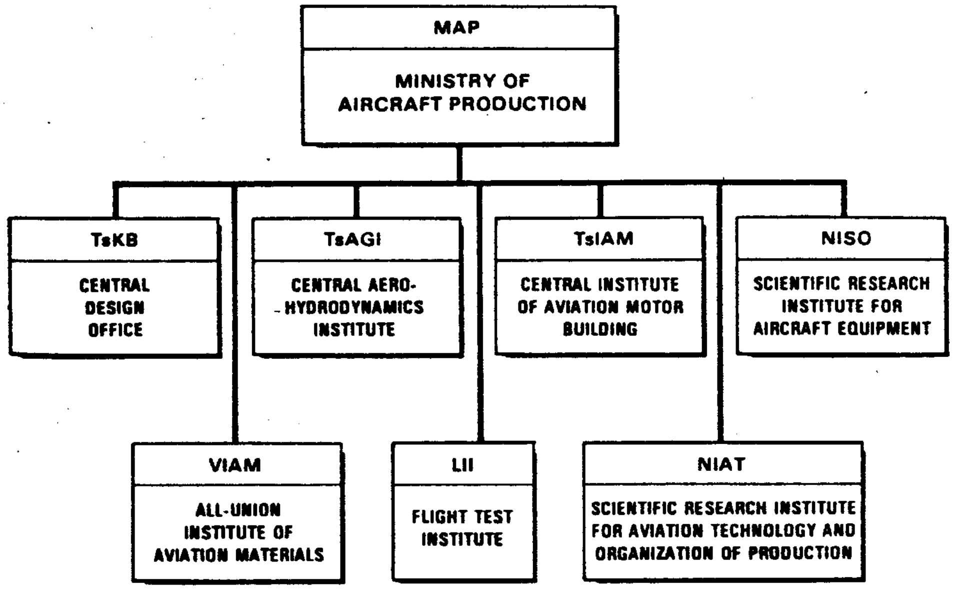
F-2 AVIATION MINISTRIES



F-3 MINISTRY OF DEFENSE



F-4 MINISTRY OF AIRCRAFT PRODUCTION



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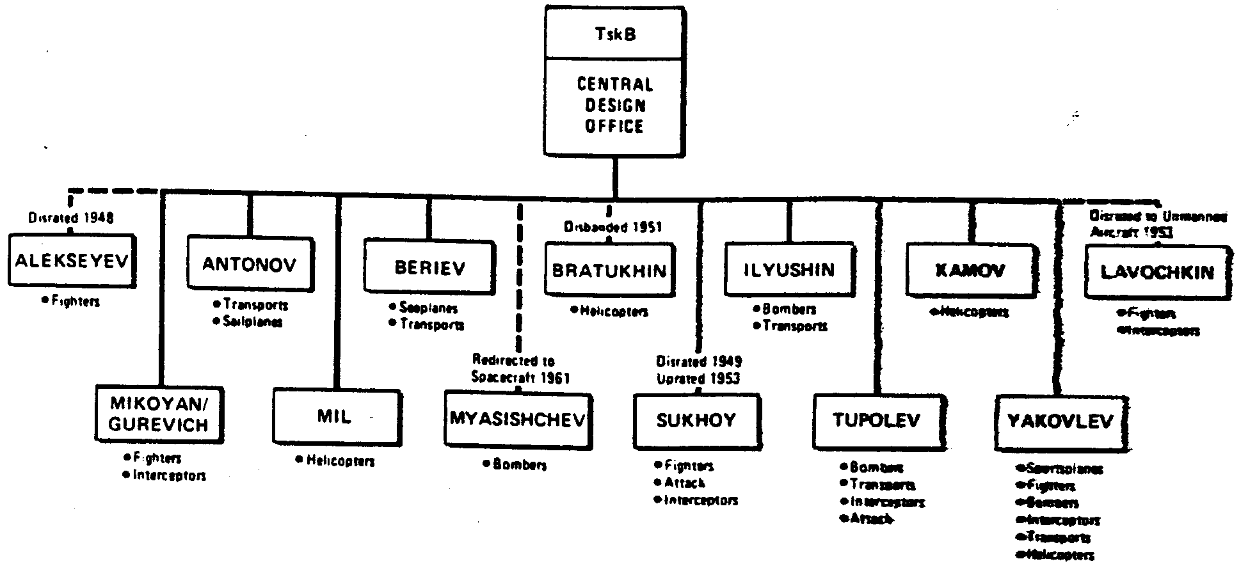
F-5 CENTRAL DESIGN OFFICE



facilities, pilots, and flight-test equipment for prototype trials.

—The Scientific Research Institute for Aviation Technology and Organization of Production (NIAT) manages the plants which produce aircraft engines, materials, and equipment and which assemble aircraft.

—The Central Institute of Aviation Motor Building (TsIAM) directs several experimental-design bureaus charged with design and development of aviation power plants.

—The Central Design Office (TsKB) (F-5) directs eight experimental-design bureaus doing detail design and construction of prototype aircraft.

Experimental-design bureau, or in Russian, *Opytno Konstruktorskoe Byuro* (OKB), describes a collective that designs and constructs engineering prototypes. A typical OKB of the TsKB, which designs and builds prototype aircraft, is organized into five principal areas—projects, technology, prototypes, flight testing, and production (F-6). The project area manages the several engineering tasks assigned to the OKB. These tasks may encompass only components development, such as wings or fuselages, or complete aircraft. The technology area administers the designers and engineers assigned as technical support to the projects. The prototype area constructs the prototypes, using highly skilled machinists working with semi-detailed drawings provided by the projects section. The flight-test area investigates the flying qualities of the prototypes to isolate and correct deficiencies before customer acceptance trials. If the acceptance trials are successful and customer acquisition approved, the production area forms a small group of engineers who accompany the prototype and its drawings to the MAP-assigned NIAT factory to assist in the preparation of the aircraft for production. Concurrently, a separate liaison engineering branch is organized to be advisors to NIAT production engineers whenever questions arise and during model changes. This second group is maintained at the OKB throughout the production run.

All five areas of an OKB work toward a single end, to design and build prototype aircraft. The OKB has very limited research capabilities and must rely heavily on the other MAP institutes for research data. Also, the OKB is not accountable for

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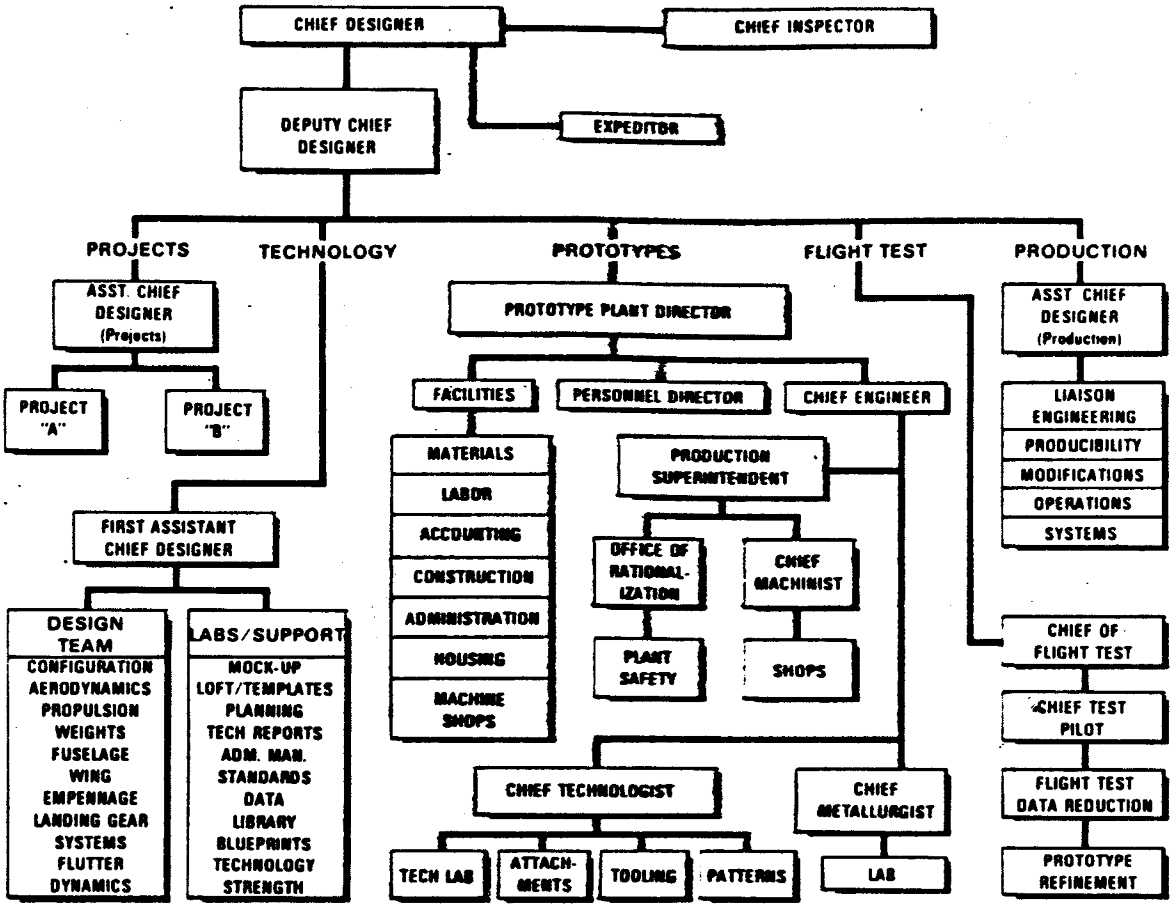
F-6 EXPERIMENTAL-DESIGN BUREAU



production of its prototype, only its initial producibility, with NIAT being responsible for all production.

Technical Tasks Are Divided Among Several Organizations: This division of responsibilities among several agencies makes it easier for a highly structured organization such as the Soviet government to oversee the several functions of the aircraft industry. Such close scrutiny is required because centralized long-range economic planning, inherent in Communist economic policy, has difficulty in adjusting to unexpected changes that can happen when a product fails or its development is protracted. To minimize disruptions, individual or group initiative is restricted; technical developments are made through controlled collective efforts. This control is accomplished by spreading the tasks for research, design, and construction among several smaller, and therefore more visible, bureaucracies. However, even though the functions of the OKB are highly regulated, it must still meet the stringent performance requirements demanded by the military customer. As a means of ensuring that an OKB will dispense its best product, the government uses an incentive system that rewards members of successful design teams with cash bonuses, and official recognition. As noted earlier, the Soviet Union allows military customers to determine the quality of the products they buy, but this granting of incentive premiums helps ensure military products of the highest standards.

Obviously, conflicts can arise between the restraints imposed by long-range centralized planning to assure low-risk programs and the inclination of any design team to resort to high-risk technology to raise the performance of its concept. Therefore, prototype design and construction methods have become strictly regulated by standards set down in "handbooks" provided by the research, test, and production offices of MAP. These handbooks are design standards cataloging proven and approved technology. The contributing offices

A NOTE ON TERMINOLOGY

Parts of the aircraft designator, MiG-21PFMA Fishbed-F, have the following meanings. *MiG* abbreviates the name of the design bureau responsible for this configuration. (All Soviet aircraft designators include the name of the design bureau's originator.) In this case, *MiG* stands for the design team of *Mi*koyan and Gurevich. Other examples—SU: Sukhoy, TU: Tupolev, and YAK: Yakovlev. The 21 is the model number of the production aircraft (odd numbers depict fighters, even numbers bombers and transports). *PFS* is the progressive development suffix: *P* indicating interceptor version; *F*, boosted, and *S*, boundary layer blowing. Other examples—*M*: modified, *B*: attack or bomber version, *A*: aerodynamic refinement. *Fishbed* is the identifying code name assigned the MiG-21 by NATO. All important Soviet aircraft models are named as soon as identified by photographs from a hand-held camera. The first letter of the name identifies the aircraft type (*F*—fighter, *B*—bomber, *C*—cargo/transport, *H*—helicopter). A name of one syllable means propeller powered; two means jet-powered. *F* indicates the point in the sequence in which this version of the Fishbed was identified by NATO.

will only release for use methods and items that they have thoroughly tested and have decided are compatible with the design proficiency of the OKB, the industry's production capability, and the environment in which the aircraft will operate.

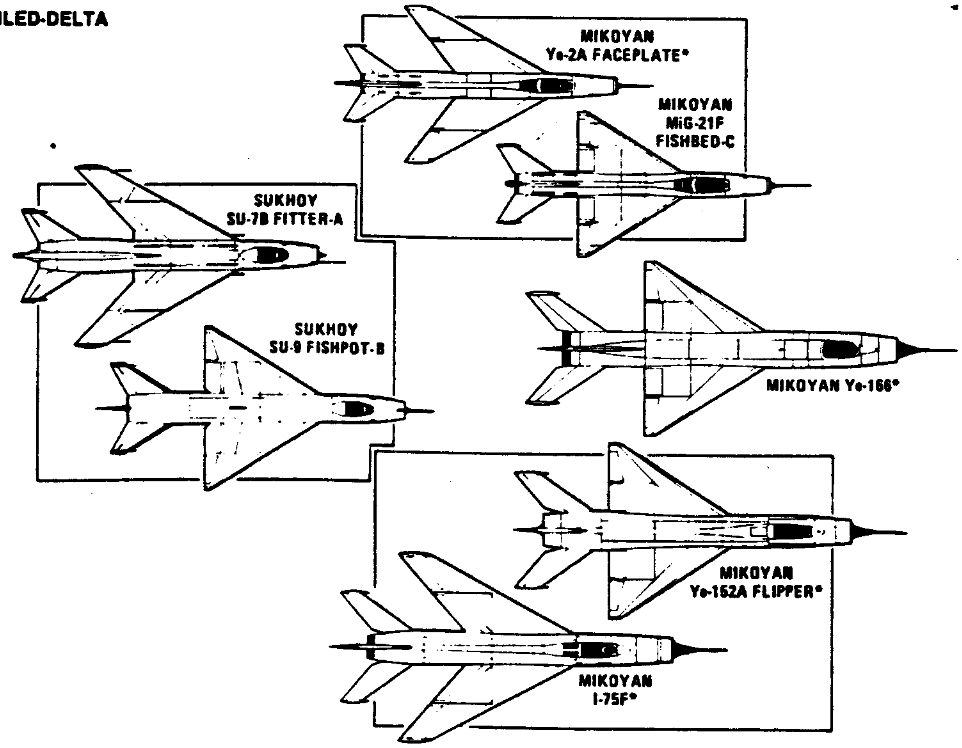
The control these handbooks have on the design bureaus can be seen in some of the aerodynamic forms that have been approved by TsAGI. In the 1950s two principal planforms were authorized for use in high-performance aircraft, the swept wing and the tailed delta. As shown in F-7, these two layouts were incorporated into the majority of the supersonic fighter designs of that era and in most cases were even adapted to a common fuselage. The wing geometry of aircraft employing each of the two wing types are very similar, illustrating the rigid compliance dictated by the TsAGI handbook.

Later, in the 1960s, the variable-sweep wing was authorized for military application and subsequently developed in two forms.

The first was to improve the performance of existing aircraft by adding variable-geometry panels: The SU-7B Fitter-A modified into the Fitter-B and the TU-22 Blinder highly modified into the Backfire-A (F-8).

The second application was the high-wing variable-geometry configuration such as the Mikoyan Ye-231 Flogger-A and the Sukhoy (also spelled Sukhoi) SU-24 Fencer-A (F-9). The Flogger-A is a MiG-21 Fishbed replacement developed from a prototype all-weather fighter, the fixed-wing Mi-

F-7 ARROW-WING AND TAILED-DELTA CONFIGURATIONS OF THE '50s
*** Prototype only.**



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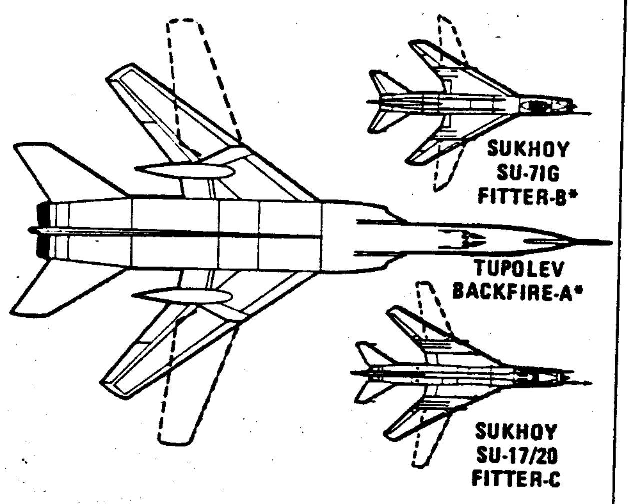
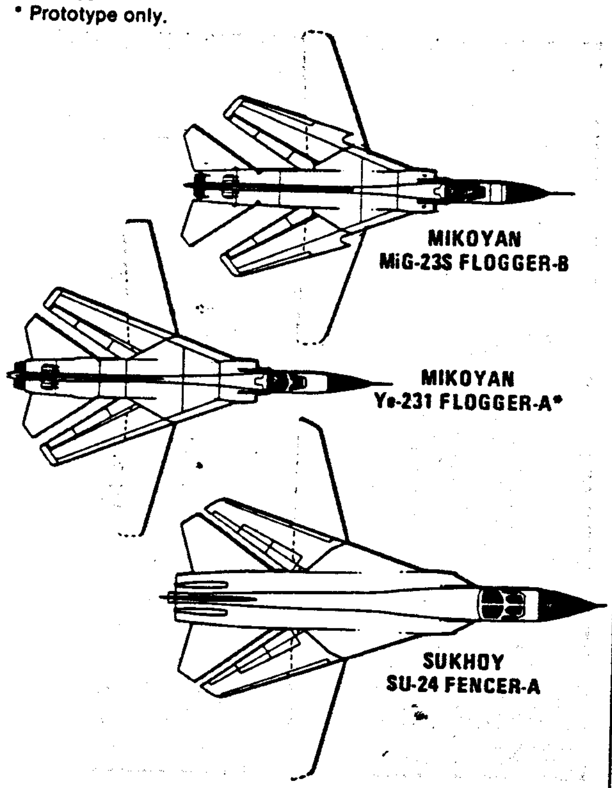
F-8 VARIABLE-GEOMETRY DEVELOPMENTS OF THE '70s
* Prototype only.



F-9 HIGH-WING VARIABLE-GEOMETRY DEVELOPMENTS OF THE '70s
* Prototype only.



koyan Faithless. The Fencer is intended as a supersonic follow-on to current subsonic light bombers, but there is no information on a Sukhoy counterpart to the Faithless prototype.

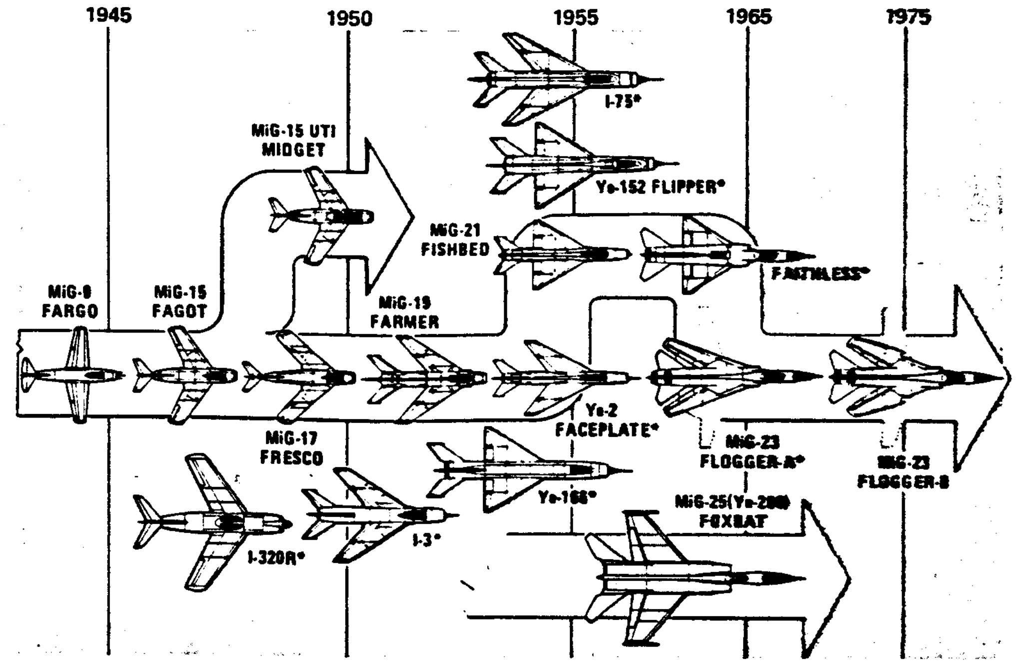
In all, four separate variable-sweep-wing aircraft eventually entered production. As can be seen by the examples of common planforms (swept wing, tailed delta, or variable sweep) used by different design bureaus, when a given technology is approved by a research institute, it will be applied across the full spectrum of aircraft types.

Other examples likewise illustrate the controls exerted by the other MAP institutes. The first inlets approved by the TsIAM for jet fighters were nose-mounted and subsequently included on virtually all fighters for the next 15 years. Equipment commonality among aircraft, as controlled by NISO, can be seen in the universal use of the same IFF equipment, including external antennas, on fighters, transports (civilian and military), bombers, and helicopters. (The antenna for this system has three vertical blades. An example can be seen mounted on the vertical tail of the MiG-21 Fishbed in F-11). Materials control by VIAM is illustrated by their reluctance in the 1960s to allow titanium as primary aircraft structure even though the Soviet Union has the world's largest reserve. Finally, the producibility of aircraft as controlled by NIAT is shown by the instance in which only those types of steels that are relatively easy to weld were permitted in the airframe of the Mikoyan MiG-25 Foxbat. Producibility outweighed strength. And as stated earlier in the description of the institutes of MAP, each has a specific discipline to control, and the vehicle of this control is the technical handbooks. These constraints, as imposed on the OKB, have created a unique design environment.

The Designer Works in a Highly Structured Environment: The designer is judged by his innovation in applying approved technology in the lowest-risk method and not by the amount of advanced technology he can incorporate. In the same vein, he is also greatly concerned with program costs since overruns are seldom tolerated. The resulting design doctrine, as followed by the OKB designers, is to maintain as uncomplicated a prototype program as possible, and thus the least costly. The principal concepts he uses to ensure that he will be able to comply with this doctrine are design heredity and component commonality.

Design heredity is a method used by designers to develop new configurations by extrapolating from their past efforts. Examples of this doctrine can be seen in the progression of prototypes from a typical OKB. F-10 shows chronologically the principal fighter prototypes produced by the Mikoyan/Gurevich OKB since World War II. The relationship of each design to the previous is evident. Design heredity likewise appears in the MiG-21 and MiG-23/27 series, where technological improvements were made during the production of one model. In the Fishbed series (F-11) at least ten major changes or improvements were made before the

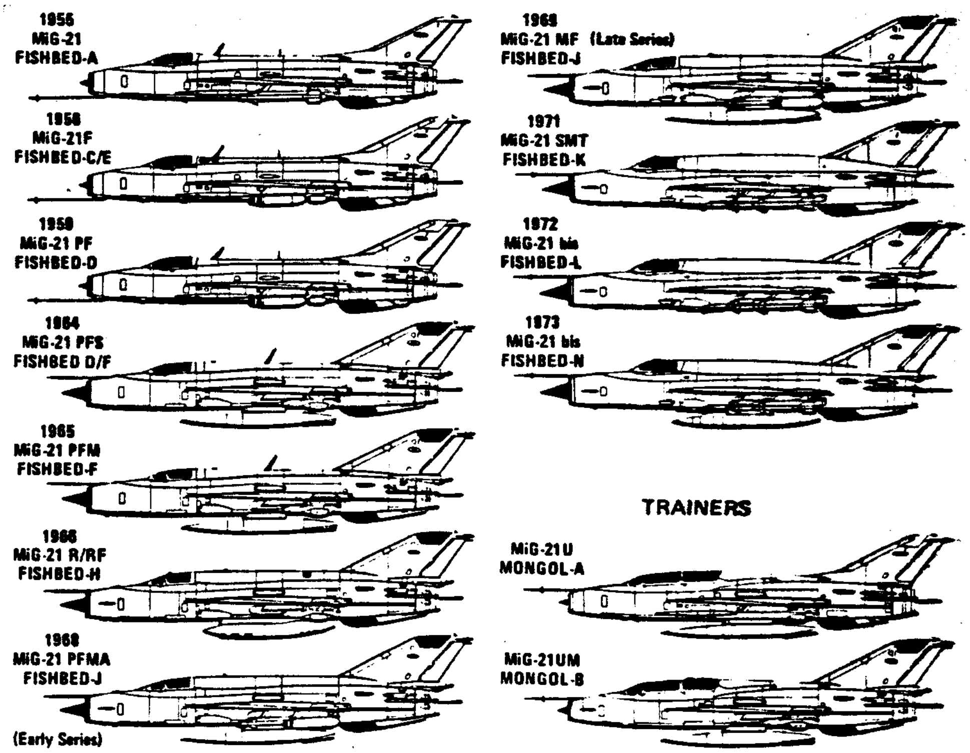
F-10 MIKOYAN/GUREVICH DESIGN BUREAU PRODUCTS SINCE 1945
* Prototype only.



replacement model, the MiG-23 Flogger, was widely introduced. Within the Flogger series (F-12) seven derivatives have evolved so far.

Component commonality applies parts of concurrent and previous designs on current prototypes to reduce the number of new systems per program. Classic examples of this concept can be found in a series of designs from the Sukhoy OKB (F-13). The fuselage and empennage of the S-1 and T-3 are almost identical, the only major difference being in the wing planform. The Fitter-B variable-sweep technology demonstrator used the same fuselage and empennage as the Fitter-A. The Flagon-A used the same wing, tails, and cockpit as the Fishpot-C.

F-11 MIG-21 PRODUCTION HISTORY



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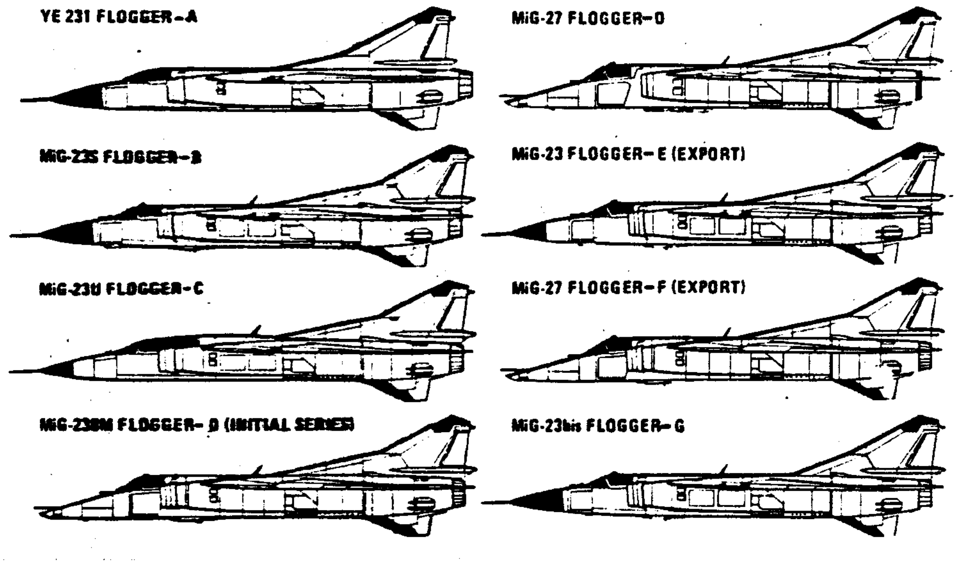
F-12 FLOGGER PRODUCTION HISTORY
* Prototype only.

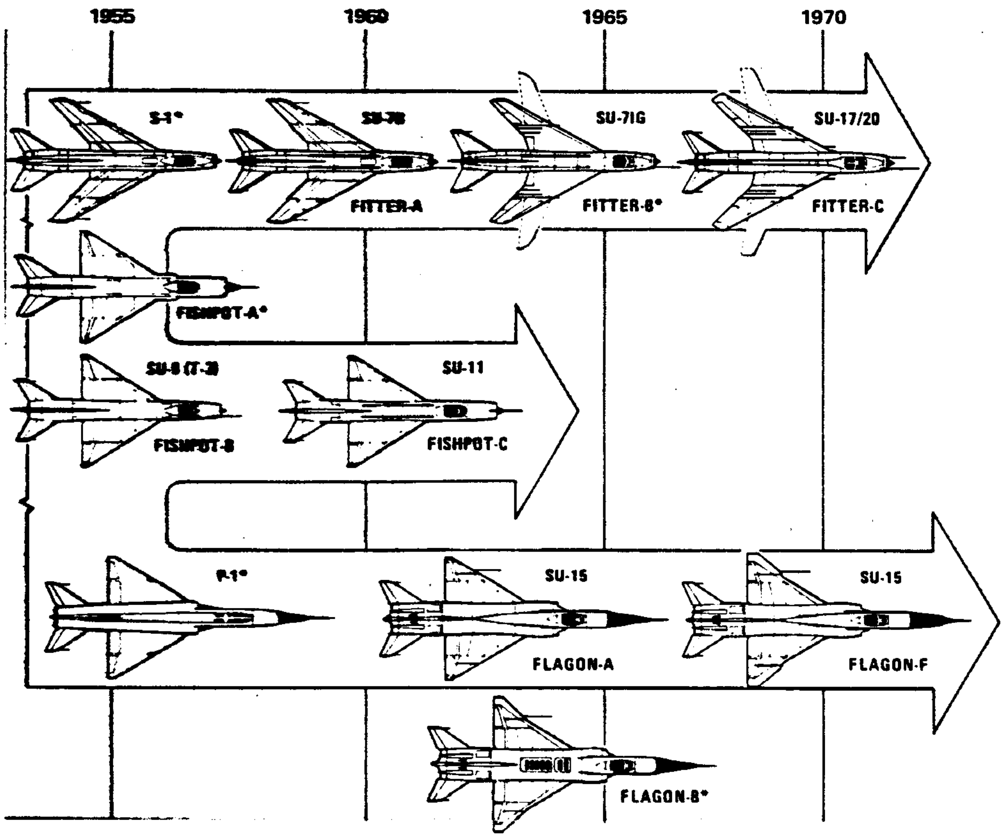


After a prototype has been accepted for production, modifications are made as needed, but the basic concepts have been developed with a minimum of new systems.

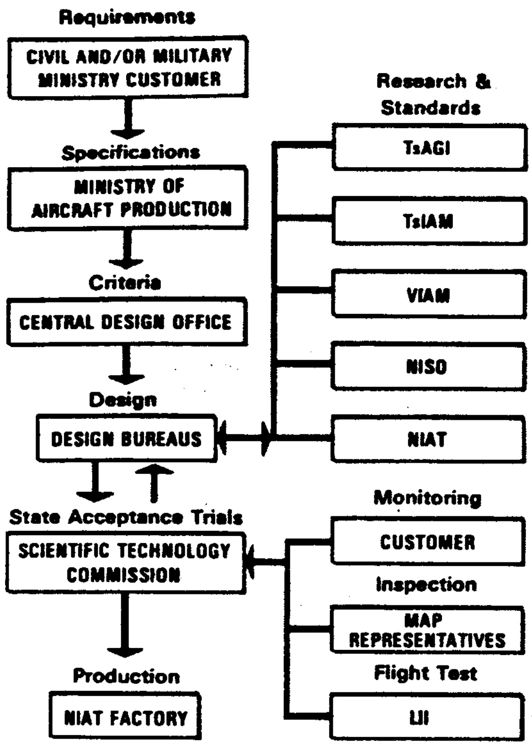
These two methods of design, heredity and commonality, are so extensively used because of the nature of a Communist economy.

As previously discussed, the Soviet leadership believes that the best way to minimize disruptions to the doctrine of long-range economic planning is by allowing only limited amounts of advanced, and therefore high-risk, technology in any one prototype. Consequently, the only way the designer can construct his new prototype is by adding the

F-13 PAST DESIGNATORS OF THE SUKHOY DESIGN BUREAU
* Prototype only.



F-14 AIRCRAFT PROCUREMENT PROCEDURE



metered technological advances to his last work. In other words, rather than making wholly new designs at protracted intervals, the designer makes incremental improvements at more frequent intervals. The main advantage of this system is the lessening of development risk. It entails at least one important disadvantage: In the incremental design process, much of each new model is still based on the older technologies of the previous models. This carry-over limits aircraft to less than optimum performance.

Methods of Procurement Are Still Evolving: These several aspects of the aviation industry—the organization, the controls, and the design methods—determine the procurement procedures. An abbreviated outline of the current system, illustrated in F-14, shows that the requirement for an aircraft is generated by the customer ministry, refined by MAP, and further refined by TsKB, which in turn instructs an OKB to design a prototype. If these designs, which may include mockups, are approved by the Scientific Technology Commission, the OKB then builds the prototype. All the efforts of the OKB are supported by research and regulated by handbooks from the MAP institutes. After the prototype is tested and refined by the OKB it is then passed on to the commission for state acceptance trials. If the prototype trials are acceptable and the government leadership approves, the aircraft enters production. This system of procuring aircraft has been altered substantially since the inception of the original system in 1920, but one needs only examine this process from the end of WW II to understand how the current procurement system evolved.

Because of the critical need for combat aircraft during the war, the major effort of Soviet industry was directed toward production rather than research. With the end of the war, the government made a concerted effort to modernize the aircraft industry. It attacked deficiencies in three ways: assimilation of foreign technology, application of the new technologies to development aircraft, and intra-bureau competition to produce advanced aircraft designs.

Assimilation of foreign technology is exemplified by three separate events: occupation of eastern Germany in 1945, which provided modern research and production facilities as well as examples of several advanced aircraft; the internment of four American B-29 bombers in 1944-45, which provided excellent examples of advanced systems and materials; and purchase of two different sizes of British turbojet engines in 1947, which provided advanced propulsion technology.

To apply these new technologies to industry, several special aircraft were constructed. In the three years following the war, for example, at least seventeen fighter prototypes were tested (F-15).

When the industry had adapted the foreign technology through the use of these aircraft, it was able to embark on its own advanced programs through the use of prototype competitions. For example, a series held between 1948 and 1952 developed a progression of day fighters (F-16). The first two winners, the MiG-15 Fagot and the MiG-17 Fresco, were powered by close copies of one of the purchased British-designed turbojets. The transonic MiG-19 Farmer was powered by two Soviet-designed turbojets. Of particular interest, one case illustrates all three modernization methods (assimilation, development, and competition)—the long-range-bomber program (F-17).

The Soviets had acquired the four B-29 aircraft by interning them at a time when the USSR was a neutral in the Pacific War. These bombers, unable to return to their bases after missions over Japan, had been diverted to Siberia. The Tupolev OKB was instructed to make direct copies of the aircraft to be followed by a limited production run. After the successful construction of these copies, a program was begun to develop indigenous long-range bombers. The Tupolev design bureau started with a large piston-powered aircraft (the TU-85) as precursor of

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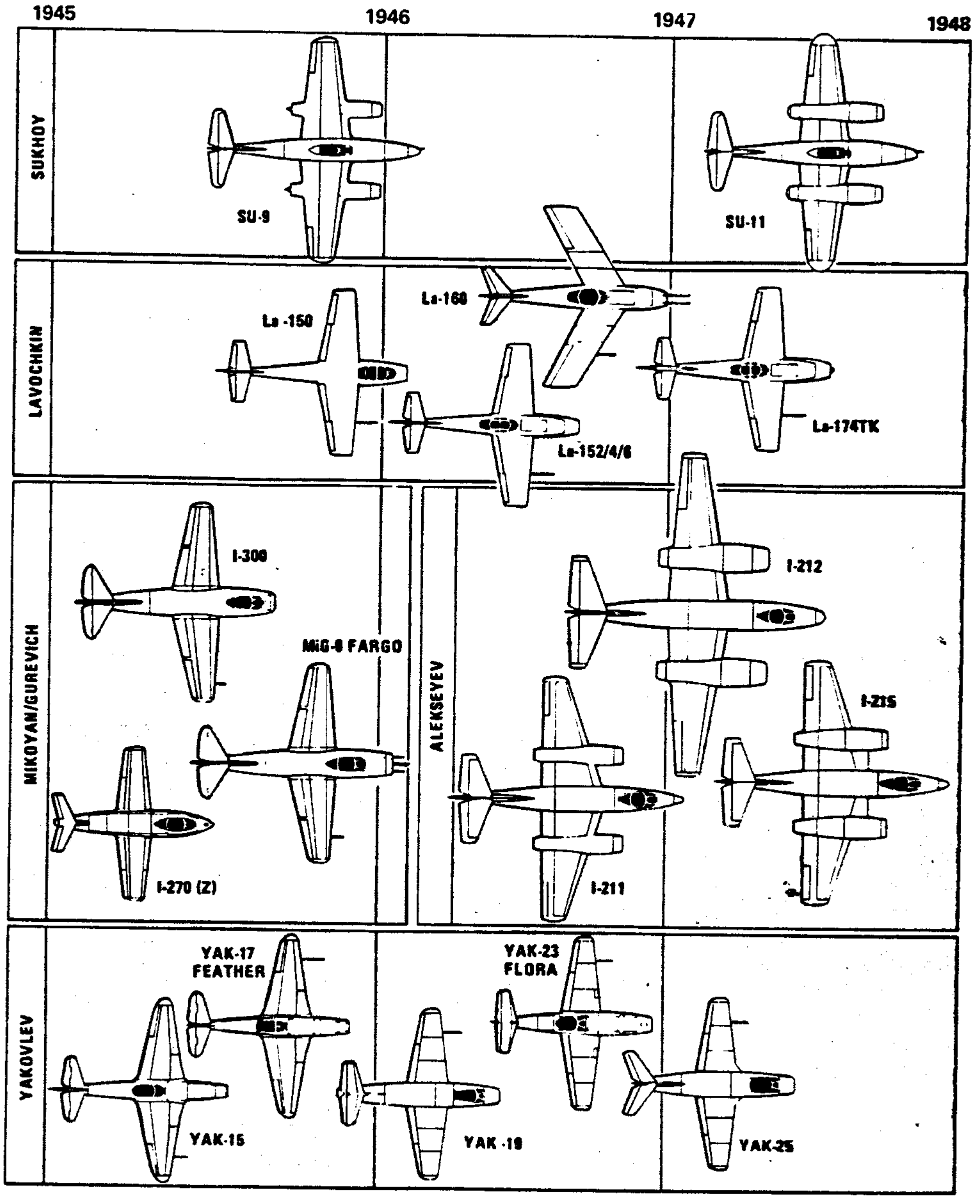
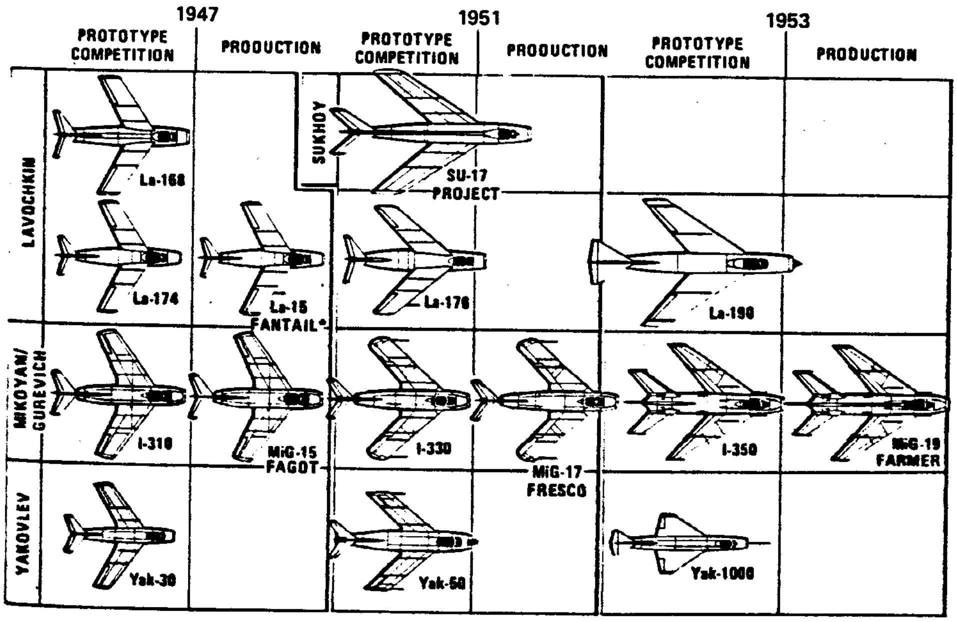
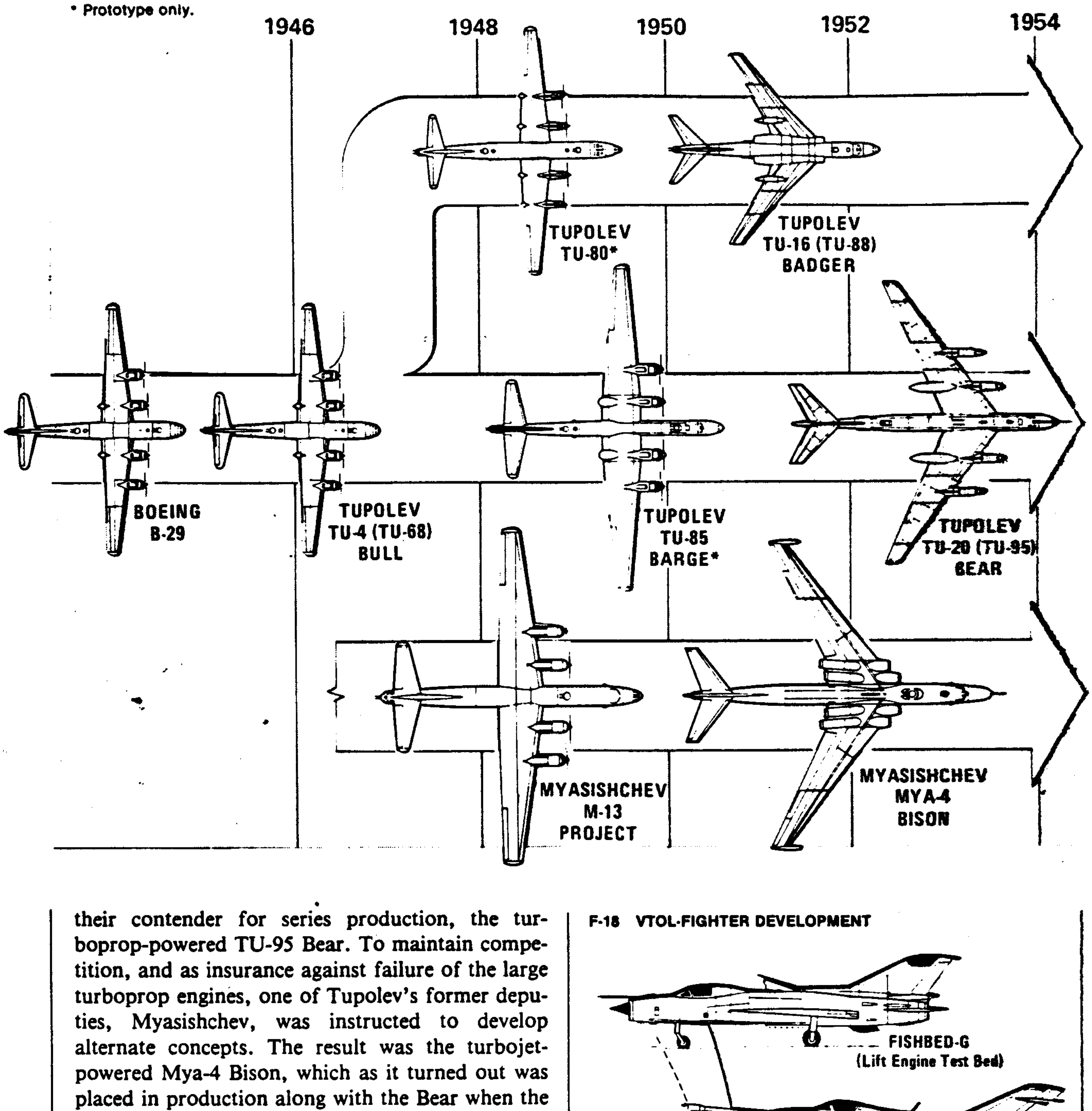
F-15 FIGHTERS DEVELOPED IMMEDIATELY AFTER WWII





F-16 DAY-FIGHTER COMPETITIONS: 1948-54
* Limited production.

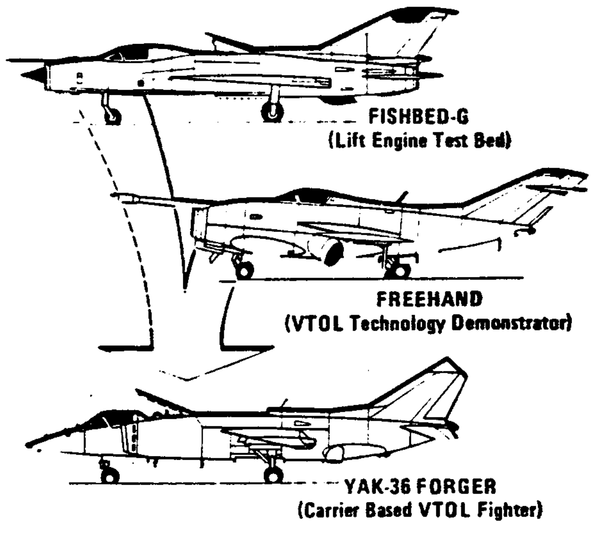
F-17 LONG-RANGE-BOMBER DEVELOPMENT: 1945-54
* Prototype only.



their contender for series production, the turboprop-powered TU-95 Bear. To maintain competition, and as insurance against failure of the large turboprop engines, one of Tupolev's former deputies, Myasishchev, was instructed to develop alternate concepts. The result was the turbojet-powered Mya-4 Bison, which as it turned out was placed in production along with the Bear when the Ministry of Defense concluded that each had merits. Thus, the aircraft industry absorbed foreign technology (B-29 to TU-4 Bull) by constructing technology-application aircraft (TU-85) and acted on new capabilities by developing operational aircraft through competition (Bear vs. Bison).

No sooner had the aircraft industry settled on prototype competitions as the means to choose the best designs for production than the new element of supersonic flight disrupted the system. Supersonics required such radical departures from established

F-18 VTOL-FIGHTER DEVELOPMENT



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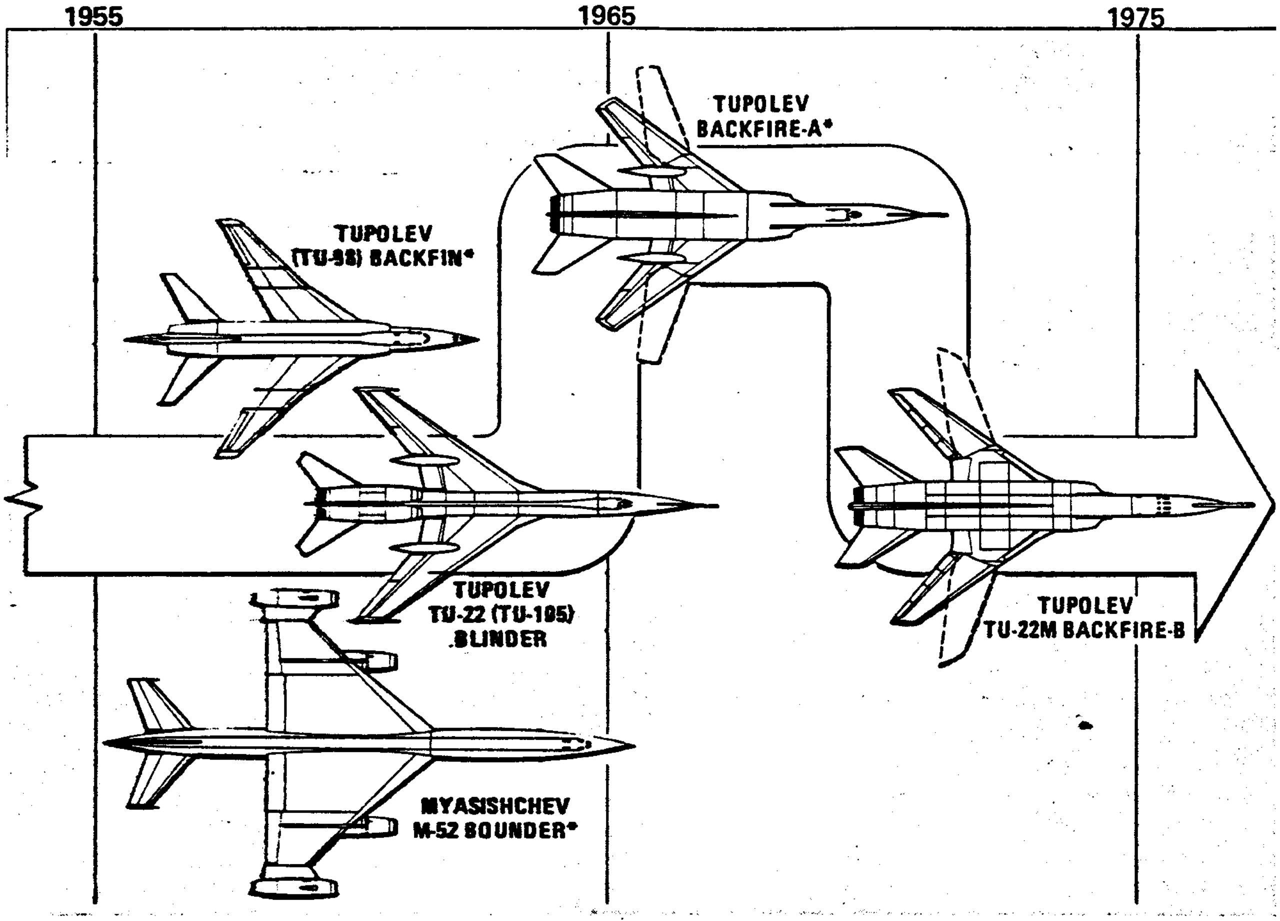
F-19 LONG-RANGE-BOMBER DEVELOPMENT: 1955 TO PRESENT
* Prototype only.



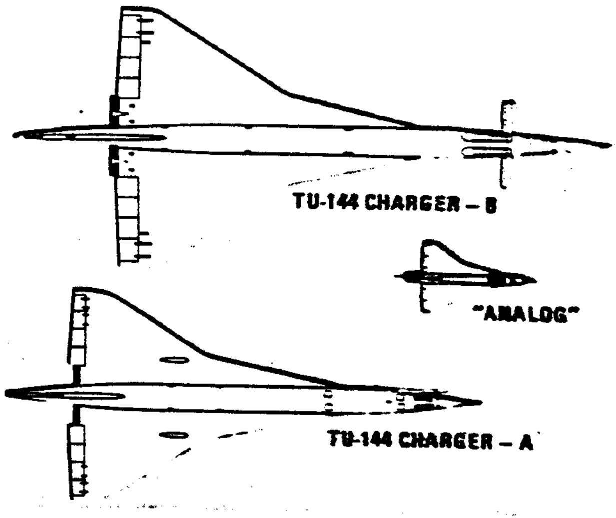
concepts that little could be adapted from the previous subsonic designs for the development of a new generation of fighters and bombers. In the design of fighter aircraft, several methods were tried in attempts to overcome this barrier. As can be noted in the 1953 fighter design competition, the different design bureaus were given great leeway in their approaches to the problem, contrary to previous restrictions (note F-16). The result of this latitude was the relatively limited production of the mediocre MiG-19 Farmer as well as the curtailing of most future prototype flyoffs between bureaus. A different method of fighter procurement was instigated: technology competitions.

Prototype programs were changed from determining winners to determining the best technology, with comparative designs from one bureau being used rather than competitive designs from different design bureaus. As a representative case, STOL characteristics were compared between the variable-sweep Flogger-A and the lift-engine-equipped Faithless. Both aircraft were products of the Mikoyan OKB. (Before that, the swept wing had been flown against the delta in an earlier Mikoyan effort, the Faceplate/Fishbed program (F-10).

The use of inter-bureau flyoff competitions had never been an end in itself. Along with the obvious means of determining the best of several designs, it was also used to eliminate less-competent design teams as well as to determine which type of aircraft the remaining OKBs were best at producing. In this way each OKB was directed toward areas of specialization, having the effect of reducing the number of bureaus building similar aircraft.

However, the technology flyoffs for fighter aircraft proved to have limitations in their ability to evaluate radical advances, especially when new technologies had no previous or alternate concepts to which they could be compared. These radical, and therefore high-risk, technologies had to be developed through a series of technology-demonstrator aircraft. As an example, two technology demonstrators were used in the vertical takeoff and landing (VTOL) fighter program for the Soviet Navy (F-18). The first indication to the West of their existence came when a pair of seemingly unre-

F-20 SST DEVELOPMENT PROGRAM



lated experimental aircraft were revealed in a 1967 air show. One was the Fishbed-G, a Fishbed-F with two lift engines buried in the fuselage, and the other was the Yakovlev Freehand, a twin-engine VTOL design with vectorable nozzles. Though the Fishbed-G may have been primarily a test bed for the STOL version of the Mikoyan Faithless, the technologies developed with it would have been readily accessible to all the other Soviet design efforts because the different design bureaus are part of the same organization—MAP. After completion of the R&D programs, the two types of propulsion systems were combined into one airframe, the YAK-36 Forger (first seen by the West in 1976 in operations from the cruiser/carrier *Kiev*).

The design of supersonic bombers was approached by methods different from those used for fighter aircraft (F-19). As with fighter procurement, competition between bureaus was discontinued but replaced not with technology flyoffs but by the construction of only one design per program. The initial efforts of this method were somewhat disappointing, resulting in mediocre aircraft (e.g. the Bounder and Blinder). Because of these setbacks, MAP reevaluated the single-prototype method, and instead initiated multistage programs using "preprototype" aircraft to develop and evaluate the high-risk technologies to be incorporated into production aircraft. The Backfire bomber offers a good example of the use of multistage development. The initial stage used an extensively modified TU-22 Blinder to gain experience with large variable-sweep aircraft and to test the side-mounted inlet and new engine. This aircraft, the Backfire-A, was then used for systems and service evaluation in anticipation of the production of the much improved TU-22M Backfire-B. There were no flyoffs, just stepped development by one design bureau. (This program was unusual in that the new aircraft continued to be called a modification, TU-22M, instead of being given its own numerical designator.)

Another use of stepped development was the supersonic transport program, an enterprise of the Ministry of Civil Aviation but developed under military auspices owing to their exclusive supersonic experience. This program differed from Backfire in that it used two distinct development aircraft before the choice of a production design (F-20). The first was a modified MiG-21 employing a cranked-delta planform to investigate the flying qualities of an all-wing configuration. The next was a prototype passenger-carrying aircraft, the TU-144 Charger-A. Two years later the highly refined TU-144 Charger-B production version appeared. As seen in the Backfire and Charger, stepped development through demonstrator and analog aircraft was the dominant heavy-aircraft procurement procedure of the 1970s.

In summary, modern Soviet aircraft procurement started in the 1940s with prototype flyoffs between competing design bureaus and later turned to flyoffs of aircraft with comparative technologies to determine the superior design. Finally, these flyoffs were phased out and technology demonstrators and preprototype analogs were used as antecedents of production aircraft. Current procurement methods are less clear. It appears that MAP will bypass all or most of the prototype methods by going initially into a program of pre-production aircraft. This method will have come about, not because the Soviet aircraft designers will have become less fallible in their efforts, but because they will be better able to define the extent of their limitations.

The Soviet aircraft-procurement system has always had the full support of the nation's leadership. The USSR's aviation industry has been able to rely on this consistent national policy which has included the commitment of the manpower and materials to achieve procurement goals. And with this upper-level backing, the aviation industry has progressively expanded its R&D base. This growing technological foundation appears to be giving the Soviet aircraft designer the means to apply advanced concepts faster and to produce each new military aircraft with a progressively larger proportion of advanced technology.

Acknowledgement

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Re-engining the 737

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Aerospace Congress & Exposition
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October 25-28, 1982

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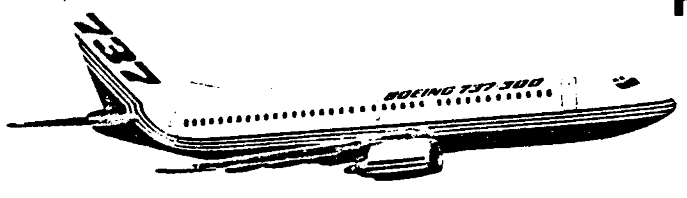

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Background

Much has been said about higher fuel prices and how this has contributed to a decline in airline profits. Various ways to reduce fuel consumption have been studied and many of them tried (Ref. 1), but the availability of more efficient engines offers the largest potential gain of any practical alternative known today. Numerous airplanes have been considered as logical candidates for engine retrofit. After low fuel-burn, high-bypass-ratio engines were successfully flown on the 707/CFM56-2 (Fig 1); it was decided to not only apply them to the KC-135 (Fig 2) and the DC-8 but to install smaller engines of this general type on the 737-200 airliner, with a fall, 1984, airplane certification target (Fig 3). This was an opportunity to not only reduce fuel consumption but to simultaneously reduce community noise by a substantial amount.

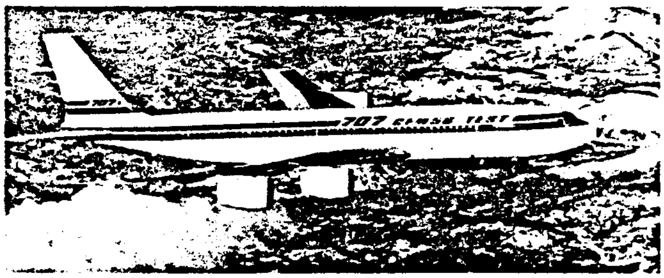

Figure 1. The 707/CFM56-2 Flight Test

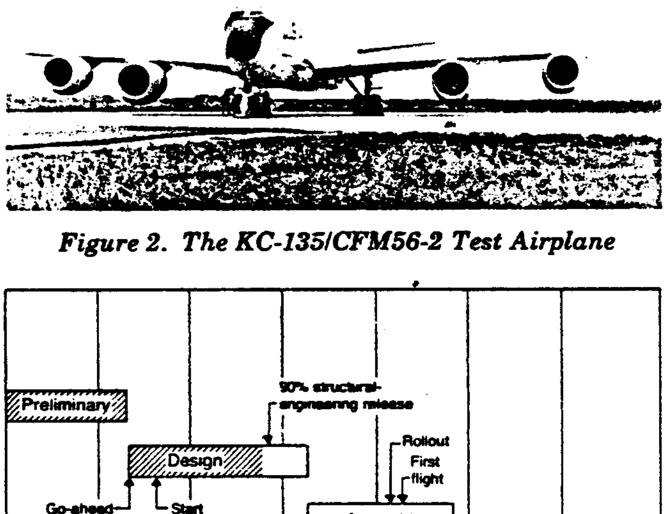

Figure 2. The KC-135/CFM56-2 Test Airplane

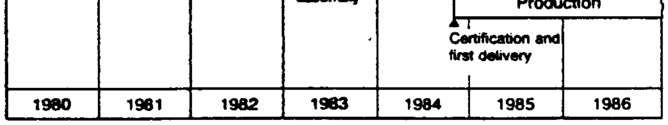

Figure 3. 737-300 Program Schedule

ABSTRACT

Opportunities exist to re-engine commercial airliners with new high-bypass-ratio engines. One such example is the Model 737-300, a derivative of the Advanced 737-200 airplane with two CFM56-3 engines. The ground clearance of the 737 and the increased engine diameter present a unique design problem. A close coupled nacelle/wing, a new leading edge slat, and a horizontal tail extension are required along with increased operational weights. About 20 passenger seats have been added. Noise levels and fuel consumption have been substantially reduced.

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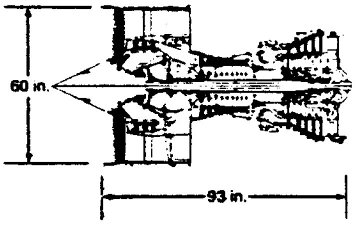
The Engine

The CFM56-3 engine is being developed by CFM International, a company jointly owned by Snecma of France and General Electric of the United States.

The CFM56-3 is a derivative of the somewhat larger CFM56-2 (Fig 4). The CFM56-3 has a modified fan, based on CF6-80 technology, and a side-mounted gearbox (Fig 5). The CFM56-2 and -3 use the same core and low pressure turbine; in fact, approximately 90% of the parts are common between the -2 and -3 (Fig 6). The CFM56 engines have design features which include active turbine clearance control and electronic power management control (PMC) (Fig 7).



Engine ground test	April 1982
Flying test bed (707)	Jan 1983
Airplane certification	Nov 1984

	CFM56-2	CFM56-3
Thrust	24 000 lb	20 000 lb
Bypass ratio	6	5
Flat-rated temperature	86° F	86° F
Weight	4 610 lb	4 278 lb
Fan diameter	68.3 in.	60 in.

Figure 4. CFM56-3 Engine Characteristics

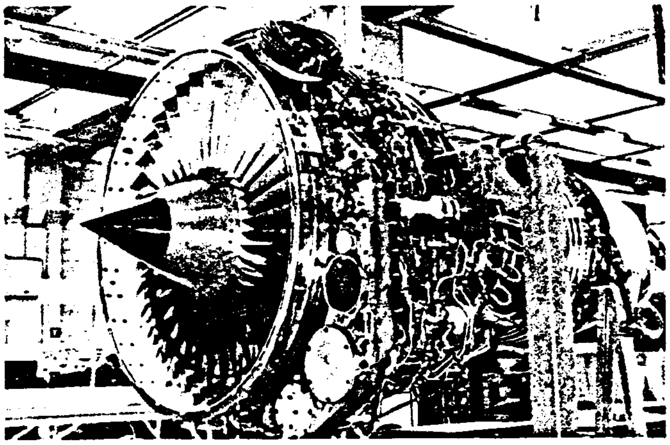


Figure 5. The CFM56-3 Test Engine

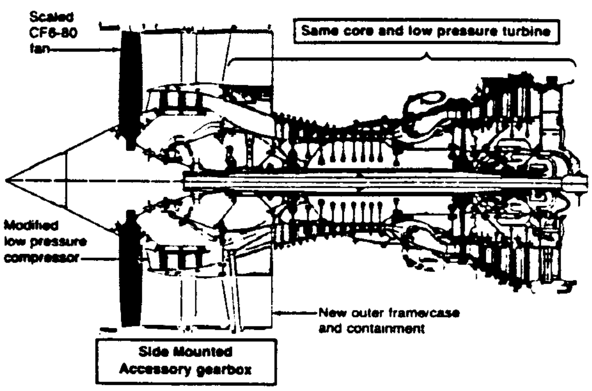
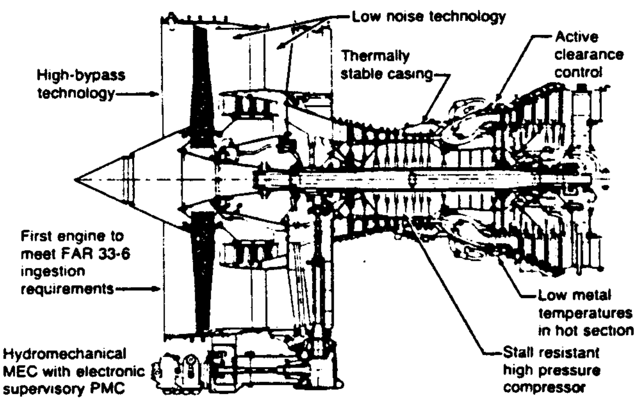


Figure 6. CFM56-3 Commonality with CFM56-2



(MEC - Main engine control)
(PMC - Power management control)

Figure 7. CFM56 Design Features

The CFM56-2 engine has undergone an extensive test program including flight testing on a Boeing 707 during 1979 and 1980. The CFM56-2 is also used on the re-engined DC-8 and KC-135. When the 737-300 enters service, one million engine in-service operating cycles will have been accumulated on the CFM56-2 (Fig 8). This amounts to over two million engine in-service operating hours.

As fuel costs become a larger portion of direct operating costs, engine fuel consumption becomes more important. The uninstalled fuel consumption of the CFM56-3 is 10-20% lower than older engines in its thrust class (Fig 9).

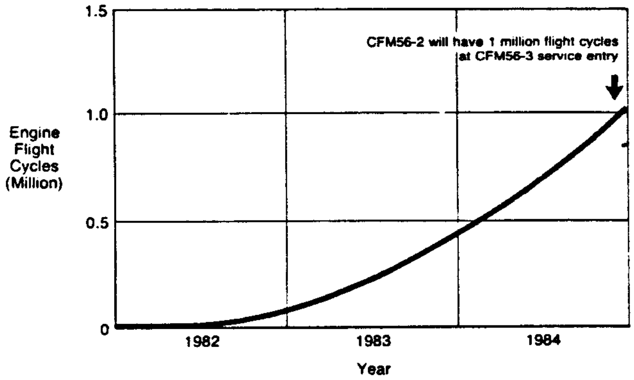


Figure 8. CFM56-2 In-Service Experience

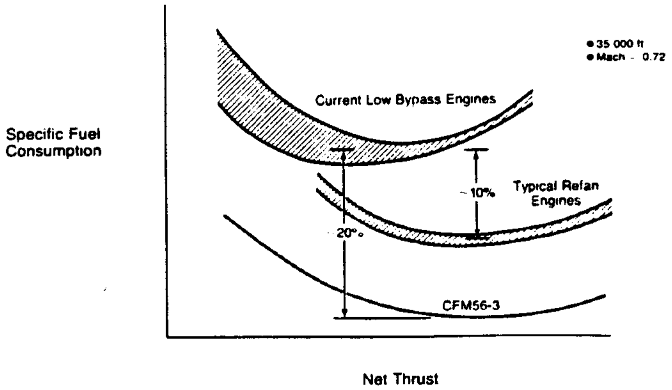


Figure 9. Uninstalled Specific Fuel Consumption

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The Airframe

The 737-300 is a derivative of the Advanced 737-200 airplane with high-bypass-ratio engines. It has increased capacity – approximately 20 additional passengers in the all-tourist 32-inch seat pitch configuration. A major design rule was to preserve the main landing gear, at least as far as length was concerned. The gear is in many ways the heart of an airplane, and a change in the center section of an airplane can be very expensive.

The preliminary design team was expected to "hold the line" against unnecessary improvements, "gold plating," etc., and really did so. However, it was not possible to simply add new engines – partly because of weight and balance considerations. The repositioned higher weight engines produced a nose-down moment which was counteracted by adding more body length aft of the wing than forward, as shown in Fig 10. The increased size (Fig 11) and capacity are in the proper direction considering increased traffic growth trends in many short-medium range, low density markets.

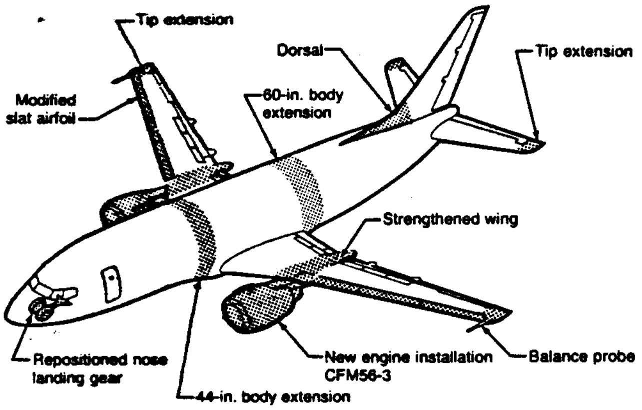


Figure 10. Principal Changes From the Advanced 737-200

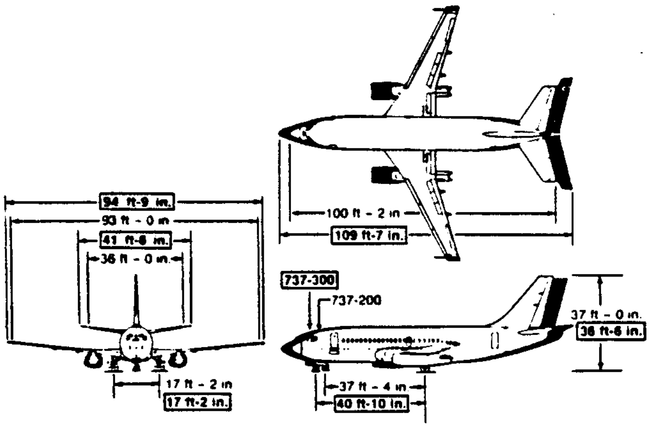


Figure 11. 737-300 and 737-200 Size Comparison

New engines from CFMI, General Electric, Pratt & Whitney, and Rolls Royce offered considerable promise. The CFMI CFM56-3 was selected as the basic engine for the program, after thorough study. It is considered the best "match" available in the program time frame. It is a derivative of the CFM56-2 engine, with a reduced diameter fan and a lower rated thrust. However, the 60-inch diameter is considerably greater than the current 41-inch diameter JT8D engine on the 737-200 airplane (Fig 12). Because of this, a unique nacelle and strut design was necessary for this airplane. If a conventional nacelle/strut design were used, with no change in landing gear length, the new (737-300) airplane would look something like Fig 13!

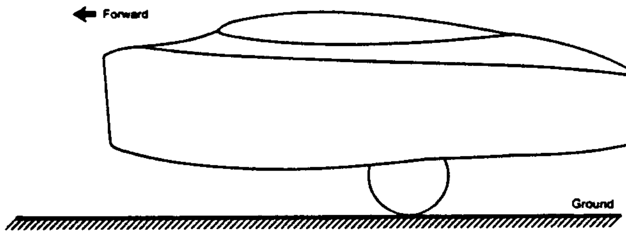


Figure 12. The 737-200 JT8D Nacelle

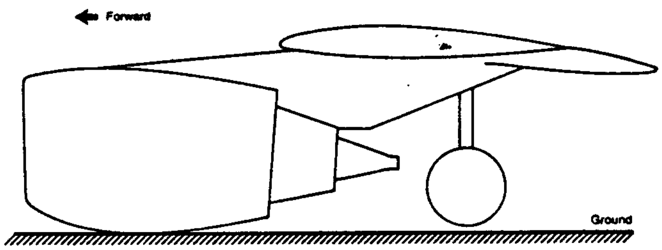


Figure 13. The 737-300 CFM56-3 Nacelle Using Conventional Design

In the mid 1970s, Boeing anticipated the need to be able to mount a large-diameter nacelle close to the wing. Research was done using advanced numerical computer techniques to learn how to successfully achieve such an installation. The opportunity to prove the resulting design concepts was provided by the 707/CFM56-2.

The 707/CFM56-2 had low interference drag partly because of the shape of the exhaust nozzle and partly because of the aerodynamic contour of the strut. Obviously, the 737-300 nacelle had to be tucked up close to the wing and the 707/CFM56-2 provided the demonstration as to how such an installation could be done. The top of the 737-300 nacelle is about even with the top of the wing. Figures 14 and 15 show the nacelle and the resulting smooth unseparated airflow.

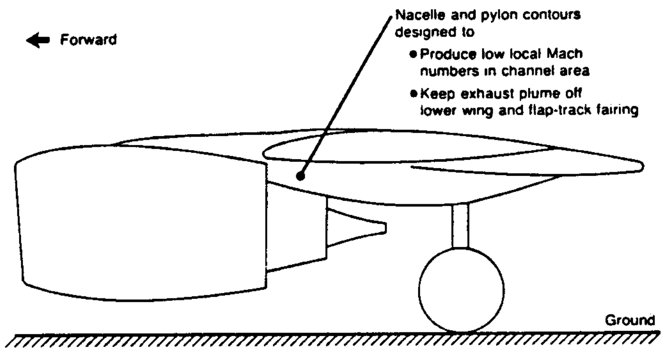


Figure 14. The 737-300 CFM56-3 Nacelle Using Close-Mounted Design

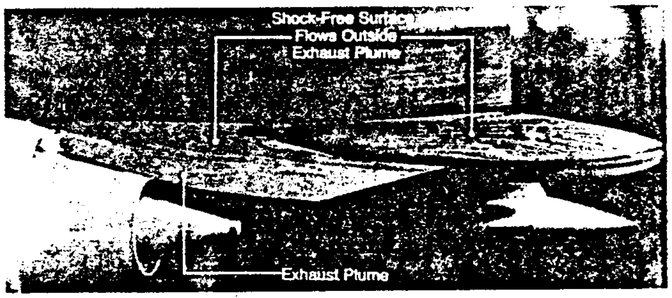


Figure 15. The 737-300/CFM56-3 Inboard Nacelle/Pylon Flow Field. (Cruise Conditions)

Approach and landing speeds were kept down by virtue of a modified slat airfoil outboard of the nacelles (Fig 16) and by the addition of vortex control devices (VCDs), or vanes on the inboard sides of the nacelles. These devices improved the airflow and the efficiency of the wing. In addition, the new slat, which increases the wing chord 4.4%, improves the initial cruise altitude capability by over 3,000 feet at the long range cruise (LRC) Mach number (Fig 17) and reduces fuel burn (Fig 18), while maintaining 737-200 handling characteristics.

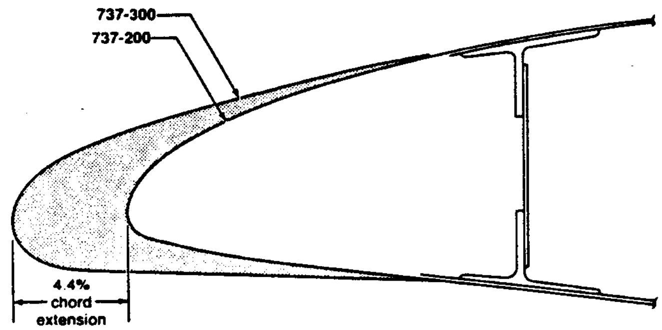


Figure 16. Leading Edge Slat Revision

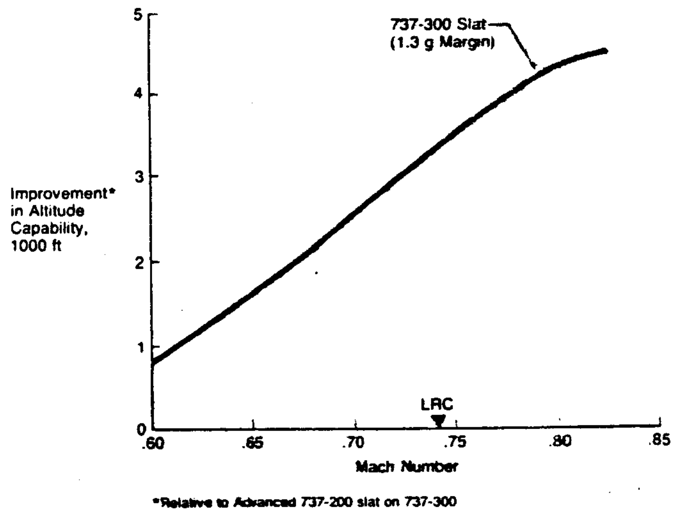


Figure 17. Buffet Boundary Improvement Due to Slat Revision

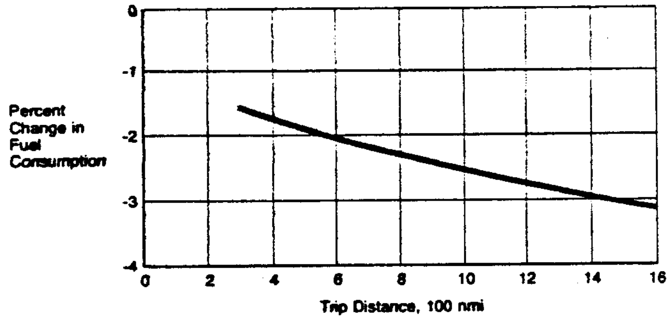


Figure 18. Fuel Consumption Reduction Due to Slat Revision

The nacelle had several design constraints which necessitated an innovative design approach to attain the proper shape and contour lines. Ground clearance considerations required the engine manufacturer to place the gearbox and accessories on the sides of the engine or cheeks of the nacelle (Fig 19), with the drive removed from the bottom centerline. This tended to flatten the bottom of the nacelle and influenced the shape of the inlet lip to a near circular upper half and a near elliptical lower half. A large radius lower lip leading edge was provided which was shown by wind tunnel tests to have excellent pressure recovery. The nacelle has a unique shape to fit the demands of the 737-300 airplane. It may be that future airplanes can use this close coupled design to reduce the size of the landing gear and reduce airplane weight.

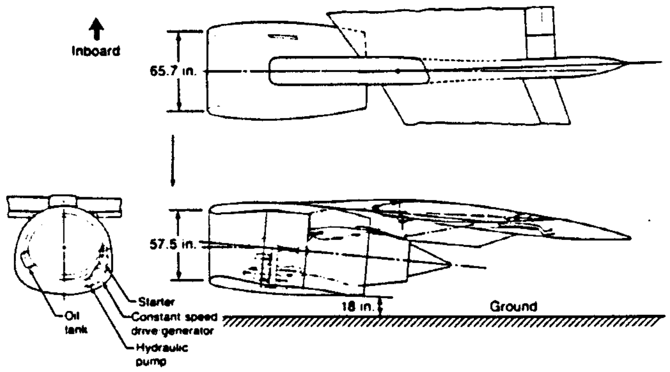


Figure 19. The 737/CFM56-3 Nacelle Installation

The nose landing gear was repositioned and the tire size increased to raise the nose 5.4 inches to achieve adequate ground clearance of the nacelle (Fig 20). The designers believe that the satisfactory nose wheel water spray pattern (Figs 21 and 22) and foreign object damage (FOD) characteristics of the 737-200 will be retained on the 737-300.

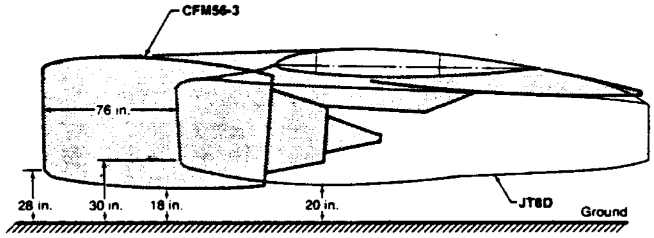


Figure 20. The 737-300/CFM56-3 and 737-200/JT8D Nacelle Relationship

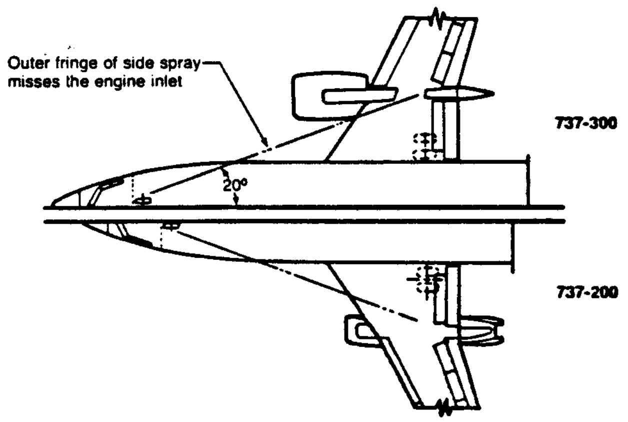


Figure 21. 737 Nose Gear Water Spray Patterns

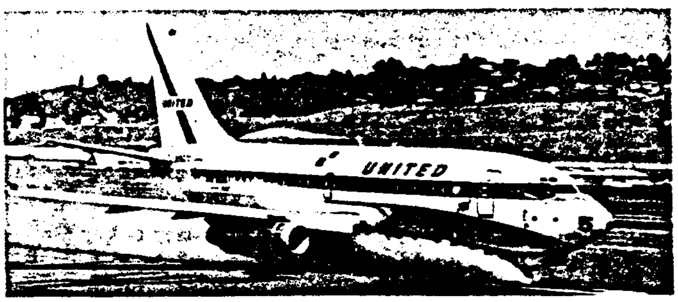


Figure 22. 737-200 Nose Gear Water Spray Testing

The 737-300 wing will have a limited number of changes due to engine relocation. Matching the engine/pylon natural frequency to the wing, to avoid flutter, is an art. A wing tip balance probe (Fig 10) was added as the most efficient means of providing acceptable flutter characteristics. A wing tip extension has been added for structural convenience. It also adds wing span which helps to reduce fuel burn by making the wing's lift efficiency higher.

Because of the larger engines/nacelles, small tip extensions were added to the horizontal tail, increasing the stabilizer span by six feet, which improves longitudinal stability.

Just a note on the flight deck – the 737-300 will be operationally common with the 737-200 airplane. The flight deck is changed to accommodate instruments unique to the CFM56-3 engine. Both models feature the increased reliability of advanced digital avionics (Fig 23). The 737-300 is to be certificated such that airline flight crews can be qualified on both models. This will be an important economic advantage to airlines with both models in their fleet.

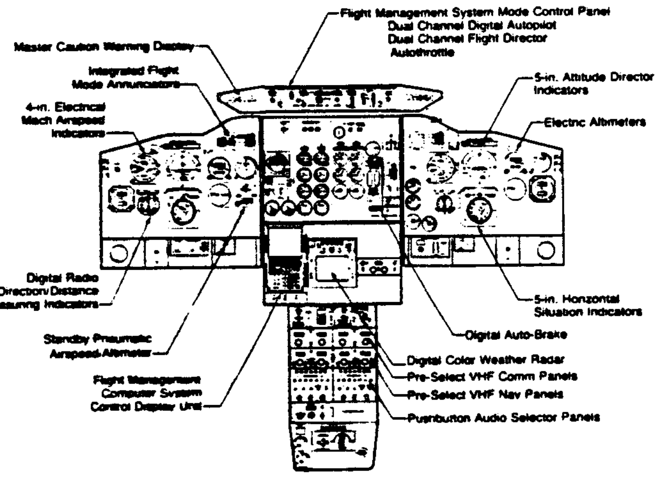


Figure 23. 737-300 Flight Deck Main Instrument Panel

Performance and Community Noise

The operational weights of the -300 have been increased relative to the -200 (Fig 24). These provide the capability to take advantage of the increased engine thrust and to accommodate the added weight due to the fuselage extension and the increase in the number of passengers.

The range capability of the -300 is increased relative to the standard -200 and is comparable to the high gross weight -200. At full passenger payload (140 passengers on the -300 and 120 on the -200), the -300 can fly 2,360 nautical miles; the basic gross weight -200, 1,750 nautical miles; and the high gross weight -200, 2,460 nautical miles.

	Adv 737-200 Basic Gross Weight	Adv 737-200 High Gross Weight	737-300 High Gross Weight
Gross Weights, lb			
Taxi	116 000	128 600	135 500
Brake Release	115 500	128 100	135 000
Landing	103 000	107 000	114 000
Zero Fuel	95 000	99 000	106 500
Passengers[1]	120	120	140
Operating Empty Wt, lb	61 490	63 170	69 400
Engines	JT8D-15A	JT8D-15A	CFM56-3
Fuel Capacity, U.S. Gal.	5 160	5 970	5 360
Lower-Hold Volume, ft³	875	640	1 068

(1) All Tourist 32-In. Pitch

Figure 24. Advanced 737-200 and 737-300 Principal Characteristics

The 737-300 has similar sea level field length-range characteristics as the basic gross weight 737-200. For 5,000 foot altitude runways, the -300 required field length is less than the -200. This translates into an improved range capability at high altitude airports such as Denver (Fig 25). For long sea level runways, such as Heathrow (London), the -300 and high gross weight -200 can fly about the same distance (Fig 26).

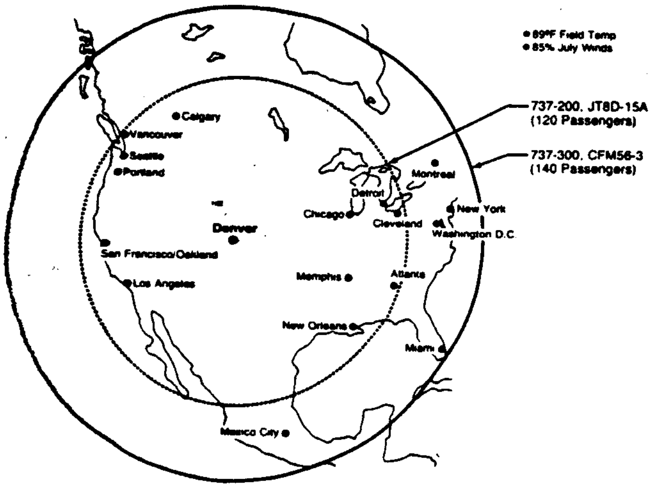


Figure 25. Range Capability From Denver

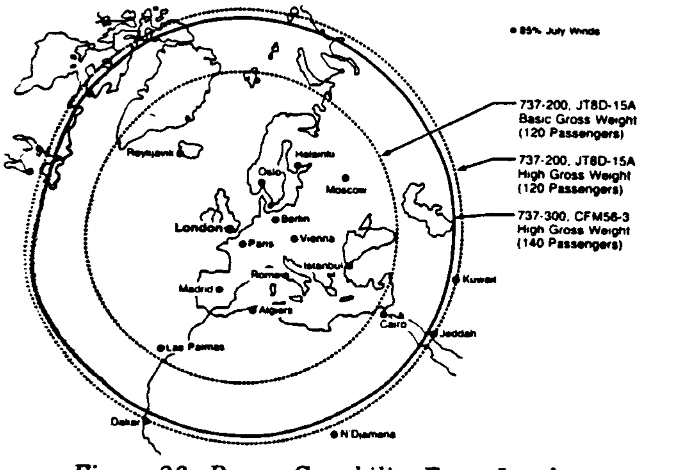


Figure 26. Range Capability From London

With the improved slat, efficient high-bypass-ratio engines and increased passenger payload, the 737-300 has an attractively low fuel burn per seat. It is lower than the JT8D-15 powered 737-200 by 25% and lower than the JT8D-15A powered 737-200 by 21% (Fig 27).

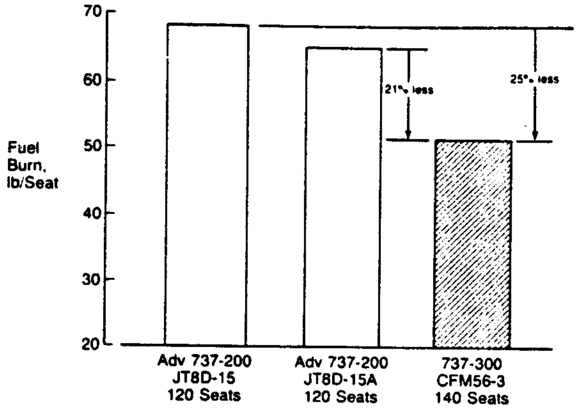


Figure 27. Fuel Burn Per Seat Comparison – 500 nmi Trip

The CFM56-3 engine was designed with noise considerations in mind. The rotors and stators have been spaced for low interaction noise and the fan duct and inlet walls are fully lined (Fig 28). The number of fan blades and fan exit guide vanes were selected and designed for low fan-tone noise. The 737-300/CFM56-3 will be quieter than the "new airplane" Stage 3 and Chapter 3 noise limits set by the U.S. Federal Aviation Administration and by the International Civil Aviation Organization (Fig 29).

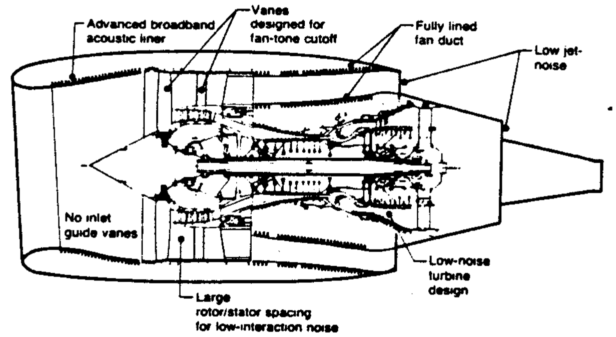


Figure 28. CFM56-3 Noise Reduction Features

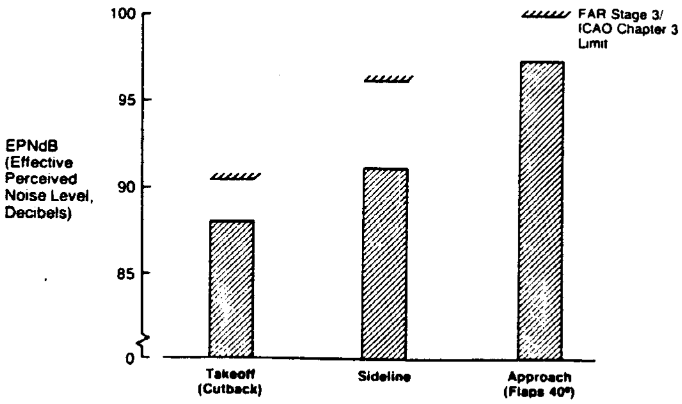


Figure 29. 737-300 Estimated Noise Levels

1.115

Economics

No matter how excellent or ingenious the design may be, the bottom line is usually "How good are the economics?" Of course, the economics will vary with assumptions such as fares, load factor, fuel price, tax considerations, and interest rates. Nevertheless, using reasonable assumptions, the 737-300 economics look good. For a 500 nautical mile trip using $1.00/gallon fuel, the 737-300 has a direct operating cost per seat 9% less than the JT8D-15 powered 737-200 airplane and 7½% less than the JT8D-15A powered airplane (Fig 30), and has a higher return on investment (ROI) than either of the older airplanes.

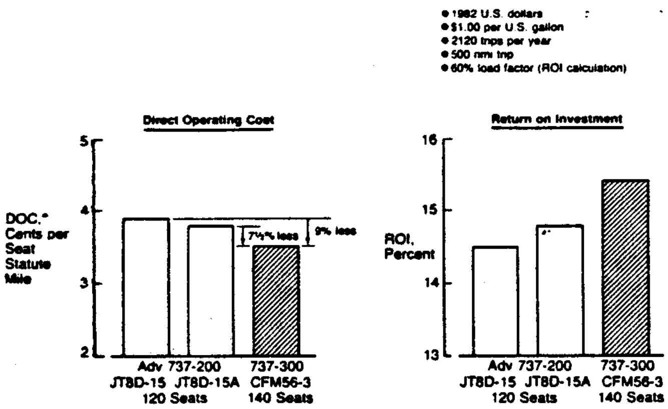


Figure 30. Operating Cost and Return on Investment

SUMMARY

Re-engining the Model 737 required a body extension, new wing leading edge, increased operational weights, and a horizontal tail span extension. Without these changes, re-engining would not have been feasible.

The 737-300 will be an example of how an existing airplane can be successfully modified by combining a high-bypass-ratio engine, with substantial in-service experience featuring improved fuel consumption with a proven, reliable airframe. It will be one of the quietest commercial turbofan powered airliners flying.

Return on investment and direct operating costs for the 737-300 are attractive. A modern cost-effective product which offers airlines low risk and excellent profit potential is the result.

Acknowledgment

The authors wish to recognize the contribution of Walter B. Gillette of the Boeing Commercial Airplane Company Aerodynamics Staff in the preparation of this paper.

Reference

1. J. C. Baer and W. M. Staab, Boeing Commercial Airplane Company, "Restoration of Performance, Models 727, 737, and 747". Paper No. 811072 presented at SAE Aerospace Congress and Exposition, Anaheim, California, October 5-8, 1981.

7 page booklet. Printed in U.S.A.

II. Design Preliminaries - Requirements (3 hours)

A. Review of Design Process

B. Mission Profiles

C. Simplified Performance Eqns.

1. Thrust or Power Required

2. Steady glide

3. Range & Endurance

4. Climb

5. Take-off, landing and stall

D. Review of Data used in Generating "Design Requirements"

E. Subsonic, Scale Dependent Drag Estimation

F. Sailplane Performance

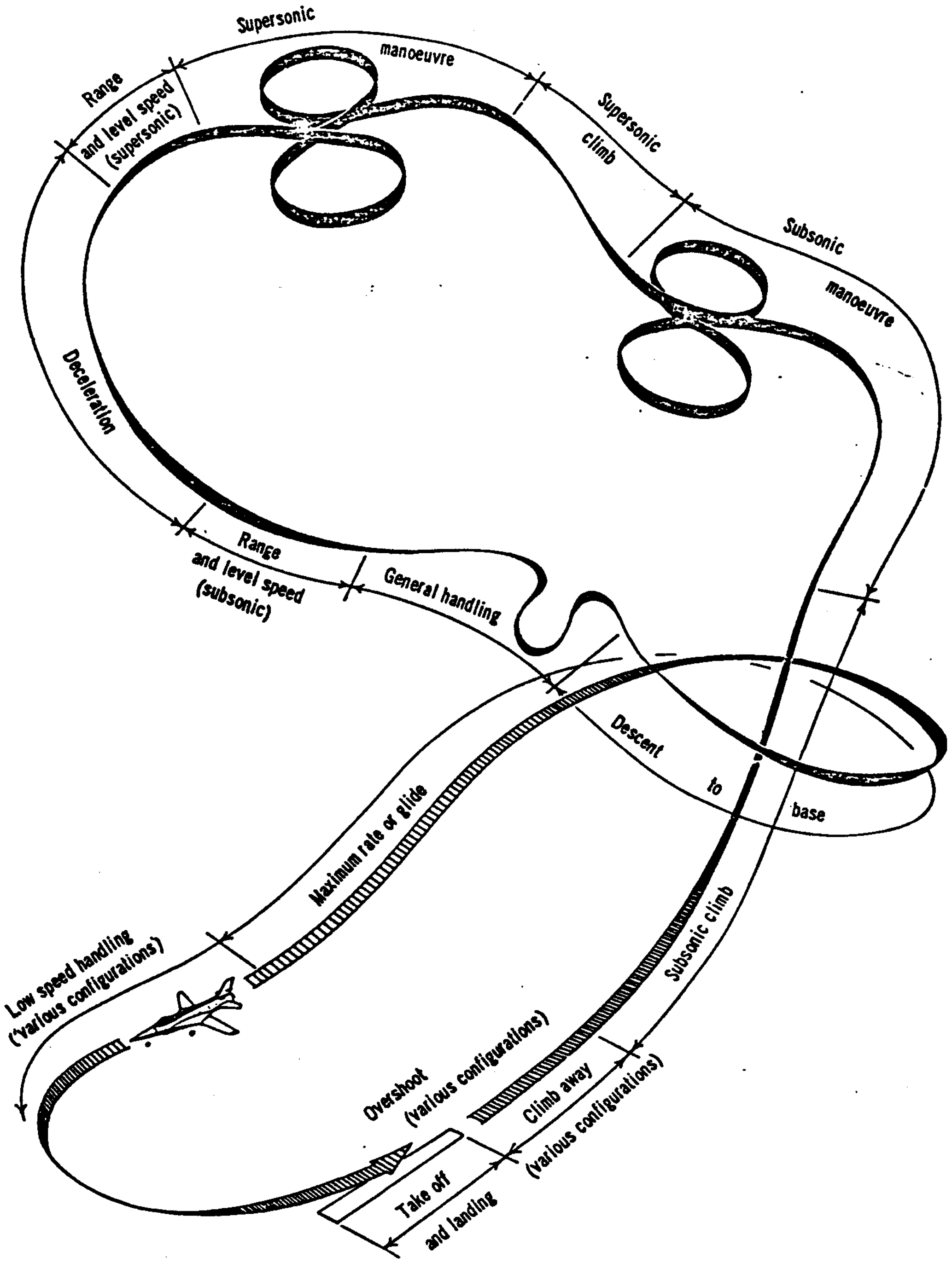
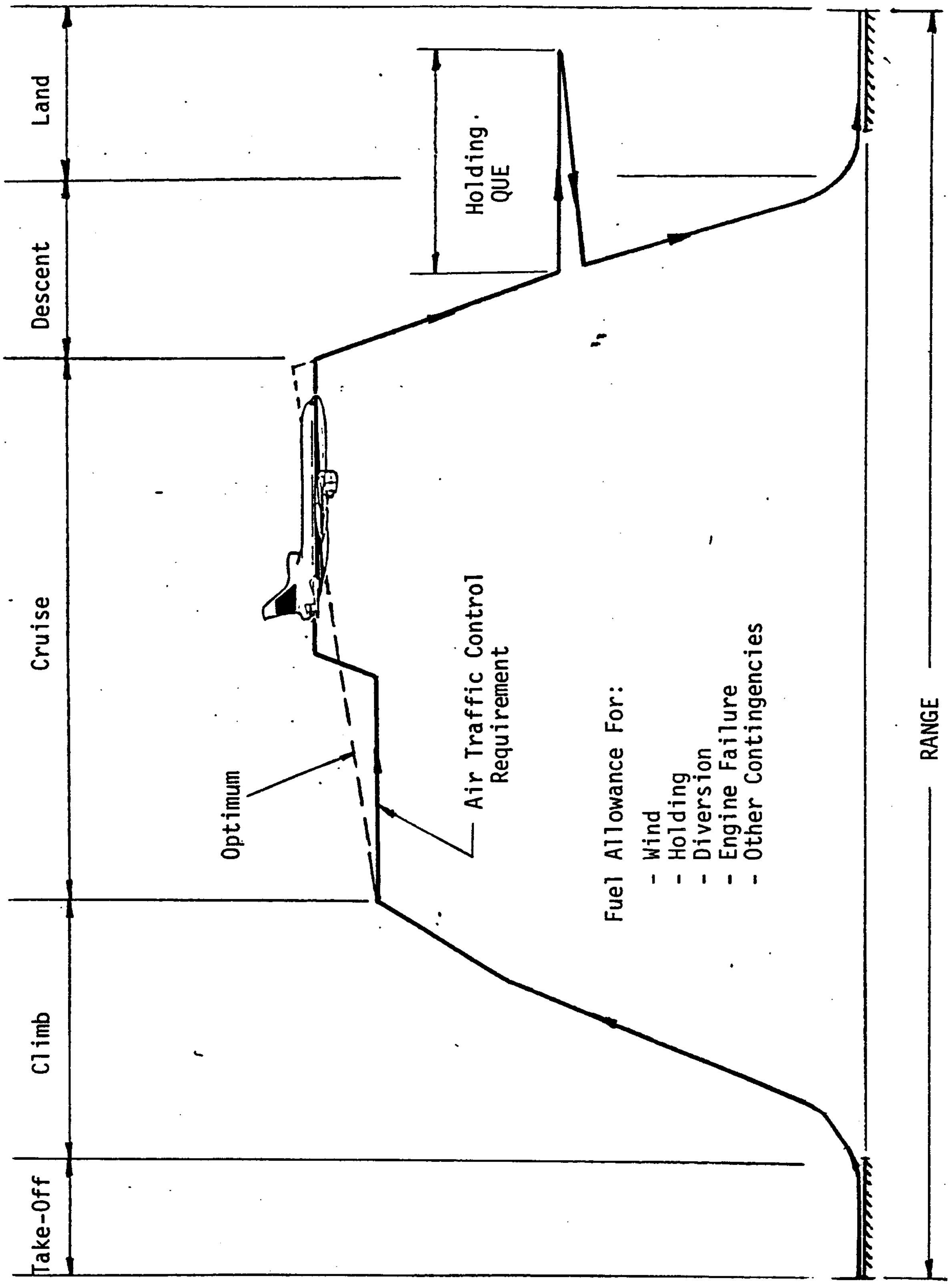


Fig. 4.1. Performance modes



CALC		9/98	REVISED	DATE	TYPICAL TRANSPORT MISSION PROFILE	
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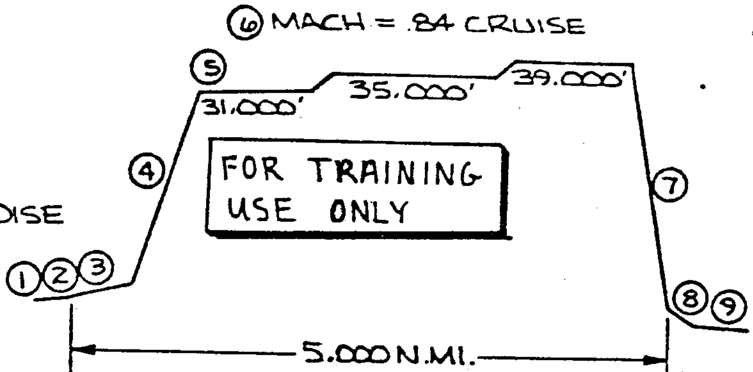
INT'L RULES
F.A.R. 121.645 (a)
AND A.T.A. 10'67

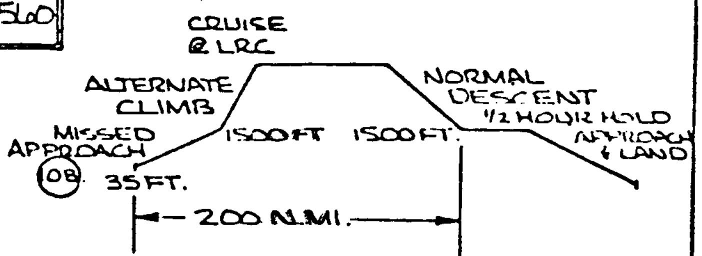
SAMPLE MISSION SUMMARY CALCULATION 747-200 JT9D ENGINES

TAXI WEIGHT 776,170 LB.
B.R.G.W. 775,000 LB.
O.E.W. 365,800 LB.
PAYLOAD 90,710 LB.
RANGE 5,000 N.MI.
FUEL LOAD 314,660 LB.
TYPE CRUISE MACH. = .84 CRUISE
TEMPERATURE STANDARD DAY
FIELD ELEVATION SEA LEVEL
BLOCK TIME 10.779 HR.
BLOCK FUEL 269,600 LB.



MISSION SEGMENTS	DATA FROM PAGE	FUEL BURNED ~LB.	FUEL REMAINING ~LB.	WEIGHT AT END OF OPERATION ~LB.	TIME ~HR	DISTANCE ~N.MI.
1. TAXI-OUT (9 MIN.)	3.24	1,170	313,490	775,000	(.15)*	—
2. TAKEOFF (S.L. TO 35 FT.)	3.25	1,190	312,300	773,810	.017	—
3. GEAR & FLAP RETRACT ACCEL (35' TO 1500' & 250 KCAS)	3.26	2,130	310,170	771,680	.033	7
WATER ALLOWANCE (5000 LB)				766,680	—	—
4. CLIMB TO 31,000 FT. (NORMAL CLIMB SCHEDULE)	5.4 5.5 5.6	24,400	285,770	742,280	.565	232
5. ACCELERATE TO CRUISE SPEED (MACH = .84)	8.6	1,670	284,100	740,610	.050	25
6. CRUISE (31-35-39,000 FT.)	9.27 9.28	235,320	48,780	505,290	9.515	4626
7. DESCENT TO 1500 FT. (NORMAL DESCENT)	10.2	1,610	47,170	503,680	.295	110
8. APPROACH & LAND	12.7	1,610	45,560	502,070	.071	—
9. TAXI-IN (5 MIN.)	12.8	(500)*			(.083)*	—
TOTAL MISSION		266,100		TRIP AIR TIME	10.546	5,000
10. RESERVES A. FUEL ALLOWANCE (10% TRIP AIR TIME)	11.6	21,360	24,200	480,710		
B. 200 N.MI. ALTERNATE WITH 1/2 HR HOLD, APPROACH & LAND	11.8	24,200	0	456,510	O.E.W & PAYLOAD	
TOTAL RESERVES		45,560				

* FOR D.O.C. COMPUTATION ONLY, NOT INCLUDED IN TOTAL MISSION. TIME INCLUDED IN BLOCK TIME. FUEL INCLUDED IN BLOCK FUEL.



CALC			REVISED	DATE	SAMPLE MISSION SUMMARY CALCULATION 747-200B JT9D	747-200
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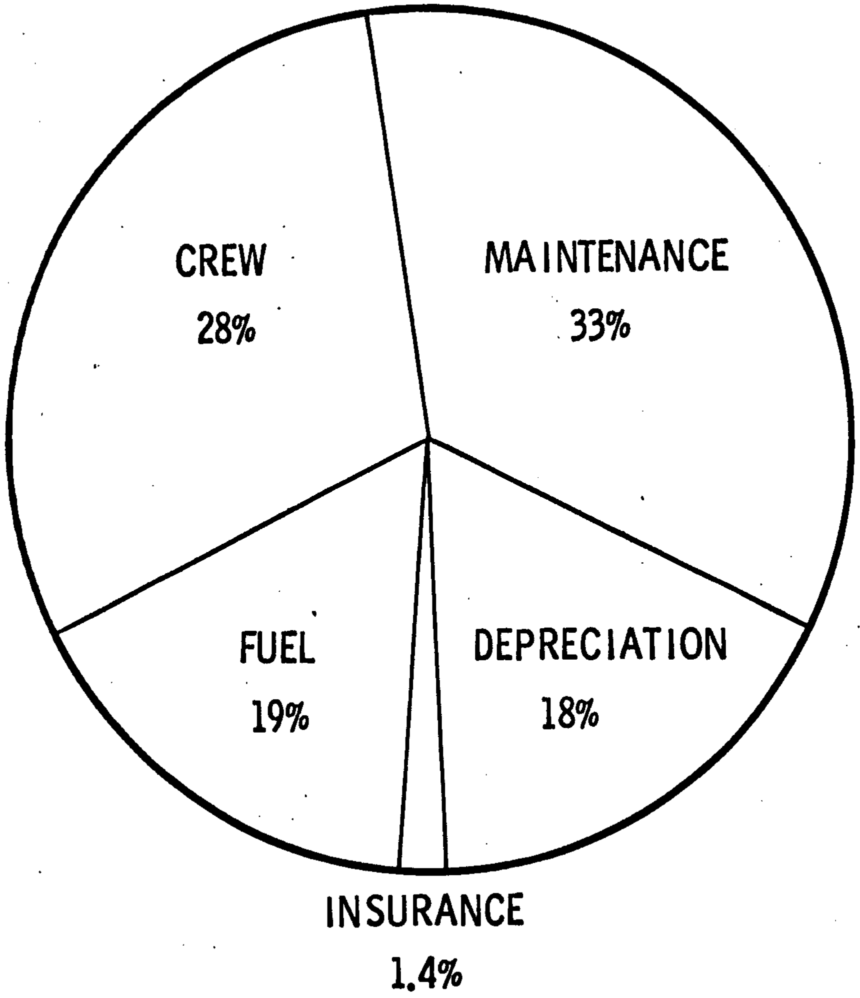
APPD						
					THE BOEING COMPANY	PAGE 3-27

D1 4100 8040 ORIG. 3/71
J18-047

Direct Operating Cost Elements

Baseline Airplane (1977 Rules - 150 nmi Trip)



NOTES:

1. MAINTENANCE COST HAS TWICE THE IMPACT OF INITIAL COST
2. HIGH UTILIZATION WITH A EFFICIENT TERMINAL ATC AND THROUGH-STOP CAPABILITIES REDUCES CREW AND FUEL OPERATING COSTS

FIGURE 1

Direct Operating Cost Computation

Notation:

C_{AP}	Airplane cost ($C_{AF} + C_{ENG}$)
C_{AF}	Airframe cost (\$)
C_{ENG}	Engine cost (per unit) (assume high by-pass turbofan)
C_{CREW}	Crew cost (assume 3 man crew)
C_{FUEL}	Fuel cost
C_{INSUR}	Insurance cost
C_{DEPR}	Depreciation (assume 15 year life)
C_{LABOR}	Labor Cost (airframe or engine)
C_{MAINT}	Maintenance cost (materials)
C_{TRIP}	Total Trip Cost
F_{NR}	Non-revenue factor = 1.02
M_{CR}	Cruise Mach number (assume ≤ 0.9)
N_E	Number of Engines
N_{PASS}	Number of Passengers
P_{CREW}	Crew pay rate
P_{LABOR}	Labor pay rate
R_{ATA}	Range (statute miles)
T_{FLT}	Flight Time
T_{BLK}	Block Time (gate-to-gate)
T_{SLS}	Sea Level Static Thrust of Engine
UTIL	Utilization Factor (hr/year) $= \dfrac{4000}{1 + \dfrac{1}{T_{BLK} + .5}} + 630$

2.06

V_{CR} Cruise Speed $= 715\, M_{CR} - 75\, M_{CR}^4$

W_{AF} Airframe Weight (w/o engines)
($W_{AF} = MEW - N_E W_{ENG}$)

MEW Manufactures Empty Weight

W_{ENG} Engine Weight (each)

W_{FUEL} Fuel Weight (assume 6.7 lb/gal).

W_{TO} Take-off Weight

$$\boxed{D.O.C. = \text{direct operating cost} \left(\frac{\text{cents}}{\text{seat st. mile}}\right) = 100\, \frac{C_{TRIP}}{N_{PASS} \times R_{ATA}}}$$

$$* \quad C_{TRIP} = C_{CREW} + C_{INSUR} + C_{DEPR} + \underset{\text{Total}}{C_{LABOR}} + \underset{\text{Total}}{C_{MAINT}} + \underset{\text{Burden}}{C_{MAINT}} + C_{FUEL}$$

<u>Cost Breakdown:</u>

$$C_{AF} = C_1 \times 10^6 + C_2\, W_{AF} \qquad \begin{cases} C_1 = 5 \\ C_2 = 86 \end{cases}$$

$$C_{ENG} = \left[C_3 \times 10^3 + C_4\, T_{SLS} \right] N_E \qquad \begin{cases} C_3 = 700 \\ C_4 = 17.76 \end{cases}$$

$$\therefore \quad \boxed{C_{AP} = \text{Total Airplane Cost (in millions of \$)} = \frac{C_{AF}}{10^6} + \frac{C_{ENG}}{10^3}}$$

$$C_{\substack{CREW\\PER\\BLOCK\\HOUR}} = CPHR = x_1 \left(V_{CR} \; \frac{W_{TO}}{10^6} \right)^{0.3} + x_2$$

$$x_1 = 29.792$$

$$x_2 = \begin{cases} 30.750 & \text{domestic (600 hr/year)} \\ 62.757 & \text{internat. (550 hr/year)} \end{cases}$$

$$\boxed{C_{CREW} = (CPHR) \times T_{BLK}}$$

$$\boxed{C_{FUEL} = (\text{Fuel Price per gallon}) \times (\text{Fuel - lbs/blk}) \times \frac{F_{NR}}{6.7}}$$

$$\boxed{C_{INSUR} = (0.01) \left[\frac{C_{AP} \times T_{BLK}}{UTIL \times 100} \right]}$$

$$C_{DEPR} = (1 - \text{Residual}) \left[(1 + Spare_{AF}) C_{AF} + (1 + Spare_{ENG}) C_{ENG} \right] \times$$

$$\times \left[\frac{T_{BLK} \times UTIL}{Year_{DEPR.}} \right]$$

$\text{Residual} = 0.10 \quad \sim 10\%$

$Spare_{AF} = 0.06 \quad \sim 6\%$

$Spare_{ENG} = 0.30 \quad \sim 30\%$

$Year_{DEPR} = 15$ years

$$C_{LABOR} = \left[C_{\substack{LABOR\\per\\cycle}} + C_{\substack{LABOR\\per\\hour}} \times T_{FLT} \right] \times F_{NR} \times P_{LABOR}$$

@ x \$/hr.

$$C_{MAINT} = \left[C_{\substack{MAINT\\per\\cycle}} + C_{\substack{MAINT\\per\\hour}} \times T_{FLT} \right] \times F_{NR}$$

$$C_{\substack{MAINT\\BURDEN}} = (2.0) \times C_{LABOR}$$

$$C_{LABOR \text{ per cycle}} = \begin{cases} \dfrac{W_{AF}}{1000} \Big/ \left(0.0419\ \dfrac{W_{AF}}{1000} + 28.159\right) & \text{airframe} \\[2ex] \left(0.0244 \times \dfrac{T_{SLS}}{1000} + 0.220\right) N_E & \text{engine} \end{cases}$$

$$C_{LABOR \text{ per hour}} = \begin{cases} \dfrac{W_{AF}}{1000} \Big/ \left(0.1035\ \dfrac{W_{AF}}{1000} + 17.919\right) & \text{airframe} \\[2ex] \left(0.0183 \times \dfrac{T_{SLS}}{1000} + 0.178\right) N_E & \text{engine} \end{cases}$$

$$C_{MAINT \text{ per cycle}} = \begin{cases} 1.235 + 2.261 \left(\dfrac{C_{AF}}{10^6}\right) & \text{airframe} \\[2ex] \left(16.0 \times C_{ENG} + 19.5\right) N_E & \text{engine} \end{cases}$$

$$C_{MAINT \text{ per hr}} = \begin{cases} 2.508 + 1.736 \left(\dfrac{C_{AF}}{10^6}\right) & \text{airframe} \\[2ex] \left(10.256 \times C_{ENG} + 18.115\right) N_E & \text{engine} \end{cases}$$

Relative Maintenance Cost

1974 Boeing 737 Line Replacement Data

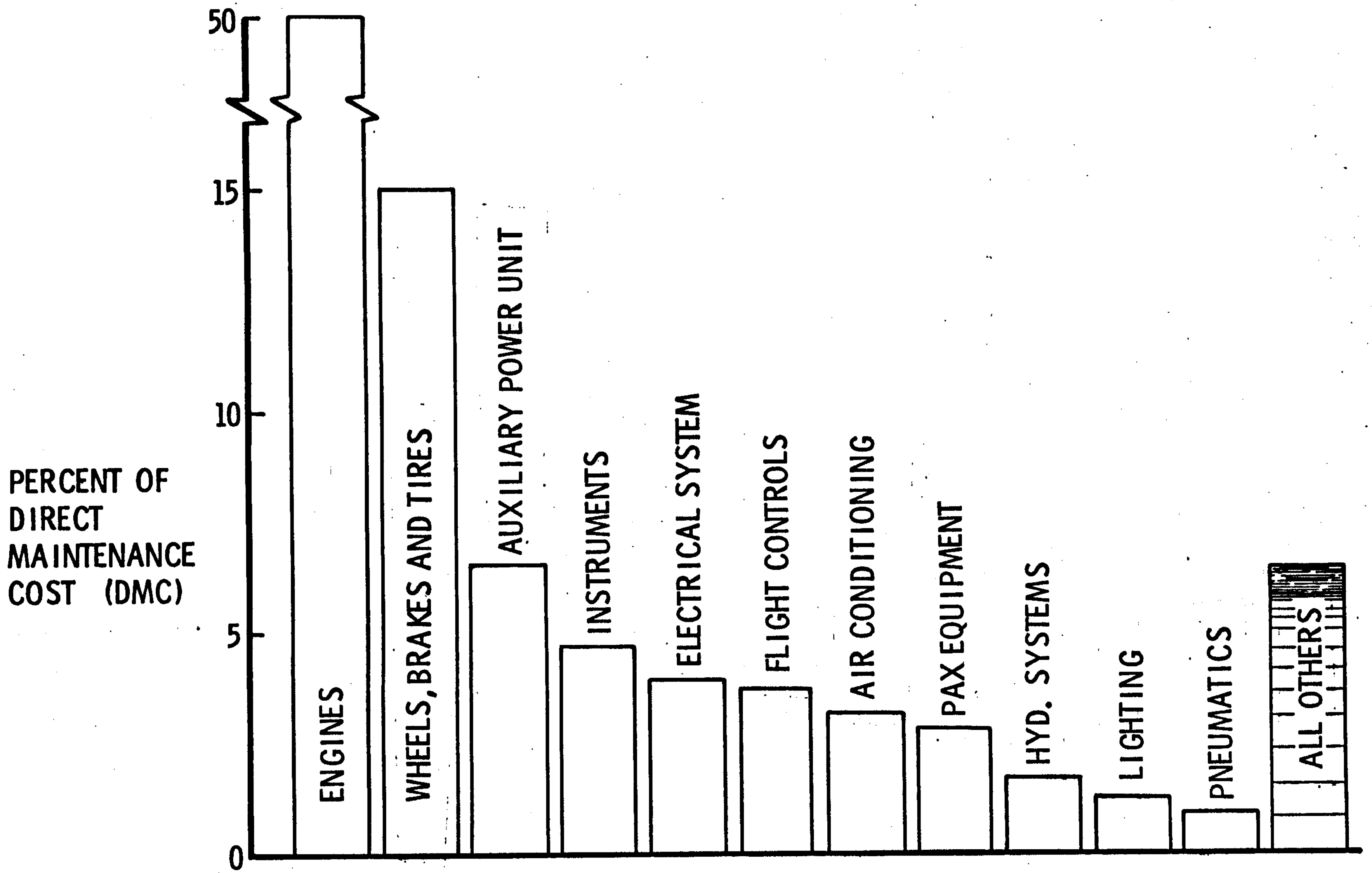
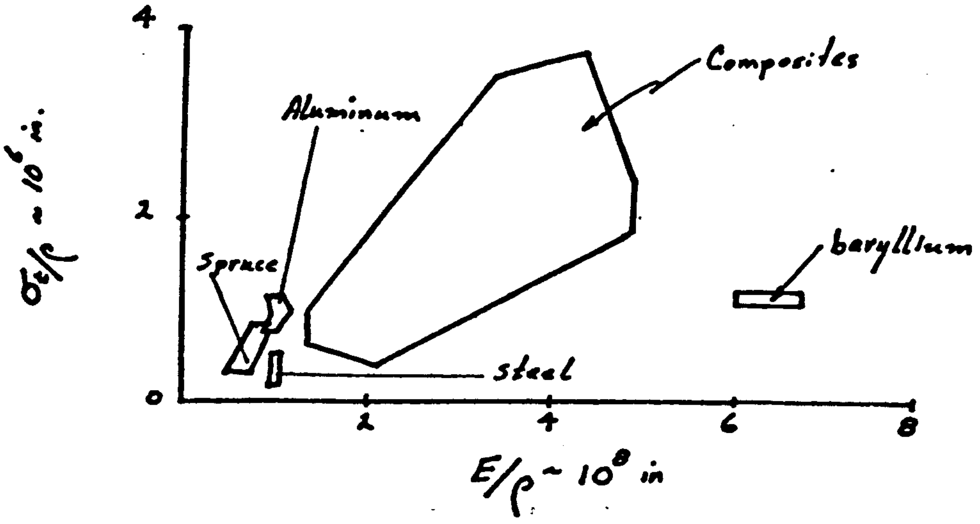
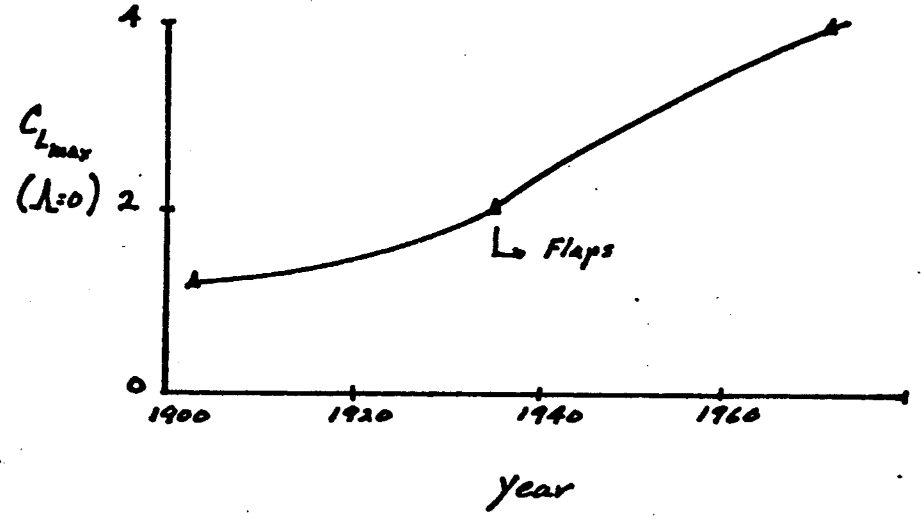


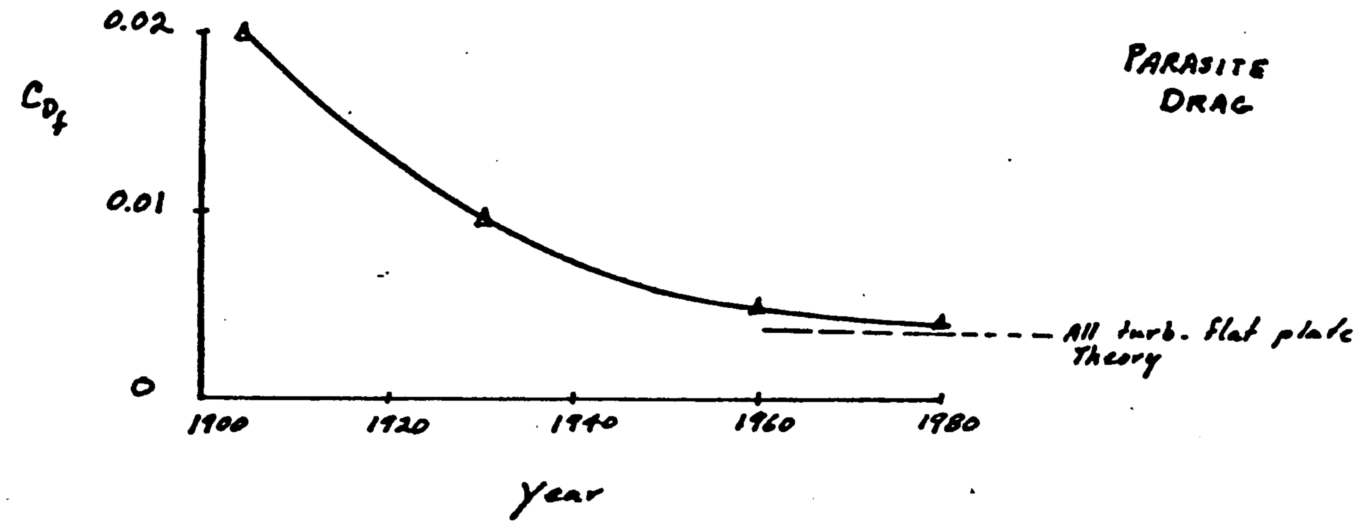
FIGURE 2



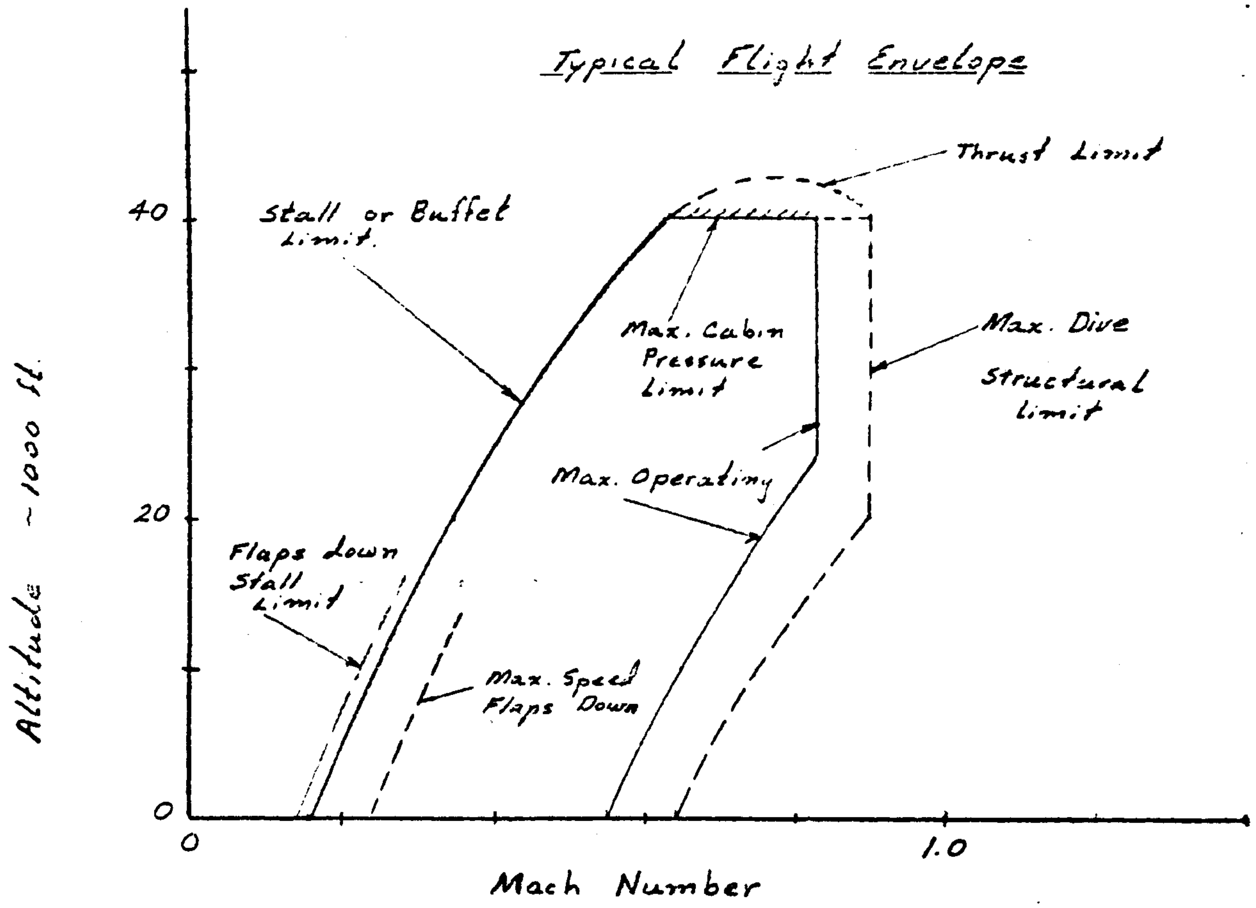
MATERIALS

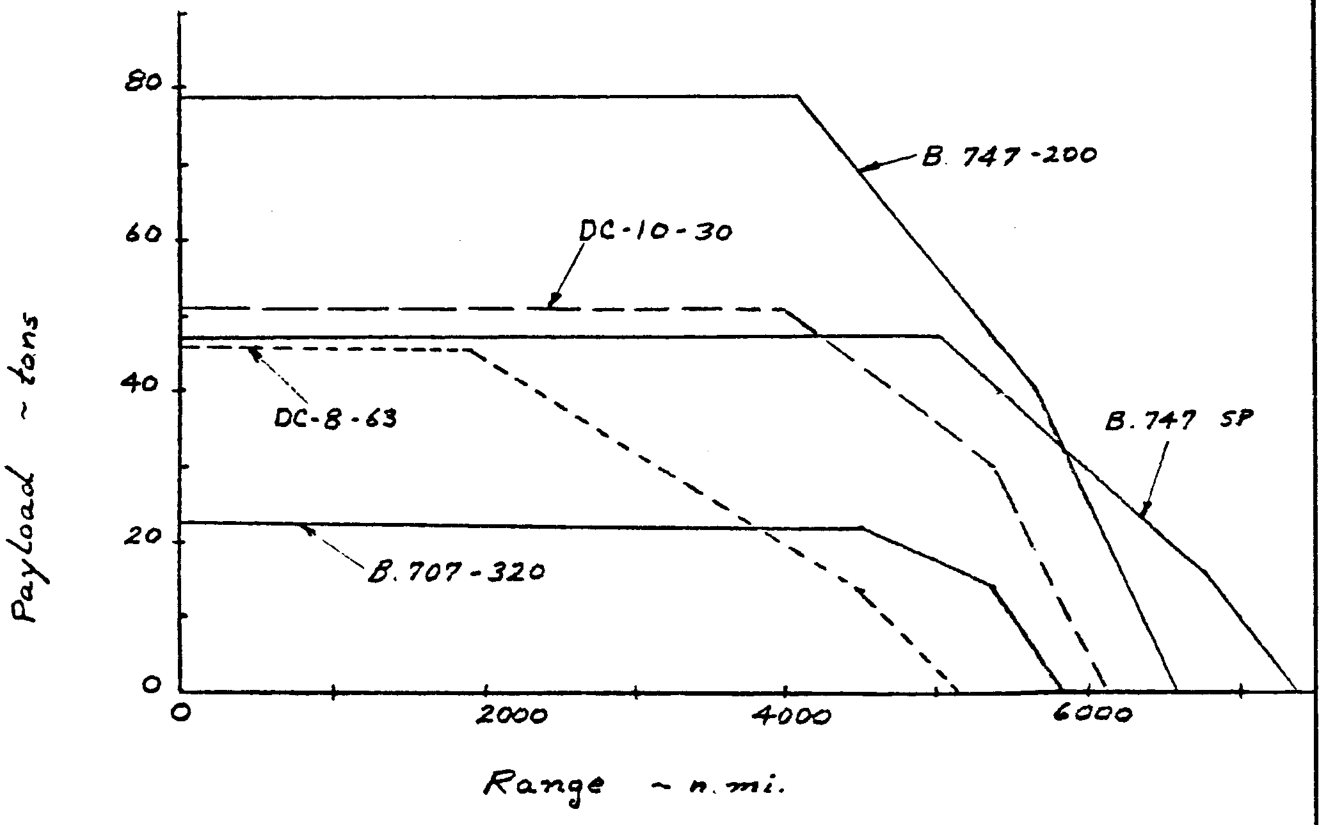


MAX. LIFT



PARASITE DRAG





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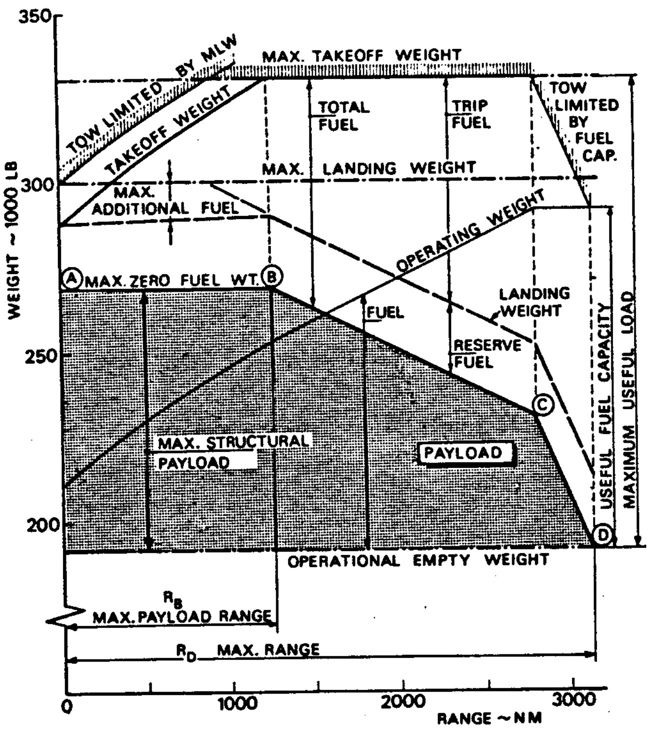
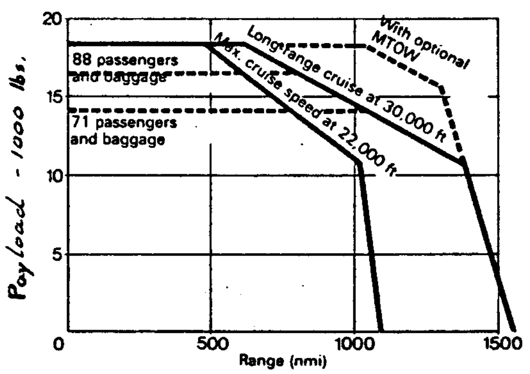
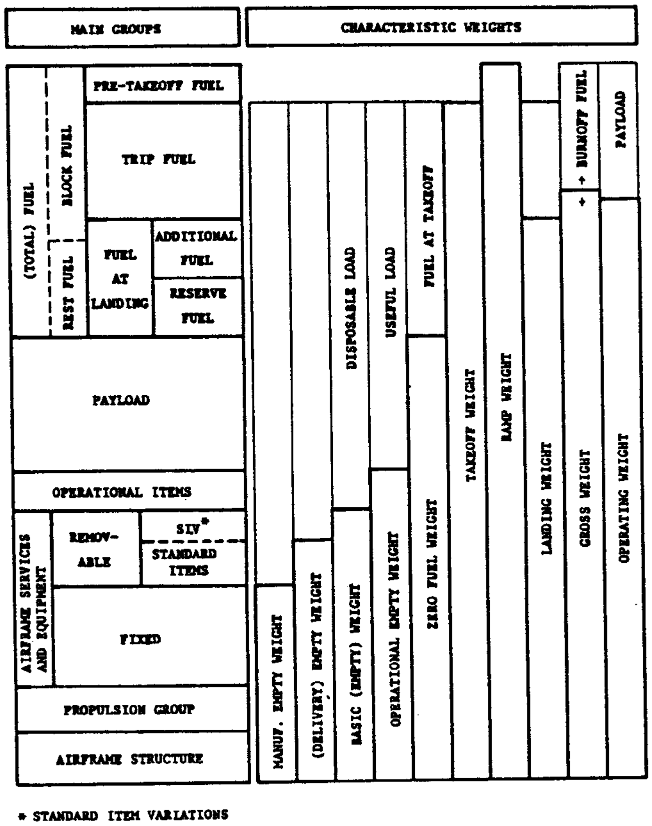


Derivation of the payload-range diagram.



Payload-range diagram for the Series 100 aircraft, assuming ISA conditions, reserves for a 150 nmi (280 km) diversion and 45 minutes hold at 1,500 m (5,000 ft), and a standard tankage of 11,540 litres (2,540 Imp gal).



Weight groups and characteristic weight terminology (not to scale)

WEIGHT TERMINOLOGY

MTOW - Max. Take-off Weight (max. wt. at instant of brake release)

MLW - Max. Landing Weight (max. wt. at instant of touch down)

OEW - Operating Empty Weight (no payload or fuel)

ZFW - Zero Fuel Weight (OEW plus payload)

MEW - Manufacturers Empty Weight (basic empty weight of "dry" airplane as it first rolls out the door)

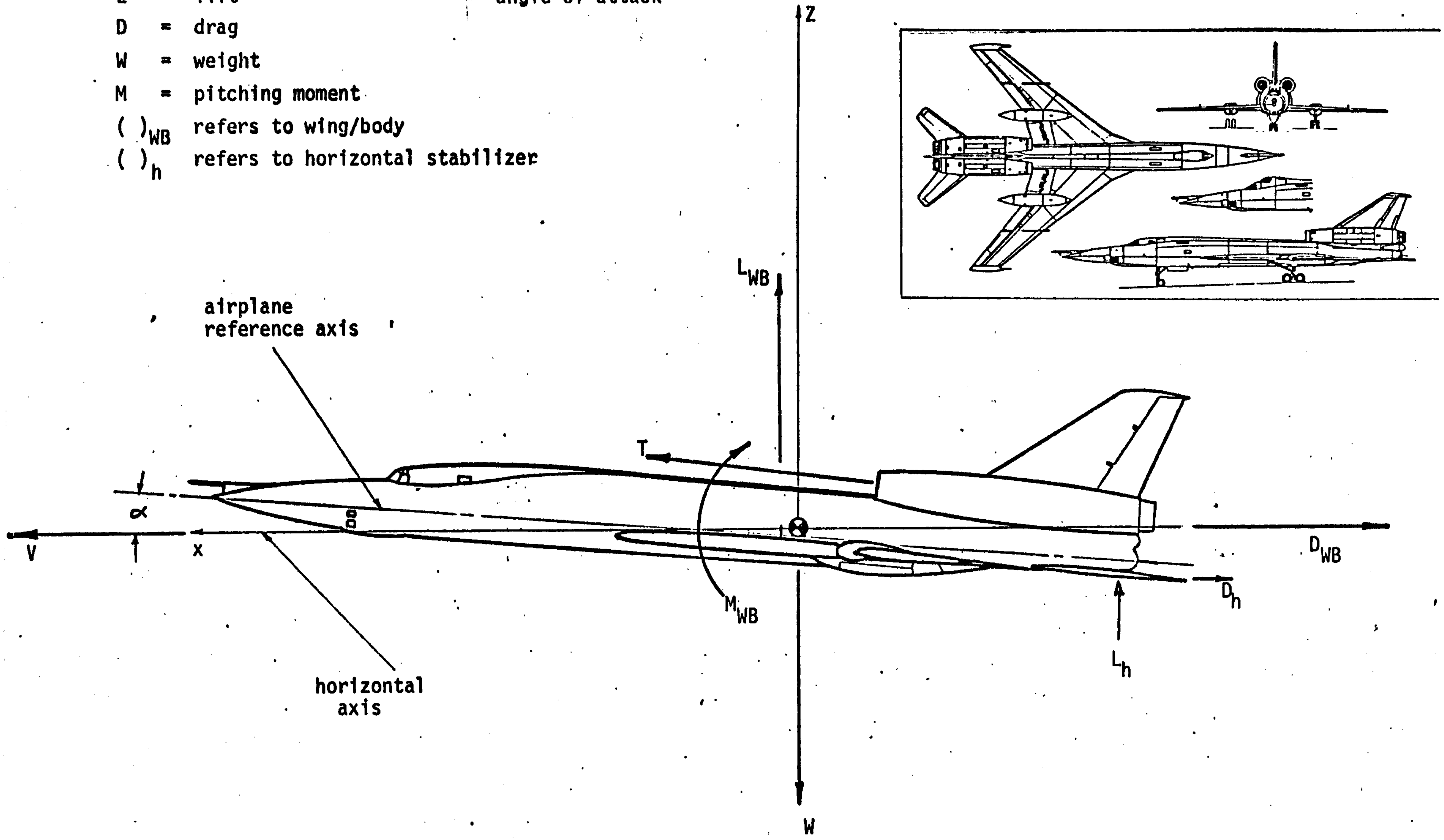
Wg - Gross Weight (total weight at any given time)

Ramp Weight - TOW plus fuel for engine run-up and taxi

Range - Payload Diagrams.

T = thrust
L = lift
D = drag
W = weight
M = pitching moment
$(\ )_{WB}$ refers to wing/body
$(\ )_h$ refers to horizontal stabilizer

V = velocity
α = angle of attack



Simplified Forces and Moments on an Airplane in Steady Level Flight

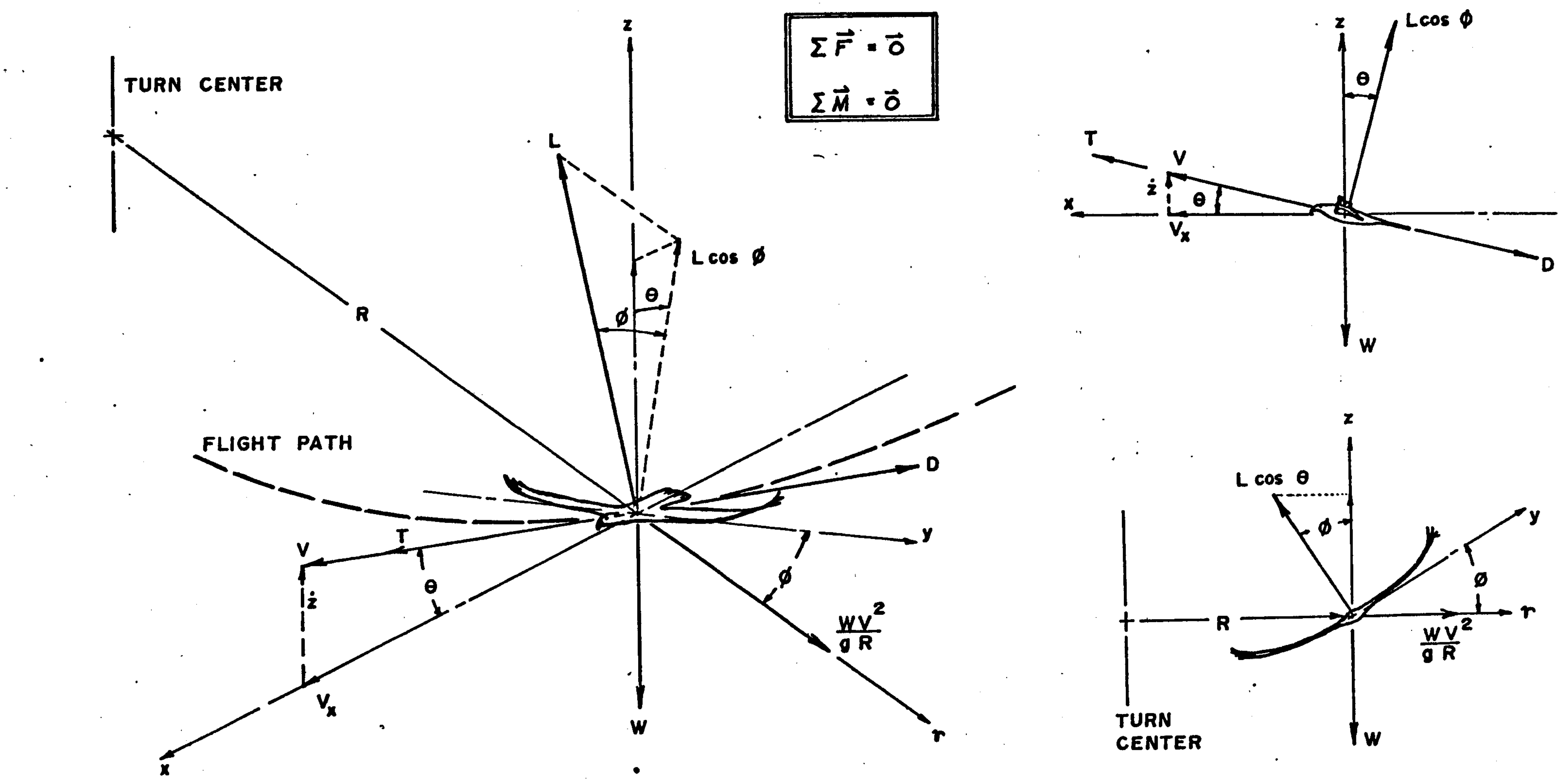


FIGURE 27. FORCES ON AN AIRCRAFT IN A STEADY CLIMBING OR GLIDING TURN

2.15

Basic Aircraft Performance
(Subsonic)

1. <u>Thrust or Power Required</u>

In cruise $\Sigma \vec{F} = \vec{0} \longrightarrow$ $T = D$, $L = W$ $\qquad P \equiv TV$

$$L = \tfrac{1}{2}\rho V^2 C_L S$$

$$D = \tfrac{1}{2}\rho V^2 C_D S \longrightarrow C_D = C_{D_0} + \frac{K C_L^2}{\pi A\!R} \; ; \; A\!R = b^2/S$$

$$\therefore \boxed{\begin{aligned} T = D &= \frac{\rho}{2}(C_{D_0} S) V^2 + \left(\frac{2K}{\rho\pi}\right)\left(\frac{W}{b}\right)^2 \cdot \frac{1}{V^2} \\ P_{req.} \equiv TV &= \frac{\rho}{2}(C_{D_0} S) V^3 + \left(\frac{2K}{\rho\pi}\right)\left(\frac{W}{b}\right)^2 \cdot \frac{1}{V} \leq P_{\substack{available \\ max.}} \end{aligned}}$$

Brake power $= BP \equiv \frac{TV}{\eta} = \frac{DV}{\eta} = \frac{WV}{\eta(L/D)}$ since $L = W$

$$\text{optima} \begin{cases} T_{min} \Rightarrow C_{D_0} = \frac{K C_L^2}{\pi A\!R} \longrightarrow \frac{L}{D}_{max} \\ P_{min} \Rightarrow 3C_{D_0} = \frac{K C_L^2}{\pi A\!R} \end{cases}$$

2. <u>Climb</u>

$$\dot{h} = \text{rate of climb} = \frac{\eta(BP_{avial.} - BP_{req.})}{W}$$

3. <u>Range & Endurance</u>

Propeller driven aircraft

$$\text{Range (st. miles)} = 375 \frac{\eta}{C_p}\left(\frac{L}{D}\right) \log_e \frac{W_0}{W_1}$$

$$\text{Endurance (hours)} = \sqrt{2\rho S}\, \frac{\eta}{C_p}\left(\frac{C_L^{3/2}}{C_D}\right)\left[\frac{1}{\sqrt{W_1}} - \frac{1}{\sqrt{W_0}}\right]$$

Jet Aircraft

$$R(\text{miles}) = \frac{1.929}{C_t \sqrt{\rho S}} \cdot \left(\frac{C_L^{1/2}}{C_D}\right)\left(\sqrt{W_0} - \sqrt{W_1}\right)$$

$$E\ (\text{hrs}) = \frac{1}{C_t}\left(\frac{L}{D}\right) \log_e\left(\frac{W_0}{W_1}\right)$$

where W_0 = initial weight (lbs)

W_1 = final weight (lbs)

c_p = specific fuel consumption $\frac{lb_{fuel}}{BHP \cdot hr.}$

c_t = " " " " $\frac{lb_{fuel}}{lb_{thrust} \cdot hr.}$

η = propeller efficiency

4. <u>Take-off and Minimum Speed.</u>

$$V_{stall} = \left\{\frac{W/S}{\rho/2\ C_{L_{max}}}\right\}^{1/2}$$

$$S_{TO} = \text{take-off ground run} \cong \frac{1}{2g}\frac{V_{TO}^2}{\overline{T}_N/W}$$

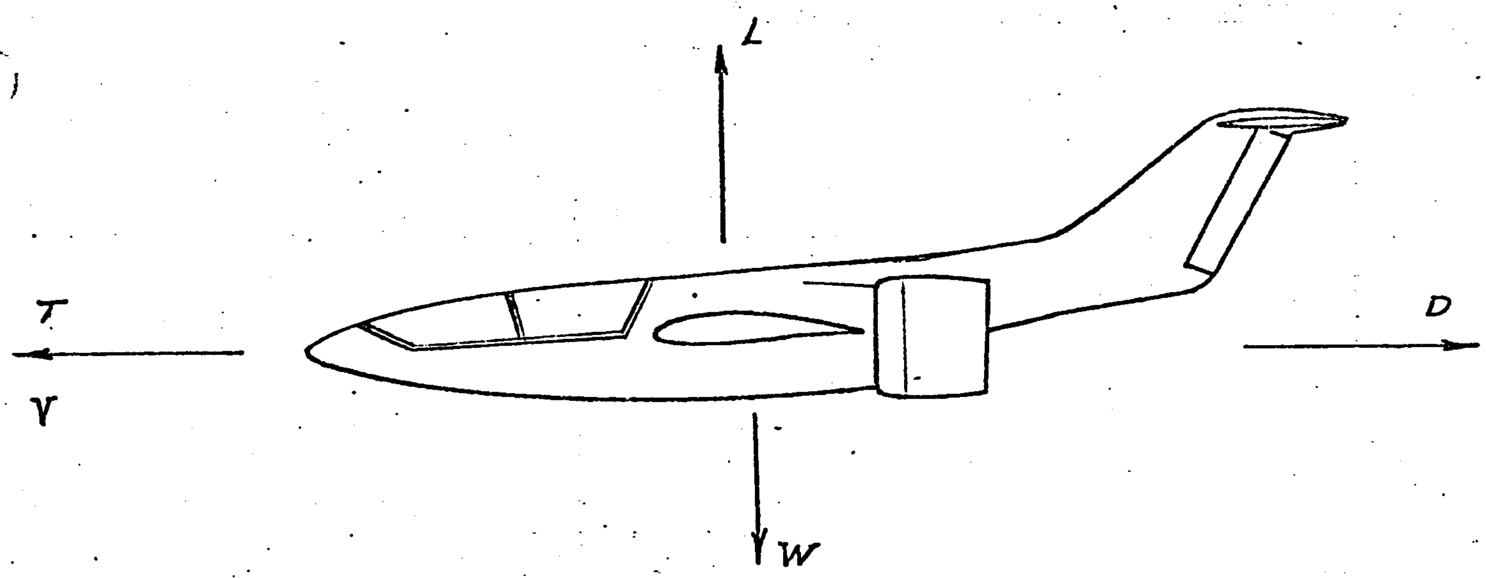
where $\overline{T}_N$ = average accelerating force

$= \overline{T} - \overline{D} - \overline{R}$

$\overline{R} = \mu(W - \overline{L})$

μ = rolling friction coeff.

Level Flight Performance Summary



Lift $= L \cong$ Weight $= W$ $\qquad L = \frac{1}{2}\rho V^2 C_L S \longrightarrow V = \sqrt{\frac{2W/S}{\rho C_L}}$

Thrust $= T \cong$ Drag $= D$ $\qquad D = \frac{1}{2}\rho V^2 C_D S$

$$C_D = C_{D_{tr}} + C_{D_p} + C_{D_i} + C_{D_t} \cong C_{D_0} + \frac{kC_L^2}{\pi A\!R} \; ; \; C_{L_1} \le C_L \le C_{L_2}$$

$$A\!R = \frac{b^2}{S} \doteq (\text{span})^2/\text{area}$$

$BP_{req.}$ = brake power required $= \dfrac{TP}{\eta} = \dfrac{\text{Thrust Power Required}}{\text{Propeller Efficiency}}$

$$\text{Thrust Required} \cong \frac{\rho}{2}V^2 C_D S = \frac{\rho}{2}(C_{D_0}S)V^2 + \left(\frac{2K}{\pi\rho}\right)\left(\frac{W}{b}\right)^2 \cdot \frac{1}{V^2}$$

$$BP_{required} \cong \frac{\rho}{2\eta}V^3 C_D S = \frac{1}{\eta}\left\{\frac{\rho}{2}(C_{D_0}S)V^3 + \left(\frac{2K}{\pi\rho}\right)\left(\frac{W}{b}\right)^2 \cdot \frac{1}{V}\right\}$$

$$BP_{required} = \frac{WV}{\eta(L/D)} = BP_{available} \qquad \text{(steady, level, flight)}$$

Range $= R$ $\qquad$ [W_0 = gross wt. at start of cruise
W_1 = gross wt. at end of cruise]

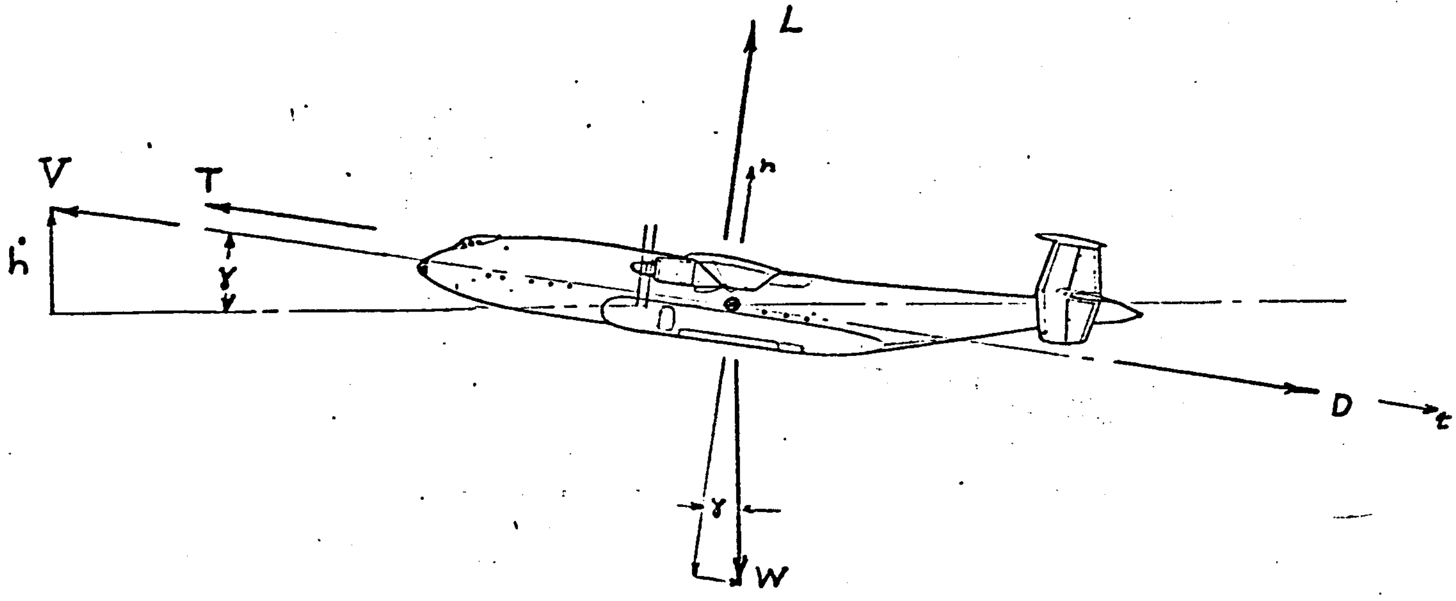
$$R_{Prop.} = C_0 \frac{\eta}{c_p}(L/D) \log_e W_0/W_1$$

c_t, c_p ~ specific fuel consumptions

$$R_{jet} = C_0' \frac{(C_L^{1/2}/C_D)}{c_t\sqrt{\rho S}}\left(\sqrt{W_0} - \sqrt{W_1}\right)$$

C_0, C_0' ~ dimensional constants

Simplified Climb Performance



$$\left.\begin{aligned}\Sigma F_n = 0 &= L - W\cos\gamma \\ \Sigma F_t = 0 &= T - D - W\sin\gamma\end{aligned}\right\} \longrightarrow$$

if $\dot{h} \ll V$

$L \cong W$

$\gamma \cong \dfrac{\dot{h}}{V} \cong \dfrac{T-D}{W}$

$$\therefore \boxed{\begin{aligned}\gamma &\cong \frac{T}{W} - \frac{1}{(L/D)} \\ \dot{h} &\cong V\left[\frac{T}{W} - \frac{1}{L/D}\right]\end{aligned}}$$

$\dot{h}$ = rate of climb

γ = climb angle

BP = Brake Power $\cong \dfrac{TV}{\eta}$

η = propeller efficiency

$BP_{available} \geqslant BP_{required}$ for level flight or climb

$$\therefore \quad BP_{required} \cong \frac{W\dot{h}}{\eta} + \frac{DV}{\eta} = \frac{W\dot{h}}{\eta} + BP\Big|_{\dot{h}=0}$$

$$= \frac{W}{\eta}\left[\dot{h} + \frac{1}{L/D}\right]$$

$$\dot{h} \cong \frac{\eta}{W}\left[BP_{avail.} - BP_{req.}\Big|_{\dot{h}=0}\right]$$

PERFORMANCE SUMMARY

Assume:

- Steady, level Flight in still Air (No. Wind)
- W = weight $\cong L$ = lift $= \frac{1}{2}\rho V^2 C_L S$
- T = thrust $\cong D$ = drag $= \frac{1}{2}\rho V^2 C_D S$
- $C_D \begin{cases} = C_{D_0} + K C_L^2/\pi A\!\!R \\ = C_{D_0} + C_{D_i} \end{cases}$ (Parabolic drag polar approximation)

$(\ )^*$ ~ optimum value

$V = \sqrt{\dfrac{2\,W/S}{\rho C_L}}$

$A\!\!R = b^2/S$

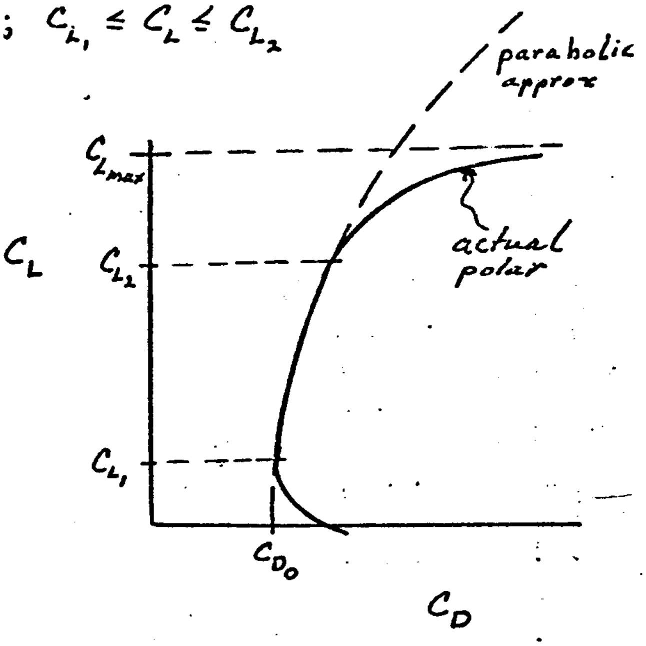
Condition	Drag Optimum	Optimum Performance: Propeller	Optimum Performance: Jet	Optimum speed (V^*)	Optimum Lift Coefficient (C_L^*)
$C_L/C_D\big\vert_{max} = L/D\big\vert_{max}$	$C_{D_0} = C_{D_i}$ $C_D^* = 2C_{D_0} = 2C_{D_i}$	Max. Range [Max. Glide Distance (Glider)]	Max. Endurance Min. Thrust	$= \sqrt{\dfrac{2(W/S)K}{\rho\pi C_{D_0} A\!\!R}}$	$= \sqrt{C_{D_0}\pi A\!\!R/K}$
$\dfrac{C_L^{3/2}}{C_D}\bigg\vert_{max} \Rightarrow TV\big\vert_{min}$	$3C_{D_0} = C_{D_i}$ $C_D^* = 4C_{D_0} = \frac{4}{3}C_{D_i}$	Min. Power Max. Endurance Max. Rate of Climb [Min. Sink Rate (glider)]	——	$= \sqrt{\dfrac{2(W/S)K}{3\rho\pi C_{D_0} A\!\!R}}$ $\geq V_{stall}$	$= \sqrt{3C_{D_0}\pi A\!\!R/K}$ $\leq C_{L_{max}}$
$\dfrac{C_L^{1/2}}{C_D}\bigg\vert_{max} \Rightarrow \dfrac{T}{V}\bigg\vert_{min}$ $\propto (M\,L/D)_{max}$	$C_{D_0} = 3C_{D_i}$ $C_D^* = \frac{4}{3}C_{D_0} = 4C_{D_i}$	——	Max. Range	$= \sqrt{\dfrac{6(W/S)K}{\rho\pi C_{D_0} A\!\!R}}$ $\leq M_{critical}$	$= \sqrt{C_{D_0}\pi A\!\!R/3K}$ $< C_{L_{buffet}}$

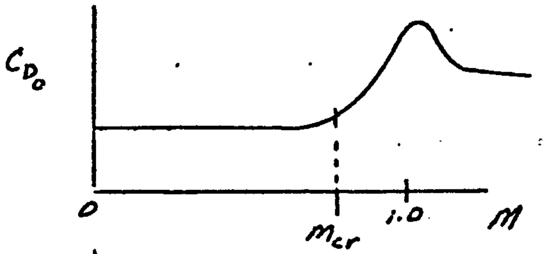
Typical Airplane Parameters

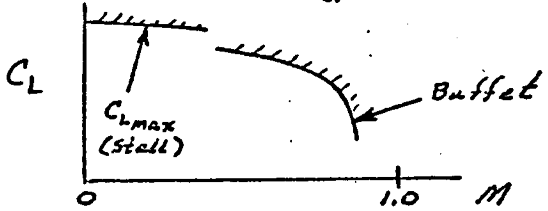
$$C_D = C_{D_0} + \frac{K C_L^2}{\pi A\!\!R} \quad ; \quad C_{L_1} \le C_L \le C_{L_2}$$

$$0 \le C_L \le C_{L_{max}}$$

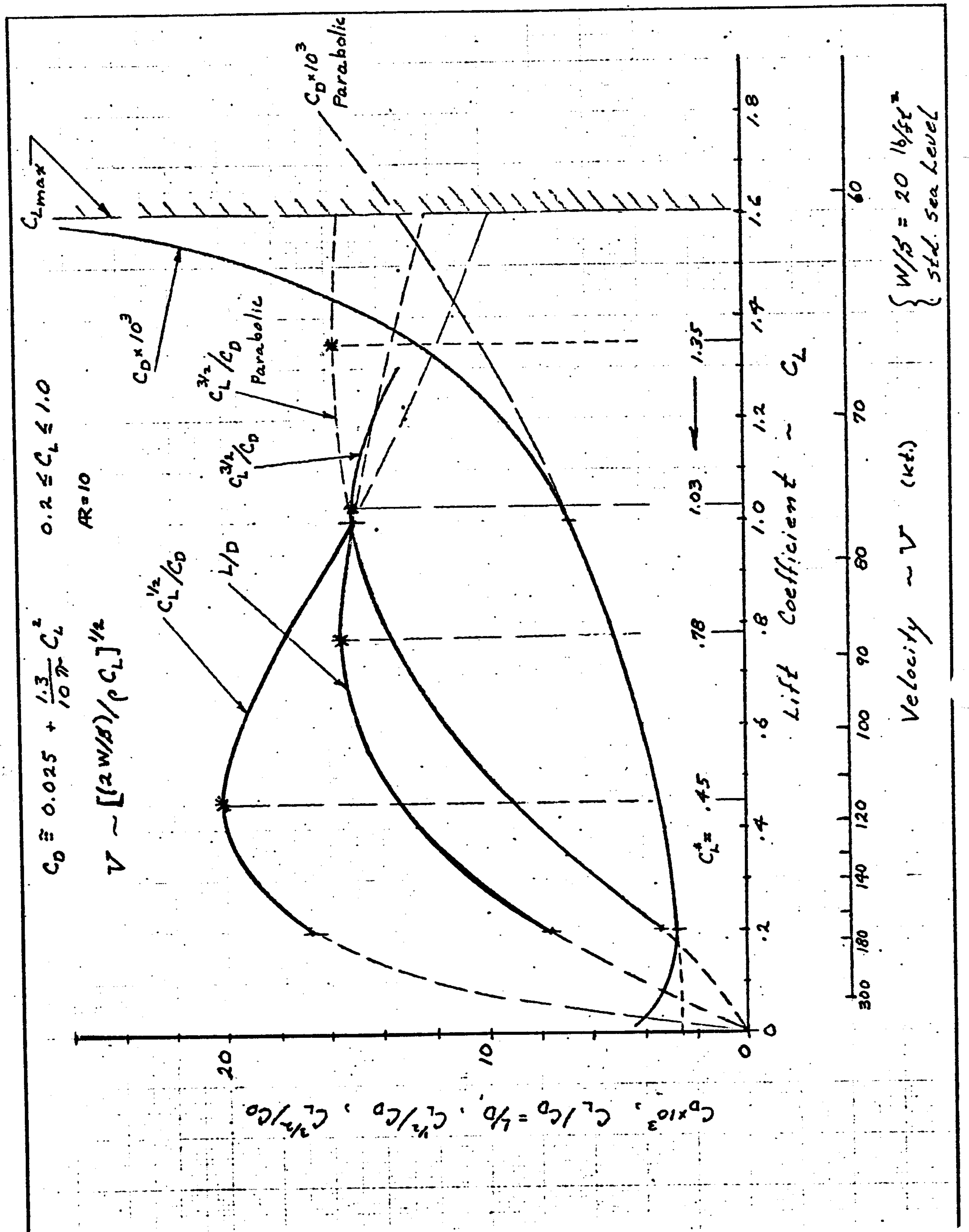
$$A\!\!R = (\text{span})^2 / \text{wing area}$$



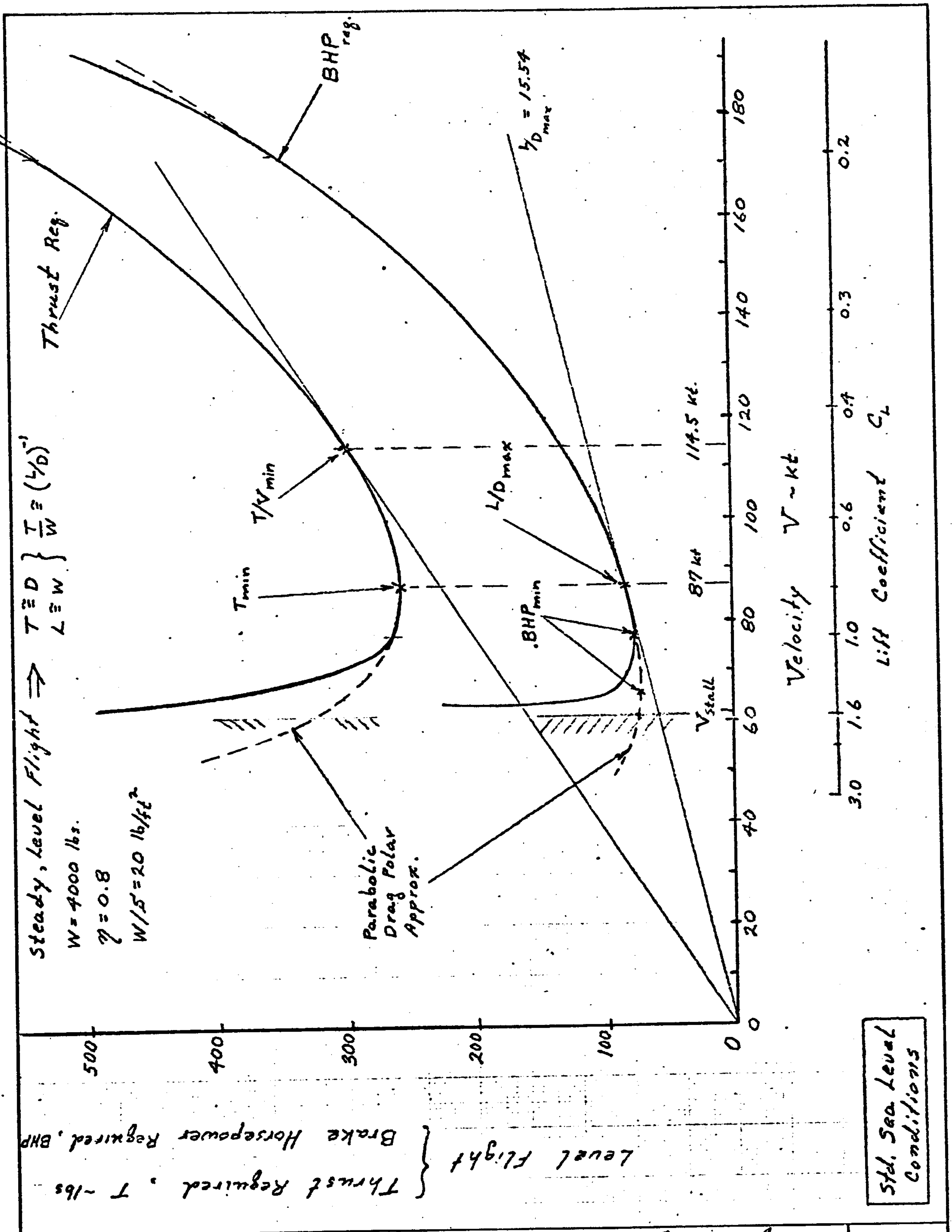




Aircraft Type	C_{D_0}	K	$A\!\!R$	$C_{L_{max}}$ (Flaps Retracted)	L/D_{max}
Rogallo Hang Glider	0.080	1.4	2.9	1.2	4-5
Sailplanes - Std Class (15 m span)	0.010	1.25-1.4	18-23	1.2-1.4	36-40
- Open Class	0.009	1.25-1.45	22-30	1.2-1.4	42-48
Single Engine G.A.	0.025-0.04	1.3-1.5	6-8	1.2-1.5	8-12
Twin Engine G.A.	0.022-0.028	1.25-1.4	6-8	1.2-1.4	9-13
Agricultural A/C (with spray equip)	0.07-0.08	1.3-1.5	6-8	1.4 ~	4-8
Subsonic Transport (prop. driven)	0.018-0.024	1.20-1.30	8-12	1.3-1.5	14-17
Subsonic Transport (jet driven)	0.014-0.02	1.25-1.35	6-9	1.3-1.5	14-18



CALC	JHMc	9/78	REVISED	DATE	Parabolic Drag Polar Results
CHECK					Small Subsonic Airplane
APR					
APR					THE BOEING COMPANY
					PAGE 2.22



CALC	JHMc	9/78	REVISED	DATE	Thrust & Power Required
CHECK					Small Subsonic Airplane
APR					
APR					THE BOEING COMPANY

Supplimentary Problems

1. For the example airplane, calculate Power & Thrust required curves as a function of speed at altitudes of 10,000 ft. ($\rho = 0.00175$ slugs/ft^3), 20,000 ft ($\rho = 0.00127$ slugs/ft^3) and 30,000 ft. ($\rho = 0.00089$). [Std. Sea Level: $\rho = 0.00238$ slugs/ft^3].

   1 BHP = 550 ft-lb/sec

2. For the example airplane shown, assume the geometry is changed such that:

$$\mathcal{R} = 20 \quad \rightarrow \quad W = \cancel{4400} \text{ lbs.}$$
$$S = 200 \text{ ft}^2$$
$$b = 63.25 \text{ ft}$$

[Note: The original values were: $\mathcal{R} = 10$, $S = 200$ ft^2; $b = 44.72$ ft., $W = 4000$ lbs].

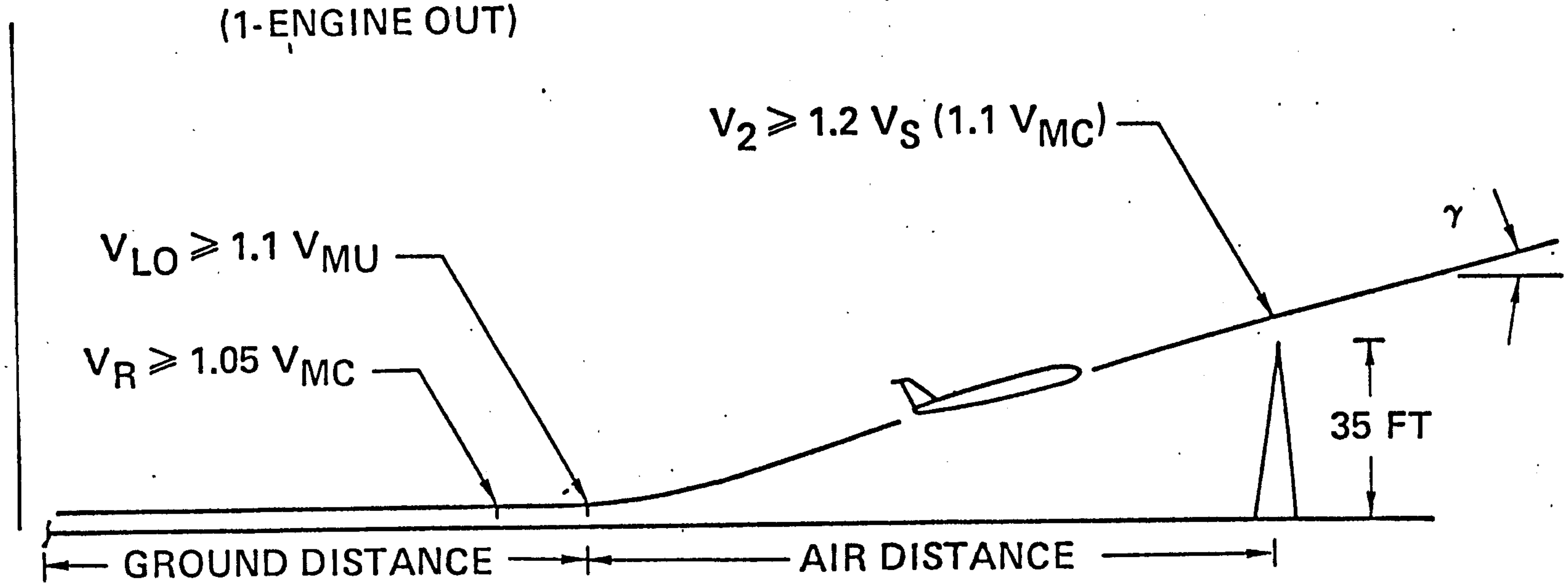
Since S does not change, assume $C_{D_0} = 0.025$. However because of the smaller chord of the higher $\mathcal{R}$ wing, assume $K = 1.35$ [due to R_n scale effects.].

Calculate new Sea Level Std. Power and Thrust required curves as functions of speed.

TAKEOFF PROFILE

GROUND RUN	CLIMBOUT
$f\left(\frac{T}{W}, \frac{W}{S}, C_{L_{MAX}} \ \& \ C_D, \mu\right)$	$f\left(\frac{T}{W}, \frac{L}{D}\right)$

SECOND SEGMENT CLIMB
TAN $\gamma \geqslant .03$
(1-ENGINE OUT)



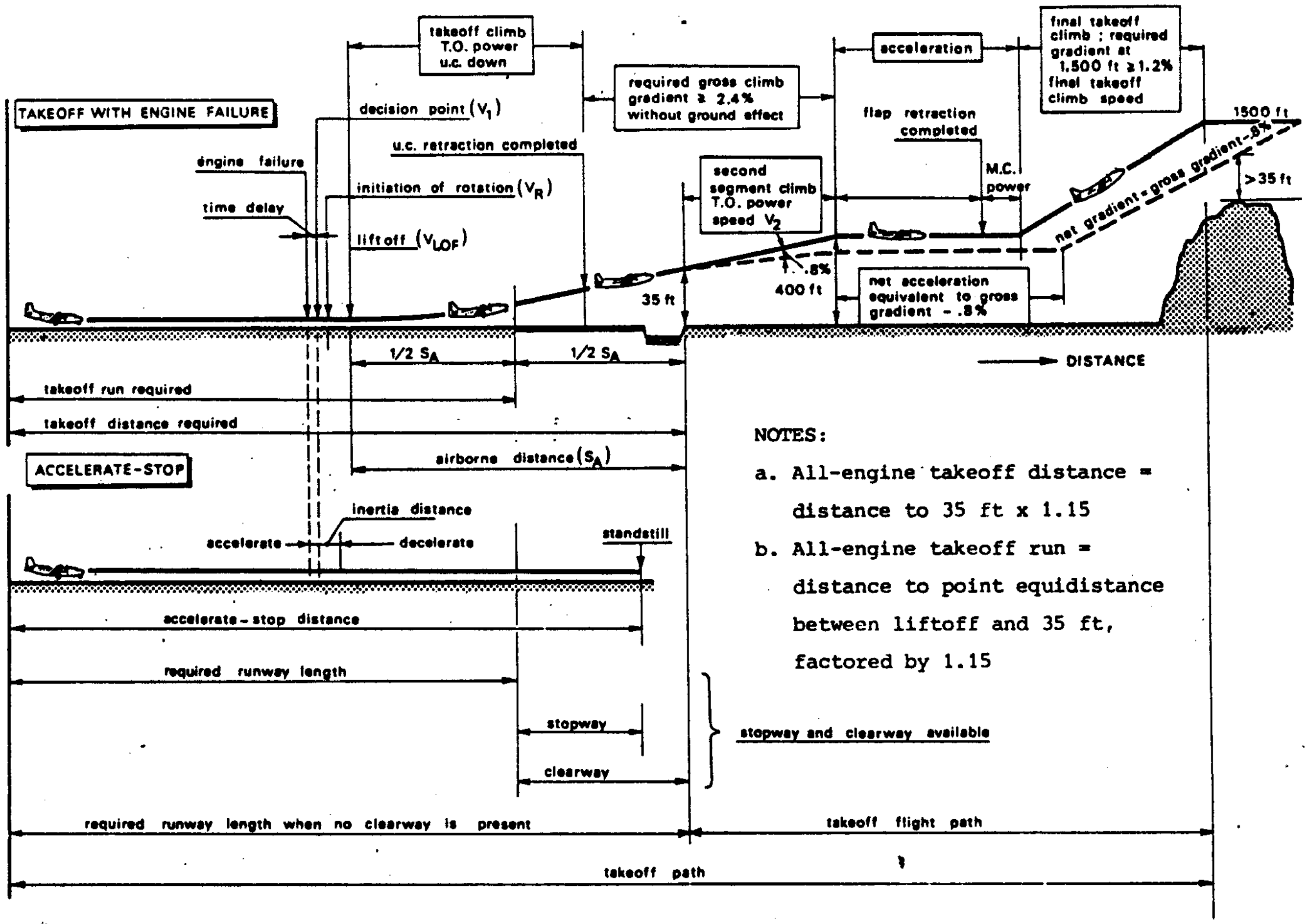


Fig. K-1. Takeoff procedures and requirements for a twin-engine civil transport aircraft

- Std. Sea Level conditions.
- Hot day at Cleveland
- Hot day at Denver
- Hot day at LaPaz

LANDING PROFILE

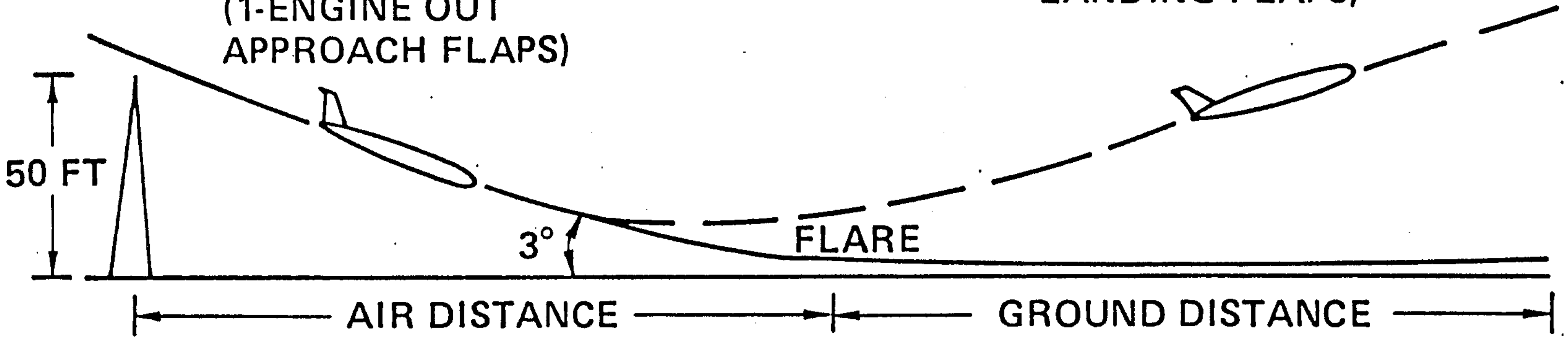
APPROACH	GROUND RUN	GO-AROUND
$f\,(C_{L_{MAX}}, \frac{W}{S}$ & L/D)	$f\,(C_{L_{MAX}}, W/S$ & $C_D, \mu, T_{REV})$	$f\left(\frac{T}{W}, \frac{L}{D}\right)$

APPROACH

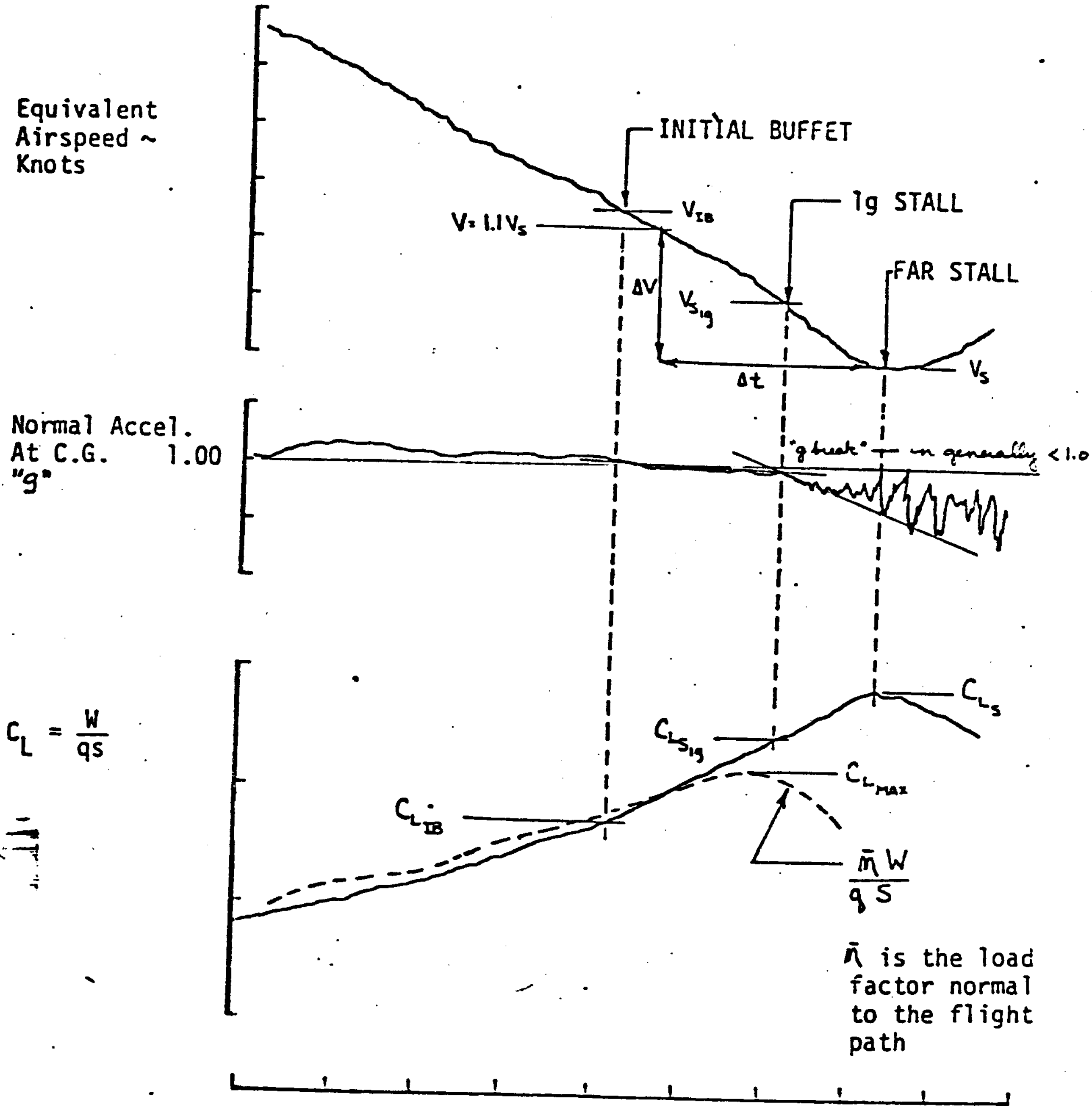
$V_{APP} = 1.3\ V_S$
TAN $\gamma_1 \geqslant .027$
(1-ENGINE OUT
APPROACH FLAPS)

GO-AROUND

TAN $\gamma_2 \geqslant .032$
(ALL ENGINE
LANDING FLAPS)



Power Off Stall (Thrust Effects Negligible)
Trim Speed 1.3 to 1.4 V_S
Wings Held Level, Speed Controlled by Elevator



- FAR STALL C_L IS THE VALUE OF C_{L_S} WHEN $\frac{\Delta V}{\Delta t} = -1$ KNOT PER SECOND.

$$C_{L_S} = \frac{W}{\frac{1}{2}\rho V_S^2 S}$$

- $C_{L_{MAX}}$ IS THE MAXIMUM VALUE OF THE TRUE AERODYNAMIC LIFT COEFFICIENT PERPENDICULAR TO THE FLIGHT PATH.

FIG. 2.3-1 TYPICAL RECORD OF A DYNAMIC STALL MANEUVER

Appendix

Sailplane Drag Estimation Procedures.

The purpose of this appendix is to review the methods used in this study for constructing certain of the detail drag estimates for an aircraft of sailplane type, wherein the drag is a strong function of the Reynolds number. These procedures are seldom adequately treated in standard aerodynamics texts and the review presented here is included for completeness for the present study and as a guide for possible future analyses. This review is limited to three basic topics:

1. Determination of sailplane total drag.
2. Induced drag of highly optimized sailplane wings.
3. Wing profile drag estimation.

Total Sailplane Drag.

The forces on a sailplane in a steady rectilinear glide are shown in Figure B-1. Summation of forces normal and tangent to the flight path yield the results.

(B-1) $$W \sin\theta = D = \frac{1}{2}\rho V^2 C_D S$$

(B-2) $$L = W \cos\theta = \frac{1}{2}\rho V^2 C_L S$$

Thus

(B-3) $$\frac{L}{D} = \cot\theta = C_L/C_D$$

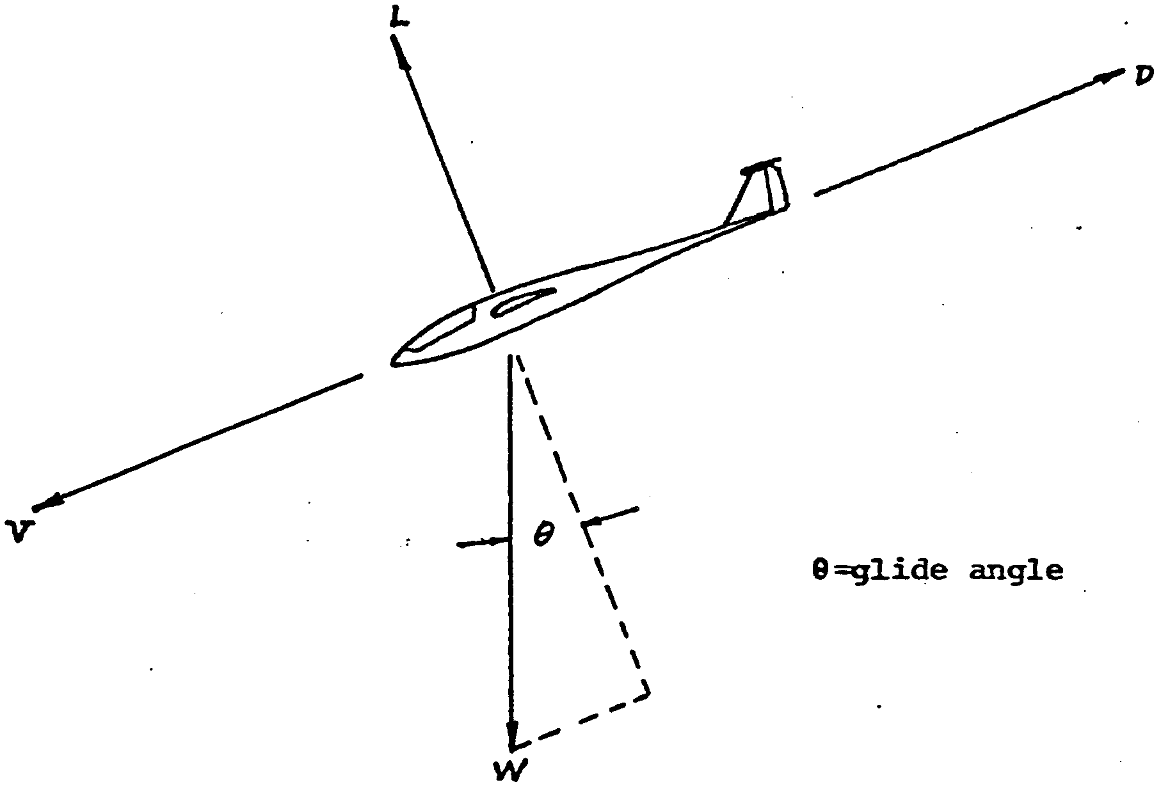
For values of $L/D > 6$, θ is a small angle and the appropriate small angle aproximations can be used, yielding the basic relations:

(B-4) $$W \doteq L \doteq \frac{1}{2}\rho V^2 C_L S$$

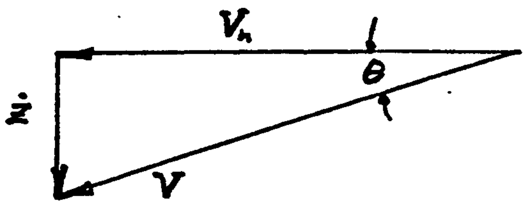
$$\theta \doteq (L/D)^{-1} = C_D/C_L$$

$$\theta \doteq \frac{\dot{z}}{V} \doteq D/L$$

Thus, knowing the glide angle or sink rate ($\dot{z}$) at a given flight speed and altitude, together with the weight of the machine, the values of lift and drag coefficient can be determined.



θ=glide angle



V=velocity along the glide path
$\dot{Z}$=sink rate
V_h =horizontal velocity

Figure B-1. Forces on a Sailplane in a Constant Velocity Rectilinear Glide.

From classic lifting line theory, the induced drag coefficient is usually written as:

(B-6) $C_{D_i} = \frac{K_w C_L^2}{\pi AR}$ where AR = geometric aspect ratio = b^2/S

K_w=1.0 for an elliptically loaded "ideal wing". Kw=wing span efficiency factor $\geqslant$ 1.0

An alternative, more accurate detailed expression for C_{D_i} is:

(B-7) $C_{D_i} = C_L^2/\pi AR + \Delta C_{D_i}$

where ΔC_{D_i} = f (C_L, planform, twist, AR)

The influence of planform and twist on ΔC_{D_i}, as a function of C_L, resulting from German analyses is shown in Figure B-2. It will be noted that the planform shown in Figure B-2 (c) which yields a very small induced drag penalty over the entire C_L range of interest is very close to that employed by the designers of the Astir. For this wing, the value of K_w in eqn. (B-6) is approximately a constant 1.04 over the range $0.2 < C_L < 1.0$ when the wing has 3° washout at the tip; this value has thus been used in this study.

Wing Profile Drag

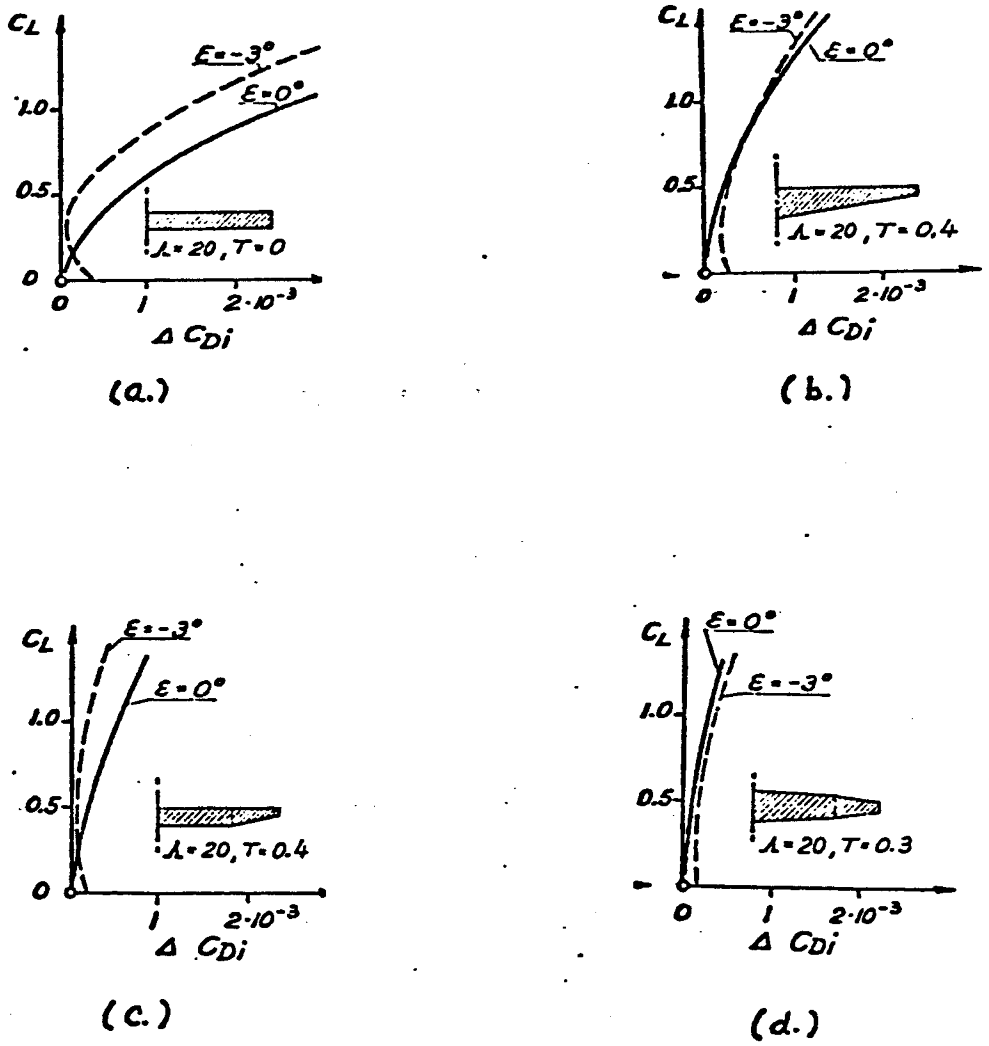
A detailed assessment of the profile drag of an unswept high aspect ratio (AR>5) wing depends on accurate knowledge of:

1. The variation of airfoil section drag coefficient (C_{d_p}) with section lift coefficient (C_l) and Reynolds number.

2. The relation between local wing section lift coefficient (C_l) and total wing C_L across the entire span.

The airfoil section (Eppler E603) used on the Astir is theoretically derived and has never been tested in the wind tunnel; thus curves of C_l versus C_{d_p} at constant Reynolds numbers are not available without very lengthy detailed computer analysis. The only hard data available are a set of boundary layer measurements made at the trailing edge of a single inboard section of the wing (semispan station approximately 20%) from which the variation in profile drag could be deduced from the formulas of Squire and Young (c.f. Schlichting pp. 620-625). These limited data yield a plot (Figure 4) of section profile drag versus wing C_L at various Reynolds numbers. The problem in accurately deducing the total wing profile drag from this depends on:

1. Relating section to wing lift coefficient values at the measured semi-span station at each measured flight condition (C_L and R_n).
2. Generalizing the airfoil section drag polar measured (and adjusted) to other spanwise stations and flight conditions.



$$C_{D_i} = \frac{K_w C_L^2}{\pi AR} = \frac{C_L^2}{\pi AR} + \Delta C_{D_i}$$

Λ = aspect ratio $\qquad$ T =taper ratio

ε = twist (positive for washin)

Figure B-2. Influence of wing planform and twist on induced drag increment.

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The total drag of the sailplane can be broken down into four basic components:

(B-5) $$D_{total} = D_{\pi} + D_{p} + D_{i} + D_{t} \equiv \frac{1}{2}\rho V^2 C_D S$$

or $$C_D = C_{D_{\pi}} + C_{D_p} + C_{D_i} + C_{D_t}$$

where $C_{D_{\pi}}$ =parasite drag coefficient of all components except the wing.

C_{D_p} =wing profile drag coefficient

C_{D_i} =wing induced drag (drag due-to-lift) coefficient

C_{D_t} =trim drag coefficient

Induced Drag of the Wing

No detailed theoretical analysis of the induced drag variation with angle of attack for the Astir wing has been made in this study. However, the modern high performace sailplane is typified by having a very highly refined wing planform and twist distribution aimed at striking an "optimum" balance between the requirements of:

1. Ease of manufacture.
2. Near optimum aerodynamic performance (minimum drag) over the entire flight range.
3. Satisfactory stall and control characteristics.

This optimization problem has been given particularly careful attention by German sailplane designers and a nearly universally accepted recipe has been adopted, and is apparently reflected in detail in the wing of the Astir. It is well know that theoretically, a planar wing carrying an elliptical distribution of lift (uniform downwash across the span) has minimum induced drag; and that if the wing is untwisted, this distribution is achieved only if the wing chord is distributed elliptically. Since a truly elliptic planform represents a major manufacturing problem, various attempts are usually made to approximate the elliptic ideal with planforms of generally trapizoidal shape. Detailed lifting line theory analysis shows that a wing of trapizoidal planform on each semispan, tapered in the ratio of about 2.5:1 can yield very nearly the minimum induced drag over a limited C_L range without any twist requirements. However for very high aspect ratio wings ($10 < AR < 35$) typical of modern sailplanes intended to operate very efficiently over a very wide C_L range ($0.1 < C_L < 1.2$) a more refined planform is usually required. These planforms usually require some form of "double taper" and some twist.

In the wind tunnel, the problem is "simple". The wing or 2-dimensional airfoil section of known dimensions is mounted in the tunnel and the forces measured at a fixed wind velocity and hence Reynolds number over a range of lift coefficient. In flight, however, assumming data are collected at steady velocity conditions, each speed corresponds to a different lift coefficient and, assumming the weight of the aircraft is constant, a given flight will cover a variety of Reynolds numbers. Without substantially altering the weight of the machine, it is impossible to measure "constant Reynolds number" polars of C_L versus C_D. To demonstrate this point quantitatively, consider the following relations:

(B-8) Reynolds number $= \frac{Vc}{\nu}$; c = wing chord

$L = W = \frac{1}{2}\rho V^2 C_L S$ in steady level fight (or small glide angle)

Thus,

(B-9) $$R_n = [2/\rho\nu^2]^{1/2} \cdot \left[\frac{Wc^2}{SC_L}\right]^{1/2} \propto C_L^{-1/2}$$

since ρ, ν, W, S, c are assumed constant.

Defining the average wing Reynolds number by:

(B-10) $$\overline{R_n} = \frac{V\bar{c}}{\nu} = (2/\rho\nu^2)^{1/2}\left(\frac{W}{AR}\right)^{1/2} \cdot C_L^{-1/2} = \text{constant} \times C_L^{-1/2}$$

where $AR = b/\bar{c} = b^2/S$

we have the result that a plot of C_D versus C_L is equivalent to a plot of C_D versus $\overline{R_n}$ for a fixed geometry sailplane flying at a "constant" height (i.e. ρ and ν values). The result is shown in the sketches in Figure B-3. The data available for this study are like that shown by the dashed curve in sketch (B-3b) and the information desired is like that shown by the dashed curve (B-3d).

The general relation between wing and section profile drag is given (assuming rectilinear flight) by:

(B-11) $$D_p = \frac{1}{2}\rho V^2 \int_{-b/2}^{+b/2} c_{d_p} c \, dy$$ where $c = c(y)$, $c_{d_p} = f(C_L, C_\ell)$

Which reduced to:

(B-12) C_{D_p} = wing profile drag coefficient $$= \int_{-1}^{+1} c_{d_p}\left(\frac{c}{\bar{c}}\right) d\eta$$

where $$\eta = \frac{y}{b/2}$$

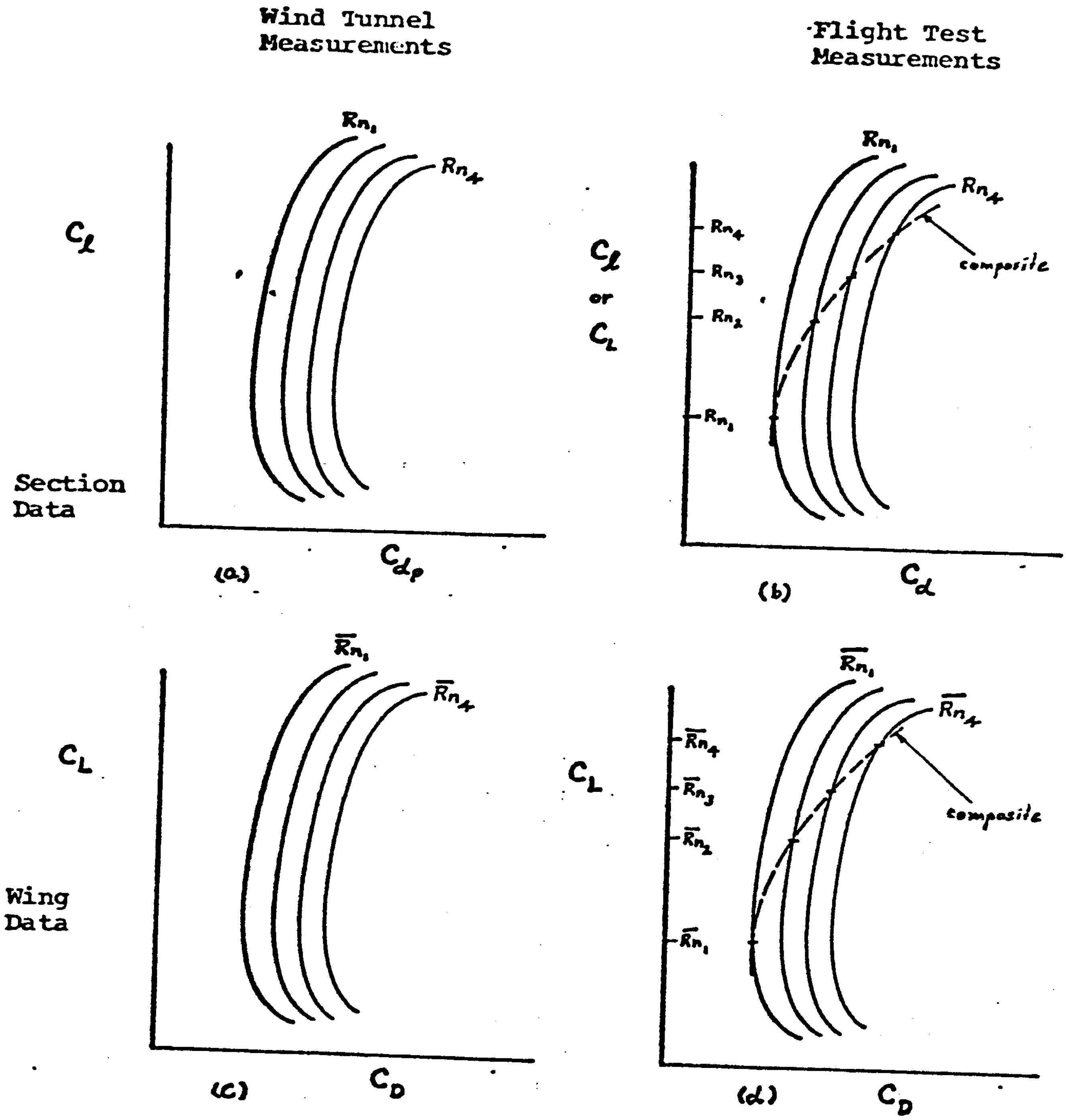

Figure B-3. Comparison of Typical Airfoil Section and Total Wing Drag Coefficients Obtained from Wind Tunnel and Flight tests.

To examine the quantitative nature of the relation between section and wing profile drag, assume:

(B-13) $C_{d_p} = k_o + k_1 C_\ell^2$ when Rn = constant

$$\frac{C_\ell c}{C_L \bar{c}} = \frac{4}{\pi}\left\{1-\eta^2\right\}^{1/2} \sim \text{elliptic lift distribution}$$

$$\frac{Rn}{\overline{Rn}} = \frac{c}{\bar{c}} \qquad c = f(\eta)$$
$$\bar{c} = S/b$$

Then, by eqn. (B-12)

(B-14) $$C_{D_p} = k_o \int_0^1 \frac{c}{\bar{c}}\, d\eta + \left(\frac{4}{\pi}\right)^2 k_1 C_L^2\, \bar{c} \int_0^1 \frac{(1-\eta^2)}{c}\, d\eta$$

where $\overline{Rn}$ = constant.

In the particular case where $c=\bar{c}$=constant ($Rn=\overline{Rn}$=constant)

(B-15) $C_{d_p} = k_o + k_1 C_L^2$ at $\eta = 0.619$ $(C_L = C_\ell)$

$C_{D_p} = k_o + (1.08)\, k_1 C_L^2$

The general results of a similar, more detailed analysis of the Astir wing, accounting explicitly for the correct spanwise chord variation, is obtained in the same fashion. In this analysis, the major assumption was made that over a limited range of section lift coefficient (at a given constant Reynolds number) the profile drag polar could be approximated by the parabolic type aproximation specified by eqn. (B-13) and that the variation of C_{d_p} with Rn at a given C_ℓ can be determined from the empirical section data. Thus:

(B-17) $C_{d_p} = \dfrac{0.95}{Rn^{0.35}} + 0.002\, C_\ell^2$ $\qquad 0.15 \le C_\ell \le 0.7$

$C_\ell c / C_L \bar{c} = \frac{4}{\pi}(1-\eta^2)^{1/2}$ $\qquad 10^6 \le R_n \le 4\times10^6$

Thus, for the Astir:

(B-18) $C_{D_p} = 0.007\, C_L^{0.175} + 0.000203\, C_L^2$

($0.15 \le C_L \le 0.7$)
w=840 lbs
Std. Sea Level Conditions.

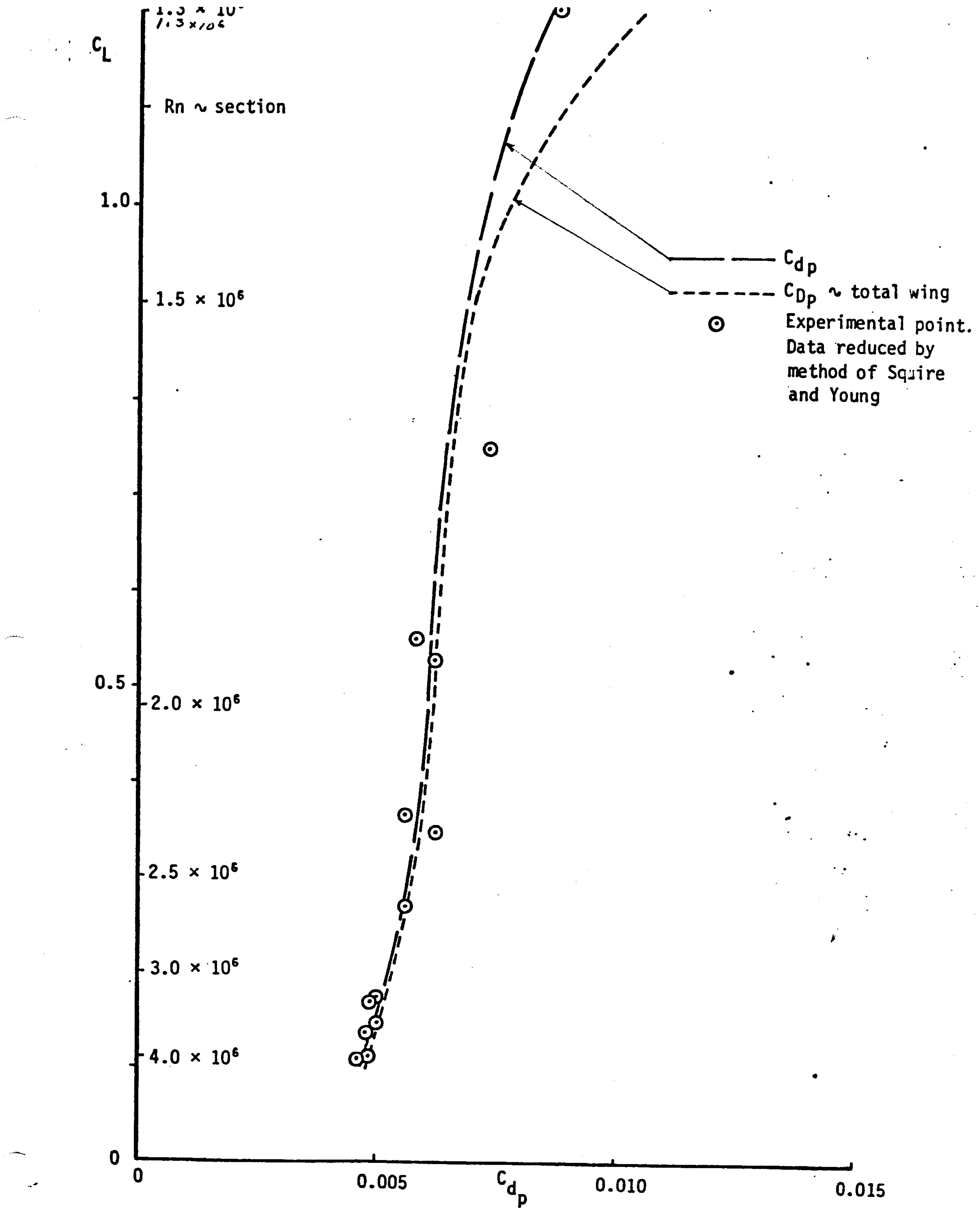

FIGURE 4 PROFILE DRAG COEFFICIENT VARIATION WITH LIFT COEFFICIENT FOR EPPLER E603 AIRFOIL

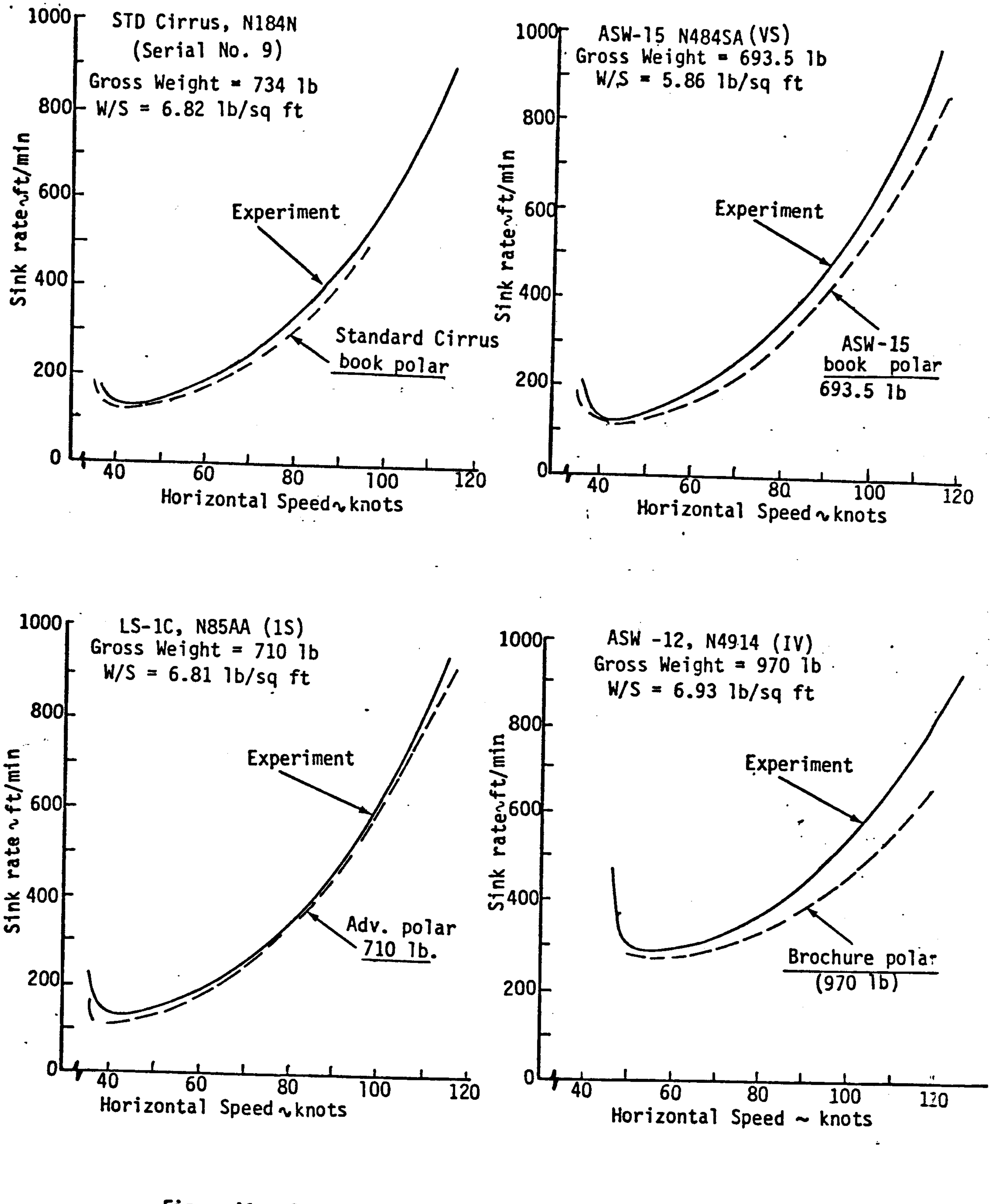

Figure A1. Comparison Between Flight Measured and Advertised Performance of Several Sailplanes

Table A-1. Sailplane Characteristics — Performance at Std. Sea Level

	Type	Origin	Wing Span (ft)	Wing Area (ft²)	Aspect Ratio	Root Airfoil	Ref Flying Weight (lbs)	Wing Loading (lbs/ft²)	Min. Sink Rate(fpm) (At v~Kt)	Measured L/D max. (at v~Kt)	Advertised L/D max (at v~Kt)	Measured v~(Kt) for 2 m/s sink rate	Advertised v~(Kt) for 2 m/s rate of sink	Ref.
1.	Schempp-Hirth "Std. Cirrus"	Germ.	49.2	107.6	22.5	FX mod.	734	6.82	134	35.2	37	85.5	88.5	3
									(42.5)	(51)	(~52)			
2.	Schneider LS-1C	Germ.	49.2	104.8	23.1	FX 66.S-196	710	6.80	134	35.0	38	84.5	85.5	3
									(42.0)	(49)	(~49)			
3.	Schleicher AS-W 15	Germ.	49.2	118.4	20.45	FX 61-163	693.5	5.86	130	35.1	39	83.5	87.5	3
									(42.5)	(50)	(~51)			
4.	Glasflugel "Std.Libelle"	Germ.	49.2	102.3	23.6	FX mod.	633	6.18	134	34.5	37	82.5	85.5	3
									(43)	(50)	(~51)			
5.	Laister LP-49	U.S.	49.2	143	17	NACA 64_3618	812	5.68	149	31	36	80	88	
									(43)	(48.5)	(~49)			
6.	Schweizer 1-34	U.S.	49.2	151	16	FX 61-163	829	5.49	147	31.3	34.5	77	80	3
									(42.5)	(48.5)	(~49)			
7.	Schleicher AS-W 12	Germ.	60.0	140	25.8	FX 62-131K	909	6.50	109	43.3	48	91	100	3
									(43)	(48)	(~52)			
8.	Siamant 18	Swiss	59.0	153.5	22.7	FX-mod.	917	5.98	114	40.9	42	87	91	3
									(42.5)	(51)	(~54)			
9.	Schreder T-6 (HP-14)	U.S.	57.0	142.5	22.8	FX61-163	830	5.82	127	36.3		86.5		3
									(43.5)	(49)				
10.	Schempp-Hirth "Nimbus 2"	Germ.	66.6	154.9	28.6	FX67-K-150	950	6.13	100	47.2	46	94		5
									(45)	(52)	(49)			
11.	Schleicher AS-W 17	Germ.	65.6	159.7	27.0	FX62-K-131	1114	7.05	95	47.4	48.5	99		
									(40)	(52)	(54)			6
12.	Glasflugel 604	Germ.	72.2	175.0	29.8	FX67-K-170	1200	6.86	105	48.5	49	91		8
									(47)	(53)	(~53)			
13.	"Jantar 1"	Poland	62.3	143.9	27.0		872	6.06	~105	43		85		9
									(43)	(45)				
14.	"Jantar 2A"	Poland	67.2	153.4	29.5		945	6.16	105	45		92		9
									(43)	(51)				
15.	Schempp-Hirth "Std Cirrus B"	Germ.	49.2	107.5	22.5	FX mod	729	6.78	130	35.9	36	87		4
									(43)	(51)	(50)			
16.	PIK 20	Finland	49.2	107.5	22.5	FX67-K-150	719	6.69	125	38	40.5	87.5		7
									(45)	(51)	(53)			
17.	Schweizer 1-35	U.S.	49.2	103.8	23.3	FX67-K-170	695	6.70	125	36.8	38	81		9
									(40)	(50)	(48)			
0	Grob "Astir CS"	Germ.	49.2	133.5	18.2	Eppler E603	772	5.78	*118		37.3		86.9	Pilot Manual
									*(40.5)		(51.3)			
							992	7.43	*138		38		93.9	Pilot Manual
									*(46)		(56.7)			
							840	6.29	+135	+36.2		+87		Fig. A-7

*Advertised +Assumed Value

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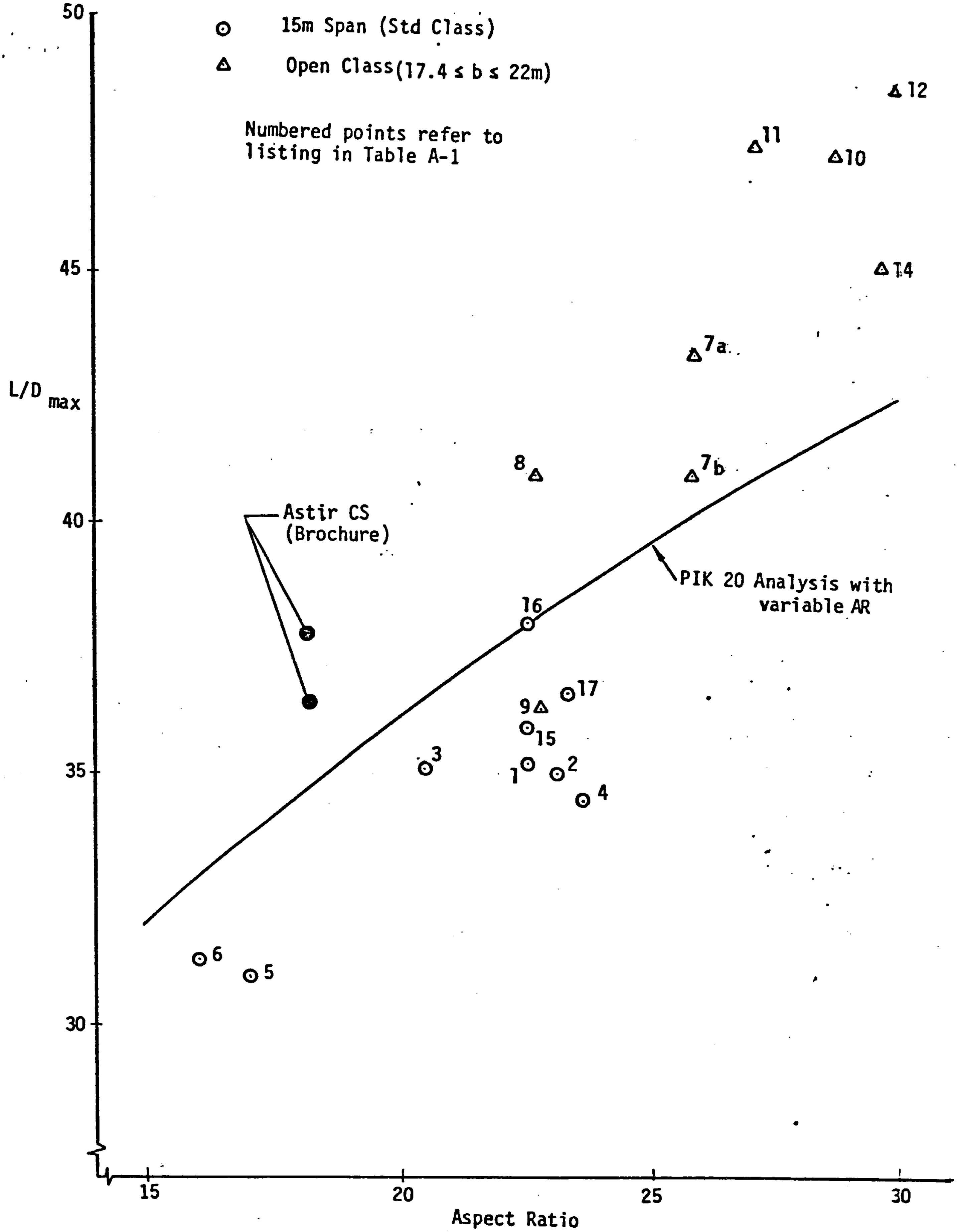


Figure A2. Maximum Lift-to-Drag Ratio Versus Aspect Ratio for Eighteen Modern Sailplanes

2.41

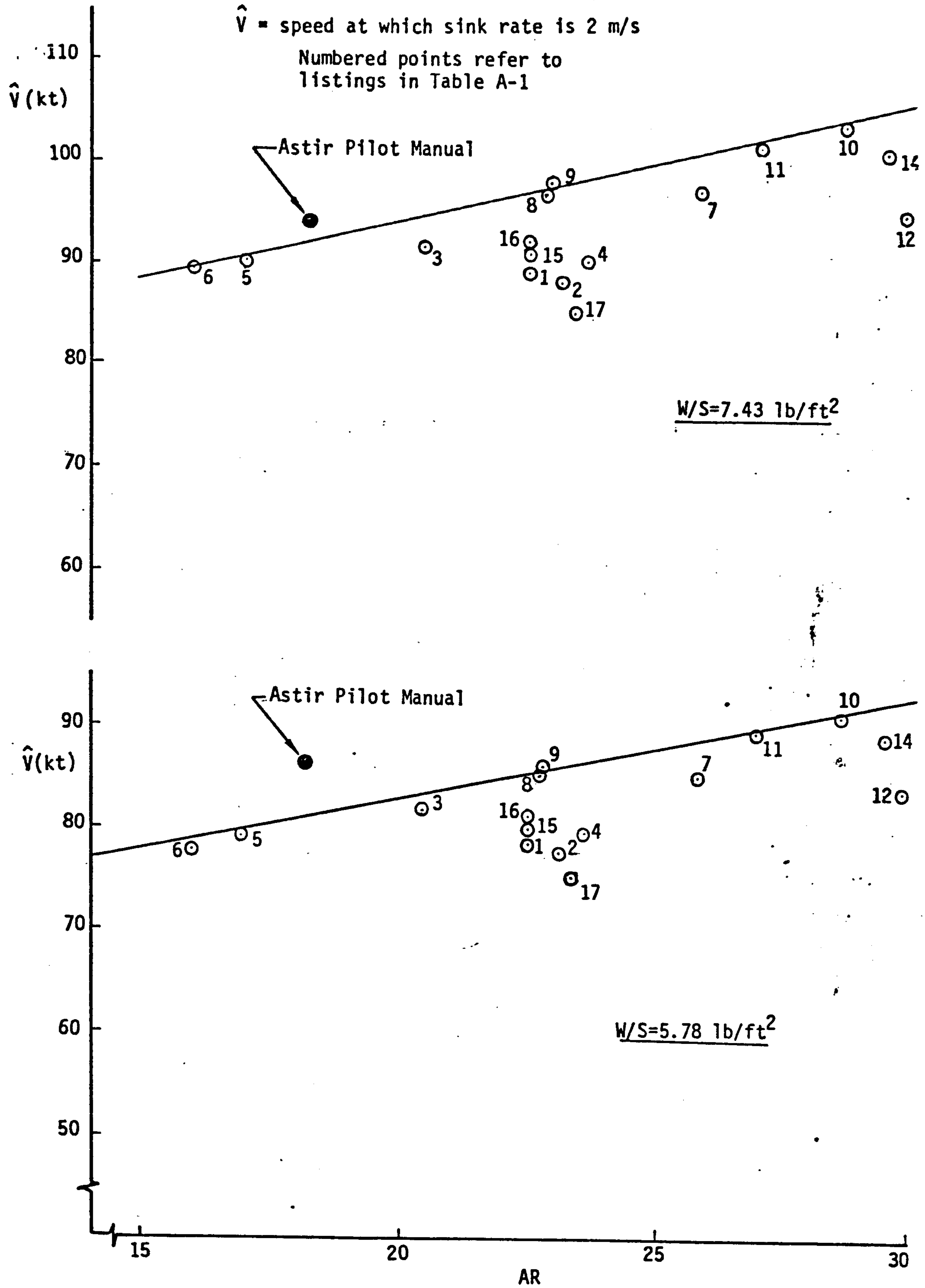


Figure A3. Speed for a 2 m/s Rate of Sink Versus Aspect Ratio for Eighteen Modern Sailplanes

2.42

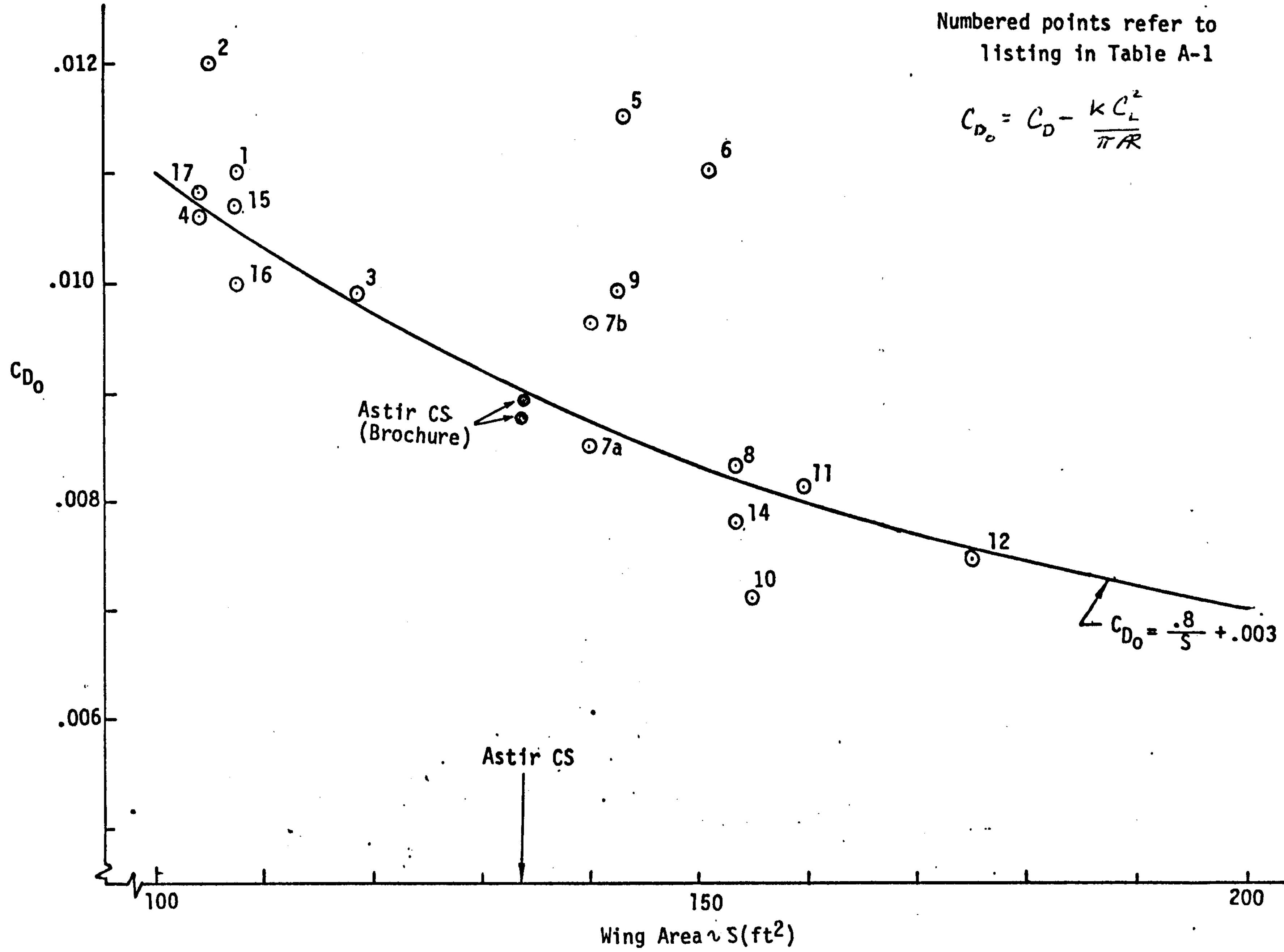


Figure A4. Zero Lift Drag Coefficient Versus Wing Area for Eighteen Modern Sailplanes

2.43

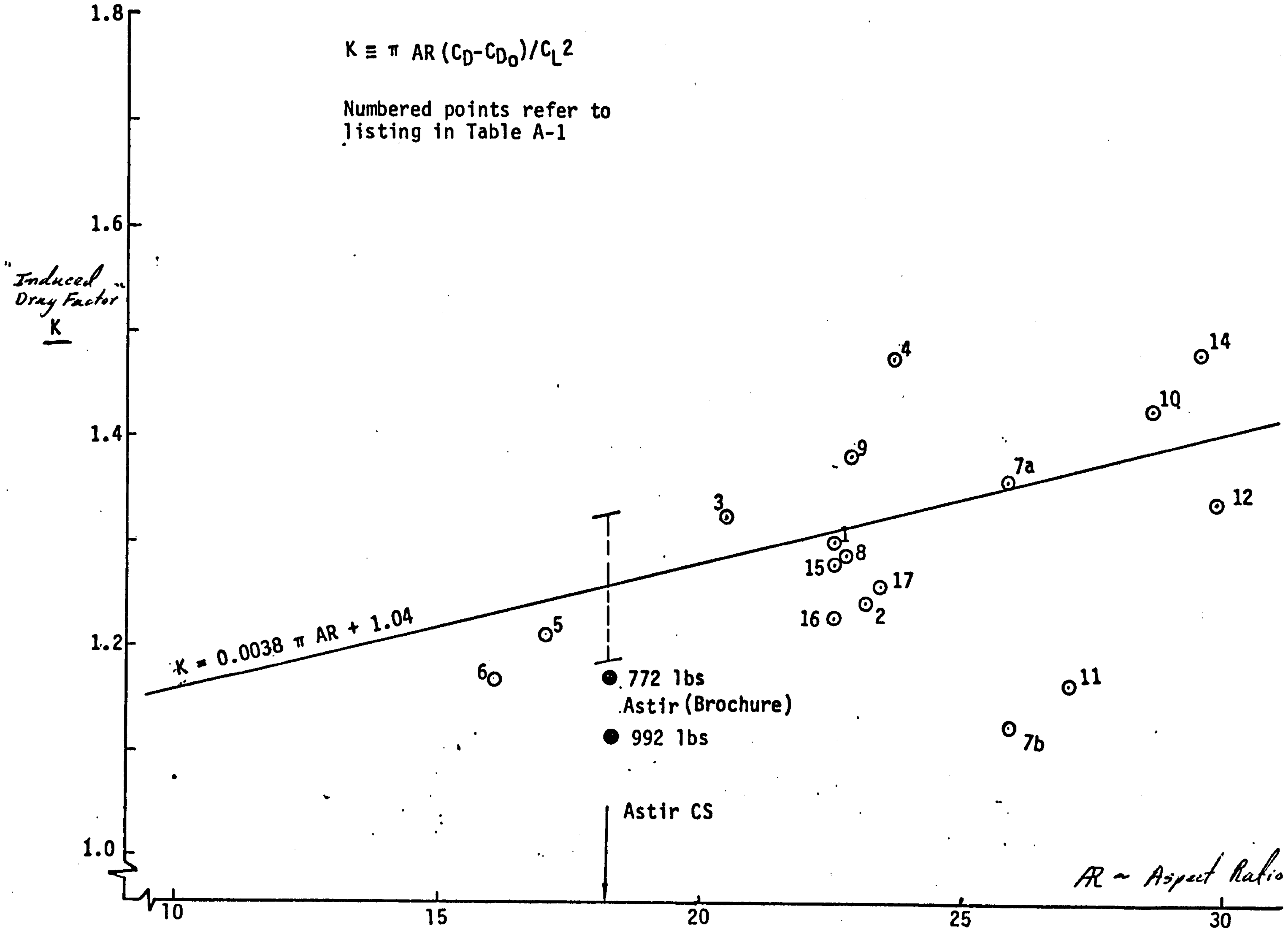
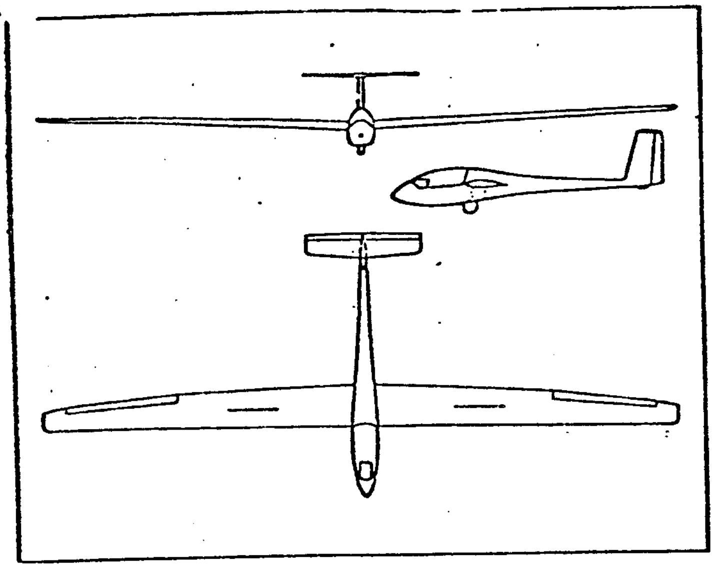
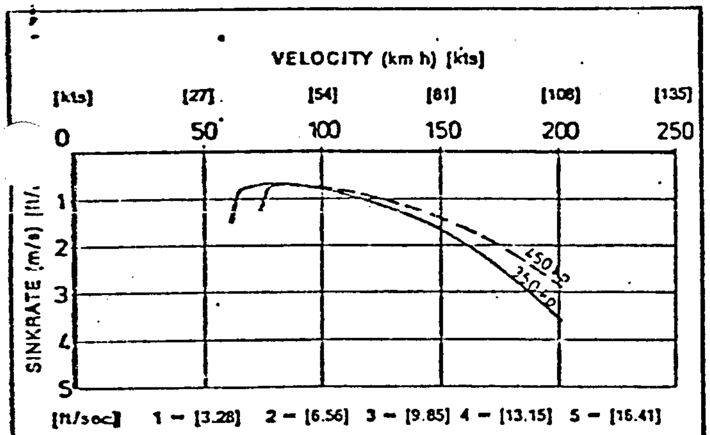


Figure A5. Induced Drag Factor Versus Aspect Ratio for Eighteen Modern Sailplanes

2.44

ASTIR CS





FLIGHT POLAR
G 102 ASTIR CS

Technical Data

Airfoil Eppler E 603

Wing

wing span	15 m
wing area	12.4 m² (133,5 sq. ft.)
aspect ratio	18.2

Fuselage

length	6.47 m (21.2 ft.)
T-tail, height	1.4 m (4.5 ft.)
cockpit height	0.87 m (2.8 ft.)
cockpit width	0.64 m (2.0 ft.)
max. height of pilot	1.95 m (6.3 ft.)

Weight

empty weight	250 kg (552 lbs.)
pay load	120 kg (264 lbs.)
water ballast (max.)	100 kg (220 lbs.)
max a.u.w. no ballast	370 kg (814 lbs.)
max a.u.w. inc ballast	450 kg (990 lbs.)
min wing loading	25 kg/m² (5.1 lbs./sq.ft
max wing loading	36,5 kg/m² (7.5 lbs./sq.ft

Air Speeds

max permissible	250 km/h (136 kts)
stalling speed	60 km/h (32.5 kts)
max winch launch	120 km/h (65 kts)
max aerotow	170 km/h (92 kts)
max airbrakes open,	250 km/h (136 kts)
wheel down	250 km/h (136 kts)

Flight Performance

Min sink without ballast	0.6 m/s at 75 km/h (1.2 kts at 40.5 kts)
Min sink with full ballast	0.7 m/s at 85 km/h (1.4 kts at 46 kts
Best L/D without ballast	37.3 at 95 km/h (51.5 kts)
Best L/D with full ballast	38.0 at 105 km/h (57 kts)

FIGURE PILOT MANUAL DATA ON ASTIR CS SAILPLANE

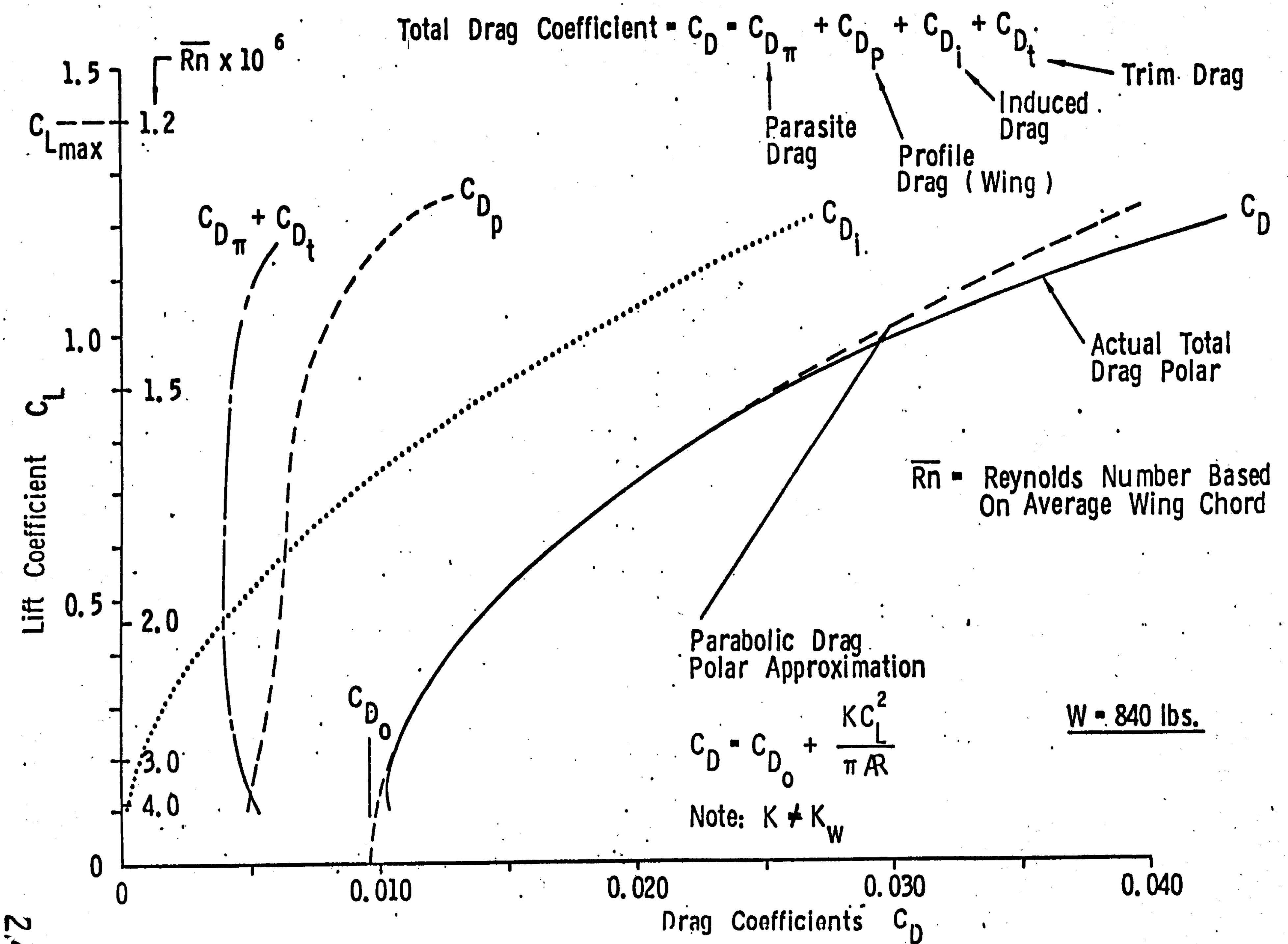


Figure 2 DETAIL DRAG BREAKDOWN FOR THE ASTIR CS SAILPLANE

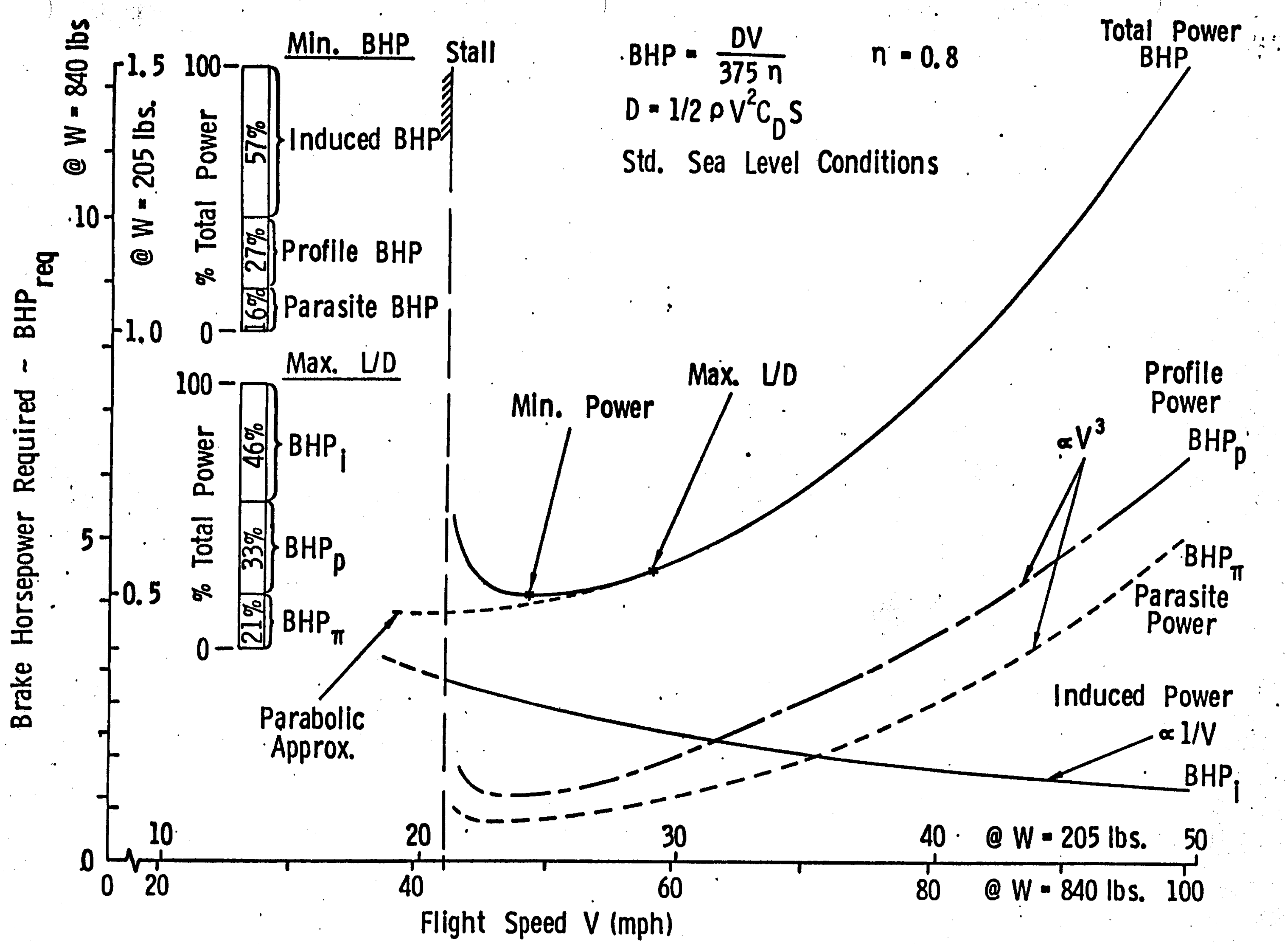


Figure 3 THE EPPLER/GROB "ASTIR CS" SAILPLANE AS A MOTOR GLIDER

McMASTERS

PROCEEDINGS OF THE NASA • INDUSTRY • UNIVERSITY GENERAL AVIATION DRAG REDUCTION WORKSHOP

Edited by: Jan Roskam

July 14-16, 1975

Space Technology Center
University of Kansas
Lawrence, Kansas 66045

2.48

9.1 Possible Applications Of Soaring Technology To Drag Reduction In Powered General Aviation Aircraft

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and
George M. Palmer
Purdue University

Introduction

The term "General Aviation" usually brings to mind the range of powered aircraft encompassing the Piper Cub through executive jet transport aircraft. Depending on one's definitions and biases, however, a case can be made for inclusion of other types of aerodynamically supported vehicles such as the sailplane and their powered derivatives (self-launched sailplanes or motor gliders) and perhaps even the lowly hang glider. While participation in soaring in this country is rather limited and the economic impact of sailplane manufacture is miniscule, the current level of technology in this branch of light aviation is extraordinary - particularly in the areas of aerodynamic efficiency and utilization of advanced materials and fabrication techniques. The purpose of this brief discussion is to outline the present state-of-the-art in soaring performance and review some of the techniques (particularly in the area of drag reduction) used to achieve this performance. It can legitimately be objected that the performance requirements of sailplanes and light powered aircraft are quite different and that sailplane manufactures are not bound by the same economic constraints as their counter parts in powered flight. However, to ignore the aerodynamic lessons learned in sailplane development would be, in our view, a serious oversight. In view of the fact that sailplane technical literature is infrequently consulted by many aeronautical engineers, particularly those at universities, this brief review is considered appropriate.

State-of-the-Art

Most modern soaring aircraft are pure sporting devices, the most elegant and advanced of which are optimized for competition - which today implies racing. The classic design problem is one of optimizing an aircraft for two design points: (1) low speed (minimum sink rate) flight in a rectilinear or banked turning attitude to maximize rate of climb and (2) minimum glide angle (or maximum lift-drag ratio) in high speed rectilinear cruise. In racing performance, however, absolute maximum L/D

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is less important than maintaining a "low" sink rate (e.g. 2m/sec) at the highest possible speed.*

At present two major types of competition sailplanes are in wide spread use: Standard Class, with spans limited to 15m (49.2 ft) with water balast (to increase wing loading in strong lift conditions) and only simple hinged flaps not connected to the ailerons permitted, and Open Class where anything is permitted. Under pressure mainly from European designers, the Standard Class will be divided into two classes for international competition after 1976, with one branch becoming an "unlimited" class keeping only the 15m span limit and the other basically retaining the present Standard Class rules.

The other category of soaring device of interest in this discussion, the "motor glider", is slowly becoming more popular in Europe and the United States. It is basically a moderate performance sailplane fitted with an engine providing it with a self-launch and out-landing retrieval capability.

Some typical modern sailplanes and motor gliders are shown in Figure 1. Performance and geometric data for several typical types are listed in Table 1. Performance capabilities are further clarified in Figure 2. Also shown for comparison in Figure 2 are glide polars for several other types of low speed flying device from (1). There are few standard handbook type references available on sailplanes and soaring technology. Probably the best sources of information are Soaring magazine, Technical Soaring (12) and the publications of the Organization Scientifique et Technique Internationale du Vol-a-Voile (OSTIV) available from the Soaring Society of America (SSA). Important recent material is available in (5.6).

The basic configuration of the high performance sailplane was well established prior to WWII. Performance increases since that time have been very large, however, due mainly to three factors:

1. A greater appreciation of the importance of the quality of the aerodynamic surfaces and the necessity of sealing gaps and flow leakage in reducing drag.

2. Advanced airfoil designs with greatly improved (compared with Göttingen and NACA 4 and 5 digit airfoils) characteristic in the Reynolds number range characteristic of sailplane operation.

3. The introduction of fiberglass construction.

These factors will be discussed in more detail later. Some comparison, based on data from Table 1, for typical sailplanes of good pre-war technical vintage and modern technology are presented in Table 2.

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* The ideal sailplane glide polar would be as "flat" as possible over the widest possible speed range. Sailplanes, as in the case of most other aircraft types, seldom "cruise" at the speed for L/D_{max}.

2.50

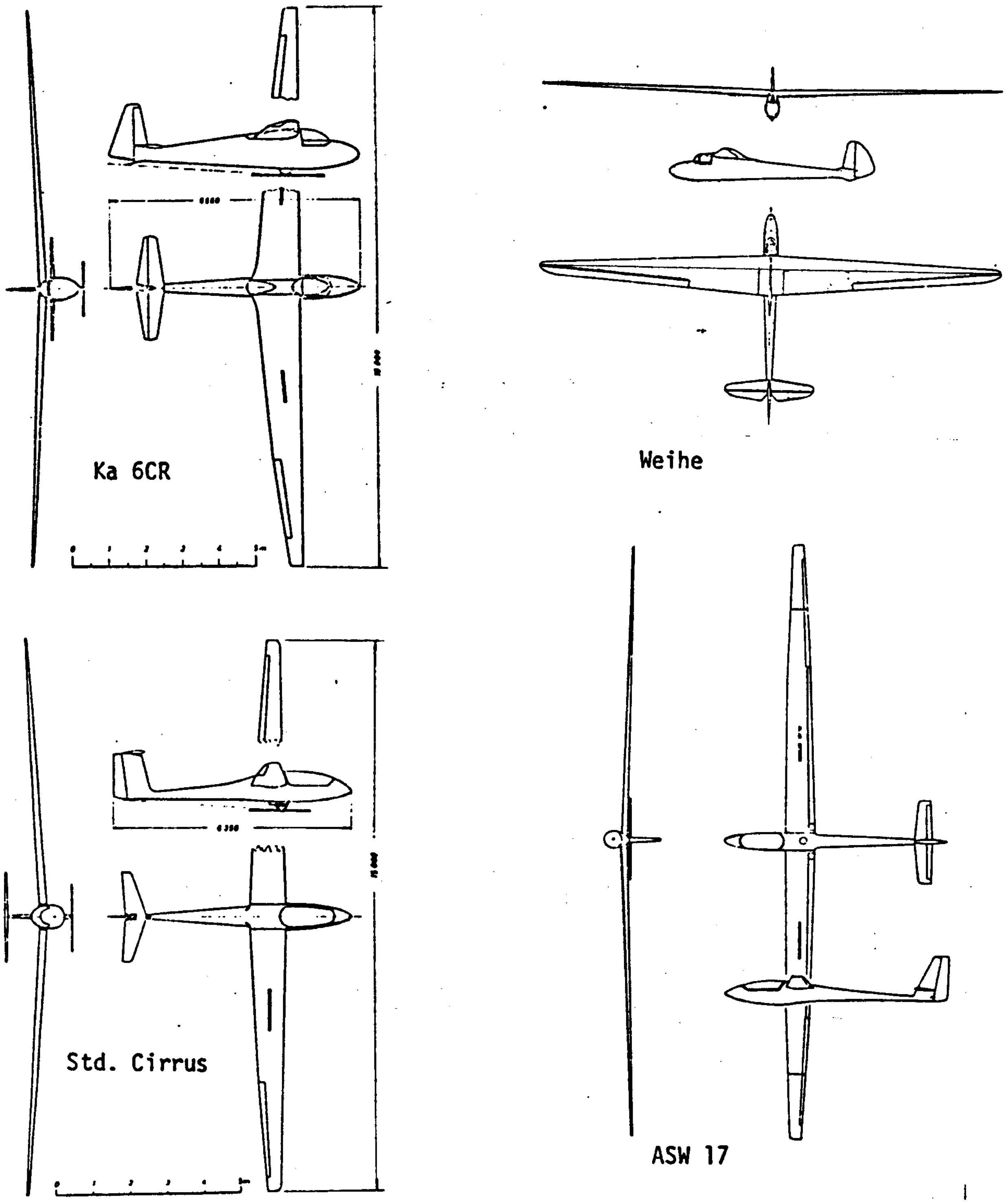

Figure 1. Sailplanes and Motor Gliders

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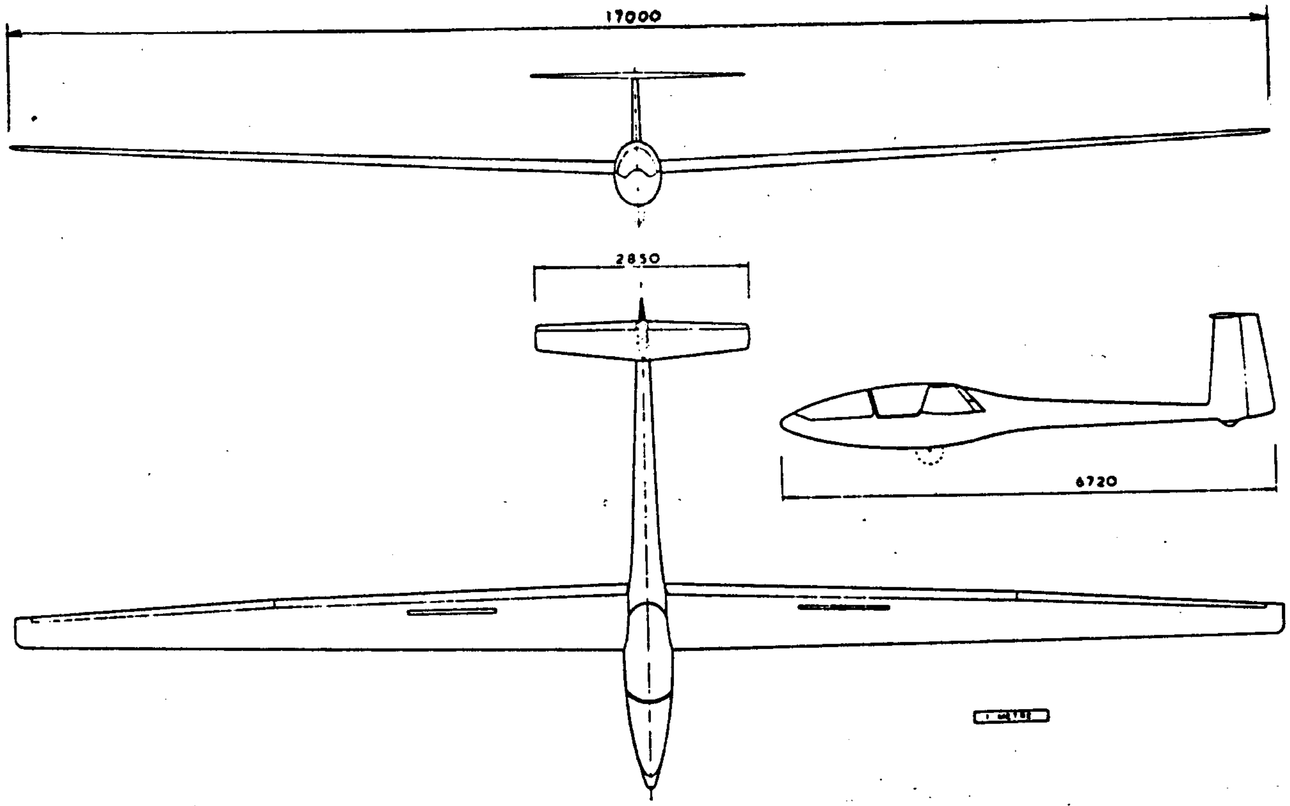
GLASFLUGEL KESTREL



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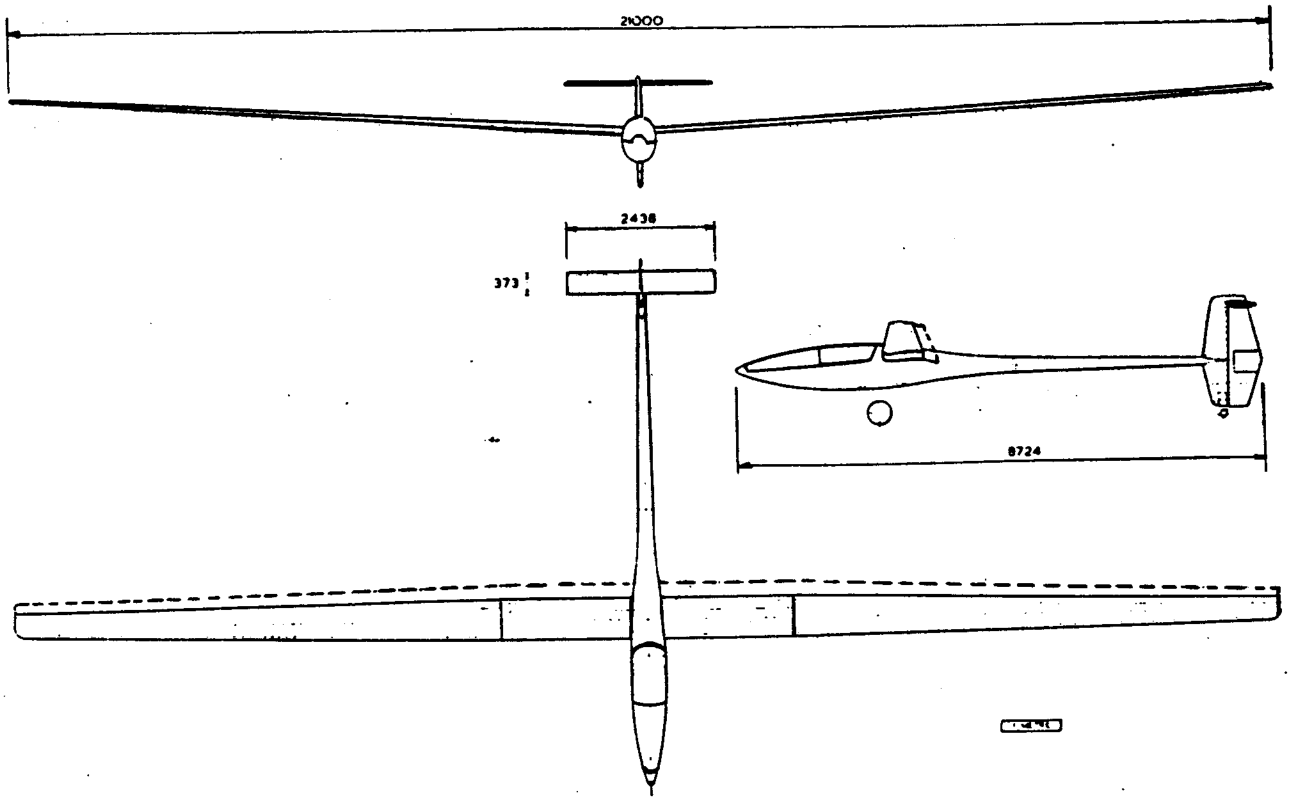
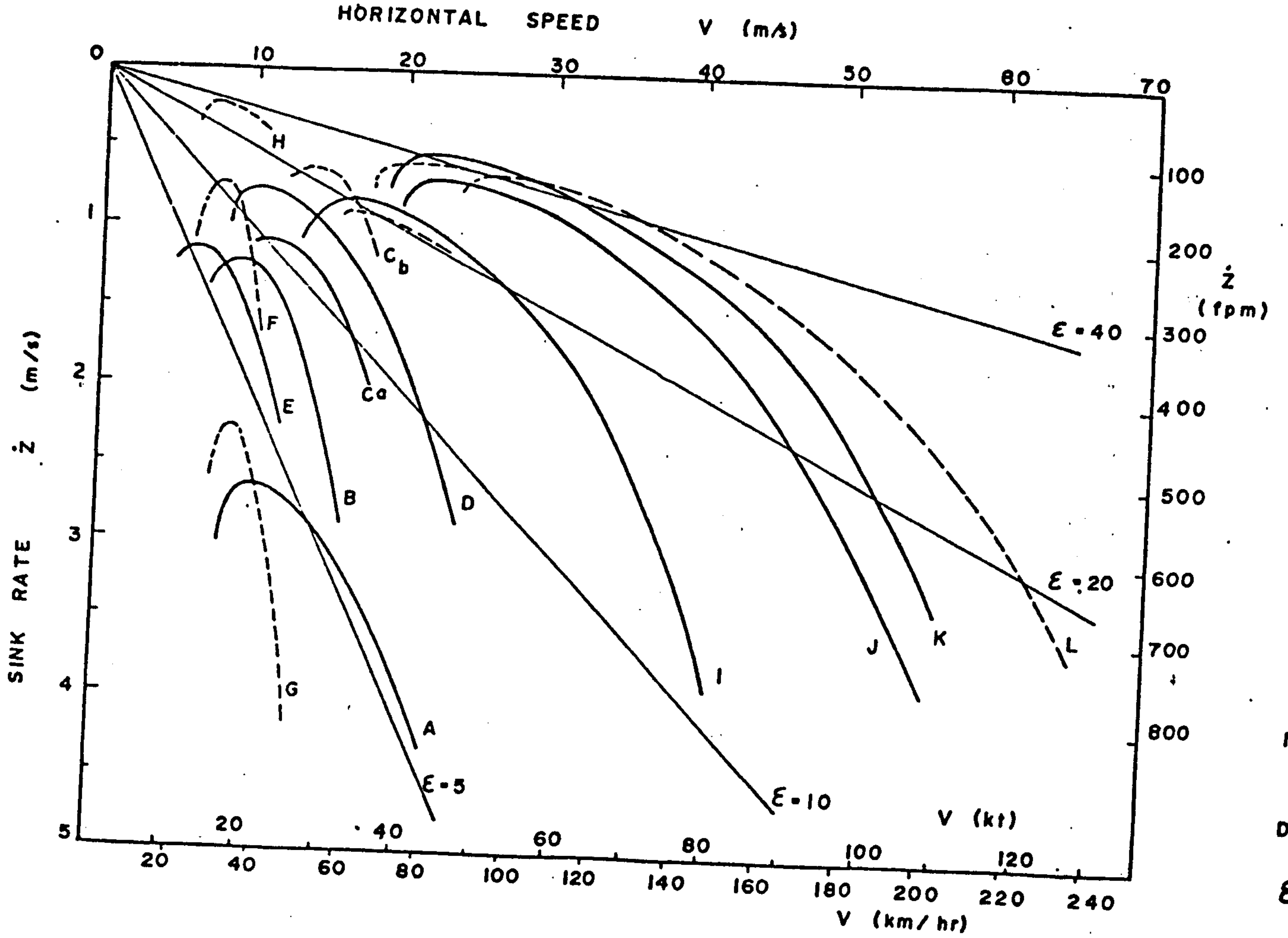


Figure 1. (continued) Sailplanes and Motor Gliders

2.52

Table 1. Sailplane & Motor Glider Characteristics

Type	First Flt.	Wing Span (ft)	Wing Area (ft^2)	Aspect Ratio	Weights (lb) Empty	Weights (lb) Loaded	Wing Load (lb/ft^2)	Airfoil	$\dot{z}$ (fpm) at (V - kt)	L/D_{max} at (V - kt)	V (kt) at $\dot{z}$ = 2 m/s	Power (BHP)	Ref.
Schleicher													
Ka 6 CR	1955	49.2	134	18.1	408	610	4.55	NACA63-618	134 (36)	29 (42)	70		3
ASW 12	1964	60	140	26	680	909	6.5	FX62-K-131	109 (43)	43.3 (48)	91		2
ASW 17	1972	65.6	159.7	27	893	1257	6.7 - 7.9	FX62-K-131 mod.	97	48 (54)			4
Schempp-Hirth													
Std. Cirrus	1969	49.2	107.5	22.5	466	734	6.82	FX66-S-196	134 (42.5)	35.2 (51)	85.5		2
Nimbus 2	1971	66.6	154.9	28.6	760	954	6.16	FX67-K-150	102 (43)	46 (51)	92		3
Glassflügel													
Libelle H-201	1964	49.2	102.3	23.6	397	633	6.18	Hütter	134 (43)	34.5 (50)	82.5		2
Kestrel H-401	1968	55.7	123.7	25.1	638	803	6.5	FX 67-K-170	124 (45)	38 (52)	92		2
Laister													
Nugget	1973	49.2	109	22.2	425	900	5.7 - 8.25	FX67-K-170	128 (42.5)	36 (50.5)	~88		4
Sigma	1973	68.9	131.2 (177.2)	36.2 (26.8)	1000	1300	9.9 (7.33)	FX67-VC-170/136	108 (37.5)	~50 (55.5)			
Jacobs "Weihe"	1938	59.1	198	17.6	508	738	3.7	Gö 549	120 (35)	31.5 (41)	~60		4
Schweizer 1-26	1954	40	160	10	433	593	3.7	NACA 43012A	165 (32.5)	21.5 (42)	64		2
Fornier RF-5	1968	45.0	162	12.5	1035	1430	8.85	NACA 23015	300 (51.5)	18 (57)	67.5	68	
Scheibe SF27M	1968	49.2	130	18.6	617	815	6.25	FX 61-184	152 (45)	31 (51.5)	82.5	26	
Schleicher ASK-14	1967	46.8	137.6	16.2	507	793	5.85	NACA 63-618	148 (39)	28 (45)		26	



A. PIGEON (5)
B. FULMAR (6)
C. BLACK VULTURE (7) (a. Parrot, b. Raspet)
D. WHITE-BACKED VULT. (8)
E. DOG-FACED BAT (11)
F. PTERANODON (12 b)
G. ROGALLO HANG GLD. (14)
H. PUFFIN II MPA (18) (outside grd. effect)
I. I-26 SAILPLANE (22)
J. STD. CIRRUS (23)
K. ASW-12 SAILPLANE (24)
L. VVG SAILPLANE (25)

Numbers in ( ) refer to entries in Table I, Ref. I

Dashed curves are estimated.

$\varepsilon = L/D$,

Figure 2. Still Air Sea Level Glide Polars For Several Natural And Man-Made Flying Devices

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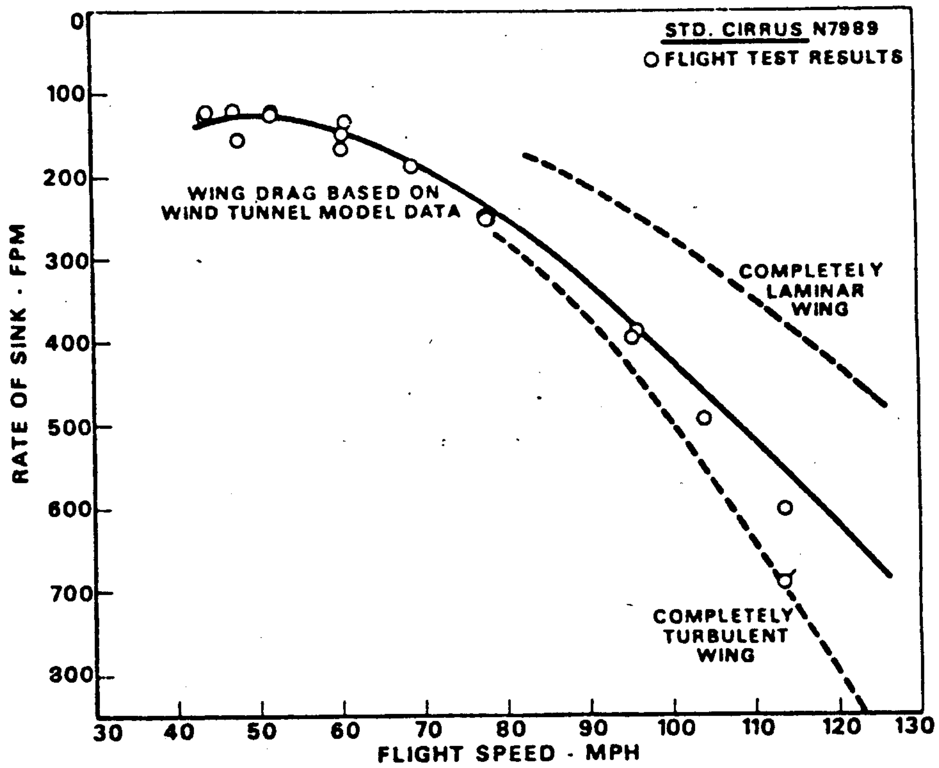
COMPARISON OF FLIGHT TEST RESULTS WITH THEORY



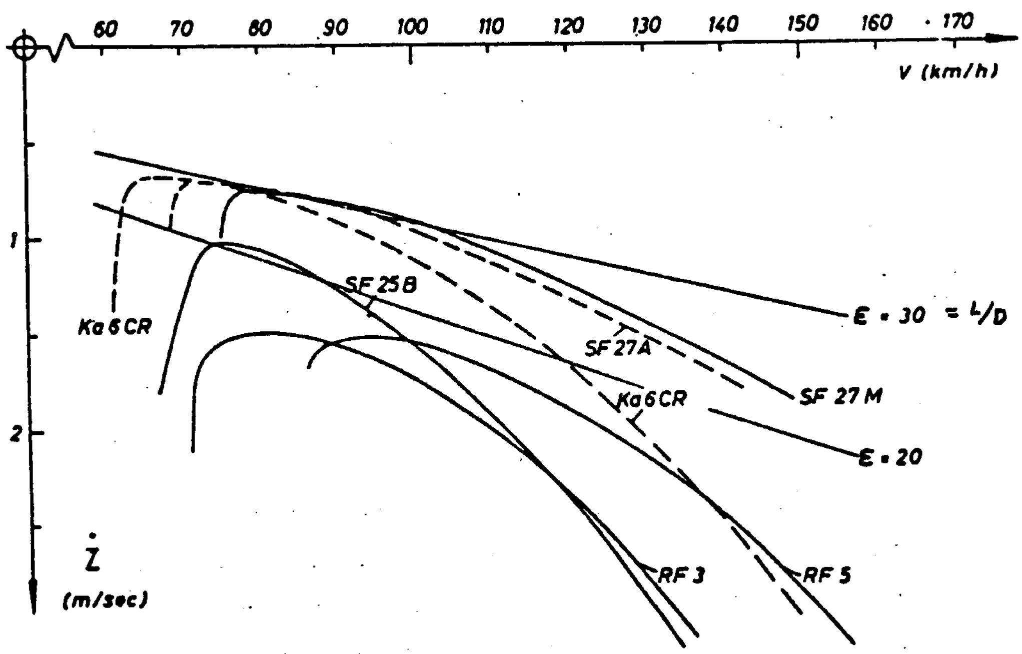


Figure 2.(continued) Still Air Sea Level Glide Polars for Several Natural And Man-Made Flying Devices

2.55

Table 2. Performance Improvement Comparison

	Weihe (1938)	ASW-12 (1964)	% Improv.
Min. $\dot{Z}$	120 fpm	109 fpm	+ 9%
V at $\dot{Z}_{min}$	35 kt	43 kt	-23%
L/D_{max}	31.5	43.3	+38%
V at L/D_{max}	41 kt	48 kt	+17%
V at $\dot{Z}$ = 6 fps	62.5 kt	88 kt	
L/D at $\dot{Z}$ = 6 fps	17.6	24.7	+41%

	Ka 6 CR (1955)	Std. Cirrus (1969)	% Improve.
Min. $\dot{Z}$	134 fpm	134 fpm	0%
V at $\dot{Z}_{min}$	36 kt	42.5 kt	-18%
L/D_{max}	29	35.2	+21%
V at L/D_{max}	42 kt	51 kt	+21%
V at $\dot{Z}$ = 2m/s	70 kt	85.5 kt	
L/D at $\dot{Z}$ = 2m/s	18	21.9	+22%

It should be noted that these gains in aerodynamic efficiency have not been accompanied by serious deterioration in stability, control or safety.

Technical Considerations

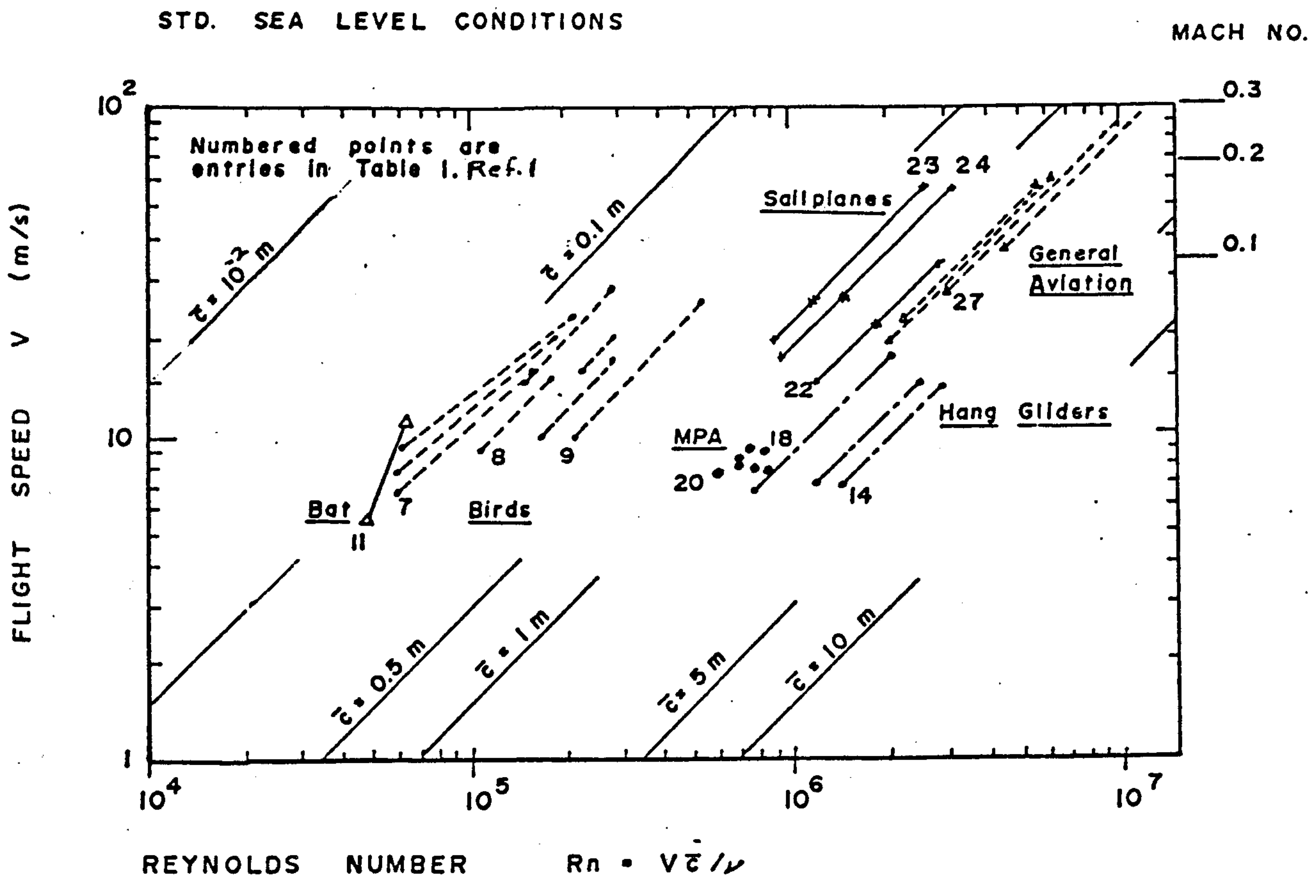
A number of practical factors make the sailplane design problem difficult, not all of which are directly related to the absence of an engine. For good climb performance (low sink rate) a low wing loading, low weight and excellent aerodynamic efficiency are desired. Further, if climbing is to be done predominantly in thermals, trim drag in moderately steep turns must be low and the speed for minimum sink rate (maximum climb rate) should be low to minimize turn radius (which, for a given bank angle, varies directly with speed squared). On the other hand, for high speed cruise the main concern is aerodynamic efficiency (high L/D). In a first order analysis (i.e. neglecting Reynolds number effects), L/D is independent of weight and thus for a given wing area, a "high" wing loading is desired. The obvious solution of use of variable geometry (e.g. Fowler flaps) to ameliorate the wing loading conflict is limited by several factors (e.g. class rules, economic and/or drag considerations, manufacturing difficulties) some of which will be discussed later. The demands of high aerodynamic efficiency in both climb and cruise require that great care be taken to minimize both parasite and induced drag. The latter is "easily" accomplished by use of high aspect ratio wings of near ideal planform. Given presently achievable values of parasite drag coefficient (about 0.010 based on wing area), the optimum compromise aspect ratios for Standard Class (span limited) sailplanes are between 18 and 22. Corresponding values for Open Class machines are between 25 and 30. *

The use of high aspect ratio wings of moderate area at typical sailplane speeds, means that the wing operates in a Reynolds number range well below that of conventional GA aircraft. For example, assuming a machine with an aspect ratio of 22 and wing area of 110 ft^2, with a useful speed range of 40 to 100 kt, the corresponding Reynolds number range (based on average wing chord) is 1.0 to 2.4×10^6 at sea level. If the machine weighed 700 lbs. loaded, the corresponding lift coefficient range at sea level would be $C_L = 1.18$ to 0.19. The general speed/Rn ranges for several types of low speed flying machines are shown in Figure 3. Sailplane experience indicates that with a little care, GA aircraft designers need not be overly concerned about the adverse influence of lowered Reynolds numbers on wing drag when large reductions in wing chord are contemplated.

Parasite Drag Reduction

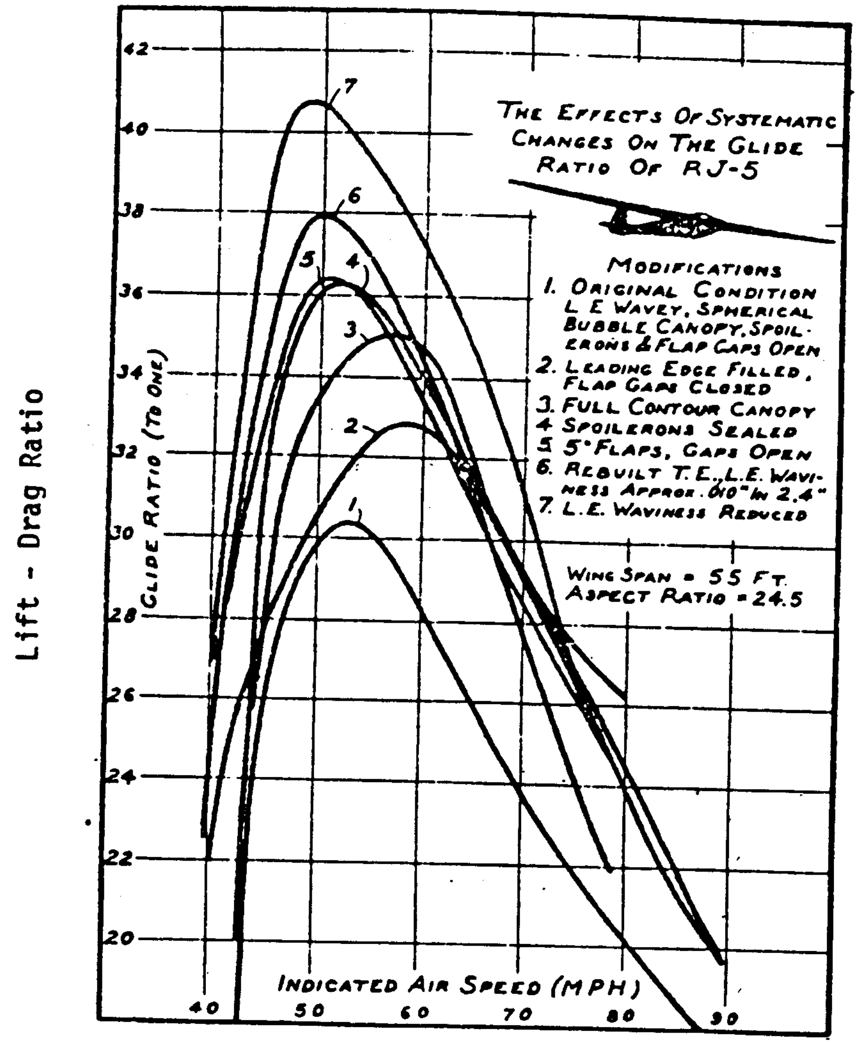
Post-war advances in sailplane performance began when Raspet (7) demonstrated the startling performance gains achievable by systematically cleaning up a machine of initially good aerodynamic layout. The machine used was Dick Johnson's one-of-a-kind RJ-5 Open Class sailplane which was of conventional layout and construction (largely wood) employing an NACA 6-series laminar flow airfoil. The results of the successive improvements resulting from careful sealing of gaps and leaks, and reduction of wing waviness and roughness are shown in the now classic Figure 4. The total cleanup resulted in a 25% reduction in parasite drag at L/D_{max}, about 40% of which was achieved by simple sealing and smoothing. A slightly more modern discussion of these effects has been presented by Wortmann (8). The impressive results obtained by Wil Schuemann in a general cleanup of an H-301 Libelle are discussed in (9).

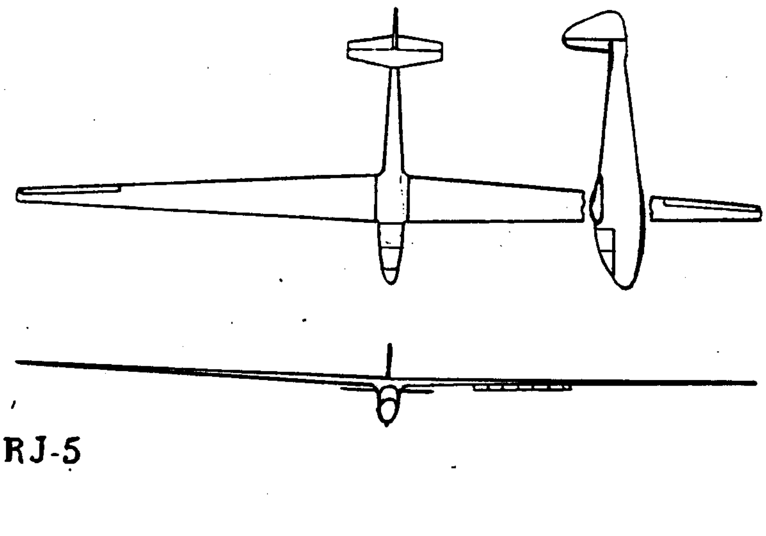
* The "optimum" in this case is not really clear, although practice indicates that machines with span greater than about 22 meters encounter serious flight and ground handling problems. Required wing area depends on the extent to which variable geometry can be achieved and desired wing loading. Thus overall operational consideration defining span and area limit optimum aspect ratios based on achievement of pure maximum L/D in both Standard and Open Class machines.



Note: Reynolds number based on average wing chord

Figure 3. Flight Speed/Reynolds Number Range for Various Low-Speed Flying Devices





RJ-5

Figure 4. Results of Drag Clean-up on the RJ-5 Sailplane

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Low-Speed Airfoils

The requirements of low drag over a reasonably wide lift coefficient range and the generally low values of operating Reynolds number make airfoil selection for sailplanes somewhat difficult. Pre-war sailplane designers relied primarily on Göttingen and NACA 4 and 5 digit airfoil, some of the former type (e.g. Gö 549) being specifically tested for sailplane applications. The advent of the NACA 6-series airfoils held promise of substantial drag reduction over at least the C_L range of the "laminar bucket" and, provided care was taken in manufacture, there appeared hope of obtaining the "bucket" in practice. A number of very successful designs were thus produced in the late 1940's and 1950's using various NACA 6x - 4xx and 6x - 6xx airfoils; the moderate camber of these sections representing a reasonable compromise for centering the "bucket" between the high and low speed extremes in required C_L.

The theoretical work of R. Eppler and F.X. Wortmann in Germany, beginning in the 1950's, showed that by careful contouring of the airfoil envelope and camber line, the transition point on low-to-moderate speed laminar airfoils could be controlled with some precision. This work lead to a family of Wortmann airfoils (the FX or Franz Xavier series) which have been almost universally adopted in sailplanes designed during the last decade. Wortmann's work is well summarized in his paper in (5) and his airfoils have produced something of a revolution in modern sailplane performance.

Wortmann has shown that by carefully contouring the upper surface of a fairly highly cambered airfoil, the upper end of the laminar flow range can be extended to section C_L values required for low sink rate.

When a highly cambered airfoil is operated at low C_L values, however, the airfoil is frequently flying at a negative geometric angle of attack, and thus the lower surface of the airfoil is the one on which transition (and/or separation) is of primary concern in maintaining low profile drag. Thus, by careful contouring of both the upper and lower surfaces, the low drag "bucket" can be significantly extended (in operating C_L range) compared to NACA 6-series sections of similar thickness and minimum drag. The extent of the bucket can often be further increased by adjusting the camber line with a small chord (10 - 20%) simple hinged flap at the trailing edge. Examples of the possible improvement are shown in Figure 5. Several typical sailplane and related airfoils are shown in Figure 6, and the general trend in maximum section lift-drag ratio with Reynolds number for several Wortmann and NACA Sections are shown in Figure 7.

While Wortmann's results are impressive the limited data available on the new Liebeck (10) sections appears spectacular. Whether such airfoils, which appear to approach some sort of theoretical limit in single element airfoil lift-drag ratio, can perform in practice when built into a practical wing remains at present an open question. Wortmann's investigations of the same type airfoils is reported in his paper in (6).

High Lift Devices

In the modern gospel of sailplanes airfoil design according to Dr. Wortmann the wing contour must be absolutely smooth and unbroken as far aft as possible. Thus, leading edge high lift devices, wide chord flaps or ailerons and particularly conventional Fowler flaps of significant chord are out of the question in sailplane design. Thus the designers choice of high lift devices is severely limited. As one example of a way to circumvent this problem, and provide the performance benefits theoretically available from use of area changing flaps *, Wortmann tailored a unique airfoil/flap system specifically for the very advanced British "Sigma" sailplane project (see Figure 1). The FX 67-VC-170/136 section for "Sigma" is fully described in (11) and the combined polar at $Rn=3x10^6$ is shown in Figure 5. The flap of this airfoil is "hidden" inside the basic FX 67-VC-170 airfoil when retracted, thus avoiding flow disruption at high speed. When extended, it produces a 36% increase in chord. An even more exotic scheme has been proposed and tested by Wortmann (11) which involves deploying a large sheet of sailcloth (e.g. dacron) allowing chord extensions of greater than 50% in the high C_L range.

Structures

The third component in the post-war revolution in sailplane performance has been the introduction of fiberglass as the main construction material; as pioneered by Nägele, Eppler, Stender and Hänle in Germany. The use of fiberglass wing skins allows fabrication of relatively wave free surfaces of unexcelled smoothness. A further consequence of the use of relatively low modulus of elasticity fiberglass is that in order to maintain desired levels of torsional and bending stiffness, wing skins must be quite thick and correspondingly stronger than required by existing sailplane airworthiness standards. One thus finds fiberglass sailplanes with load factors approaching those of modern fighter aircraft with little weight penalty (due to the low specific gravity of the fiberglass). Considerable room for further improvements in structures exists by use of advanced composites and materials such as DuPont Kevlar (PRD-49) with nearly three times the modulus of elasticity of existing E-glass systems.

* Partial span Fowler flap systems have been extensively tested on the South African BJ-series of sailplanes with generally poor results.

2.6 b

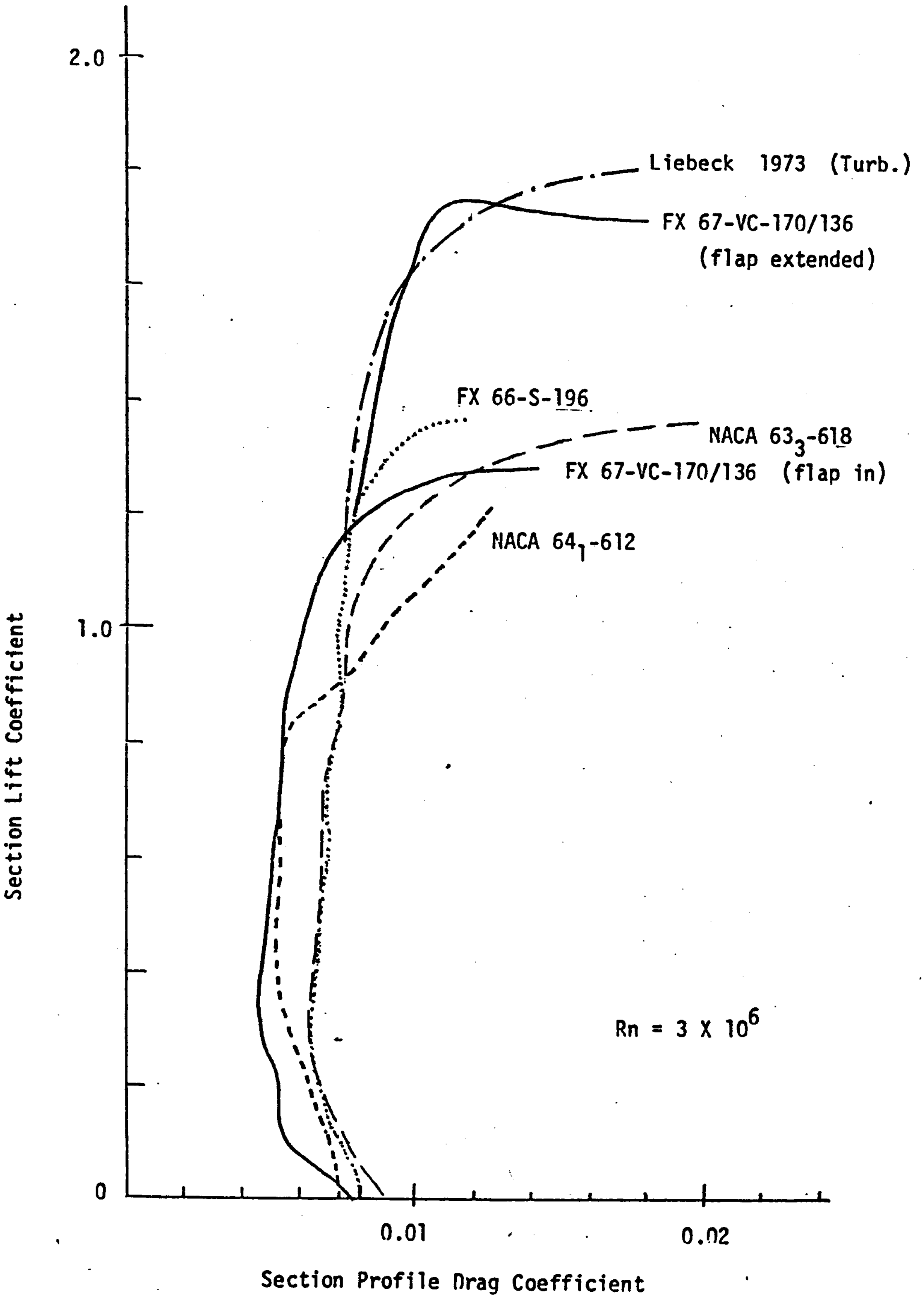

Figure 5. Several Airfoil Drag Polars at a Reynolds Number of 3 X 10^6

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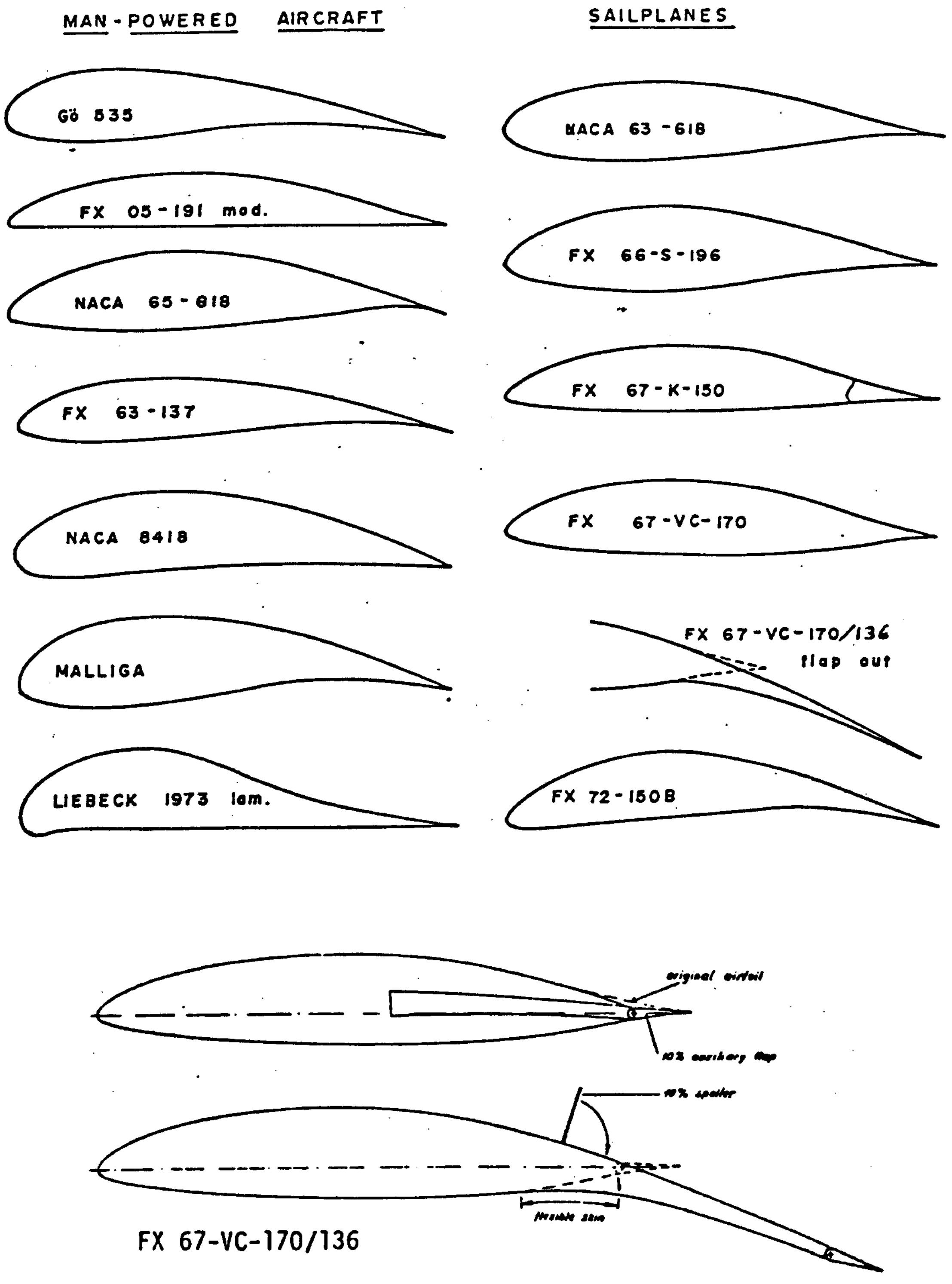

Figure 6. Typical Sailplane and Related Airfoils

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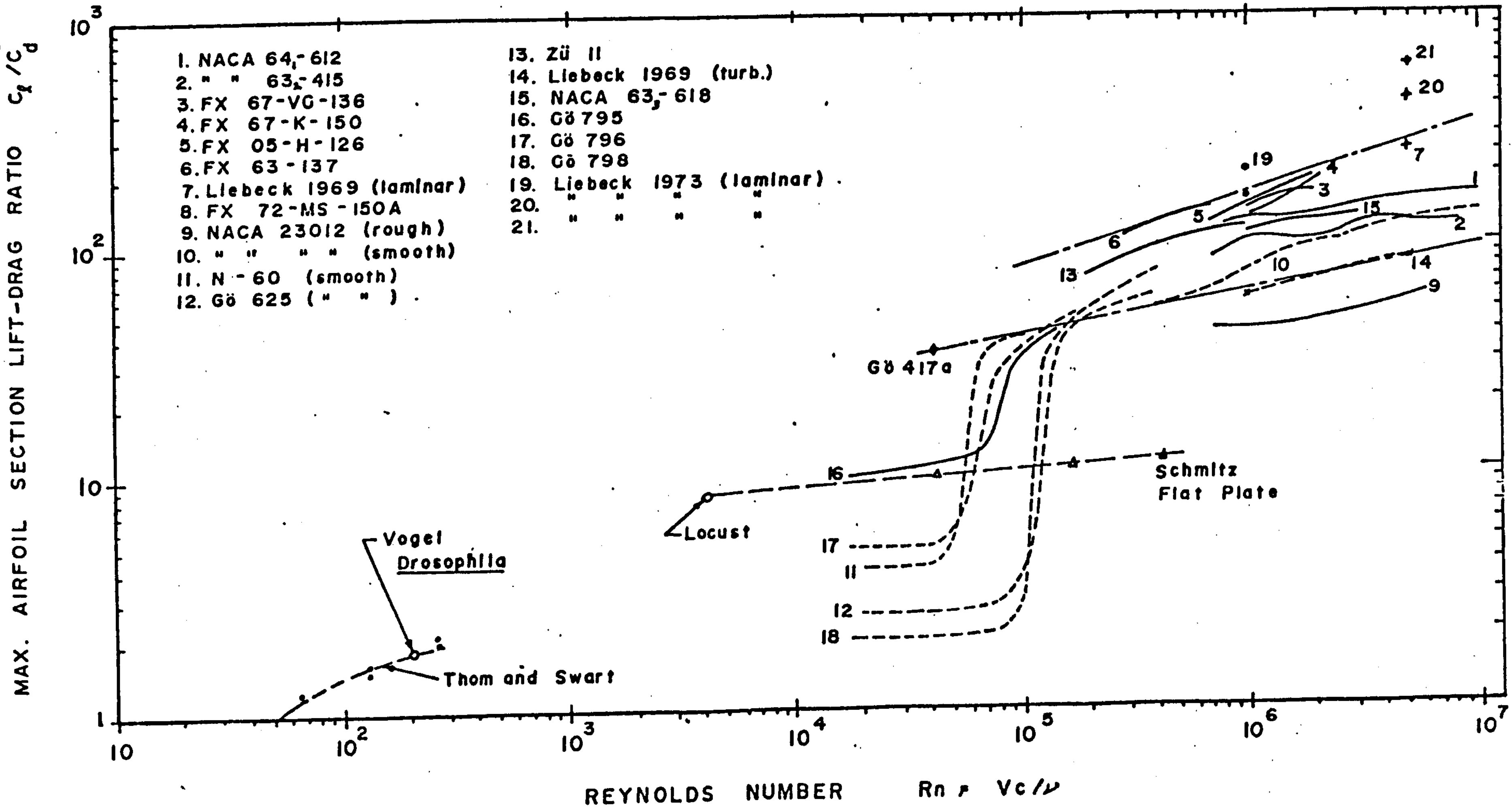

Figure 7. Variation in Maximum Airfoil Section Lift-Drag Ratio with Reynolds Number

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Research Needs

1. Performance of Wortmann type sail plane airfoils in wings with roughness, waviness and Reynolds numbers typical of light powered general aviation aircraft need to be further investigated.

2. Simple, economical methods of construction need to be found which lead to improved surface finish and manufacturing tolerance to approach the performance of sailplane types for powered general aviation aircraft. Possible examples are:
   - "Plastic" coatings over conventional structures. (e.g. Schweizer I-35 sailplane)
   - Metal bonding techniques (e.g. the Laister "Nugget")
   - Major assembly extrusions from metal and plastic

3. The optimum configuration design implications of the use of sailplane technology in powered general aviation aircraft needs to be investigated. Specifically:
   - optimum wing geometry
   - engine/thrust producing device location
   - requirements for high lift devices
   - wing/body integration (i.e. wing position and body shape) to minimize adverse interference

4. The economic and configuration implications, in light of the fuel crisis and sailplane technology, of optimizing the design of a given general aviation aircraft toward maximum "transport economy" even at the possible expense of "productivity" needs to be evaluated.

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Existing fiberglass systems have several disadvantages, however. Since most high performance sailplanes are produced in limited quantities, most fabrication is done by hand lay-up resulting in high cost and major quality control problems. Inspection for structural integrity remains a major difficulty. Questions also remain about the aging characteristics of existing fiberglass systems, and the problem of ultra-violet degradation of the structure carries the commerical disadvantage of offering the customer a choice of the basic color scheme of his aircraft. Thus, alternative fabrication schemes to preserve the beneficial aerodynamic qualities of fiberglass construction, while reducing cost, etc. continue to be explored. Notable among these schemes are the use of bonding (e.g. the Liaster "Nugget") (4), the use of "plastic" coatings over conventional aluminum structure (e.g. Schweizer 1-35) (4) and the extrusion of major structural assemblies as described by Morelli (5).

Concluding Comments

The design of modern high performance sailplanes is an enormously challenging task. The absence of an engine means that the designer cannot indulge in the "luxury" of simply fitting the machine with a larger powerplant to obscure deficiencies in aerodynamic design, weight overruns or to provide the basic design with "growth potential". Further, the machine must have very low drag values over a relatively wide lift coefficient range and these values must be achieved in a relatively low Reynolds number range. It is for these reasons that a careful study of the remarkably successful methods sailplane builders have used to achieve these goals may well repay the designers of powered General Aviation aircraft. It has not been the intention of the authors of this brief discussion to advocate or imply that all sailplane technology is applicable to General Aviation aircraft in general or that the operational and economic constraints faced by the GA designer make incorporation of applicable features feasible. However, even a brief examination of the performance figures achieved by modern soaring machines and a little reflection as the often huge disparity in L/D values between sailplanes and GA aircraft indicates that careful attention to the lessons learned in sailplane design and manufacture hold realistic promise for substantial gains in the aerodynamic efficiency of several GA types. The fuel crisis, whether transient or permanent, may force a redirection in GA design with greatly increased emphasis on operation "economy", perhaps even at the expense of speed (or productivity) and initial vehicle cost. Modern soaring technology indicates one path along which future development might progress.

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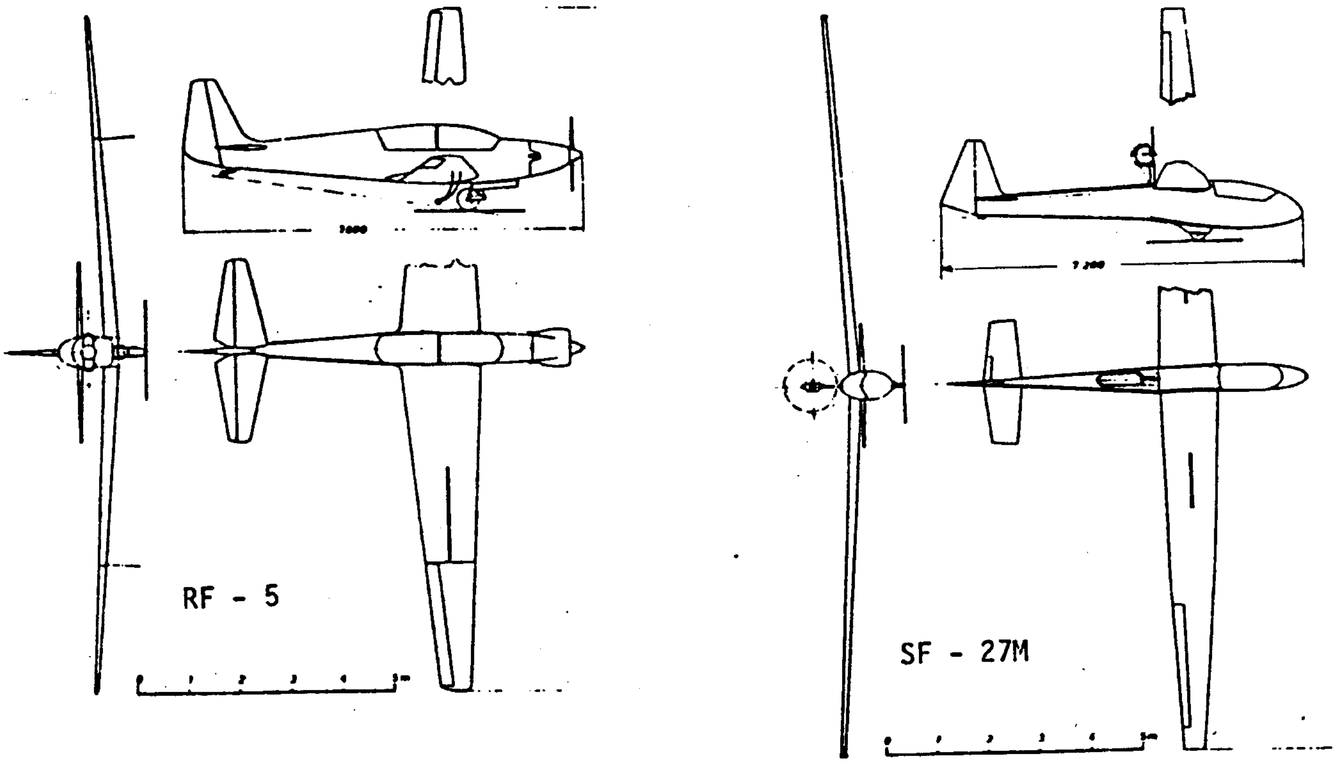
Figure 1. (continued) Sailplanes and Motor Gliders



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III. Design Preliminaries - Constraints (3 hours)

A. Government Regulations (FAR's)

B. V-n Diagram Construction

C. FAR stall & buffet limits

D. Other constraints (e.g., economic, aesthetic)

E. Introduction to Optimization - Geometric Programming

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Table 1-2. Subdivision of FAR requirements

SPECIFIC REQUIREMENT / AIRWORTHINESS STANDARD		PART 25	SFAR PART 23	PART 23
GENERAL	. weight limitation	no	⩽ 12,500 lb	⩽ 12,500 lb
	. min. number of engines	two	two	one
	. seating capacity	no restriction	more than 10 occupants	up to 9 passengers
PERFORMANCE	. engine failure req.'s in takeoff	yes	no	no
	. accelerate-stop	complete stop	limited	no
	. landing	detailed	limited	> 6000 lb: limited ⩽ 6000 lb: no
	. wet runway	yes	no	no
	. climb capability after engine failure	throughout flight	takeoff, landing	multi-engine: limited en-route
FLIGHT CHARACTERISTICS	. lateral c.g. shift	included	no	no
	. minimum control speed	related to liftoff speed and stall. speed	related to stalling speed at MTOW	
	. spin characteristics	no	limited	complete
	. maneuver load factor margin in cruise	avoid buffet onset	no	no
STRUCTURAL DESIGN, CONSTRUCTION	. maneuver and gust load envelope	yes	yes	limited for single engine
	. fatigue evaluation	fail-safe, safe life fatigue evaluation of major parts	for pressure cabin, wing and associated structure	
	. fail safe / safe life	specified throughout	for wing and carry - through structure	
	. bird-proof windshield	yes	no	no
	. limit descent velocity for landing gear loads	10 fps	dependent on landing wing loading, but ⩽ 10 fps.	
	. max. cabin pressure alt. after system failure	15,000 ft	no	no
	. special emergency provisions for pax.	yes	yes	no
	. ice protection prov.	yes	limited	no
	. restarting capability of engines	yes	yes	no
SYSTEMS, EQUIPMENT	. powerplants and related systems	complete independence	complete independence	limited independence
	. system redundancy	throughout	essential functions duplicated	no
	. equipment for adverse weather flight	yes	yes	no

Table 1-4. Differences in FAR airworthiness standards for small and transport category airplanes

British Civil Air Regulations (BCAR)

Section	Requirement	U.S. Equiv.
A	Procedure	Sub. Chap. B
C	Engine & Prop.	Pt. 33 & 35
D	Aeorplane (Large)	Pt. 25
E	Glider	Pt. 21 + Glider Criteria Handbk.
G	Rotorcraft	Pt 27 & 29
J	Electrical	Pt 37 & 41
K	Light Plane	Pt. 23
L	Licensing	Sub. Chap. D
R.	Radio	Pt. 37 & 41

Operations → Separate Air Navigation Regulations.

COUNTRY (BUREAU)		UNITED STATES OF AMERICA (F.A.A.)			GREAT BRITAIN (C.A.A.)	
GROUP	AIRPLANES PERFORMANCE GROUP	SMALL		GENERALLY LARGE	LIGHT	LARGE
	MAX. TAKEOFF WEIGHT	≤ 12,500 LB	≤ 12,500 LB		C, D, E, ≤ 12,500 LB	A >12,500 LB
CATEGORY		NORMAL, UTILITY ACROBATIC AND AGRICULTURAL (RESTR.)	NORMAL	TRANSPORT	NON-, SEMI- AND AEROBATIC AGRICULTURAL	NON-AEROBATIC
CLASS	NUMBER OF ENGINES	ONE OR MORE	TWO OR MORE	TWO OR MORE	ONE OR MORE	TWO OR MORE
	TYPE OF ENGINE	ALL TYPES*	PROPELLER ENGINES ONLY	ALL TYPES*	ALL TYPES*	ALL TYPES*
MINIMUM CREW	FLIGHT	ONE OR MORE	TWO	TWO OR MORE	ONE OR MORE	TWO OR MORE
	CABIN ATTENDANTS	NONE	<20 PASS.: NONE ≥20 PASS.: ONE	<10 PASS.: NONE ≥10 PASS.: ONE OR MORE	-	-
MAX. NUMBER OF OCCUPANTS		10	11 THRU 23	NOT RESTRICTED	-	NOT RESTRICTED
MAX. OPERATING ALTITUDE		25,000 FT	25,000 FT	NOT RESTRICTED	NOT RESTRICTED	NOT RESTRICTED
MAX. DESIGN DIVING SPEED			NOT RESTRICTED		300 KTS/M = .6	NOT RESTRICTED
APPLICABILITY				*reciprocating, turboprops, -jets and -fans		
AIRWORTHINESS STANDARDS AIRPLANES " " ENGINES " " PROPELLERS		FAR PART 23 " " 33 " " 35	SFAR PART 23 FAR " 33 " " 35	FAR PART 25 " " 33 " " 35	BCAR SECTION K " " C " " C	BCAR SECTION D " " C " " C
NOISE STANDARDS		FAR PART 36	PROP. DRIVEN: APPENDIX F	" " 36	-	-
GENERAL OPERATING AND FLIGHT RULES		FAR PART 91	FAR PART 91	FAR PART 91	LAID DOWN IN AIR NAVIGATION REGULATIONS	
OPERATIONS	DOMESTIC, FLAG AND SUPPLEMENTAL COMM. OPERATORS OF LARGE AIRCRAFT	-	-	FAR PART 121		
	AIR TRAVEL CLUBS USING LARGE AIRCRAFT	-	-	FAR PART 123		
	AIR TAXI AND COMM. OPERATORS		FAR PART 135	-		
	AGRICULTURAL AIRCRAFT	FAR PART 137	-	-		

Table 1-1. Classification of aircraft categories in the American and British airworthiness requirements

PROFILE SEGMENT	PERFORMANCE CONFIGURATION	ENGINE POWER/THRUST	ALTITUDE	SPEED	REQUIRED DATA	REMARKS
GROUND MANEUVER (gm)	MTOW/MLW flaps down	as required $\mu = .02$	0 ft	-	Time: T_{gm} 1) Fuel: F_{gm} 1) Distance: 0	1) ATA '67: 14 min. ground idle + 1 min. takeoff
TAKEOFF	MTOW t.o. flaps	takeoff, all engines	0-35 ft	0 to V_2 + 10 kts		
INITIAL CLIMB AND ACCELERATE	t.o. flaps, u.c. up	takeoff, all engines	35-400 ft	V_2 + 10 kts	Time: T_{to} Fuel: F_{to} Distance: D_{to}	2) below 10,000 ft V_{cl} may not be more than 250 kts (IAS) 3) - max. cabin rate of climb: 300 f p m - at high altitude climb at M_{cl},
	flaps retracting		400-2,000 ft	accelerate to $1.2V_S$(flaps up)		
EN ROUTE CLIMB TO CR. ALTITUDE, ACCELERATE (cl)	en route	max. operational climb	2,000 ft	accelerate to V_{cl} 2) 3)	Time: T_{cl} Fuel: F_{cl} Distance: D_{cl}	4) cruise altitude such that: $D_{cl}+D_{des}<D_{ct}$
			2,000 ft to cruise altitude	V_{cl} 2) 3) and acc. to V_{cr}		
CRUISE (cr)	en route	as required, up to max. cruise	cruise alt.; 4) long-range: max. of two step climbs	V_{cr} (M_{cr}) long-range, cost-econ. or high-speed	Time: $T_{cr}=D_{cr}/V_{cr}$ Fuel: F_{cr} Distance: D_{cr}	5) D_{cr}=1.02 x actual cruise distance + 20 n m
DESCENT AND DECELERATE (des)	en route	flight idle	cruise alt. to 15,000 ft	V_{des} 2)6)	Time: T_{des} Fuel: F_{des} Distance: D_{des}	6) max. cabin rate of descent 300 fpm $\theta_{cr}-\theta_{des} \leq 4^o$ (θ=fuselage inclination angle)
	appro. flap below 10,000ft	as required for 3°	15,000 ft to 3,500 ft	initial appro. speed		
LANDING	landing flap, u.c. down	glide slope	3,500 ft to 0 ft	final appro. speed to 0		
AIR MANEUVERS (am)	en route	as required	cruise	V_{cr}	Time: T_{am} 7) Fuel: F_{am} 8) Distance: 0	7) T_{am} = 6 min 8) $F_{am} = (T_{am}/T_{cr})\ F_{cr}$

Table 11-1. Typical basic flight profile analysis, transport aircraft (Refs.: ATA '67 method, SAWE Technical Report 619)

DOMESTIC OPERATIONS (up to 3,000 n m)	INTERNATIONAL OPERATIONS (above 3,000 n m)
ATA '67: 1. Fly for 1 hour at normal cruise altitude at fuel flow for end of cruise weight, speed for 99% of maximum range speed. 2. Exercise a missed approach and climbout at destination. 3. Fly to and land at alternate airport 200 n m distant. OTHER PROCEDURE 1. Execute missed approach at destination airport. 2. Cruise to alternate airport 200 n m distant at cruise speed used in basic mission (altitude optional). 3. Hold at alternate airport for 45 minutes, altitude 1,500 ft. 4. Descend and land at alternate airport. 5. Contingency fuel equal to 50% of holding fuel.	WITH ALTERNATE (ATA '67): 1. Continue flight for time equal to 10% of basic flight time at normal cruise altitude and speed for 99% maximum range. 2. Execute missed approach and climbout at destinate airport. 3. Fly to alternate airport 200 n m distant. 4. Hold at alternate airport for 30 minutes at 1,500 ft above the ground. 5. Descend and land at alternate airport. WITHOUT ALTERNATE: Continue basic flight profile for two hours.
	Conditions for flight to alternate airport and holding: 1. Cruise thrust or power setting may be equal to 99% of max. subsonic range, cruise speed used in basic mission (Table 11-1). 2. Holding thrust or power setting shall be for maximum endurance or 110% of min. drag or power speed, whichever is greater. 3. Cruise altitude shall be optimum for best range, except that it shall not exceed the altitude where cruise ditance equals the climb plus descent distance.

Table 11-2. Typical reserve fuel policies

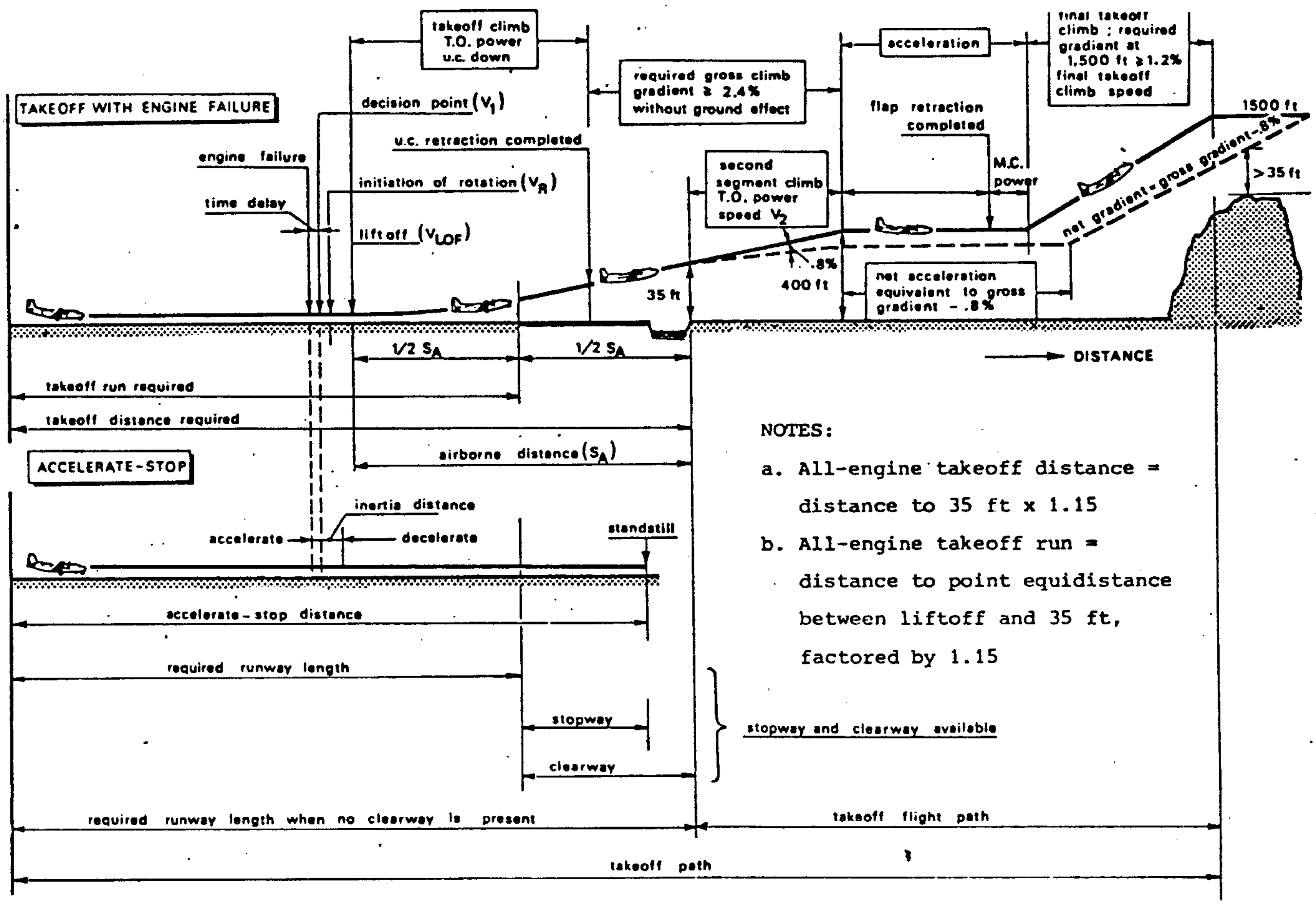
PHASE OF FLIGHT		AIRPLANE CONFIGURATION						MINIMUM CLIMB GRADIENT %		
		flap setting	u.c.	engine	thrust (power)	speed	altitude	$N_e=2$	$N_e=3$	$N_e=4$
TAKEOFF CLIMB POTENTIAL ("first segment")		t.o.	↓	one engine out	t.o.	V_{LOF}	0→h_{uu} [1]	0	.3	.5
TAKEOFF FLIGHT PATH	"second segment"	t.o.	↑	one engine out	t.o.	V_2 [2]	h_{uu}→400 ft	2.4	2.7	3.0
	final takeoff ("third segment")	en route	↑	one engine out	max. cont.	$V \geq 1.25 V_S$	400→1,500ft	1.2	1.5	1.7
APPROACH CLIMB POTENTIAL		approach [3]	↑	one engine out	t.o.	$V \leq 1.5 V_S$	0 [1]	2.1	2.4	2.7
LANDING CLIMB POTENTIAL		landing	↓	all engines	takeoff [4]	$V \leq 1.3 V_S$	0 [1]	3.2	3.2	3.2

Nomenclature:

- V_{LOF} - liftoff speed
- V_2 - takeoff safety speed
- V_R - rotation speed
- V_S - stalling speed
- u.c. - undercarriage position
- h_{uu} - height at which u.c. retraction is completed
- N_e - number of engines per a/c

1) out of ground effect
2) defined in Section 2 of Appendix K
3) flap setting such that $V_S \leq 1.10\ V_S$ for landing
4) more precisely: the engine power (thrust) available 8 seconds after throttle opening to takeoff rating
5) takeoff requirements are at actual weight, other requirements at landing (touchdown) weight

Table 11-3. Summary of climb requirements for turbine-powered transport category aircraft (FAR 25)



Takeoff procedures and requirements for a twin-engine civil transport aircraft

AN OVERVIEW OF

TECHNOLOGY RESEARCH APPLIED TO THE BOEING 757

by

J. K. Wimpress

and

J. H. McMasters

A Keynote Address Presented at the Boeing Commercial Airplane Company

1981 Engineering Research and Development Symposium

18 May 1981

SUMMARY

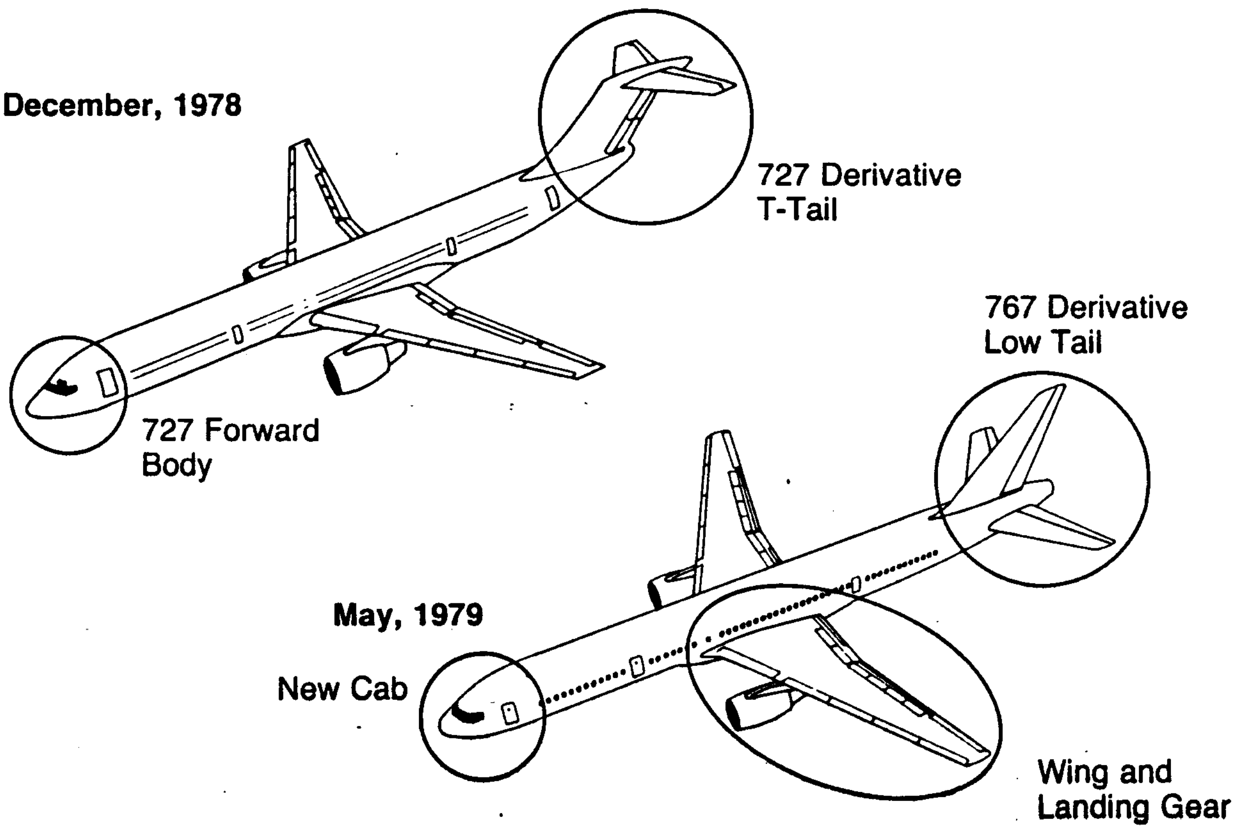
In May 1981, the Boeing Commercial Airplane Company held an R&D Symposium with the purpose of presenting to the various staffs within BCAC the technological accomplishments of the various research units since the last Symposium held in October 1977. An important secondary objective of the Symposium was to display to management the value of the sustained company funded research effort. To this latter end, the senior author was asked to prepare and deliver a keynote address outlining the direct research benefit to a current major airplane program - the 757. The result of this effort is documented in this report.

The presentation begins with a brief review of the development history and characteristics of the Boeing 757. Then, in order to place the role of research advances applied to the 757 in context, several factors which profoundly influence the modern transport airplane design problem are discussed. It is shown that the main motive forces behind the 757 generation aircraft are a desire to improve fuel efficiency and reduce noise. In addition the technology has reached a level of maturity and new constraints of increasing stringency have been added to the point where we have been forced to run harder merely to stay in place.

With this background and context established, the remainder of the presentation is directed at a discussion of selected highlights of research advance in each of several technologies (i.e., aerodynamics, flight controls, propulsion, structures, weights, systems and noise) applied directly to the 757 project.

The conclusions reached are that safety, efficiency and durability are the keys to successful 757 development. Research has, in turn, provided the means to design an aircraft which excels in these parameters. It is further concluded that one should not expect all research effort to be successful - that is not the nature of research. To be of value to an airplane project, research developed tools and methods must be ready when needed. Finally, an important benefit of the sustained research effort is the training it provides in the use of new design methods and tools.

Configuration Development



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INTRODUCTION

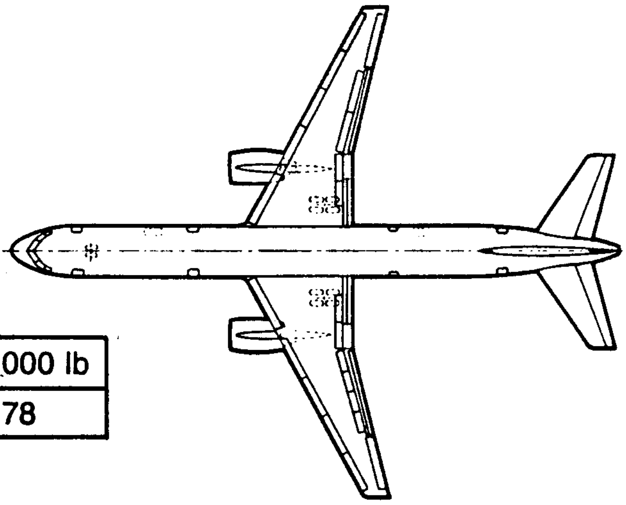
The evolution of ~~which~~ the 757 began in derivative studies of the 727. However, competitive pressure demanded a completely new design representing the application of the best knowledge of The Boeing Company today. The 757 is a thoroughly modern aircraft and is expected to remain in the forefront of its technology for the remainder of this century.

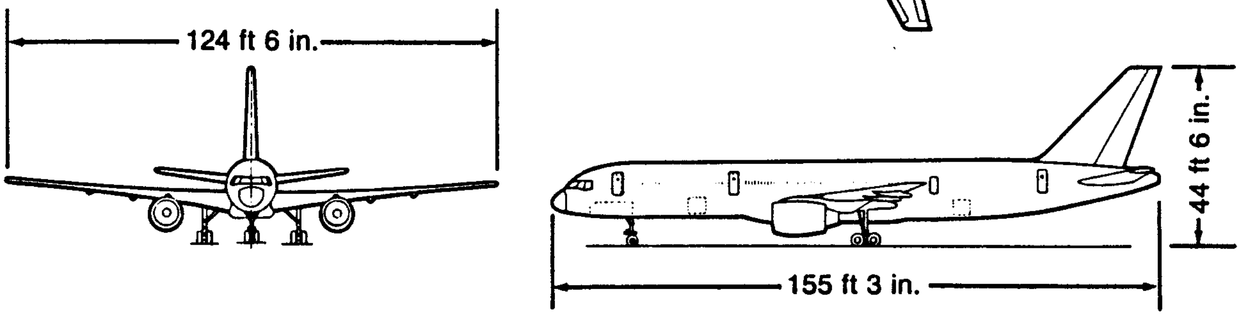
General Arrangement

Boeing 757

First Flight 2/82
First Delivery 1/83

Engine Options	
Rolls-Royce RB211-535C	
Pratt & Whitney PW2037	
Maximum Brake Release Weight	220,000 lb
Mixed Class Passengers	178





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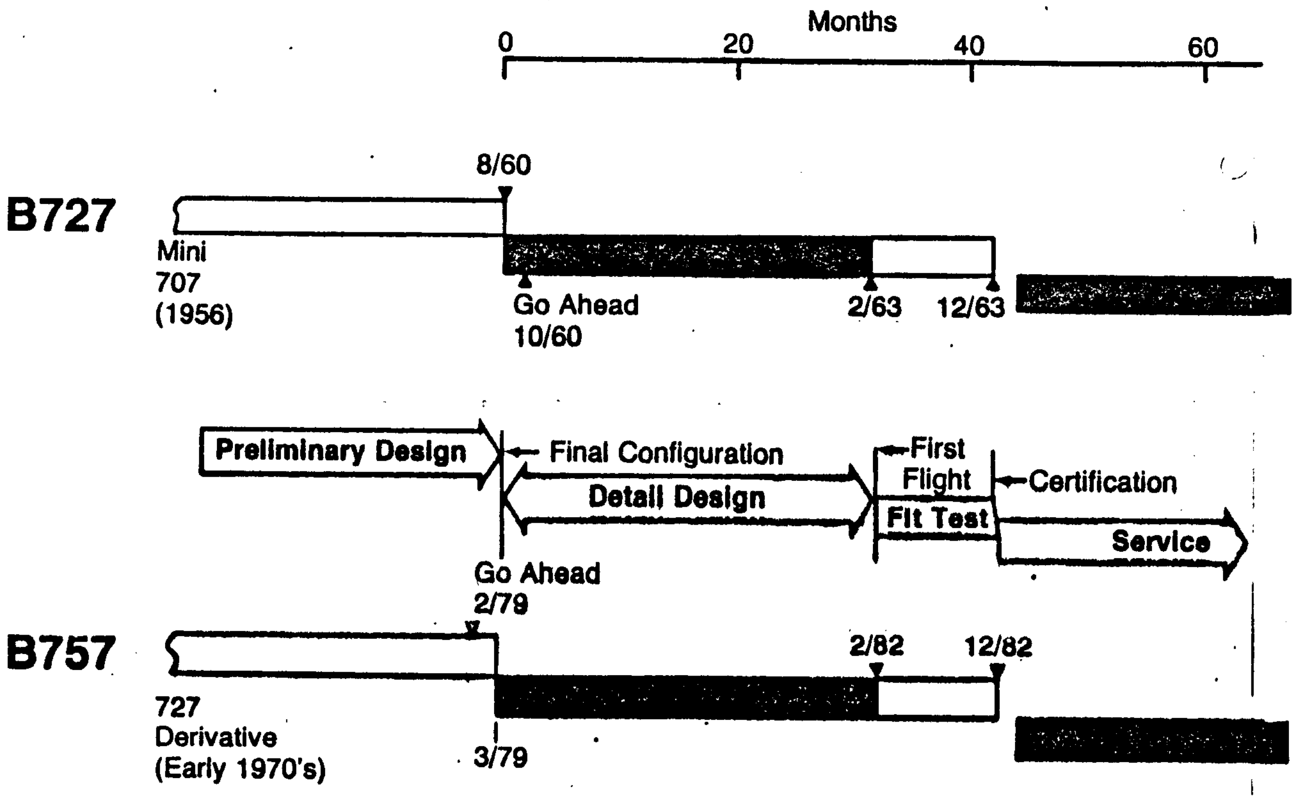
But where is the evidence of research application? The 757 has the appearance of a plain vanilla transport airplane. Its speed, altitude, range and field length performances are not unusual and appear to represent no advance over previous aircraft in its category.

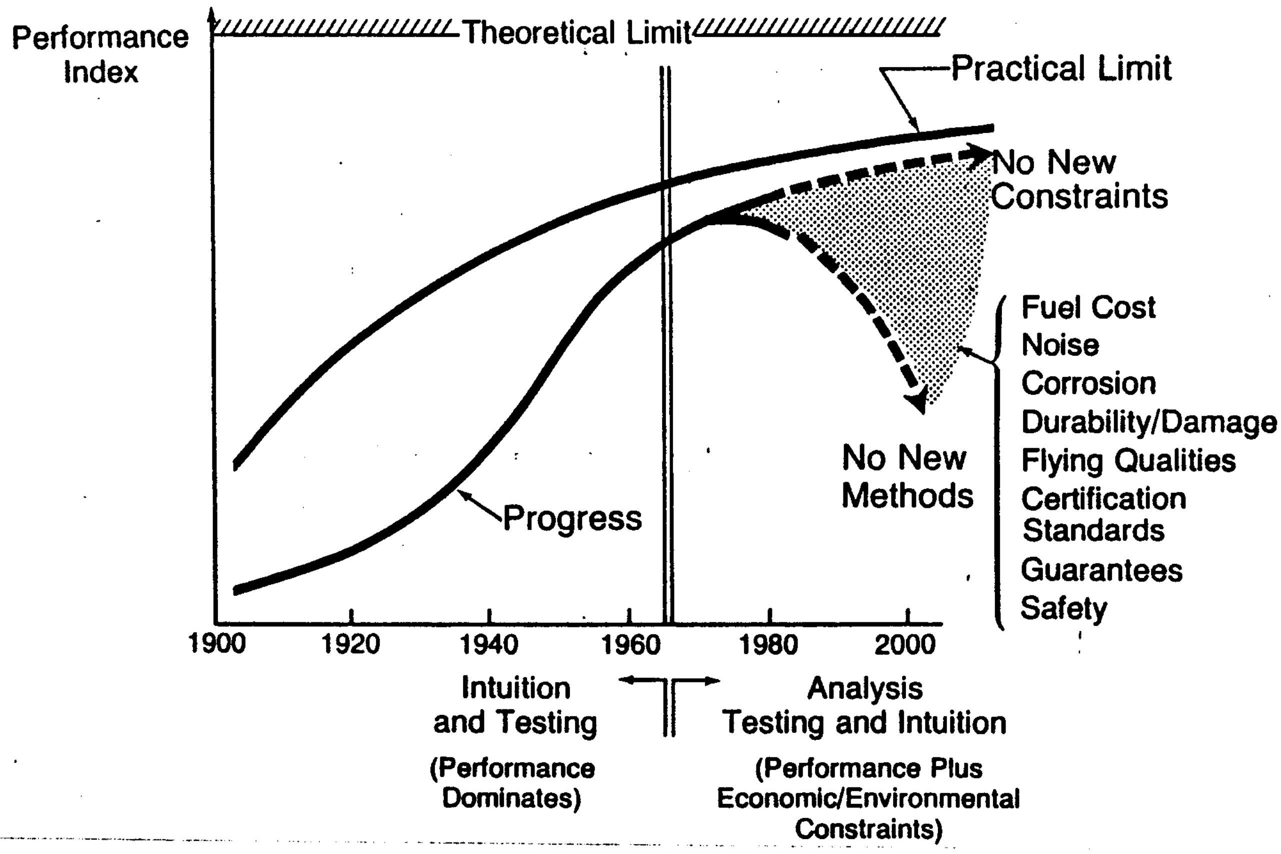
Developmental Time Scale Comparison



Even the 757's developmental time scale is about the same four year interval of its predecessor 727 of nearly two decades ago.

The fruits of Boeing funded and other research <u>are</u> embodied in the aircraft, but in many cases they are hidden. There are new electronic controls and new materials, but in many cases the advances made possible by sustained company research efforts are even more subtle.

Technology Development



In order to place the current applications of research to the 757 in context, it is first necessary to review general trends in the history of air transport development. For this purpose, one may consider a plot of improvement in some relevent index of performance (e.g., lift-drag ratio, direct operating cost, fuel burned) versus historical time. On such a plot there will be two upper bounds. One is an absolute theoretical limit (such as that on induced drag or thermodynamic efficiency) and a second, time dependent, practical bound representing the state-of-the-art in the various fundamental technologies (aerodynamic, materials, etc.) involved. One observes that in the early period of development, rapid progress can be made by large-ly cut-and-try experimentation with relatively low investment. However, as the practical limit is approached, progress lessens, and at the same time other, often non-technical influences may enter the design problem. At this point, performance would deteriorate without new technical developments and improved design methodology. The up-shot is that at the present stage in air transport development, we must run harder and harder merely to stay in place, let alone make significant advances.

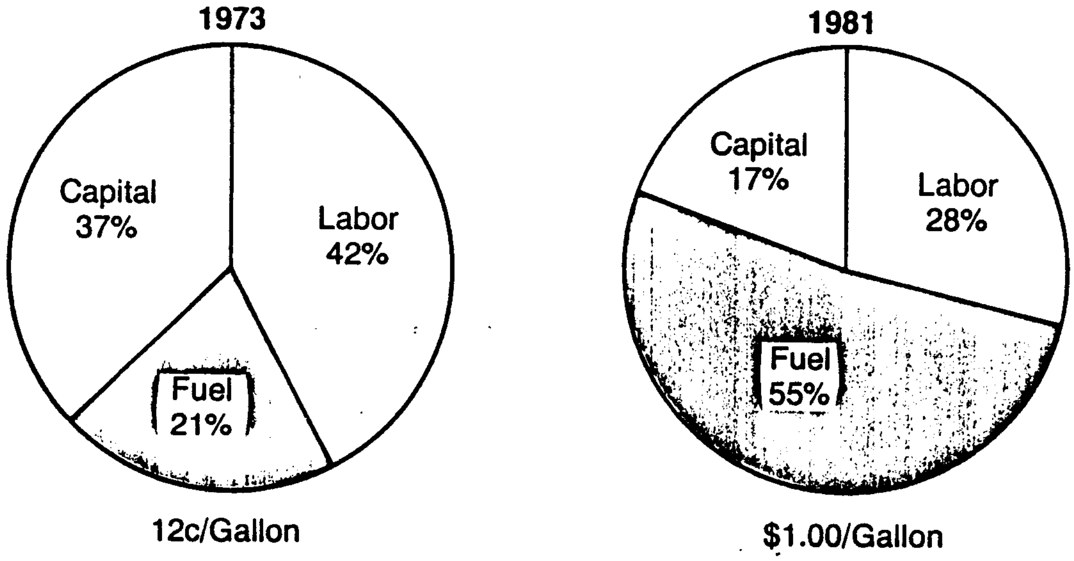
As shown, a long list of modern factors tending to decrease performance can be constructed. Among these factors are:

1. Fuel cost increases
2. Noise standards
3. Air traffic control limitations
4. Increased corrosion protection requirements
5. More stringent certification standards
6. More demanding customer guarantees
7. Safety

Let us examine several of these factors in more detail. The first two (dramatically rising fuel costs and noise) are the reasons for the 757 to exist. If it were not for these two factors alone, there would be little reason to replace the well-proven, economical (at purchase) 727.

Fuel Price History

Impact on DOC, 1000 nmi Trip



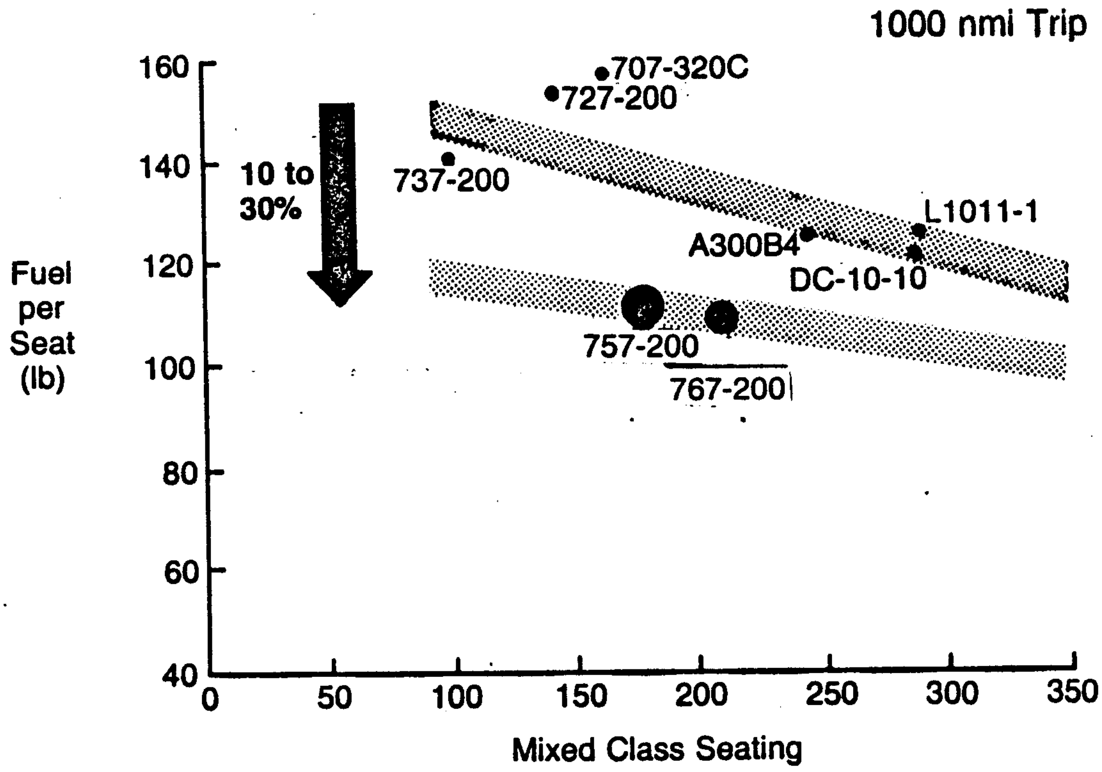
1% Drag = 33,000 Gallons/Year } Per Airplane
1% OEW = 26,000 Gallons/Year }

<u>Fuel Costs</u>

Rising fuel costs have been a topic of numerous discussions. This factor has been, by far, the largest influence of any design parameter in the 757 effort. The effect of rising fuel costs will continue to increase, and this influence on the 757 is clearly demonstrated by sensitivity analyses which show that:

1% drag = 33,000 gallons fuel burned per aircraft per year
1% OEW = 26,000 gallons fuel burned per aircraft per year
in normal anticipated aircraft utilization

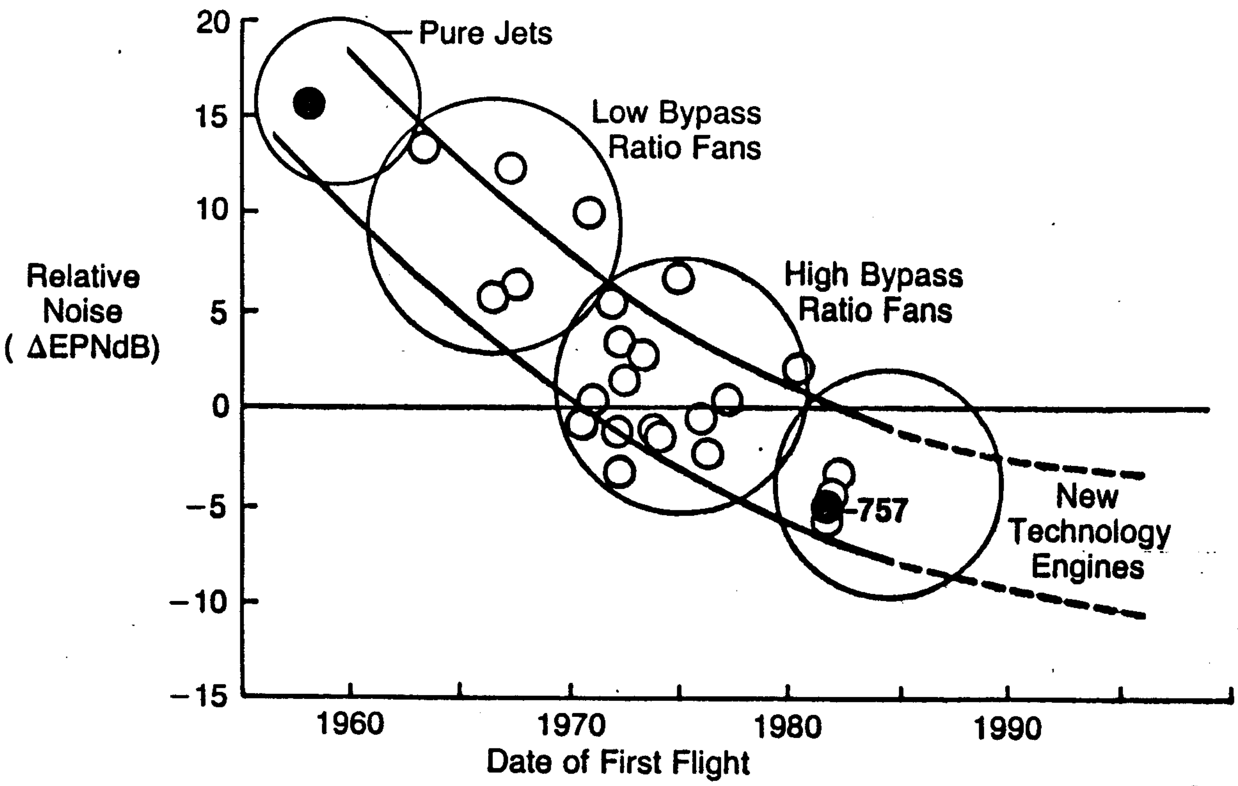
Block Fuel Comparison



⑦ RE BLUE 7

The 757 will use appreciably less fuel than its predecessor (10 - 30% less). While about two-thirds of this improvement comes from the use of new technology engines, the remaining savings come from the accumulation of refinements and advances made across the whole technology spectrum incorporated in the airplane.

Noise Reduction Trends



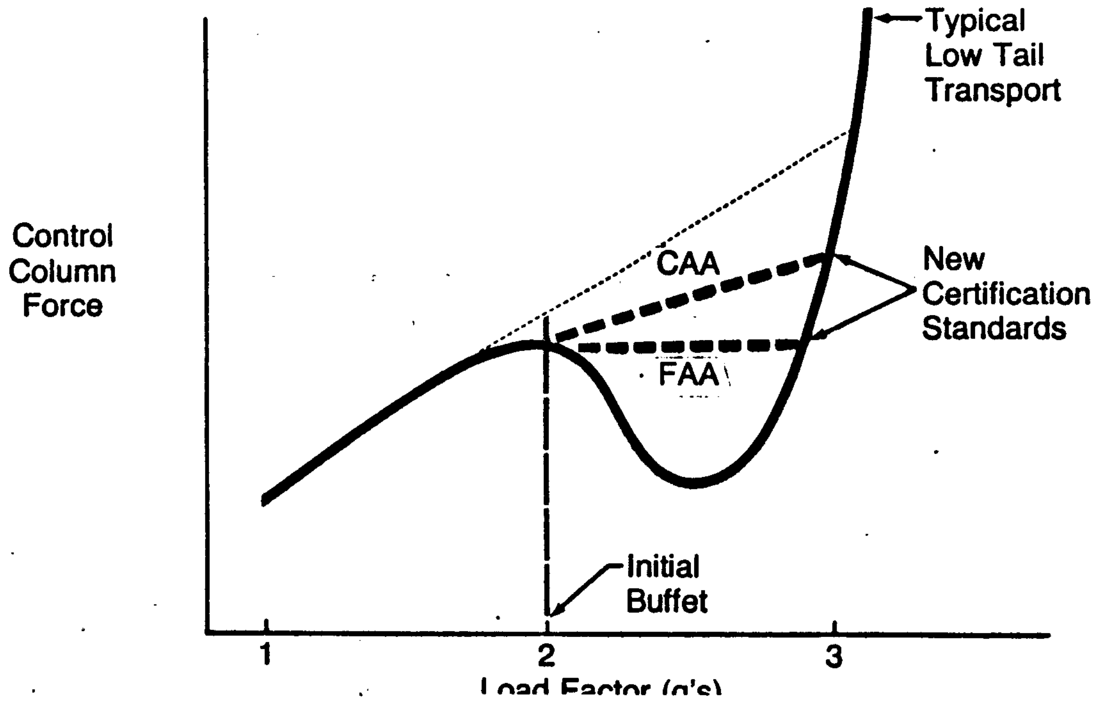
<u>Noise</u>

Reduced noise has been the second dominant parameter in the 757 design problem. It can be argued that if pure jets had remained the sole turbine option in commercial aviation there would likely have been a public rebellion. The introduction of low bypass ratio fan engines helped by reducing take-off noise, but brought them the problem of approach whine. It is ironic that Federal Aviation Administration activities directed at certifying noise, which irked Boeing initially, have probably helped us in the long run. Noise has now become a competitive issue. Economic penalties are now imposed for non-compliance at several airports such as Orange County (California) and Tokyo. Of interest here is the fact that this approach (economic penalty rather than FAA regulation) was the approach advocated by The Boeing Company from the outset.

The 757 will be markedly quieter than its predecessors. It is noteworthy here to observe that a 3dB reduction represents a halving of the noise energy.

Certification Standards

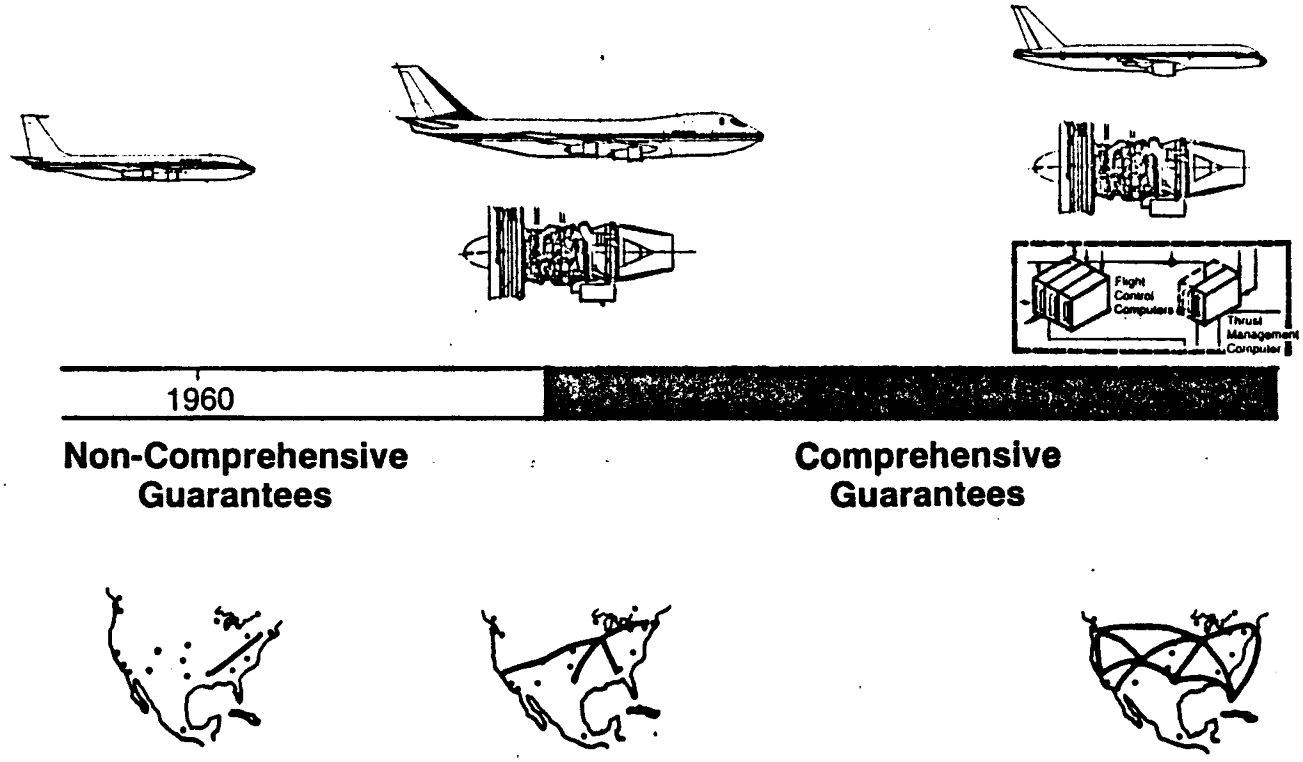
Cruise Mach Number



Certification Standards

As an example of increasingly stringent certification standards, the example of maneuver stability criteria can be cited. Essentially all low tail transport aircraft show reduced stability above initial buffet on-set. In the past this has never created a problem. Today's certification standards are far more severe, however. The U.S. FAA now specifies that control column force shall not decrease following initial buffet onset, while the British CAA suggests an even more stringent requirement in that the control column force must maintain at least half its pre-buffet slope at all post-buffet onset values of load factor. To meet these new requirements artificial stability is required to avoid the need for a very large horizontal tail.

Guarantees



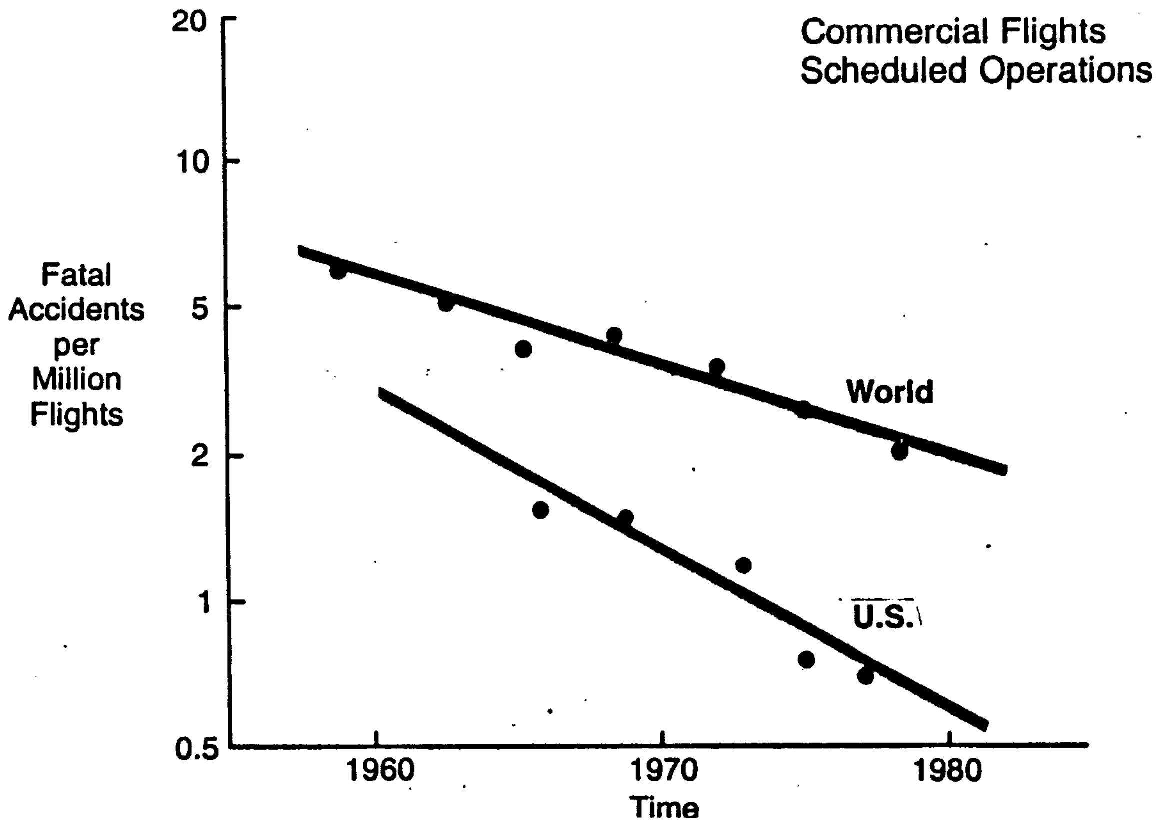
GUARANTEES

While on the one hand government certification standards have become more stringent, customer guarantees requirements have also increased. These new guarantees reflect both an increasing sophistication on the part of the whole range of customer airlines and an increasing concern for optimum performance (again driven by fuel consumption considerations) over a given route structure.

In the early days of turbine transports (707 vintage) guarantees on speed, field length and fuel mileage in general terms were all that were required. The customer demanded of Boeing only guarantees on the airframe, and engine performance was solely the responsibility of the engine manufacturer. With the advent of the 747, comprehensive mission guarantees, including engine performance, became a Boeing responsibility.

Today the guarantee situation is even more severe. Not only must the airframe manufacturer make comprehensive guarantees covering airframe and engine performance, but the comprehensive nature of the guarantees has been extended to the elaborate electronic equipment included in such items as the Flight Management System (FMS). In addition more specific guarantees over various routes must be made, as well as guarantees on compatibility with specific airports (e.g., La Guardia limits on weight and taxi speed).

Flight Safety



<u>Safety</u>

Since the advent of jet transportation in the late 1950's, there has been a very dramatic improvement in safety associated with scheduled airline operations, both world wide and domestically. In the U. S., this improvement has been an almost ten-fold reduction in fatal accidents per million scheduled flights.

In order to maintain this progress, it has become necessary to consider very many more "what ifs?" The challenge this presents is formidible in its own right.

Today's Situation

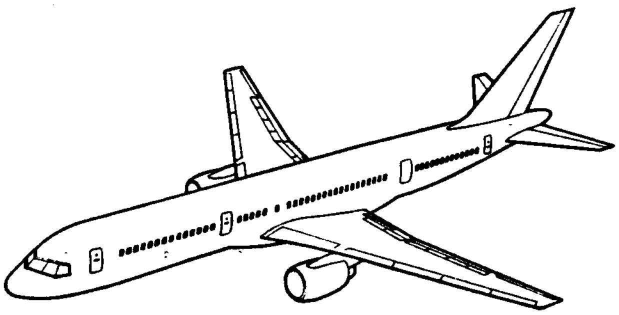
- **Work harder for smaller improvements**
  - Mature Technology
  - More complex environment
  - More non-technical constraints
  - Larger matrix of possibilities
- **Research allows advances with assurance**

Introductory Summary

Taken together, the modern constraints previously enumerated show that as far as overall performance is concerned, we have to work harder merely to stand still. We must thus compensate for the many external influences and respond to a very complex environment. The matrix of possibilities is very large and can in many cases be resolved only by use of analysis methods - repeated testing have become too costly and time consuming. What the sustained, company funded research program has allowed us to do is proceed with assurance in the face of the present challenge.

To clarify this ascertion, let us examine some specific applications of technology research to the example of the 757 (recognizing that many of the examples cited apply with equal validity to the 767). For convenience of presentation these examples will be grouped by technical discipline.

Aerodynamic Overview



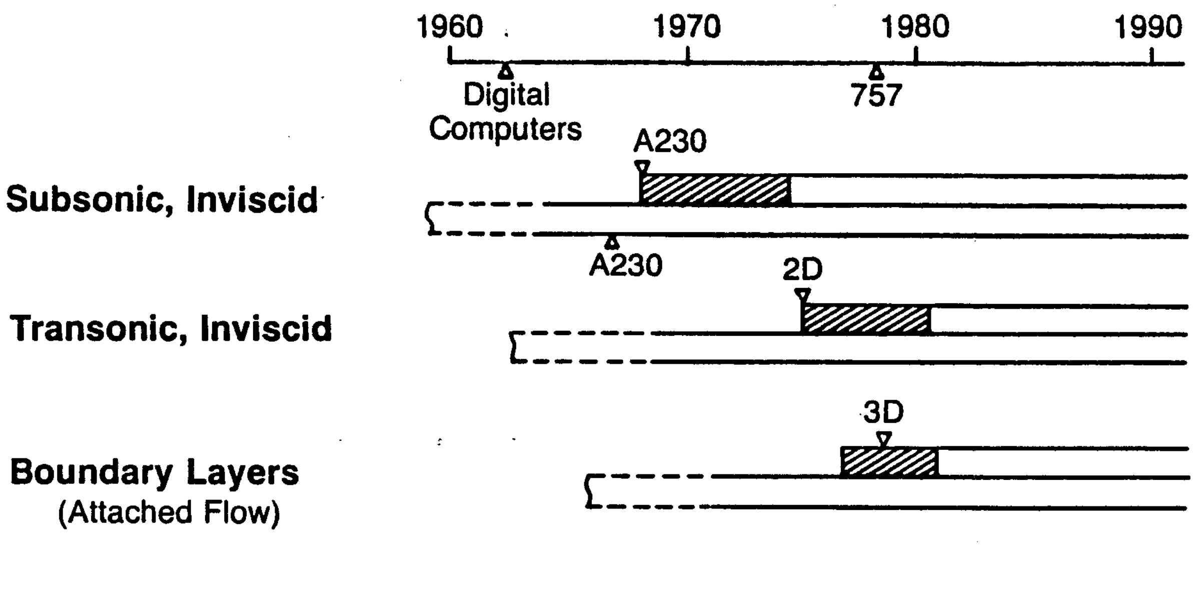
- Advanced Technology Airfoils and Wings
- Improved Aerodynamic Tailoring
- Excrescence Drag Reduction

12.1

AERODYNAMIC OVERVIEW

Aerodynamics are fundamental to airplane performance. In this overview presentation the topic discussed will be limited to the high-lights listed.

Computational Aerodynamics



Emerging Capabilities:
- Shock/Boundary Layer Interactions
- Separated Flows

<u>Computational Methods</u>

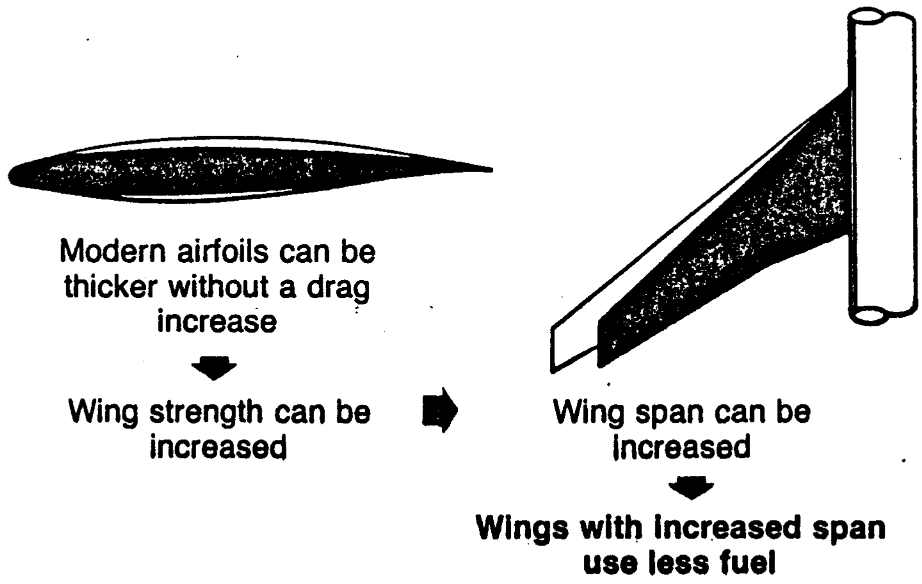
The use of computational methods in aerodynamic analysis and design has undergone something of a renaissance during the past two decades due to the advent of high-speed digital computers. The computer has made possible the solution of problems of great importance, but of such complexity that manual (slide rule & desk calculator) solutions were previously impractical.

The availability of the computer has lead to long-term efforts to develop programs for the solutions of potential, compressible and boundary layer flows. In general, there is a long period of time between research development of the tools and full acceptance of them by project groups. Usually confidence in the methods is built by iterative use in conjunction with traditional testing techniques. An example of this is the aero program A230 (subsonic potential flow simulation of full airplane configurations) which was developed in the research unit in 1966 and then underwent an extended "trial period" in various project groups. By 1975 the program was well enough accepted to allow its routine use in the development of important aspects of the 757. This pattern may be considered typical.

At the present time more of these computational tools are available; and they are becoming increasingly useful in providing guidance for wind tunnel testing. In addition, even more sophisticated methods (e.g., shock/boundary layer interaction and separated flow modeling) are under development.

Advanced Technology Airfoils and Wings

Airfoil Development



1.1.1

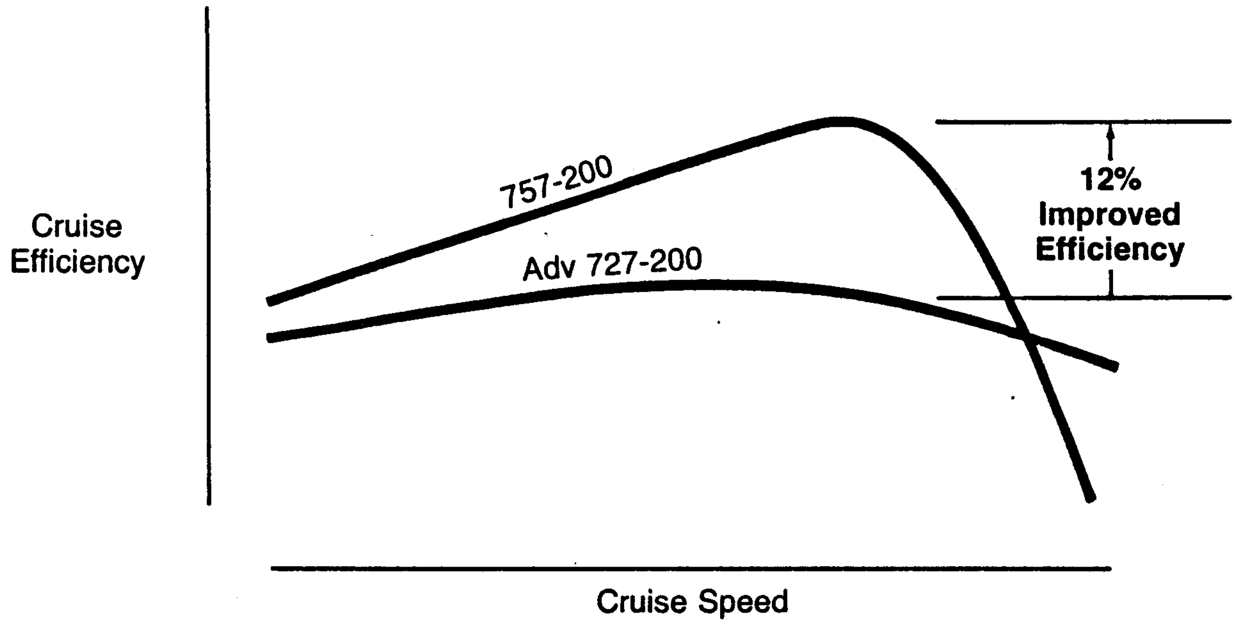
Wind Development

An important area of research advance has been the development of new transonic airfoil sections. These sections produce weaker shock waves (and hence lower drag) when employed on wings of given sweep and operating speed as compared to sections previously available.

In the 757 design effort there has been no desire to increase cruise speed performance. Thus this technology has allowed development of wings of greater thickness-chord ratio and reduced sweep compared to aircraft of earlier vintage. These advances in turn have allowed development of a wing of increased span (without substantial weight penalty) which results in decreased induced drag and hence improved fuel economy and reduced noise levels. In addition, it has been possible to reduce the weight and complexity of the wing by elimination of the inboard aileron.

Advanced Technology Airfoils and Wings

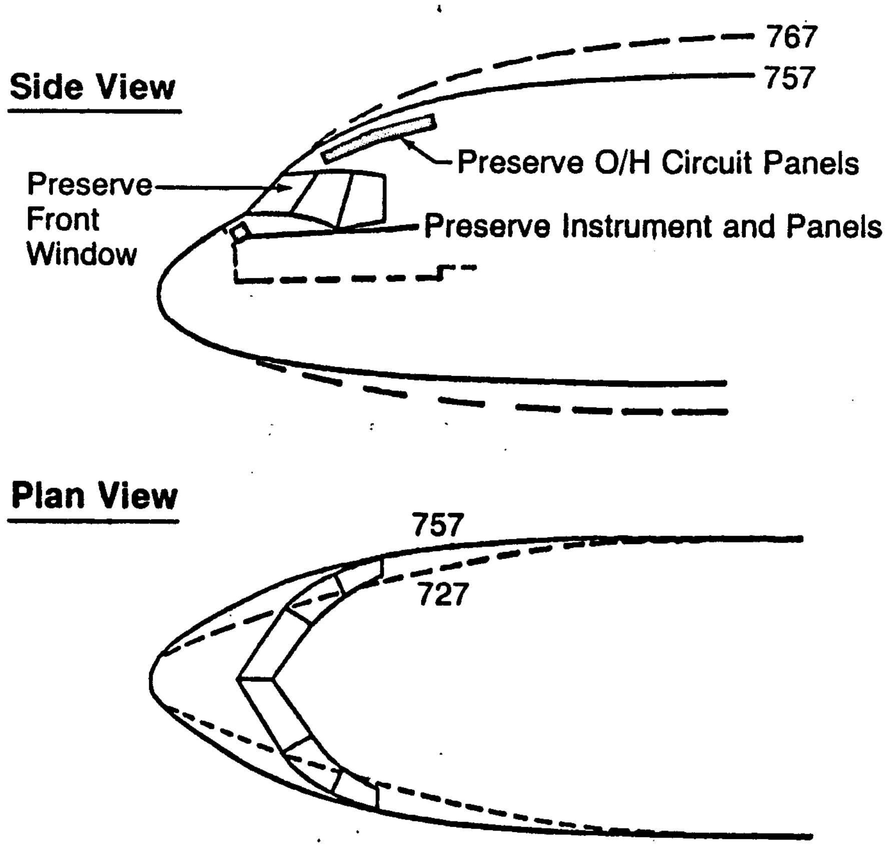
Aerodynamic Efficiency



15.1

This new generation wing is aimed more nearly at optimum performance at a specific cruise Mach number, and is less tolerant to speed variations than wings of previous similar aircraft. The result, however, has been a net 12% improvement in performance (as measured by cruise efficiency) compared to the present 727. In addition the 757 will have be a better buffet boundary. Important portions of the 727 high altitude flight envelope are limited by buffet, while due to the improved wing design, the 757 is limited mainly by cabin pressure considerations.

757 Cab Development

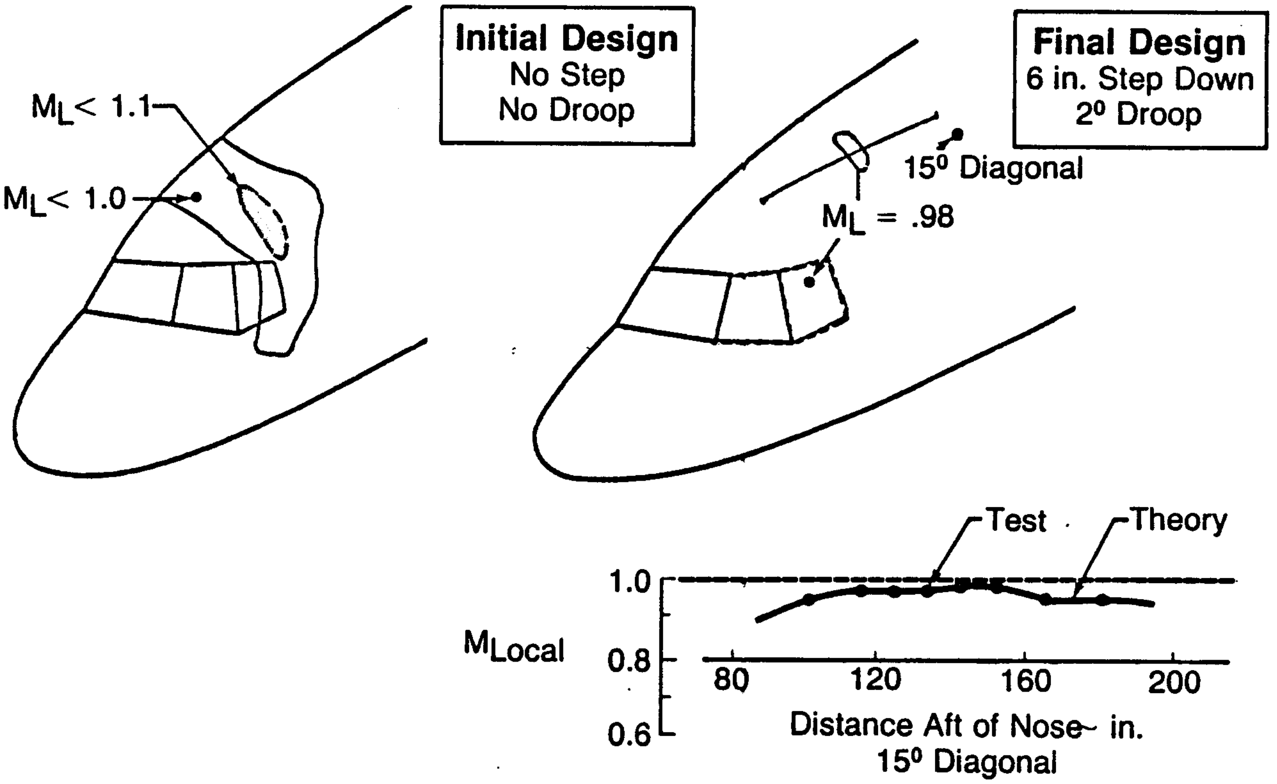


Aerodynamic Tailoring

The use of research developed computational methods has resulted in important advances in the area of aerodynamic contour refinements. A prime example of this new capability is in the cab development on the 757. The original cab was to have been similar to that of the 727, but it soon became desirable to achieve a high degree of commonality in the flight decks of the 757 and 767. This meant maintaining a common location of cockpit panels and a common windshield, while fitting the resulting cab to a narrow body fuselage, and elimination of the noisy corners on the 727 windshield.

757 Cab Development

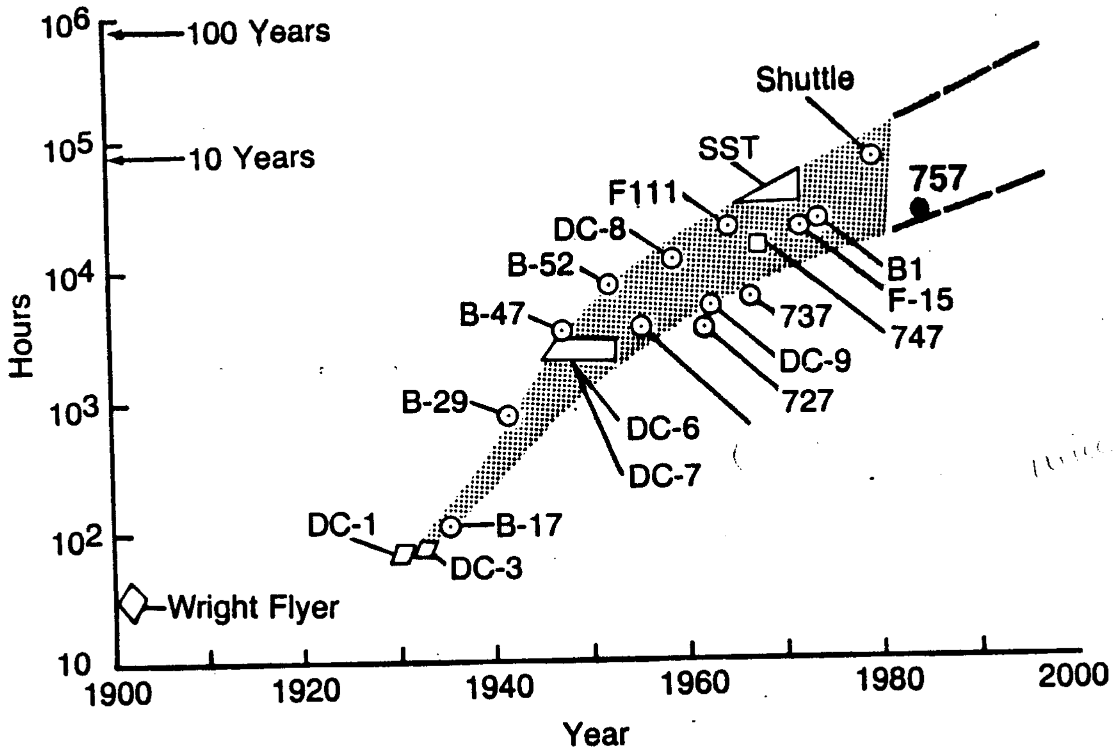
M = 0.8



17.1

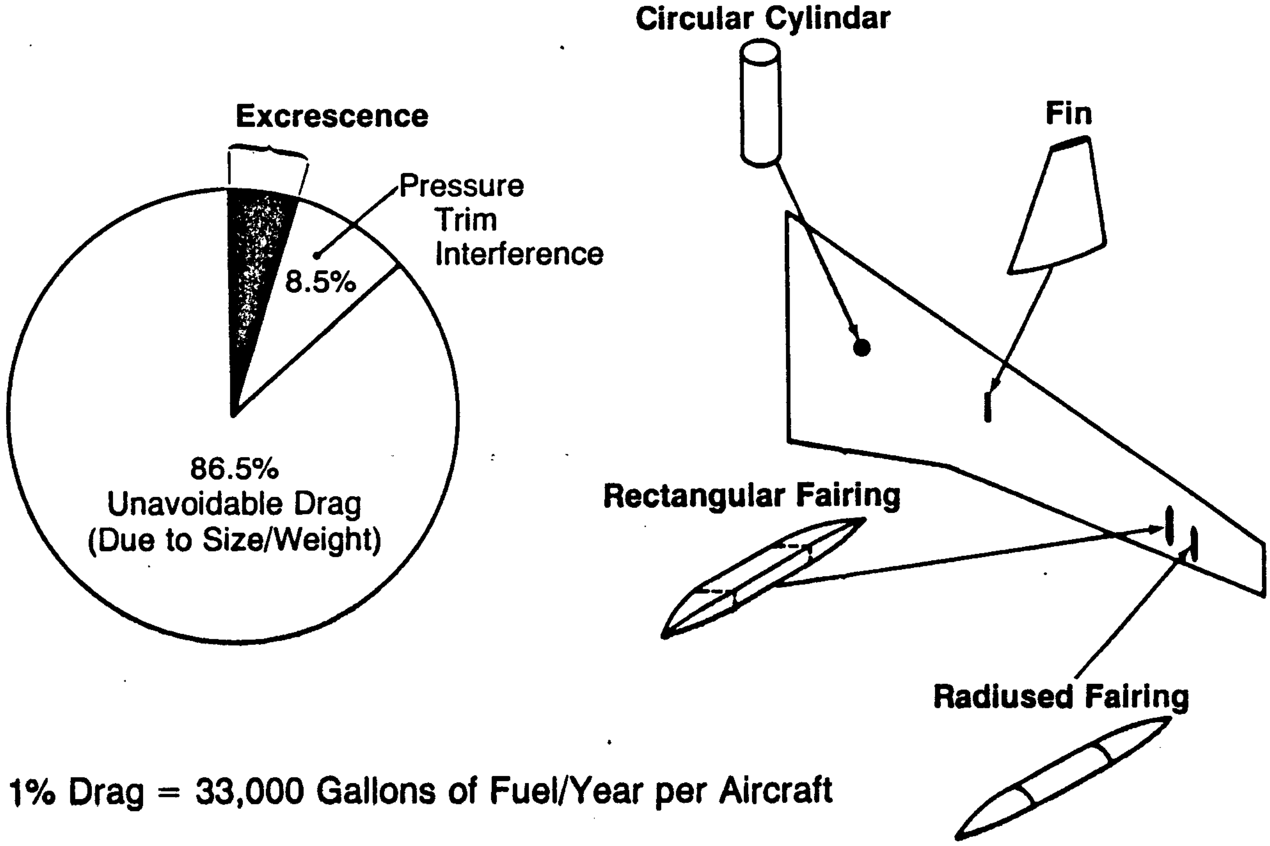
Analysis of the initial version of this revised cab showed a substantial region of supersonic flow near the crew area and a possible drag and cockpit noise problem. Based on computational results alone, the decision was made to incorporate a step-down and drooped cab on which the region of supersonic flow was eliminated. Later wind tunnel tests verified that the computational predictions were correct, as shown in a test-theory comparison of the configuration.

Wind Tunnel Test Trend



Despite such advances in computational aerodynamic capabilities, however, a great deal of development work continues to be performed in the wind tunnel. In fact the increasing difficulty of the design problem has increased the amount of testing required compared with past generations of Boeing transport aircraft. By the time the 757 makes its first flight about 18,000 hours of wind tunnel testing will have been conducted. Until even more advanced computation methods are fully developed and accepted in the project environment, this trend in wind tunnel testing time required cannot be expected to dramatically change.

Excrescence Drag



1% Drag = 33,000 Gallons of Fuel/Year per Aircraft

14.1

EXCRESCENCE DRAG

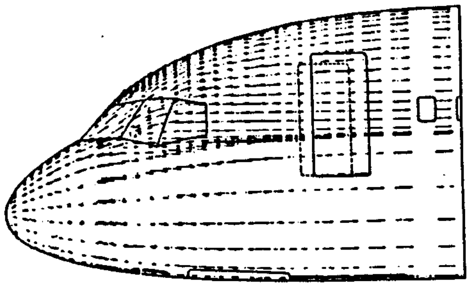
Excrescence drag is that component of the total airplane drag associated with surface roughness, leaks, skin joints, antennas, and the other "hairs and warts" on the surface of a real full scale airplane. Traditionally, excrescence drag has accounted for about 6 percent of total transport aircraft drag.

The target goal for excrescence drag on the 757 is 5 percent of total airplane drag. The importance of this "small fraction" can be seen again with reference to the drag sensitivity value:

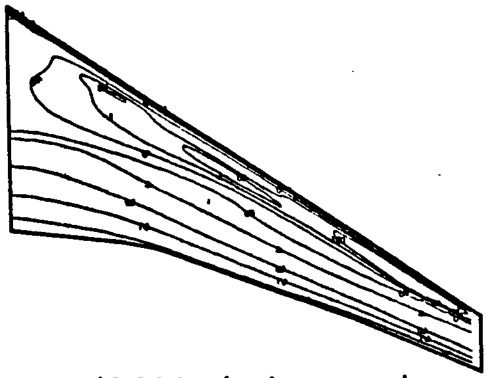
1 percent drag = 33,000 gallons of fuel per year per aircraft in normal anticipated service.

NOTE: On an aircraft like the 757 about 86.5 percent of the total cruise drag is an unavoidable consequence of the aircraft size and weight in the form of viscous skin friction drag on even a "perfect" surface and drag associated with the production of lift - i.e., induced drag. Another 8.5 percent is associated with pressure, trim and interference drag, components which can be controlled in some measure by combined application of computational method - e.g., the tailoring example previously cited - and extensive wind tunnel testing.

Excrescence Drag Control



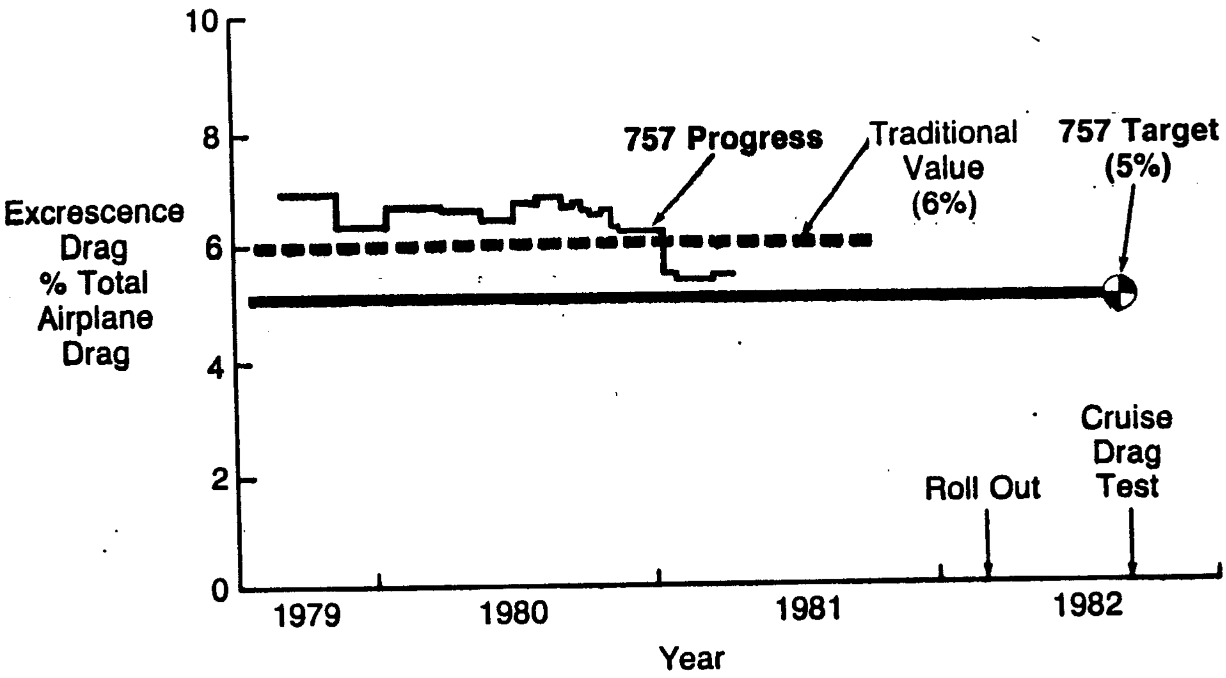
A230 Streamlines



A488/A411 Isobars and Boundary Layer Characteristics

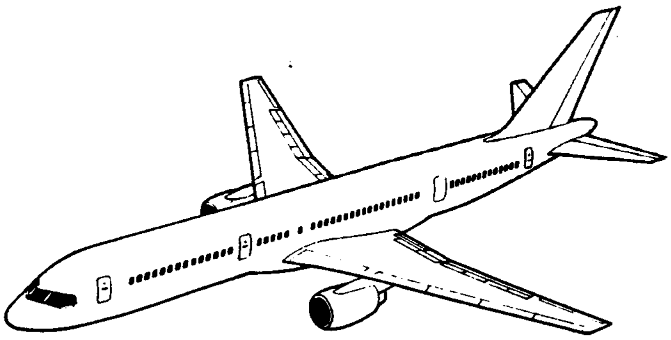
In the area of excrescence drag control, research developed computational analysis tools have proved of very significant value. Computer programs such as A230 can provide detailed information on surface streamline patterns and pressure distributions - detailed data which would be prohibitively expensive to obtain by wind tunnel testing techniques. Newer computer programs such as A488 (which incorporates program A411 for three-dimensional viscous flow analysis) can provide compariable maps of pressure distributions and boundary layer characteristics in a transonic flow. Such detailed information can then be used to place excrescence drag producing items such as antennas in low velocity areas of the flow, or allow minimum drag alignments with the local air flow. The viscous flow analyses can guide the contouring of local, critical areas of the skin surface and aid in assessing the penalties of variously roughened surfaces.

Excrescence Drag Control



Evaluation of the 757 excrescence drag control program shows that the project is nearing its 5 percent target goal. The present estimate is that the final target will be achieved if the Corogard paint smoothness problem can be solved.

Flight Controls Overview



- Digital Flight Controls and Augmentation Systems
- Control Systems and Actuators
- Parameter Identification – Simulators

FLIGHT CONTROLS OVERVIEW

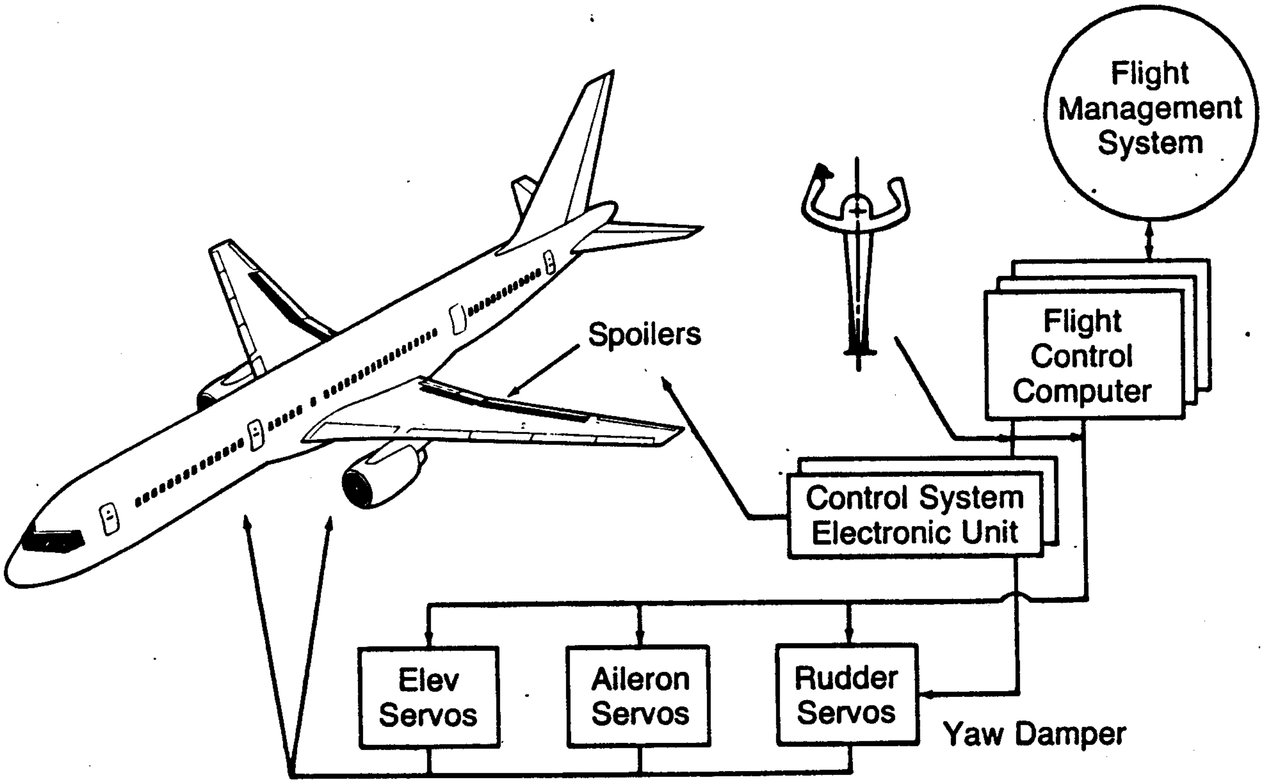
Flight controls have become increasingly sophisticated in recent designs. Again, driven by requirements for improved fuel efficiency, there have been efforts directed at reducing tail sizes and provision of adequate stability/control characteristics with farther aft center-of-gravity locations.*

In this overview presentation discussion will be limited to three areas:

- o Digital Flight Controls
- o Yaw Dampers
- o Parameter Identification & Simulators

*NOTE: The aft c.g. position of the 757 is 39 percent MAC, as compared with a more traditional value of about 35 percent MAC.

Digital Flight Controls



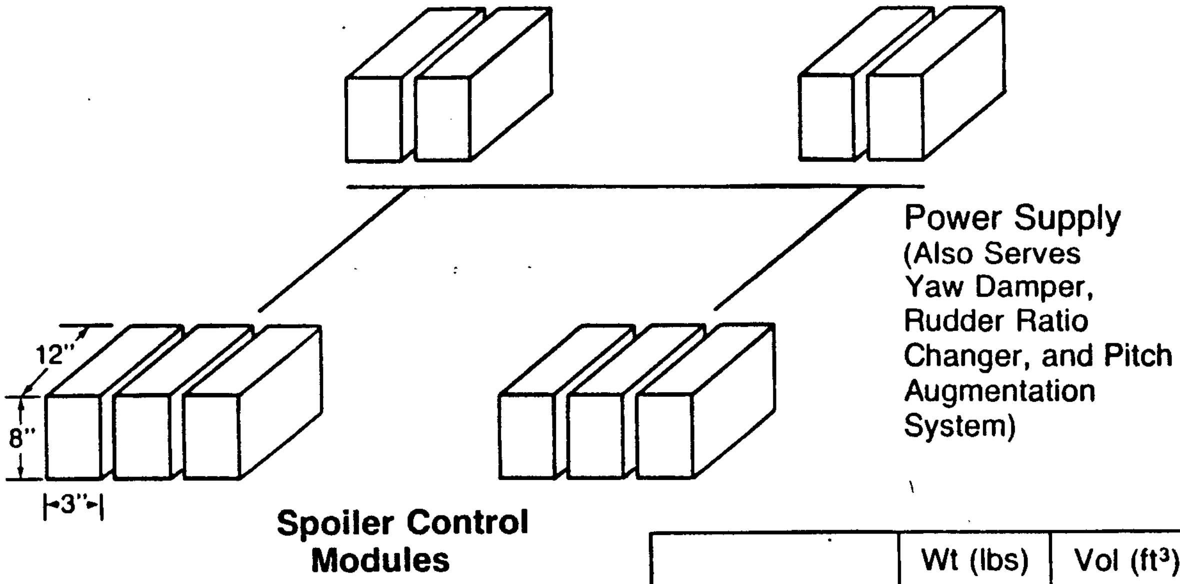
Digital Flight Controls

This aspect of the airplane system has usually been associated with autopilots. Today, however, we have a far more elaborate and sophisticated Flight Management System (FMS). Input from the overall FMS is an integral part of the 757 digital flight controls system.

As an example of an advance in the 757 flight control system, made possible by digital systems, the direct control of the spoilers is through the Control System Electronic Unit (CSEU) - with no direct manual backup link. To see why this new system represents an advance, one may contrast a previous generation mechanical system with the present digital system.

This picture shows the mechanism by which the spoilers - used for speedbrakes and lateral control - were controlled on the 747. The basic device, full of springs, pushrods, cams and cables, measured almost 5 feet across and occupied a volume of nearly 11 cubic feet. The weight was about 160 pounds.

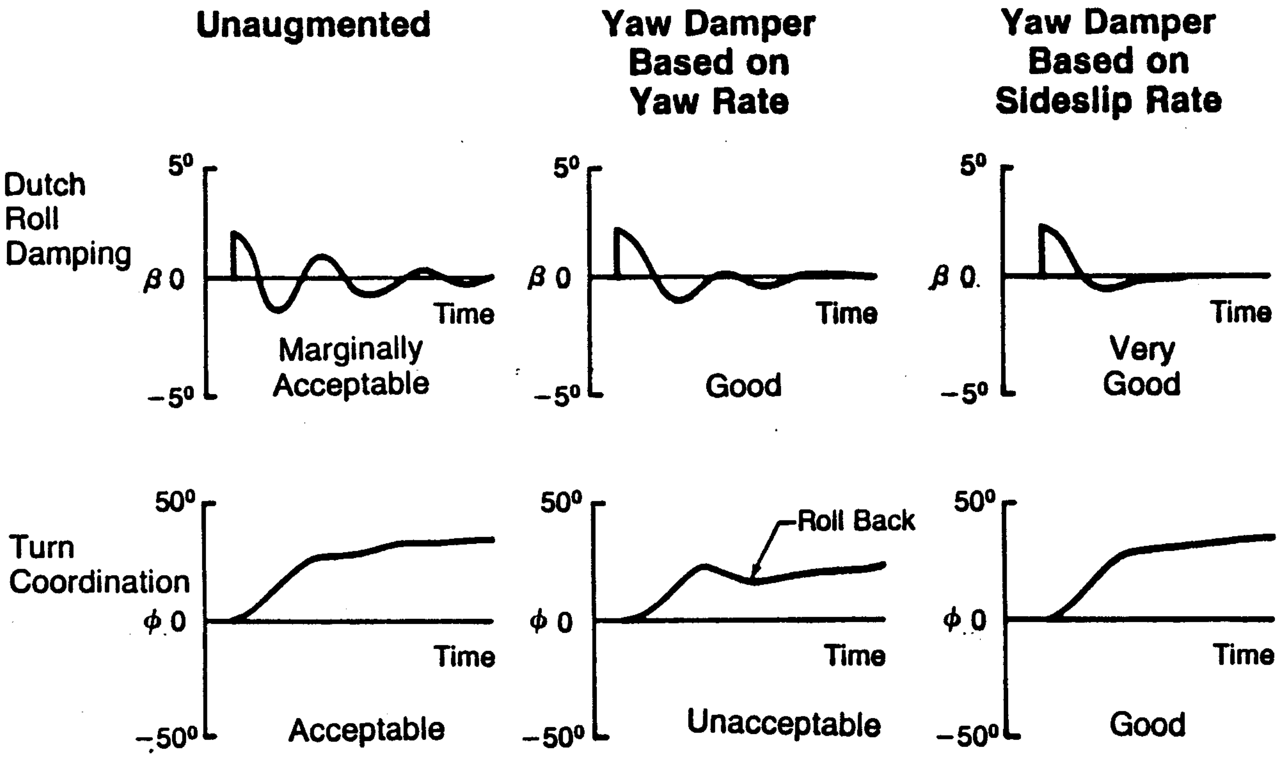
Digital Flight Controls



	Wt (lbs)	Vol (ft^3)
Spoiler Modules	46	1.0
Power Supplies	44	0.7
Total	90	1.7

This figure shows the system which controls the spoilers on the 757. In this system electric signals control hydraulic valves, which in turn move the spoilers. The advantages of the new system are its versatility and the relative ease with which spoiler deflection scheduling can be changed in light of flight test experience. As shown, the total volume of the new system (including the power supply which also serves other portions of the digital flight control system) is less then 2 cubic feet and total weight (again including power supply) is about 90 pounds.

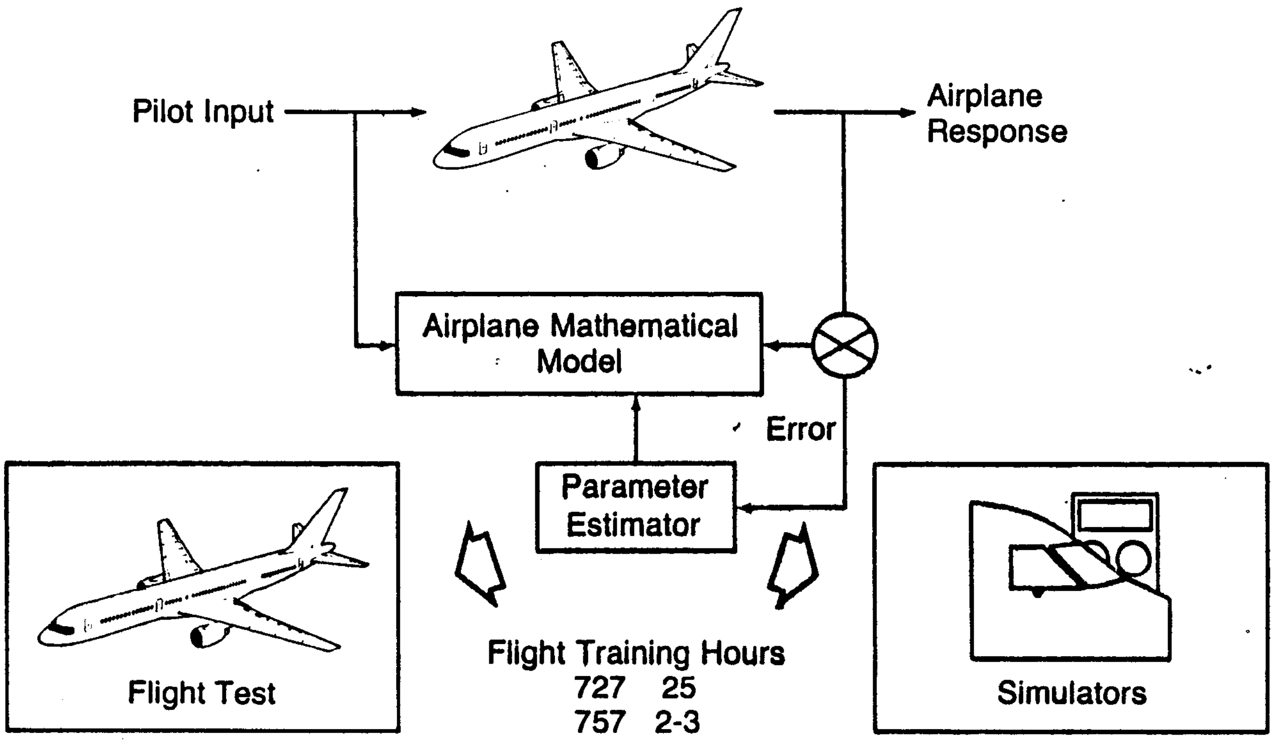
Yaw Damper – Landing



Yaw Damper

The yaw damper serves two functions: yaw damping and turn coordination. In the past, such systems have functioned on the basis of yaw rate. While such systems function well enough as yaw dampers, they are not good with respect to turn coordination. Late in the 747 program, a new system based on side slip rate was developed. The new approach resulted in a system which eliminated undesirable roll-back in low speed turns, and gave both good yaw damping and good turn coordination. This new system is the one employed on the 757.

Parameter Identification

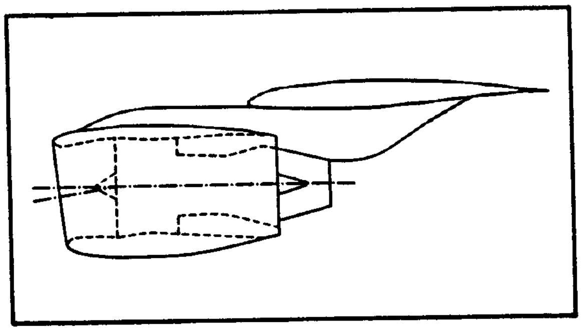


<u>Parameter Identification & Simulators</u>

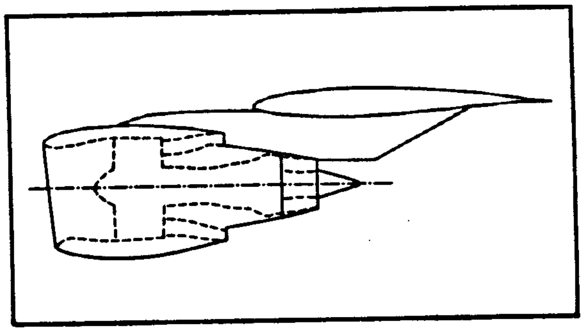
Simulators have been used for many years in crew training. Efforts have been made to make such simulators even more realistic in order to reduce the amount of increasingly expensive flight time in training. An important new technology which can be employed to both improve the diagnosis of flight test results and improve the simulator is called parameter identification.

In this technology, a mathematical model of the aircraft's dynamic behavior is generated from theory and wind tunnel test results. To be accurately representative of the actual airplane, flight test data is also required in the mathematical model. Using the technique of parameter identification, the measured motion of the real airplane is compared with the motion predicted by the wind tunnel/theory model and by application of sophisticated statistical methods an improved mathematical model can be generated. A by-product of the parameter identification is the clarification of magnitude of individual components of the forces and moments which cause a given motion. The end result is a powerful means of diagnosing in detail the results of flight testing and improving simulators. In turn flight training time can be reduced from the early 727 level of 25 hours, to a target of only 2-3 hours for the 757. Advances in simulator technology hold the promise of eventually eliminating the need for flight training in a new aircraft type prior to first revenue flight.

Propulsion Overview



Rolls-Royce
RB211-535C



Pratt & Whitney
PW2037

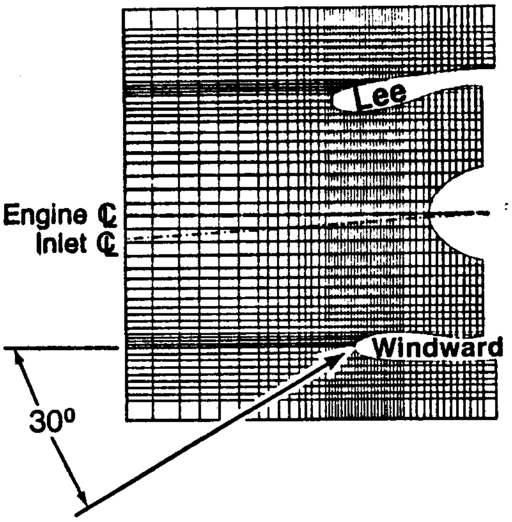
- Nacelle Contours
- Nacelle/Airframe Integration
  - Aerodynamic
  - Flutter
  - Noise
- Engine Performance Auditing

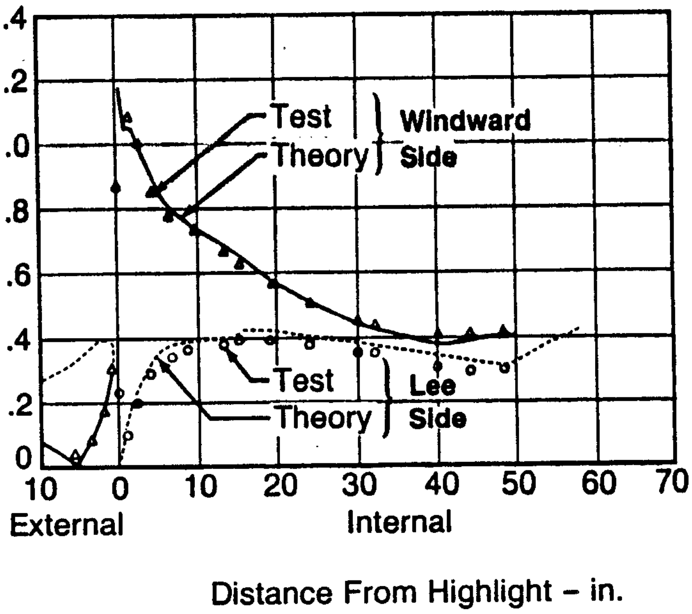
PROPULSION OVERVIEW

Three engines are currently planned for use on the 757. There are two versions of the Rolls Royce RB211 and, later, the Pratts Whitney 2037. The nacelles for these engines must be carefully integrated into the overall design. Several areas of technology involved in this integration will be discussed in this portion of the presentation including noise and structures as well as the usual problems of aerodynamics.

Nacelle Aerodynamic Analysis

M = 0.8



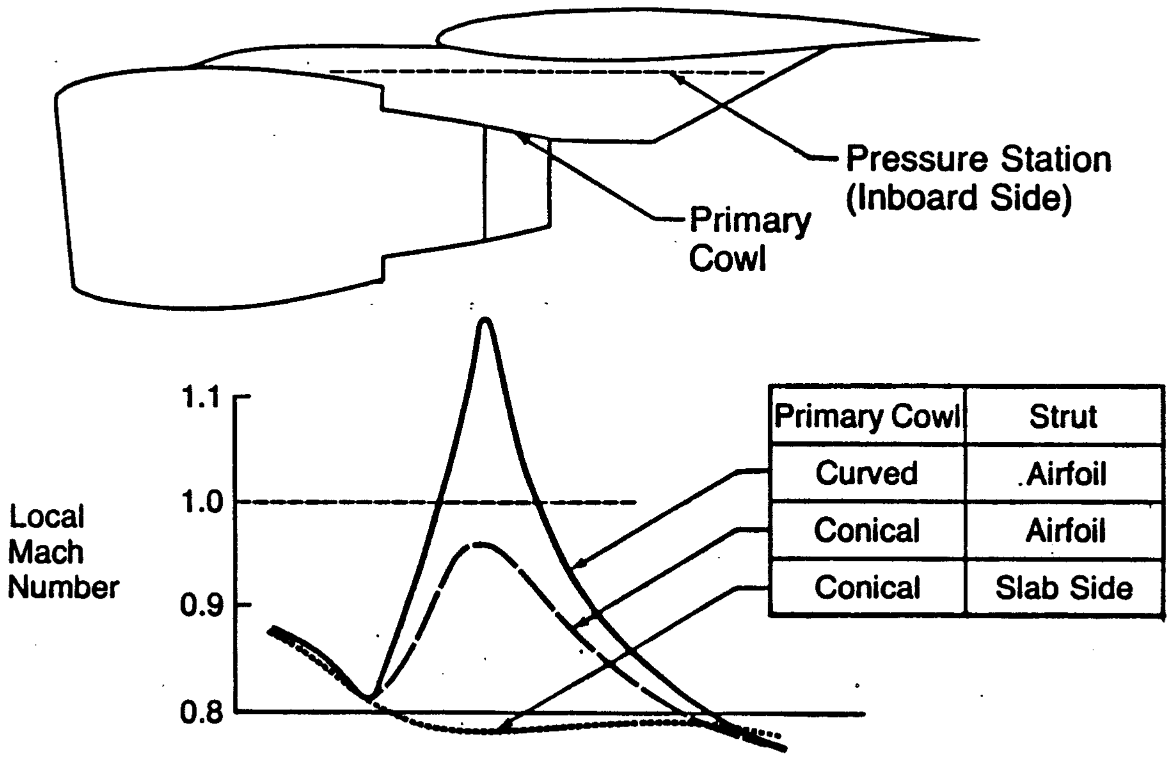


Isolated Nacelles

As in aerodynamics, the advent of the computer has made possible significant advances in our ability to analyze propulsion related problems. In the case shown, the excellent agreement between test and theory of pressures on the inlet of a nacelle is demonstrated. Today it appears that wind tunnel testing of isolated nacelles is no longer necessary in the development phase of a new program unless the geometry departs substantially from past practice or particularly severe operational conditions are anticipated.

Nacelle/Airframe Integration

Aerodynamic

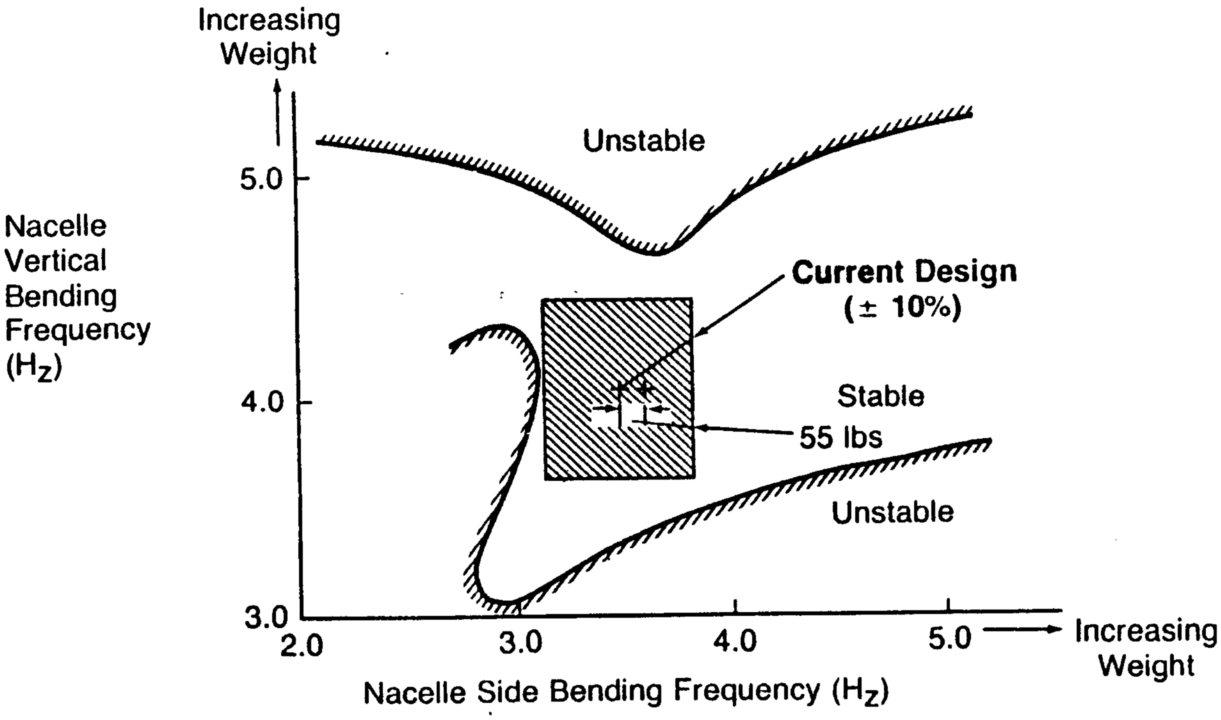


Primary Cowl	Strut
Curved	Airfoil
Conical	Airfoil
Conical	Slab Side

<u>Nacelle Aerodynamic Integration</u>

With the introduction of large high-bypass ratio turbofans, wing mounted nacelles must be more closely coupled to the wing surface due to ground clearance limitations. This leads to problems of high strut drag and possibly severe interference effects. Use of computational methods to guide wind tunnel testing can lead to contours which eliminate local shock waves, reduce drag and eliminate buffeting. This capability is of extreme importance since, for example, movement of the nacelle in relation to the wing can be very costly in weight. On the 757 this weight penalty can be as high as 100 pounds per inch of longitudinal nacelle movement.

Nacelle/Airframe Integration Flutter



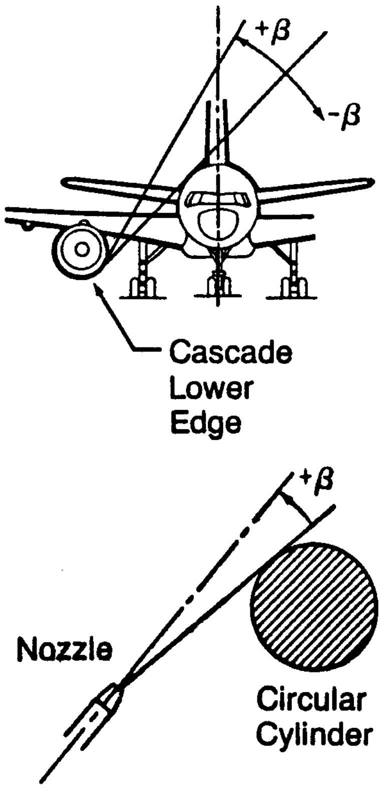
<u>Nacelle Structural Integration</u>

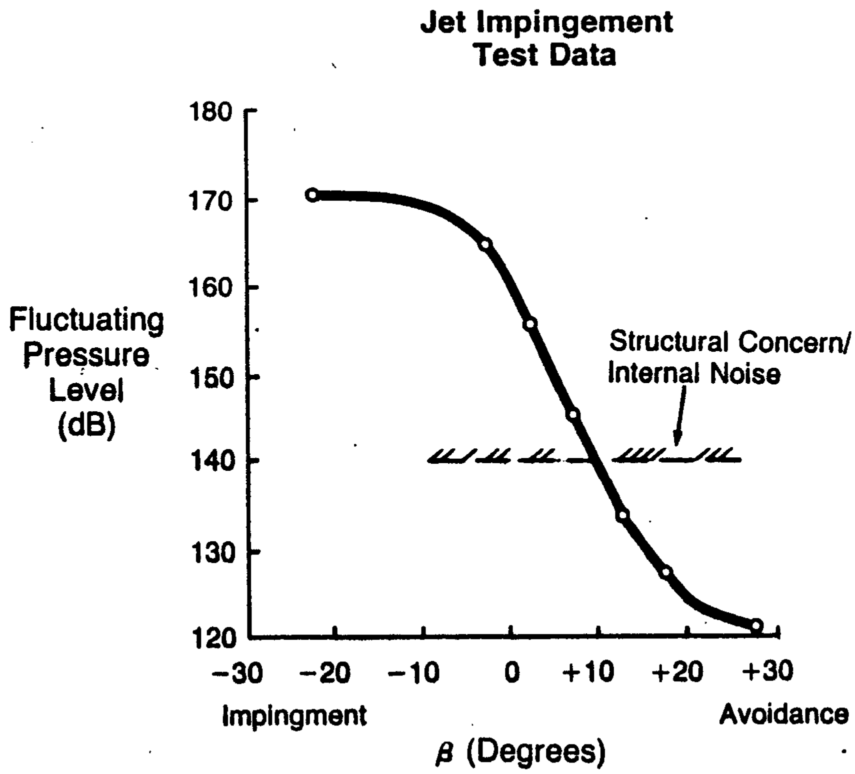
Present practice allows nacelles to be structurally located on a wing on the basis of calculations and the location is then verified by testing. As shown below there is a very narrow window between various flutter modes. In turn, providing unnecessary stiffness can be very costly. In the example shown, there is a 55 pound weight penalty associated with changing nacelle side bending frequency by an 0.1 Hertz (representing an increase of only 3 percent). Therefore very accurate prediction methods are required.

In addition, the engine picture can change rapidly. An example is the withdrawal by General Electric of its candidate 757 engine and the emergence of the Pratt & Whitney 2037. Customer guarantees for these new engine options must sometimes be made before tests can be conducted. Therefore generalized analysis methods of satisfactory accuracy are a necessity.

Nacelle/Airframe Integration

Noise



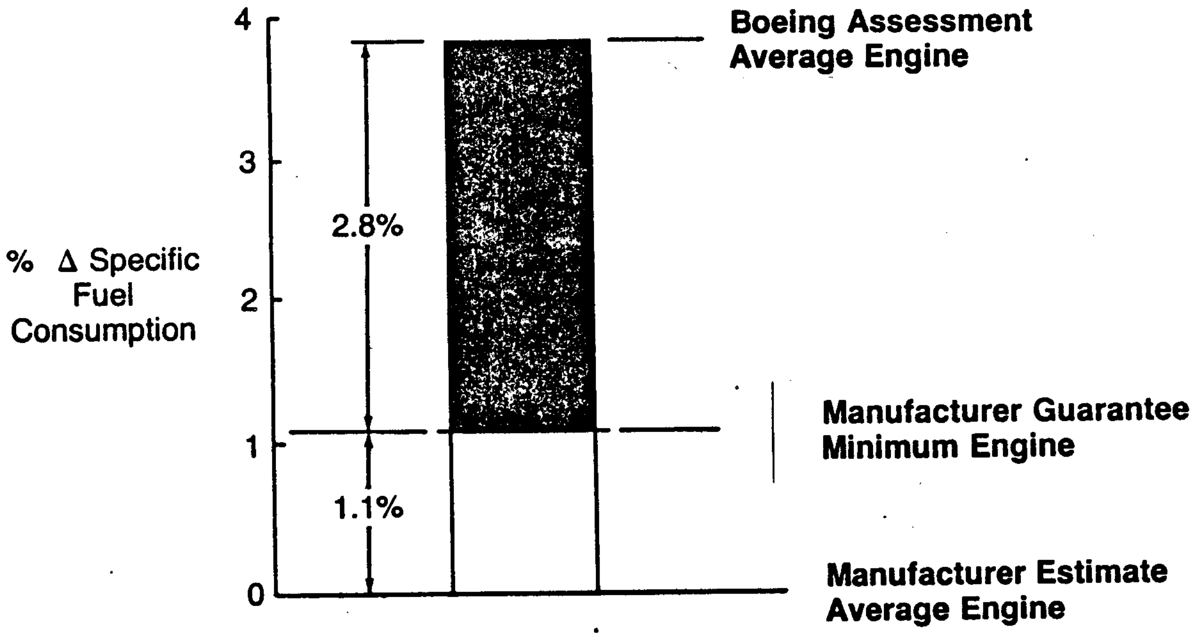


Nacelle Noise Integration

Another important area of nacelle/airframe integration involves noise, in this case that associated with thrust reversers. Thrust reversal on high-bypass ratio turbofans is a difficult problem. Thrust reverses are heavy and in order to maximize effectiveness, it would be desirable to exhaust around the full periphery of the engine. This is not possible, however, due to possible problems of thrust reverser produced bouyancy effects and jet impingement on the fuselage, which can cause structural problems and high passenger cabin noise levels.

To assess these latter problems a simple test was conducted, in this case by the project, rather than by research. It demonstrates, however, that project personnel with previous research experience, can effectively solve a difficult problem by intelligent use of their past experience. In this case a simple test was conducted by blowing a jet of air in proximity to a simple circular cylinder and measuring the fluctuating pressure level on the cylinder surface. These tests showed that the jet (representing the cascade reverser flow) had to miss the body in order to avoid unfavorable structural and noise effects. This information was then used to determine reverser cascade shape.

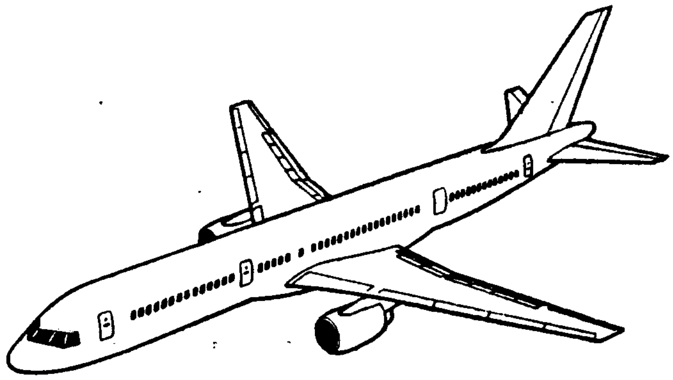
Engine Performance Audit
Installed TSFC Comparison



Engine Auditing

While Boeing is not in the engine business, the problem of making comprehensive customer guarantees has forced the company to develop, through the Propulsion Research Unit, a sophisticated in-depth capability to analyze engine performance, and thus assess engine manufacturer's performance claims. In many cases these engine company claims have been found to be optimistic - as we discovered on the 747 program, for example. Thus, a detailed in-house engine performance analysis capability has had to be developed in order to perform audits of manufacturer claims. The outcome has been the Boeing Company's ability to make its own best guess of performance for guarantee purposes, with a consequent substantial reduction in the risk that overall performance guarantees will be missed.

Structures Overview

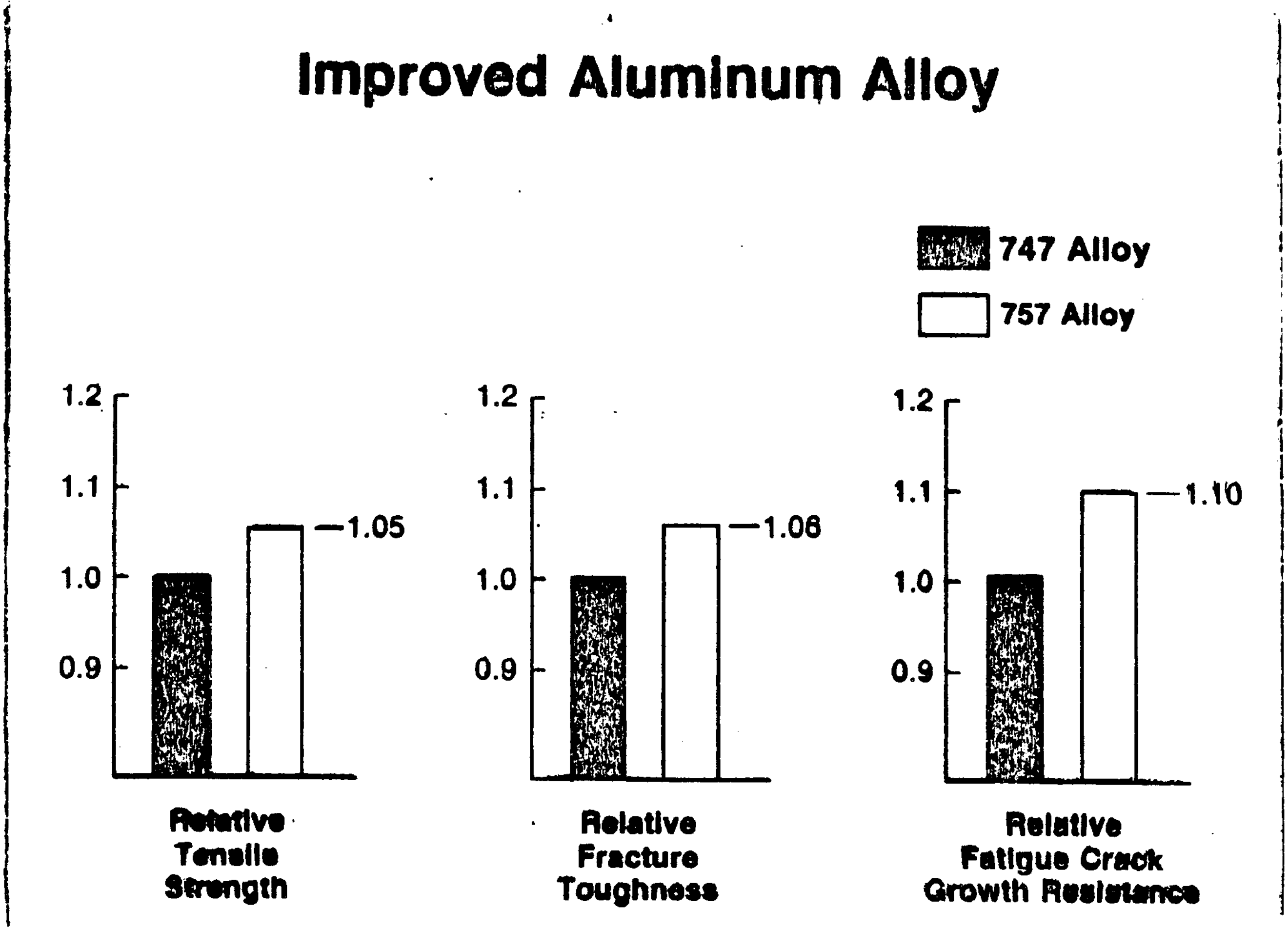


- Improved Materials
  - Aluminum Alloys
  - Composite Materials
  - Interior Materials
- Durability
- Damage Tolerance

STRUCTURES OVERVIEW

The advent of large digital computers has led to substantial advances in structural analysis capabilities. Finite element analysis methods and more complete flutter modeling have been among the benefits of increased computer availability. In the same time period there have been substantial advances in, both metals and in composite materials, technology.

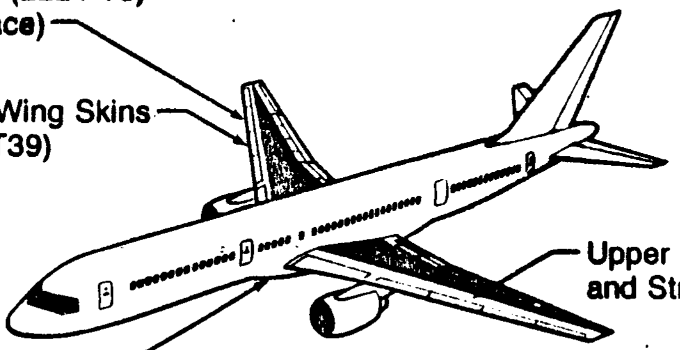
In this overview presentation, the discussion will touch on improved materials and improved structural durability and damage tolerance technology.



Improved Aluminum Alloy

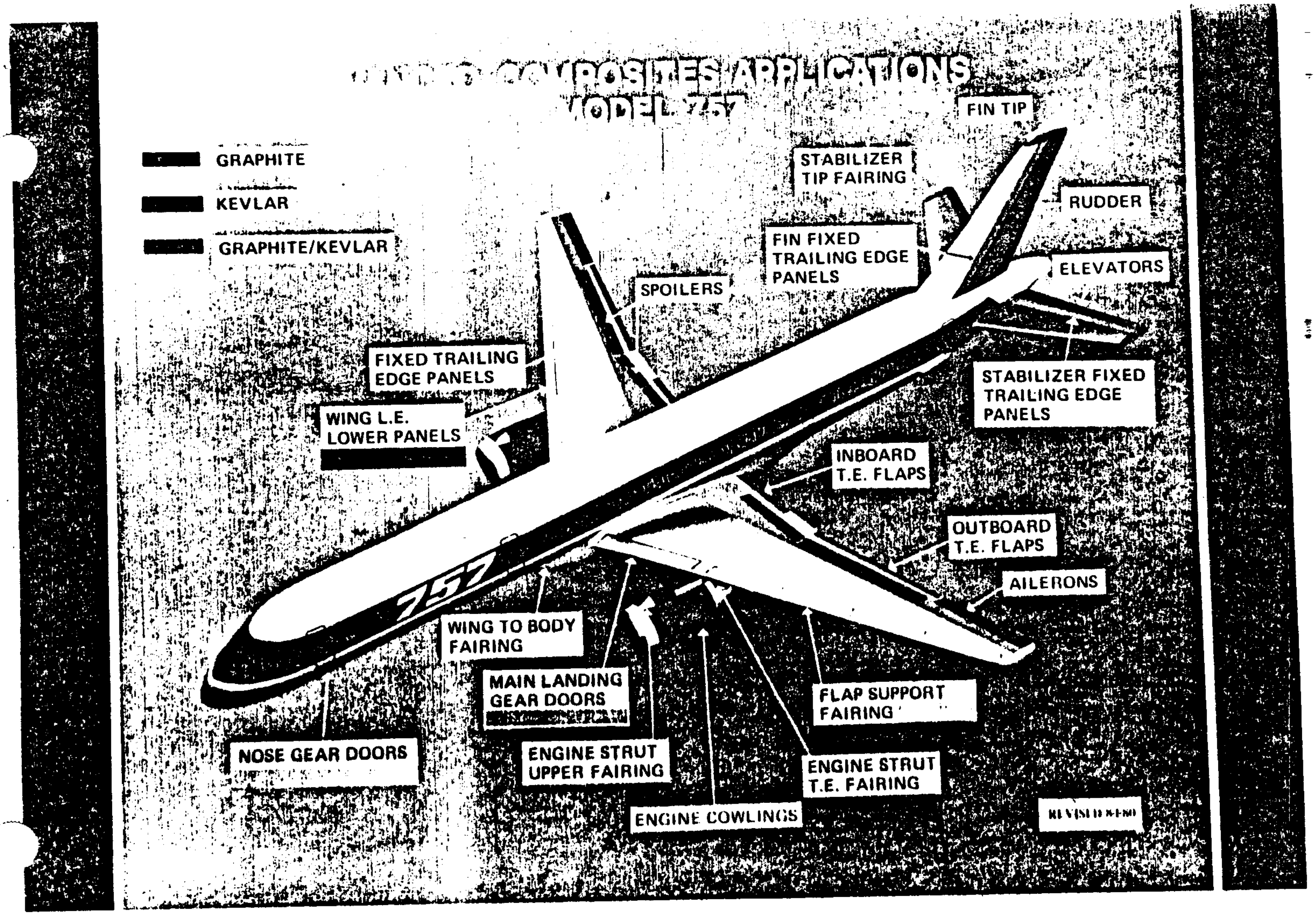
A long range research effort has shown that by purification of alloys both strength and toughness can be improved. This has resulted in development of new aluminum alloys with properties superior to those of 747 vintage alloys.

Improved Aluminum Usage



Pounds Used	Pounds Saved	Fuel Saved/Airplane per Year
11,380	610	12,200 Gallons

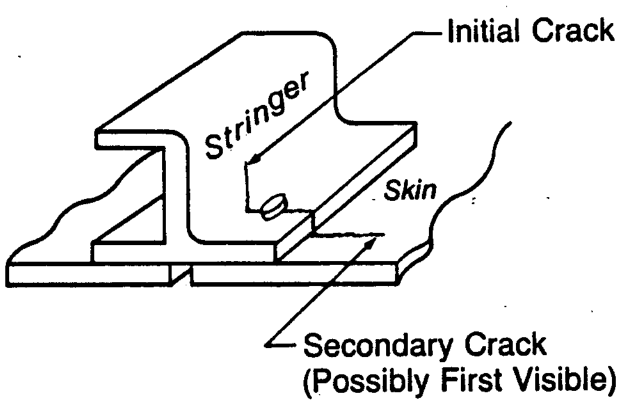
These new alloys have become standard and are used in major structural areas of both the 757 and 767, and in the case of the 757 contribute a 610 pound weight saving (equivalent to a saving of over 12,000 gallons of fuel per aircraft per year in normal anticipated airline usage).



Composite Materials

Boeing has used composite materials in the form of fiberglass for years. In the 757 and 767 programs, composite usage is being substantially extended to include use of new materials such as Kevlar and graphite for fairings, non-load carrying structure and control surfaces. The use of 3150 pounds of composite material in the 757 saves 1140 pounds of weight compared to the manufacture of these components from aluminum. In addition, manufacturing now prefers to use composites in many areas.

Damage Tolerance



Boeing Determines
- Crack Growth Rate
- Residual Strength
- Critical Crack Length
- Inspection Period

Airlines
- Perform Inspection

FAA
- Approves Boeing Data
- Assures Airlines Comply

Damage Tolerance

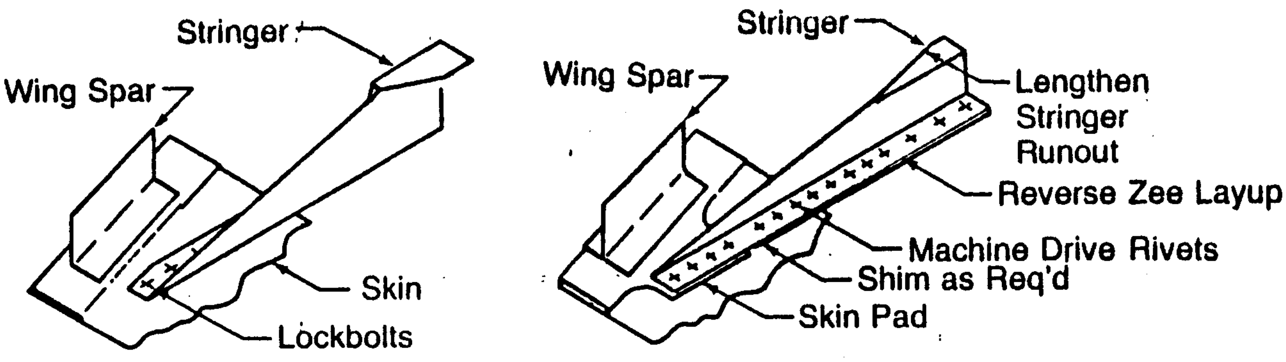
U.S. transport aircraft have generally been very damage tolerant as demonstrated by their excellent safety record over the past 25 years. Unfortunately, the few isolated failures have been very visible.

In consequence, the FAA is now demanding more substantiating data on damage tolerance capabilities and more involvement by manufacturers and the airlines, throughout the life of the airframe. At the same time, methodology has been developed which allows prediction of factors such as crack growth and residual strength. Thus inspection periods and methods of locating damage can be specified so that remedial action can be taken before the structure is no longer able to carry limit loads.

Boeing is in a position to work in partnership with the airlines, which must themselves perform the necessary inspections, and maintain safety throughout the service life of the airframe. In addition, Boeing provided data can be approved by the FAA and they can, in turn, assure that the airlines carry out required inspections.

Design For Durability

Wing Stringer Runout – Lower Surface



Inactive
DFR: <10

Acceptable
DFR: 19

Life Improvement Factor + 15

<u>Structural Durability</u>

Most military aircraft are relatively short lived and, with notable exceptions (e.g., the B-52 which has had a service life far beyond original expectations), fatigue has not been a major concern.

In early generation transports little account was taken of fatigue. Fatigue testing came late, and in some cases expensive changes and retrofits were necessary, (e.g., reskinning on the 720). These problems also caused expensive downtime for the airlines.

Methodology has now been developed which allows quantification of fatigue resistance. Structural durability methodology and standards can now be applied which greatly increase fatigue life without significant weight penalty. This has been possible by exercising intelligent (analysis guided) care in handling detail design of critical structural elements (i.e., better detail design with the same amount of material).

Damage Tolerance and Durability

Safety
(Certification)

Damage tolerance

- Residual strength
- Crack growth
- Damage detection

Economics
(Competition)

Durability

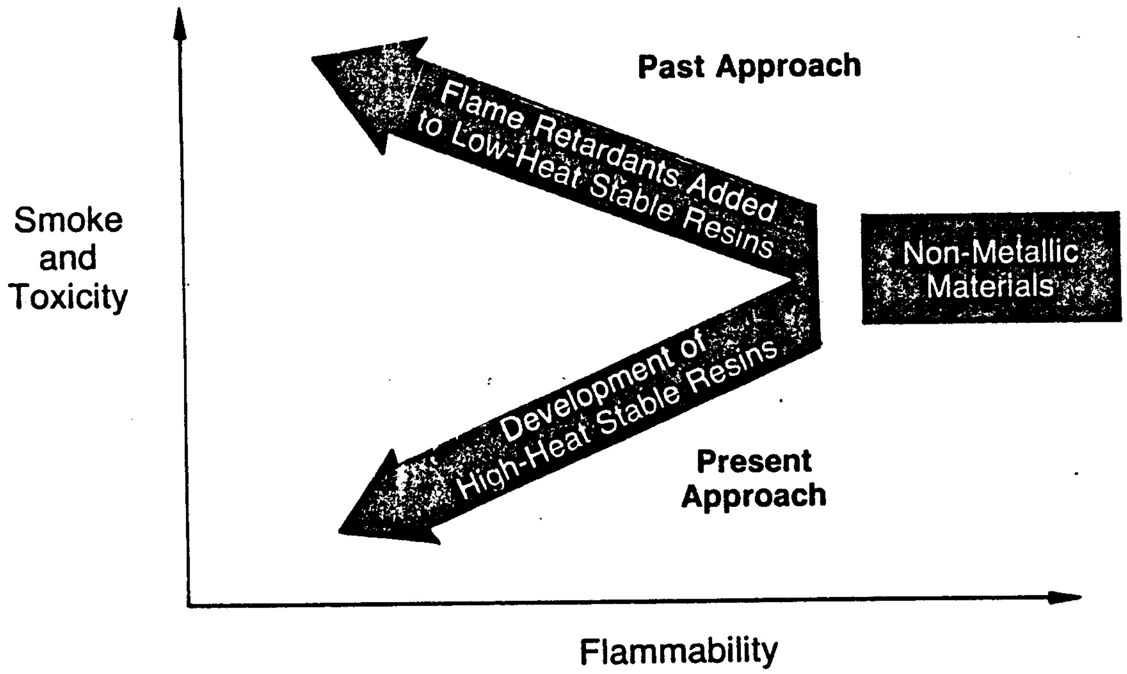
- Fatigue
- Corrosion
- Maintenance

Damage Tolerance and Durability

The final topics to be discussed in this section are damage tolerance and durability. While structural durability has been of concern for over a decade as a competitive issue, the newer concern for damage tolerance is based on safety and certification issues.

New Interior Materials

Improving Fire Characteristics

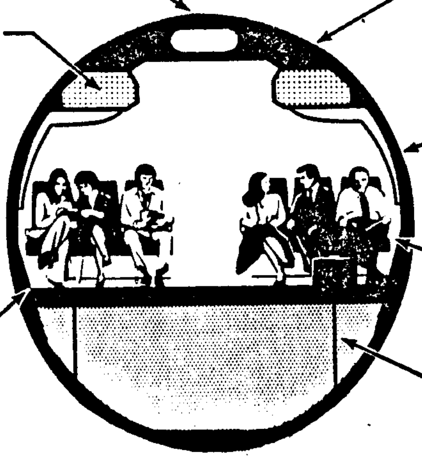


New Interior Materials

Materials technology research has also been directed towards improved non-metallic interior materials. In the past, this research was aimed primarily at reduction in flamability by adding flame retardant chemicals to the resin material. This approach lead, however, to increasingly toxic smoke generation.

Recent research have been directed instead at development of heat stable resins which result in materials which are both resistant to burning and produce little toxic smoke when burning does begin.

New Interior Materials – 757



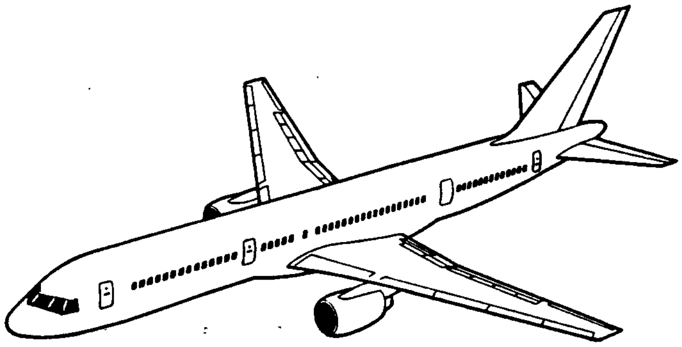
Sandwich Panels
- Class Dividers
- Partitions
- Lavatory Walls

Improvments Over 727	Sidwall/Ceiling Panels	Textiles
Flame Spread Index	– 50%	NA
Smoke Generation	– 60%	– 50%
Toxicity Levels	Reduced	Reduced
Weight Savings	≈ 50%	≈ 20%

Thus, a ten year program in the Central Technology Staff has led to both better interior materials and provision of realistic guidance to the FAA in formulating new forthcoming regulations regarding interior materials safety standards.

These new interior materials are used throughout the 757, and show less tendency to both propagate fire and generate smoke.

Weights Overview



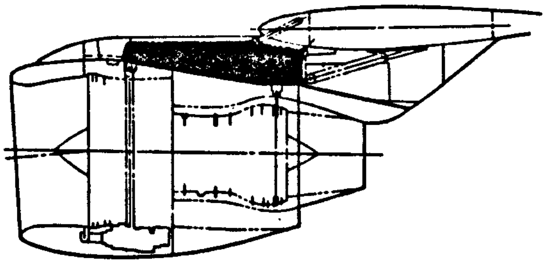
- A fundamental technology

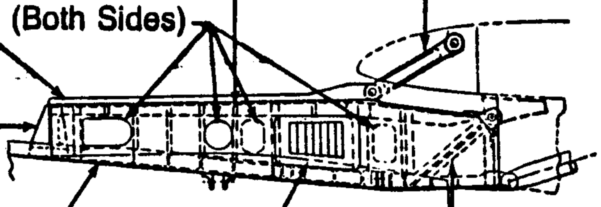
- More comprehensive analysis in same amount of time

WEIGHTS OVERVIEW

As in the cases of the previously cited technologies, the advent of the computer has had a major effect on weights technology. This fundamental technology has metamorphosed from one of simple bookkeeping to one which provides timely, comprehensive analyses that contribute to the making of major design decisions.

Nacelle Strut Weight



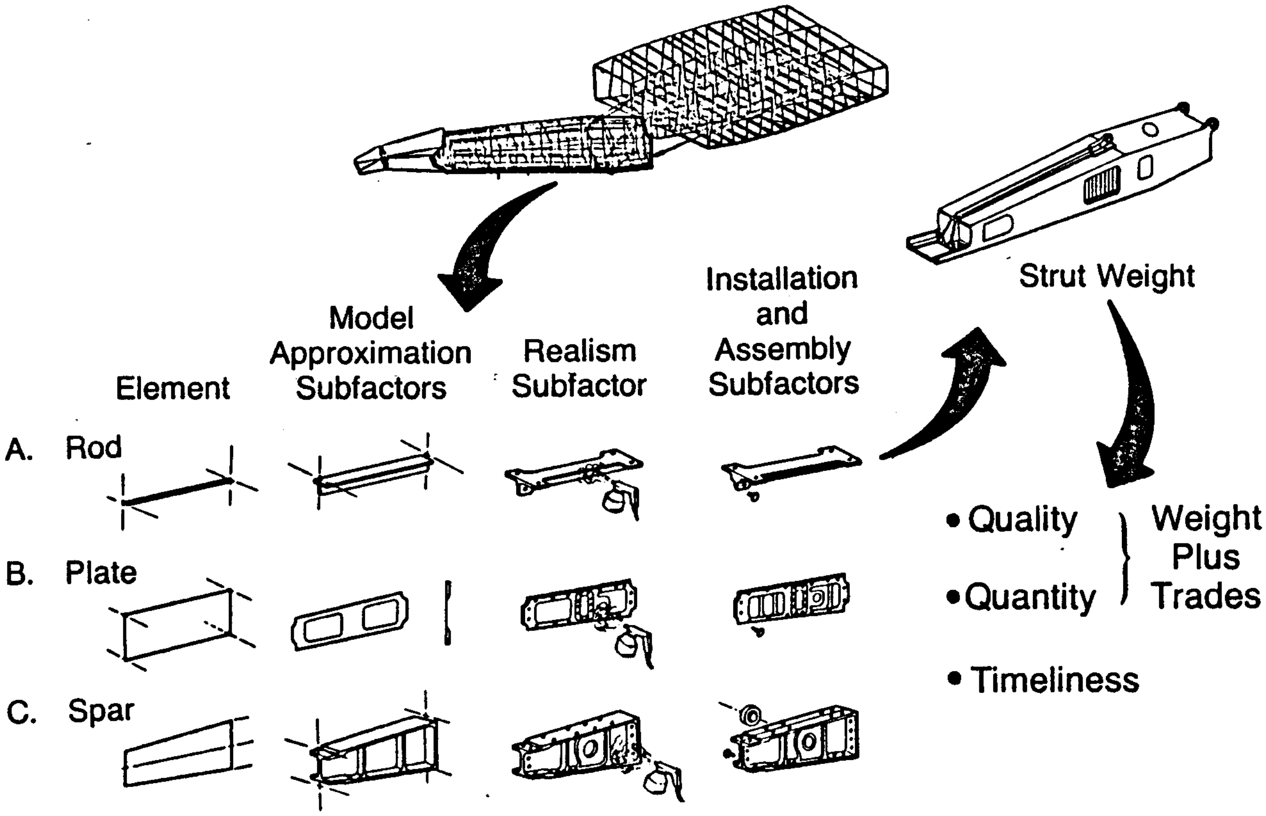


Nacelle Strut Example

As an example of Weights Technologies greatly increased capability, the example of an exercise in estimating the weight of a 757 nacelle strut assembly can be cited.

This strut is not a simple structure. It is full of holes, fittings, firewalls, etc.

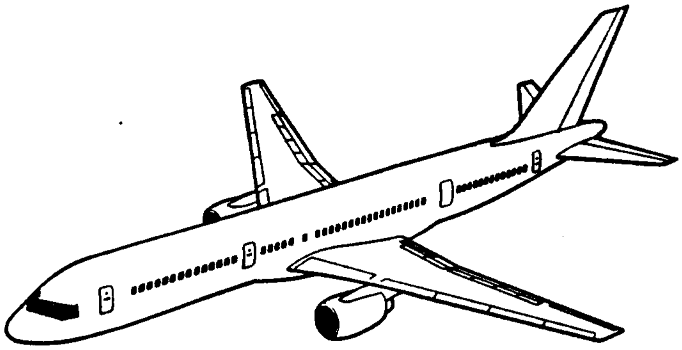
Finite Element Analysis



4.5.1

Using finite element structural analysis tools, the sizes of the basic strut elements required to carry imposed loads and provide necessary stiffness can be established. Given the size of the basic structural elements and the density of the materials to be used, the basic weight can be estimated accurately. At this point, based on experience, the several weight increments involved in transforming the idealized structure into an actual one can be added to produce a final, realistic, and accurate total weight estimate. In addition to providing a basic weight estimate for the specific assembly, information is also provided on possible weight versus strengh/stiffness trades; and all this information is made available on a time-scale sufficiently short to allow it to be used in making fundamental final design decisions.

Systems Overview



- Flight Management System
- Digital Avionics
- Electro/Mechanical Systems

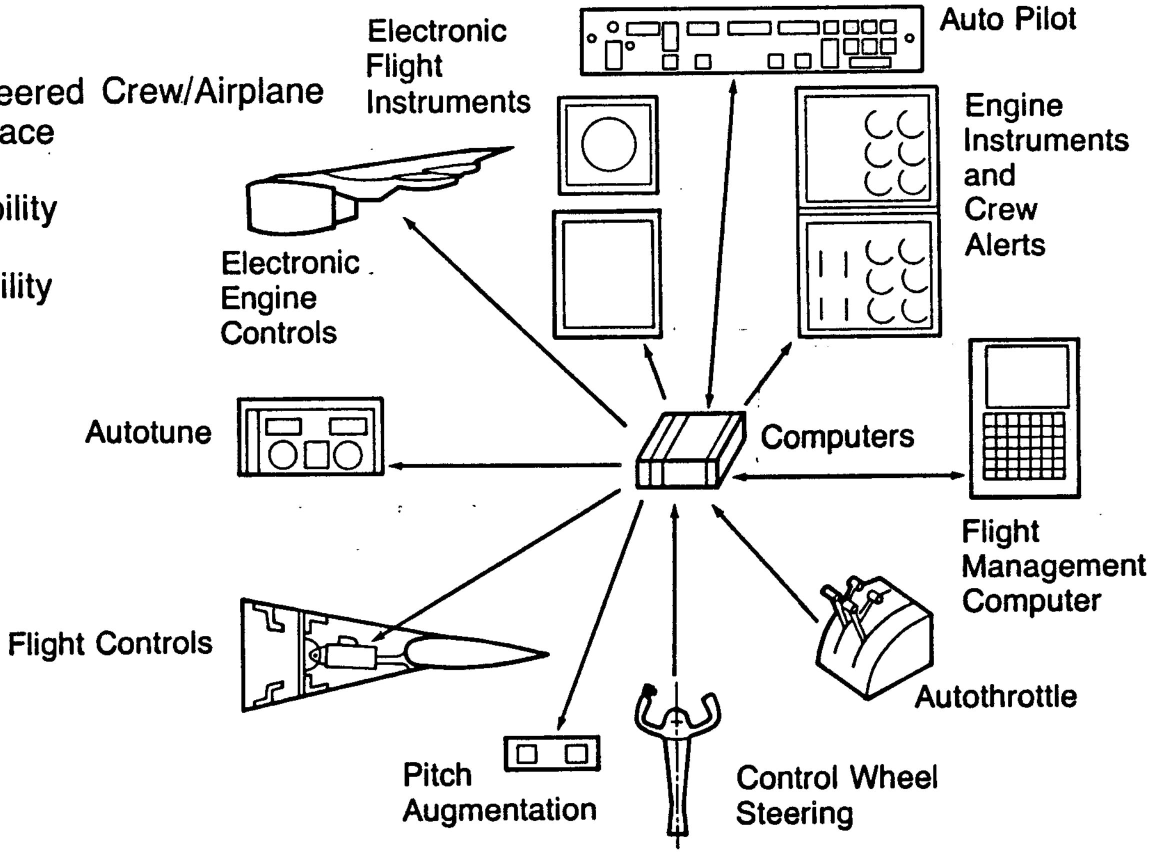
SYSTEMS OVERVIEW

In this section, four topics from systems research will be discussed:

- The Flight Management System
- Digital Avionics Displays
- Environmental Control System Improvements
- Improved Hydraulic Systems

Flight Management Systems (FMS)

- Engineered Crew/Airplane Interface
- Reliability
- Flexibility



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<u>Flight Management System</u>

One of the major advances in the new generation transport aircraft is inclusion of a comprehensive Flight Management System (FMS) based on the use of digital computers.

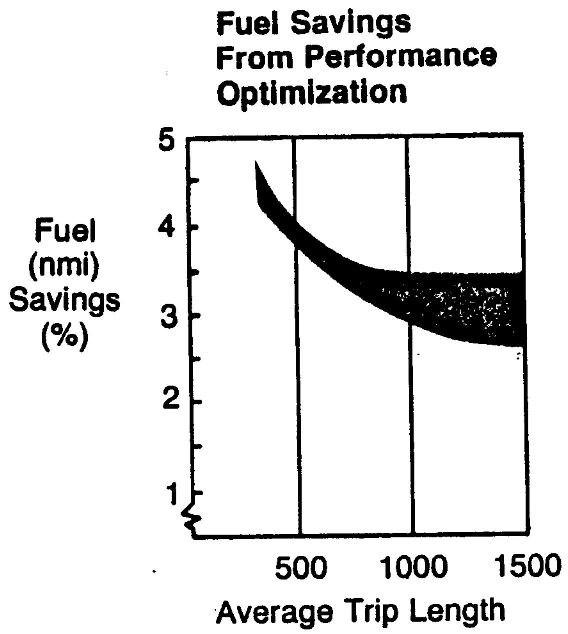
The computers form the heart of the:

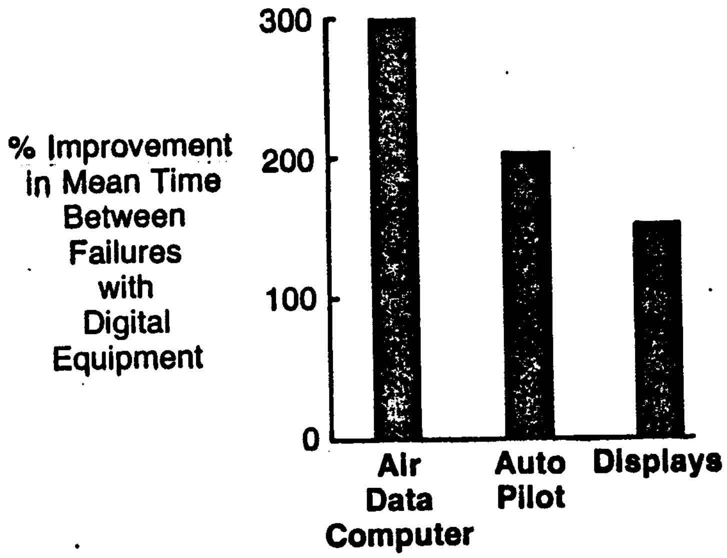
- o Flight Instruments
- o Autopilots
- o Stability Augmentation Systems
- o Engine Controls
- o Flight Path Managements

While representing a potentially major advance, this technology is full of traps. There are major problems of software control and the temptation to expand the system without limit. The FMS system probably represents the single greatest development risk in the 757 program.

Flight Management System

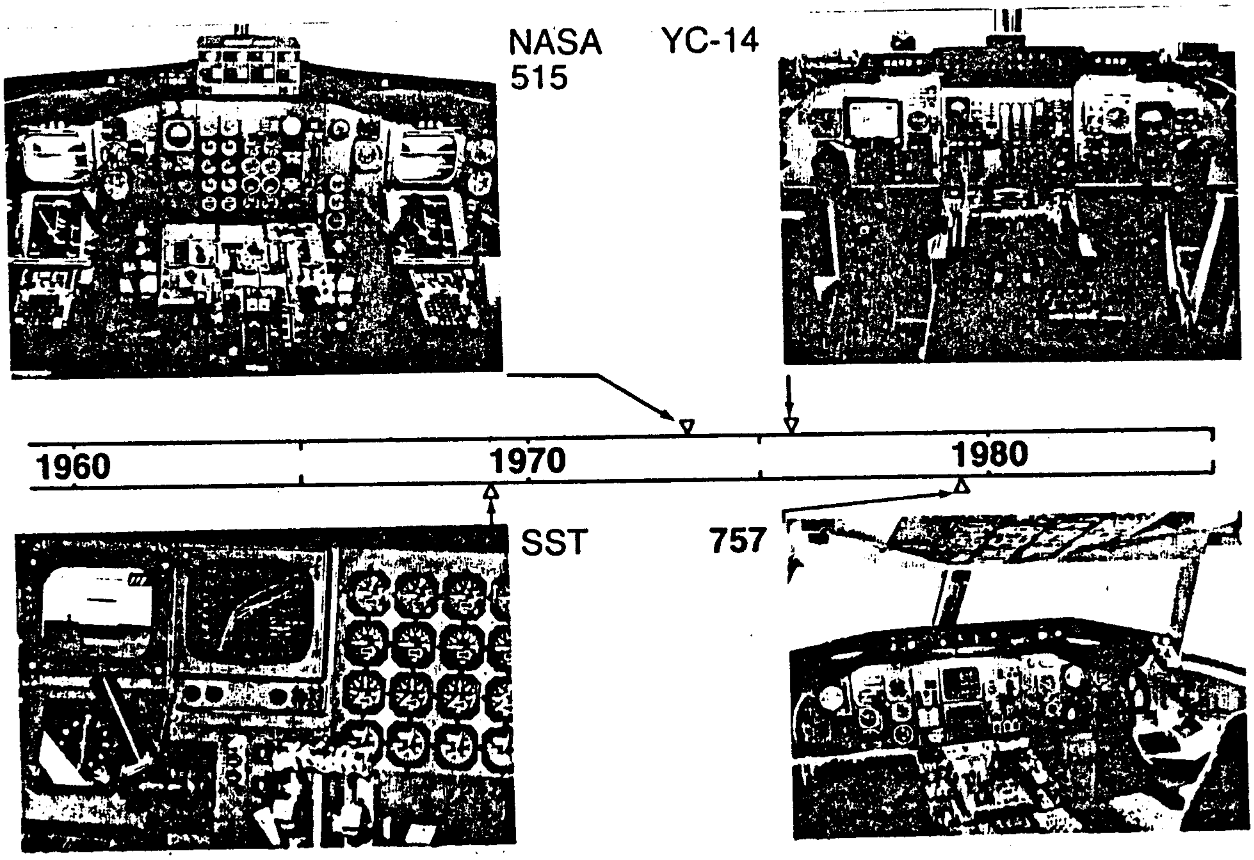
Potential Payoff





The successful incorportion of the FMS in the 757 offers potentially large payoffs, however. The FMS allows the airplane/crew interface to be truly engineered, with consequent benefits in both improved safety and flexibility. More accurate flight path control is possible with consequent reductions in fuel consumption. The FMS also holds the promise of greatly improved reliability compared with previous systems, thus promising reductions in maintenance costs and departure delays.

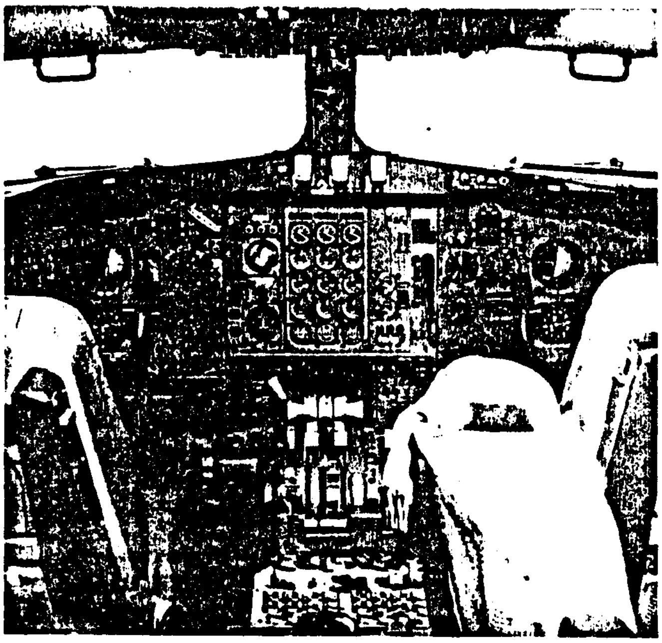
Flight Deck/Digital Avionics Evolution



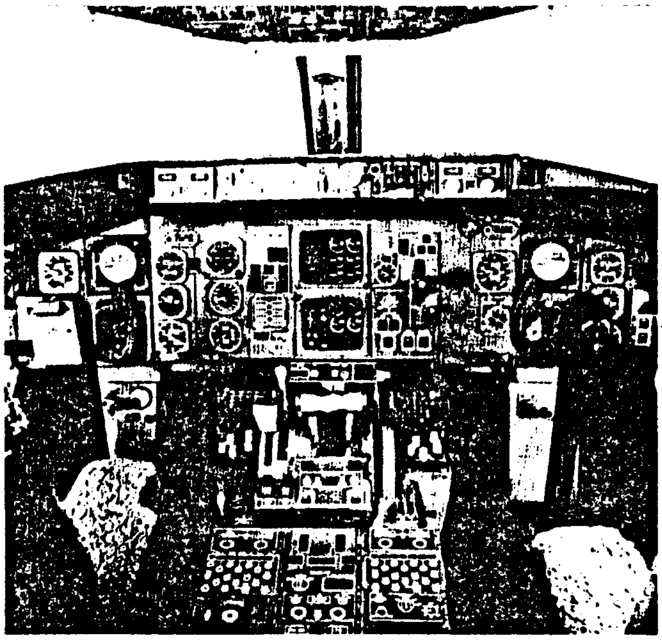
Digital Avionics & Flight Deck Displays

Development of digital cockpit controls and displays has a long history at Boeing. Efforts began in 1968 ~~om~~ in conn connection with the SST program. In this case it was judged that events would happen so fast during even a normal flight that simpler systems would be inadequate. Despite cancellation of the SST program, developments on digital avionics and displays continued and a system of this type was incorporated and its feasibility demonstrated in the NASA 515 TCV (Terminal Configuration Vehicle) program in 1974. A version was then incorporated in the YC-14 in 1976 and sufficient confidence was gained to allow incorporation in the 757/767 in 1980.

Flight Deck Evolution



Then (Circa 1960) 727-200



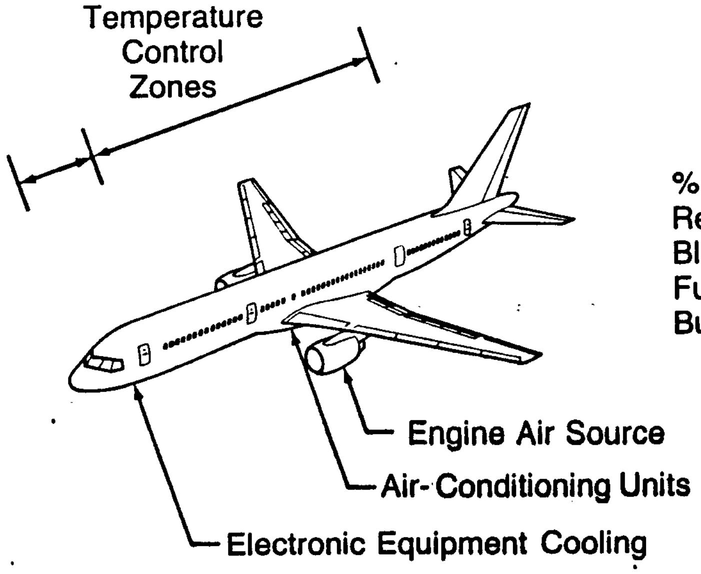
Now (1981) 757-200

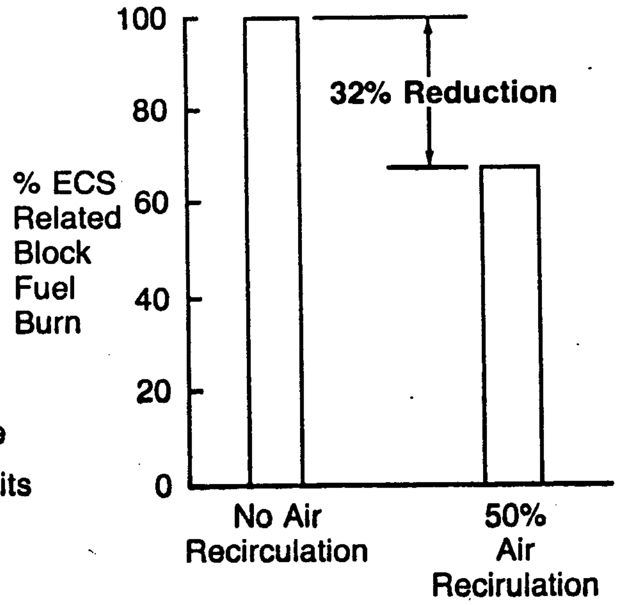
51-1

The result has been a substantial simplification of the flight deck displays in the 757 as contrasted to the earlier 727.

Most pilot control is now done automatically in normal operations. In addition, the pilot is shown only the information which he can do something about thus reducing confusion and distraction in an emergency situation.

Environmental Control System

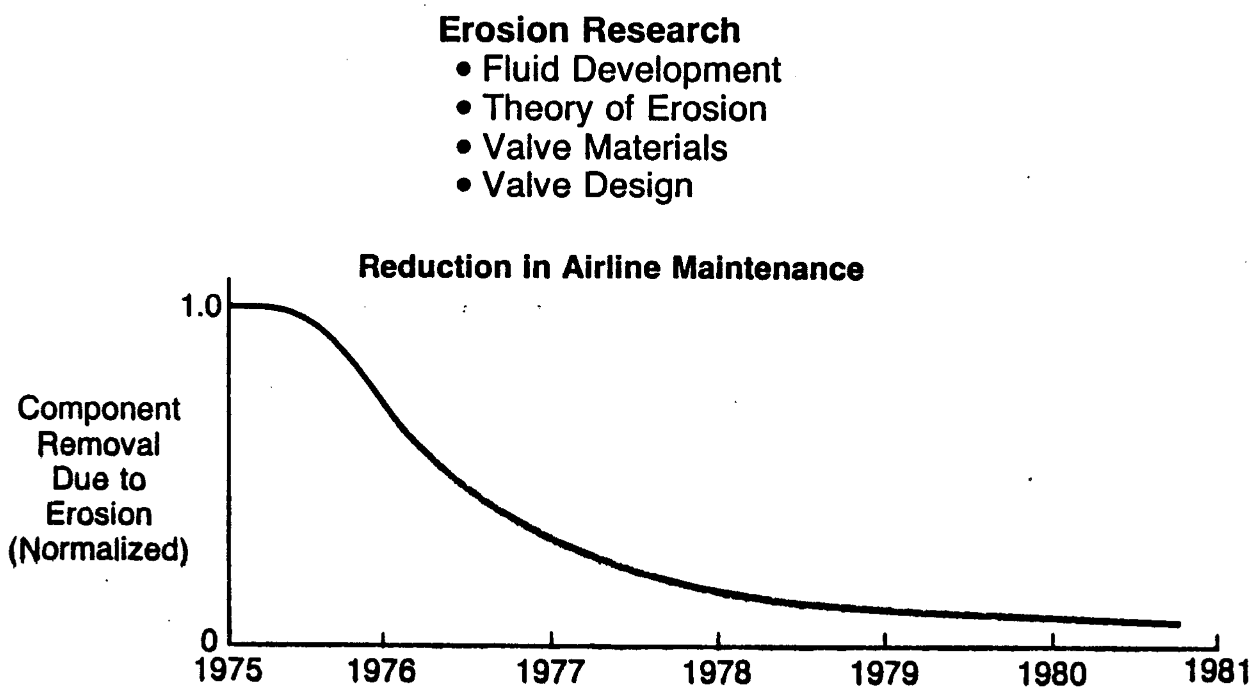




Environmental Control Systems

As part of the across the board effort to improve the fuel efficiency of the 757, as improved environmental control system (ECS) has been developed. In this new system, a better balance has been struck between location of heating and cooling sources in order to reduce fuel related requirements for both. In addition, a more efficient recirculating system is used for cabin air conditioning. This requires provision of adequate filters and in order to avoid smoke ingestion, the cockpit area is separated from the cabin area air conditioning sources. The result of this improved system is a net 1/3 reduction in ECS related fuel requirements while providing adaquate additional cooling required the increased electronic equipment usage.

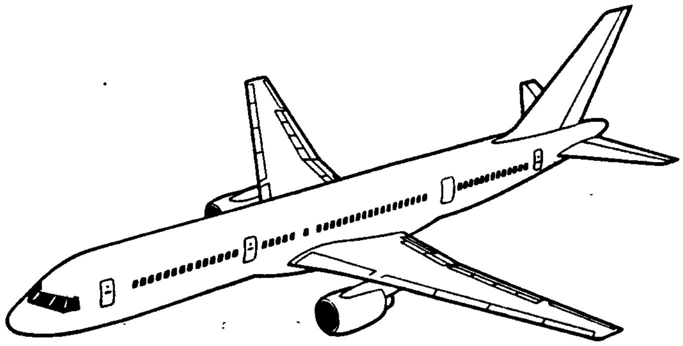
Improved Hydraulic Systems



Improved Hydraulic Systems

Safety imposed requirements for fire resistant hydraulic fluids led to significant problems of valve erosion and leakage. Mechanical system research directed at these problems has resulted in a number of advances. A theory of erosion has been developed, allowing an overall understanding of the basic problem. In addition, better hydraulic fluids have been developed as well as better valve materials and design approaches. In consequence, over the past five years, erosion related maintenance has been almost eliminated.

Noise Overview



- Noise reduction involves other technologies
- Noise Source Identification
- Nacelle design for low noise

NOISE OVERVIEW

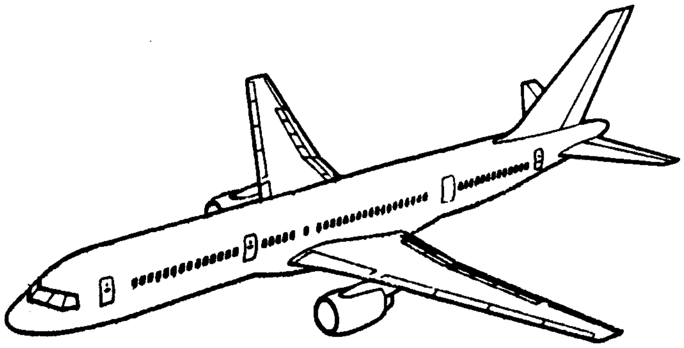
Overall noise reduction efforts involve several technologies. In this portion of the presentation ,the overall design approach taken to reduce noise is described, followed by a discussion of noise source identification progress and nacelle acoustic lining materials.

Noise Control Design

Wing Design
- Increased Span
- Reduced Drag
- Reduced Thrust

Nacelle Design
- Acoustic Linings

Power Plants
- High Bypass Technology
- Decreased Core and Jet Noise
- Fan/Guide-Vane Design Minimizes Fan Tones
- Engine Case Acoustic Linings



No dominant reducible noise source remains

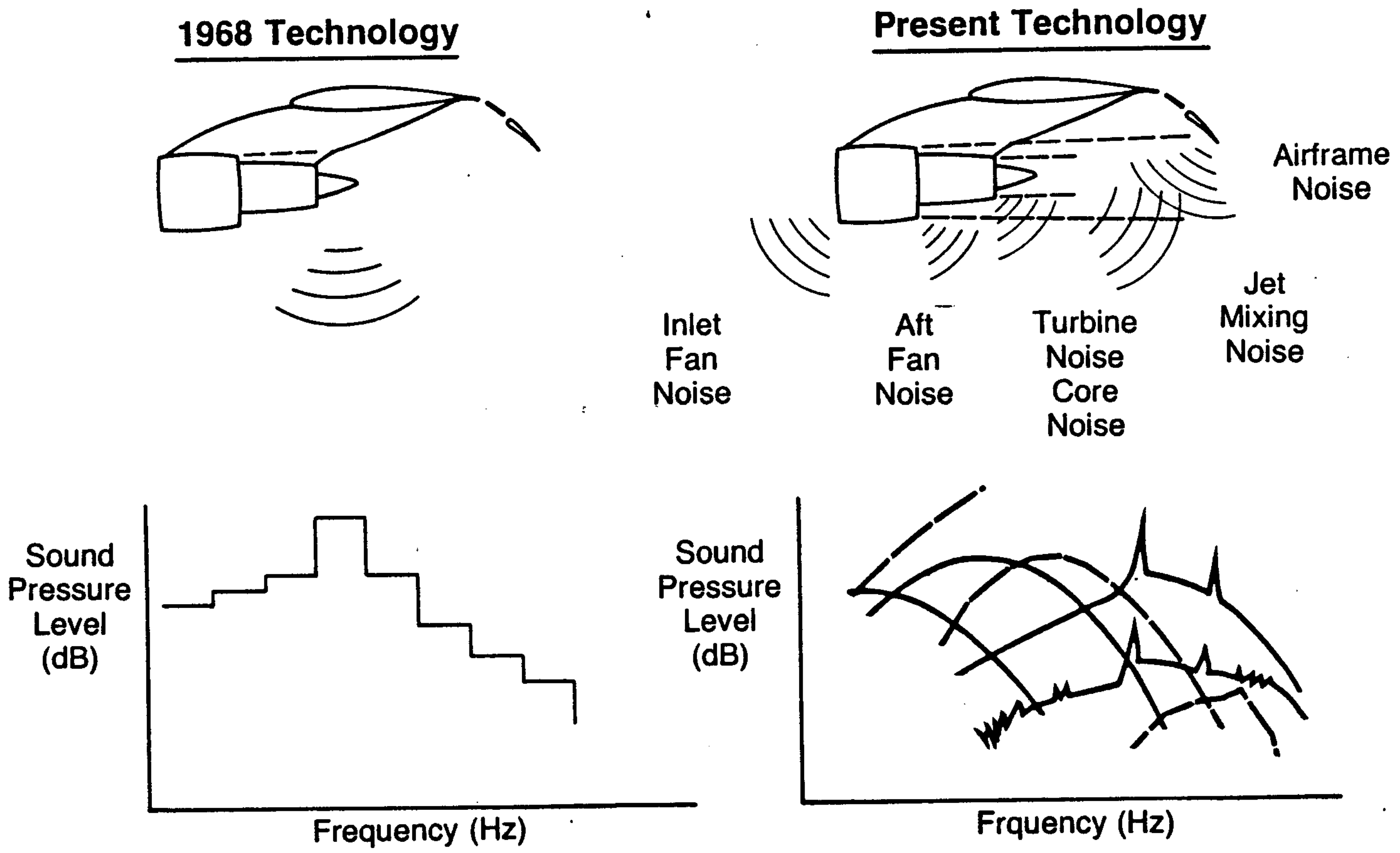
Noise Control Design

Noise reduction in the 757 has been achieved by a combination of many advances.

Larger span wings lead not only to improved fuel efficiency, but to reduced noise by reducing drag and thus thrust required. Improved acoustic linings in the nacelles as well as the inherent improvements associated with use of high bypass ratio fans further contribute to propulsion system noise reductions.

As matters now stand, no dominant reducible noise source remains in the 757.

Noise Progress



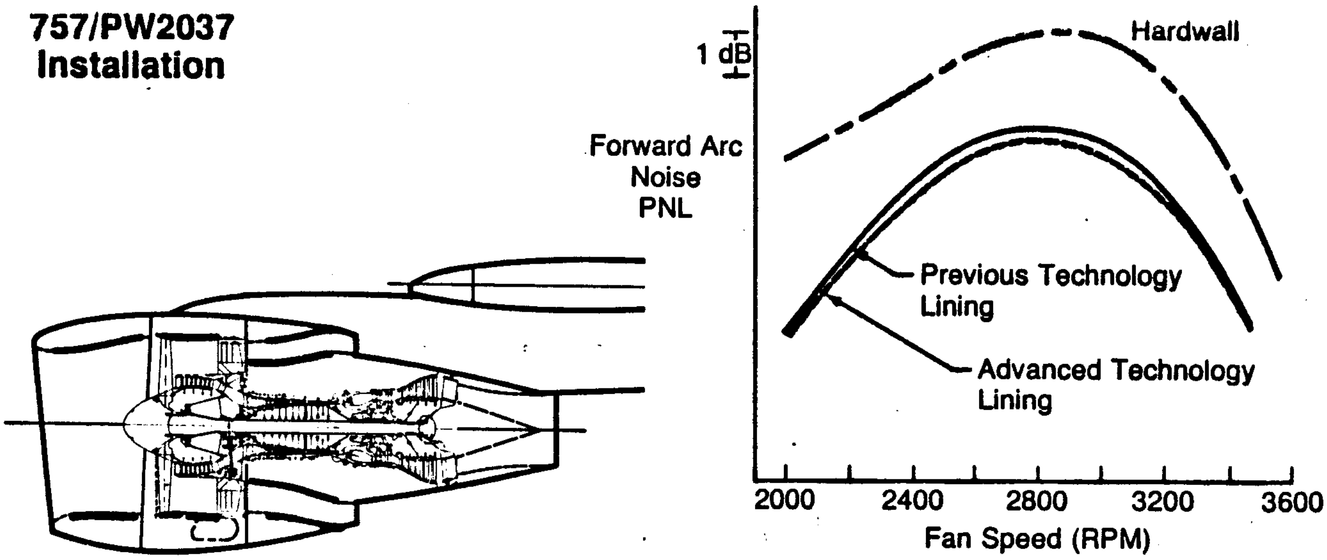
Noise Source Identification

During the development of the 747 a decade ago, only total noise levels were known, and a noise staff of fifty was required to deal with the problem.

Today, research advances have made possible the detailed identification of all major individual noise sources. It is now possible to identify which noise sources should be worked on, and those which, if modified, will not produce an overall improvement. On the 757, the noise problem can be attacked with more assurance of success and in a systematic way, by a staff half the size required on the 747.

Application of Lining Technology

757/PW2037 Installation



- 2% Certification Risk Reduction
- Save 32 lbs/Airplane
- Reduced Fabrication Complexity
- Cost Saving

<u>Nacelle Lining Technology</u>

As an example of research led advances in noise technology, nacelle design and lining technology can be cited.

On the 757, nacelles without secondary air inlet doors have been designed and fan and core walls are treated with acoustic lining materials. These lining materials are tuned to specific noise values <u>by design</u>. The resulting lining installations are thus lighter, simpler and less costly. In addition, confidence in meeting noise certification standards is increased.

Conclusions

1. **Safety, efficiency and durability – keys to success in a maturing technology**
2. **Research applied to 757 to excel in these parameters**
3. **Not all research is used. By nature, not 100% successful**
4. **Tools must be ready when needed**
    - Decisions are quick
    - Results must be dependable
5. **Research activity trains for its application**

571.1

CONCLUSIONS

An overview of some of the more significant research-inspired advances in the new generation of Boeing transports has been presented. In the mature technological environment in which these transports are being developed, safety, efficiency and durability are the keys to commercial success. The sustained, company funded, research effort of The Boeing Company allow the 757/767 to excel in these basic characteristics.

It is important to note, however, that not all research funded by the company can be applied directly to the new airplane projects. By its nature research cannot be 100% successful. Research developed tools which may be of value must be ready when needed. In the course of a normal new airplane development program decisions to be made are numerous and they must be made quickly. Research developed analysis tools used to support the design effort, must be both accepted and dependable to be of value in this process. This factor is fundamental, and must be recognized by the various research units.

Finally, in the course of developing this presentation, it has become clear that one of the most significant, non-quantifiable, benefits of the company research effort is the training it provides both the developers and users of the methods and techniques which the technology research units produce. This training allows a greatly improved level of skill and understanding of a difficult and demanding design problem - the development of the world's best commercial transport aircraft.

IV. Specification Formulation - (3 hours)

A. Mission Analysis

B. Market Analysis

C. State-of-the-Art Technical

D. Examples

1. SUPERSONIC MISSILE

KANSAS CITY MISSOURI

August 2nd,
19 32

Douglas Aircraft Corporation,
Clover Field,
Santa Monica, California.

Attention: Mr. Donald Douglas

Dear Mr. Douglas:

Transcontinental & Western Air is interested in purchasing ten or more trimotored transport planes. I am attaching our general performance specifications, covering this equipment and would appreciate your advising whether your Company is interested in this manufacturing job.

If so, approximately how long would it take to turn out the first plane for service tests?

Very truly yours,

Jack Frye

Jack Frye
Vice President
In Charge of Operations

JF/GS
Encl.

N.B. Please consider this information confidential and return specifications if you are not interested.

SAVE TIME – USE THE AIR MAIL

The letter and specifications that brought about the birth of the DC-1 which changed the concept of commercial transports for all times.

TRANSCONTINENTAL & WESTERN AIR, INC.

General Performance Specifications
Transport Plane

1. Type: All metal trimotored monoplane preferred but combination structure or biplane would be considered. Main internal structure must be metal.

2. Power: Three engines of 500 to 650 h.p. (Wasps with 10-1 supercharger; 6-1 compression O.K.).

3. Weight: Gross (maximum) 14,200 lbs.

4. Weight allowance for radio and wing mail bins 350 lbs.

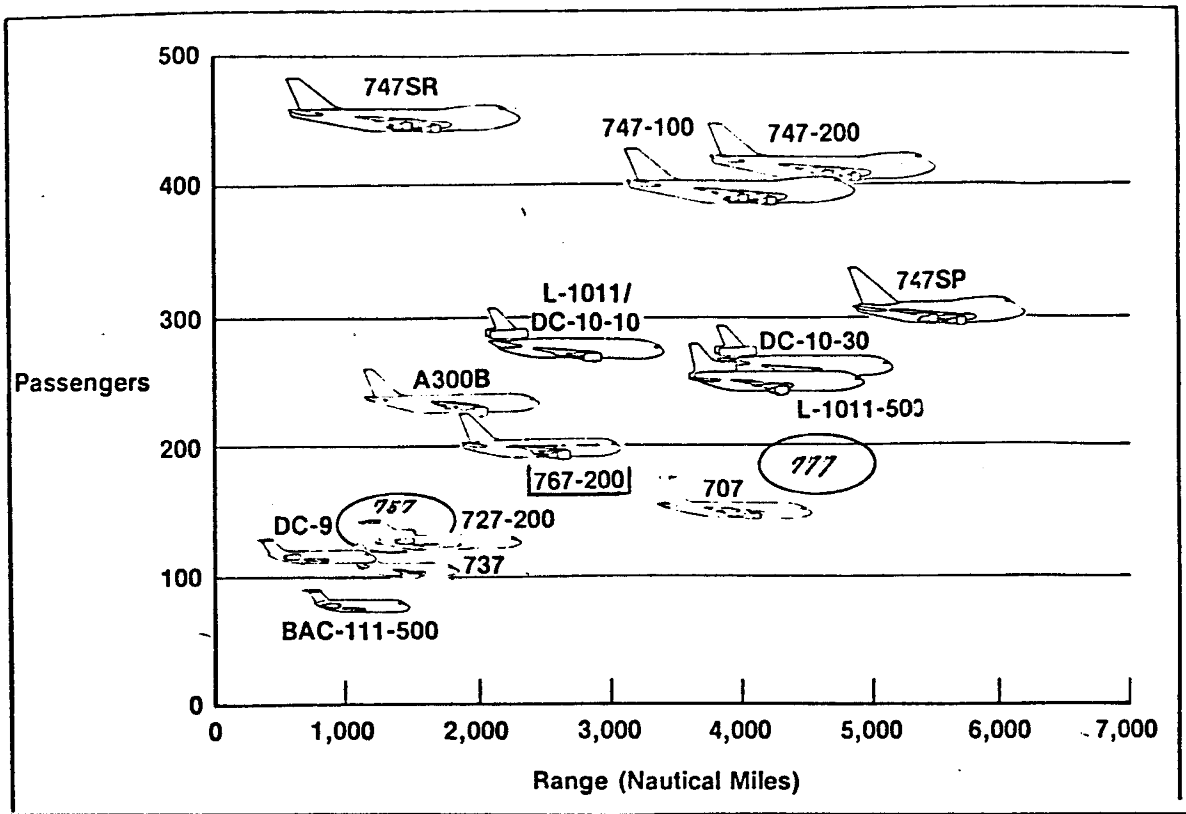
5. Weight allowance must also be made for complete instruments, night flying equipment, fuel capacity for cruising range of 1080 miles at 150 m.p.h., crew of two, at least 12 passengers with comfortable seats and ample room, and the usual miscellaneous equipment carried on a passenger plane of this type. Payload should be at least 2,300 lbs. with full equipment and fuel for maximum range.

6. Performance

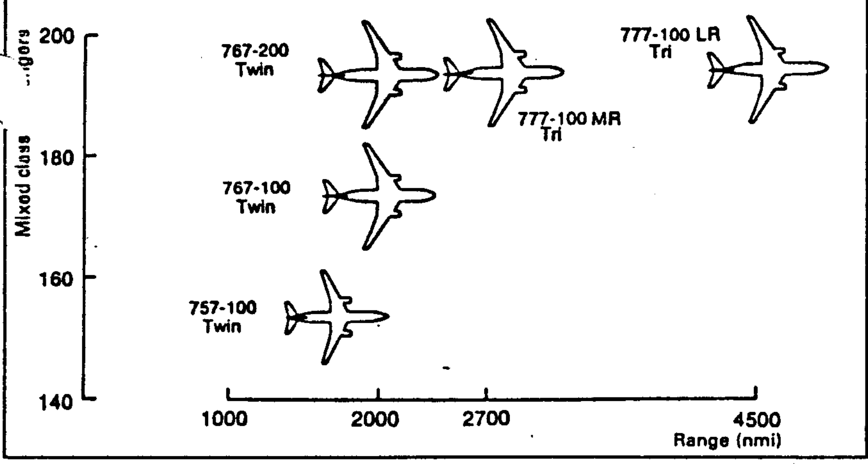
Top speed sea level (minimum)	185 m.p.h.
Cruising speed sea level - 79 % top speed	146 m.p.h. plus
Landing speed not more than	65 m.p.h.
Rate of climb sea level (minimum)	1200 ft. p.m.
Service ceiling (minimum)	21000 ft.
Service ceiling any two engines	10000 ft.

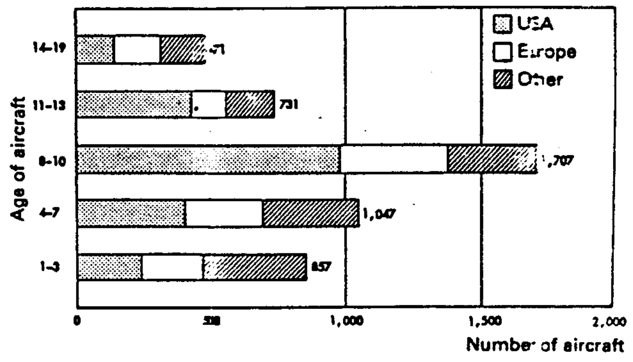
This plane, fully loaded, must make satisfactory take-offs under good control at any TWA airport on any combination of two engines.

Kansas City, Missouri.
August 2nd, 1932



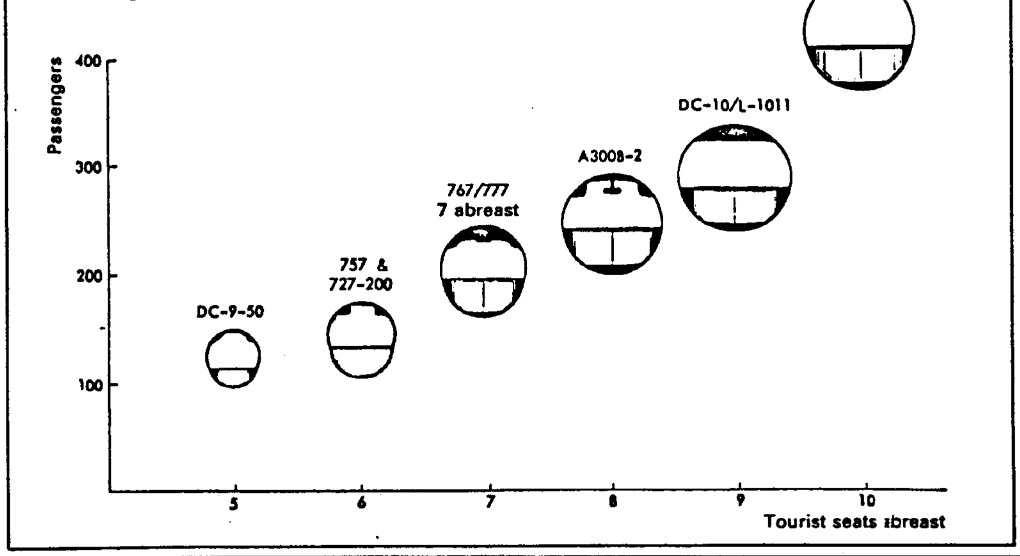
The new Boeing transport family.





World jet transport fleet age distribution, showing aircraft in service at January 1, 1977. Excludes USSR, China and non-ICAO nations, except Taiwan. (*Source:* Boeing)

Fuselage cross-sections of current civil transports compared.



How Decisions Are Made
Major Considerations for Aircraft Programs

John E. Steiner

Abstract

Aircraft programs, both civil and military, represent complex risk experiences. Success usually involves the attainment of a relatively long-term program to achieve efficient production volumes; this in the face of a constantly changing set of market conditions, competitive actions and technological alternatives. Key decision points are identified, and risk variables in finance, technology, management, and market readiness explored. Decisions are noted that can leverage long-term potentials for success and others, that once made, may become irreversible in view of program cost penalties. The author draws on his involvements and observations of program decision-making over a 40-year career span and involving over a dozen programs. While most are commercial, several fundamental observations applicable to military programs emerge. The character of decisions and influences leading to program success or lack thereof are examined to find the lessons learned and to comment on the road ahead.

I. Introduction

Over 80 years ago, the Wright brothers made decisions that led to the historical successes that this lecture commemorates. It is a great honor for me to be asked to join those who, through the years, have memorialized the Wright brothers and the achievements they attained.

Aircraft programs, both civil and military, represent complex risk ventures that are accomplished in an environment of constantly changing market conditions, competitive actions, and technological alternatives.

World aviation is moving into a new, and, I believe, an even more complex era in which affordability and internationalism are becoming major influences. While some of the historical tenets of aircraft decision making will remain the same, some will not. The objective of this paper is to discuss some of the major factors that have affected past aircraft decisions, and to consider the changes that aviation's new era may imply for the coming generation of decision makers.

We'll approach this by noting the decisionary forces evident following World War II and the major changes thereafter. We'll review technical progress made and note its future extensions. Following this will be an examination of the decisionary forces in play on some past programs and in particular, key decisions that led to their success or failure. Since aircraft program success is highly dependent upon engines, we'll also examine these decisions and their forcing factors as well. We'll then examine the decisionary forces forming the new fleet of commercial aircraft and the track of lessons learned from program decisions, and note their implications with respect to the environment of the road ahead.

II. Post War Overview

Aviation progress, of course, has been immersed in a much greater matrix of time and events, outside the scope of this paper. However, it is important to highlight the environment following World War II, since it involved a period of achievement in military and civil aviation unparalleled in later times.

Aircraft was a natural for post-war development, and a product that could readily respond to civil markets as well as the continued military concerns triggered by cold war events. This period launched a number of military and civil derivative programs, such as the B-50 from the B-29 and the DC-6 and DC-7 from the wartime C-54. Each was an incremental refinement step furthering technologies developed or proven during the war. All new jet-powered military programs were also initiated, such as the F-86 and B-47. This rapid progress turned the 1950s into a bow wave of advancements as this decade saw some 17 major military programs started plus an even larger number on the civil side. The key decision drivers for their go aheads are

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summarized in figure 1, and they will stand as key drivers for program decisions made today.

Key Decision Drivers

- Market Needs (and Timing)
- Government Actions (and Priorities)
- Competitor Actions
- Technology Readiness
- Fiscal Considerations

Figure 1

Conditions at this time were, for the most part, favorable to the fostering of competitive program starts. There was a large domestic market need (military and civil) and a large industry in place that was backed by a high quality research infrastructure. By comparison, the highly innovative European industry was constrained. This period was largely an American event in scale...it highlighted U.S. domestic markets and competitions.

Europe's industry gradually recovered and, over the following decade, laid the ground work for the many cooperative European developments that followed. Japan's aircraft industry, relatively constrained through this, has now emerged with credentials highly respected by both the U.S. and European industries. This, plus other new competitors and causative factors, has greatly changed the decision environment, as summarized in figure 2.

Changing Decision Environment

- Design Orientations
- Development Costs and Risks
- Military and Civil Priorities
- Internationalism
- Affordability

Figure 2

New requirements and new advancements have obviously affected design orientations and design decisions. Costs, risks, and priorities are not the same worldwide, nor are the affordability values that are attached to them. U.S. priorities were revalued and this has precipitated a dramatic change in the technological fiber of the nation. Nevertheless, as figure 3 illustrates, government outlays for aircraft through this transition remained surprisingly stable.

Federal Outlays for Aircraft, Missiles, and Space FY1947-77

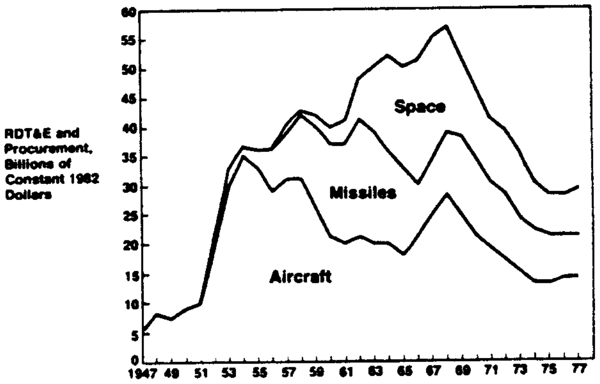
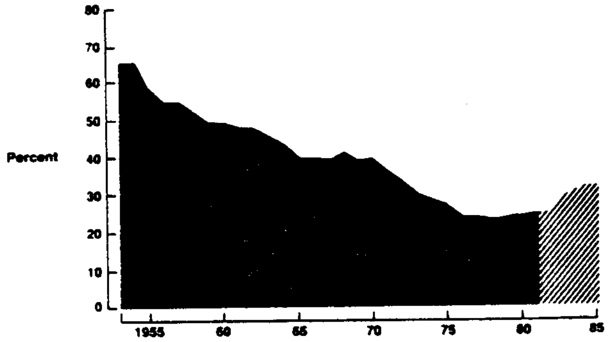


Figure 3

The downward trend that occurred in the late 1960s reflects a massive insertion for social spending with offsetting reductions in other government programs, including defense. The 40 year change in total defense outlays is shown in figure 4.

Defense Outlays: FY 1953 to 1985

(As Percentage of Federal Budget)



Sources: OMB: Budget of U.S. Government FY1978 through 1982; AIA

Figure 4

It should be noted that defense spending (in constant dollars) remained fairly stable, but total outlays increased steadily, thus reducing the defense percentage as shown. The reordering of national priorities has exerted a profound change on the momentum of U.S. aeronautical developments. Through the 1970s for example,

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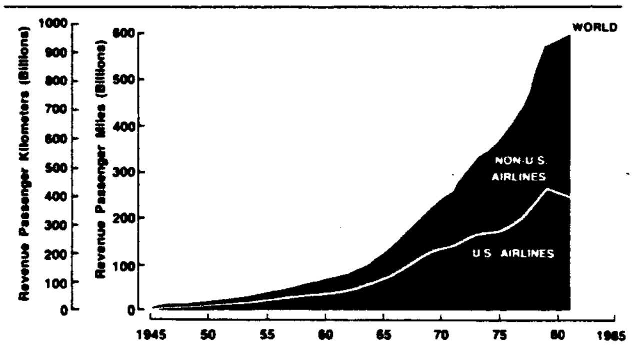
new military program starts dwindled to a few, although paper competitions and false starts were many. Much of the advancement momentum was taken over by commercial industry developments. There was good reason for this because the growth realized in world air travel since the mid-1960s was beyond all earlier predictions. The history is illustrated in figure 5.

World Air Travel Growth
Scheduled Services



Note: Excludes USSR, PRC, Non-ICAO Nations, and Charter Carriers

Figure 5

Figure 5 also illustrates the growing significance of the non-U.S. portion of the total world market. The technological achievements contributing to the creation and growth of air travel markets came from many nations to make air travel an affordable alternative, as illustrated by the air fare reduction history shown in figure 6.

Round Trip Air Fare
New York City-Los Angeles

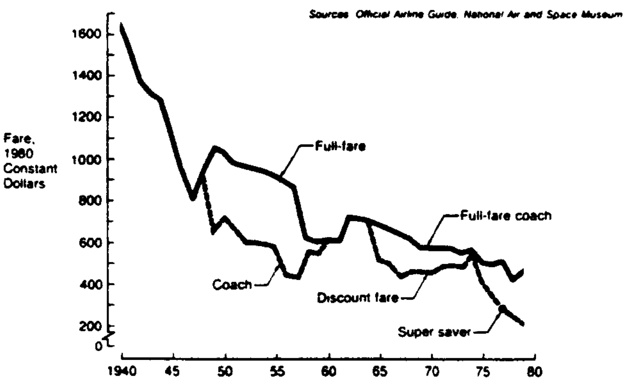


Figure 6

Our overview has shown that the U.S. maintained a healthy technological environment for some two decades beyond World War II. Since then, however, its pace slowed, and the advancement momentum in world aeronautics changed, reflective of the rapid expansions in Europe and Japan.

The pace and state of technology readiness is a major consideration in program competitions, as will be noted later. Therefore, we'll first examine the progress that has occurred in recent decades, and also the potentials that will form program considerations in the future.

III. The March of Technology

Technology decisions are relatively straightforward when the technology base is well understood, the development has been completed, the payoffs are clear and the risks are low. Such is not the usual case when new airplane programs are started or major airplane modifications undertaken. More likely is that competition will push the state of the art, shorten allowable development scheduling, and establish goals that strain credibility and involve considerable risk. It is essential that decisions don't repeat past mistakes but maximize the potential for the future.

You could ask...what causes people and organizations to strive for technology that could prove embarrassing to individuals or risk a company's existence. More and more technical advancement is demanded by the customer, forced by competition, or pressed by a public mandate to improve the environment. Other reasons are probably equally important and the pressures are inescapable. Therefore, goals and requirements must be clearly defined and understood by all participants before risk contracts are signed. Both the buyer and the seller can be seriously injured by overly ambitious dreams or impractical desires.

Aircraft technical advancements flow from many national sources and will continue to do so. The radar, jet engine, swept wing, and much of today's modern electronics are only a few examples of international contributions. Technical secrets are perishable with time, and since the period from discovery to validation and on through to application can take ten years or more, attempts to keep developments proprietary are mostly futile. It is more important that the developer make timely decisions in order to enjoy the advantage of one or more application cycles before outsiders acquire sufficient technical base to proceed with their own.

Trends

The most revolutionary advance in airplane productivity occurred virtually overnight with the introduction of the swept wing and turbojet engine. The resulting increase in speed and improvement in passenger comfort obsoleted the medium- to long-range propeller powered transports. In retrospect, the transition occurred with amazing ease to all concerned. Turbojet and turbofan engine developments have been among the biggest contributors to improvement in airplane efficiency. Figure 7 shows commercial jet engine specific fuel consumption to have decreased some 40 percent over the last 25 years.

Fuel Consumption Improvements

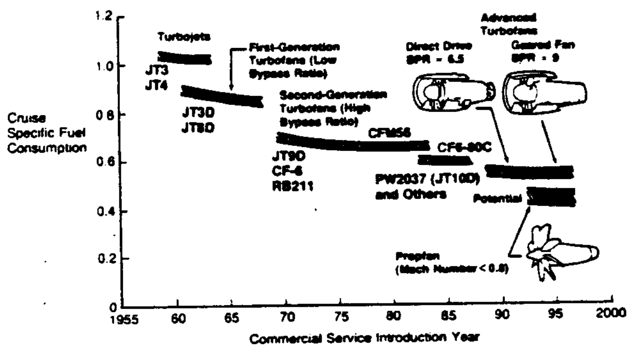


Figure 7

This phenomenal improvement occurred as a result of cooperative investments by both government and industry, but not without considerable pain to the users. Excessive parts replacement and high maintenance costs followed new engine model introductions when the validation periods were foreshortened. Further improvements in fuel efficiency are possible by using geared fans or, more radically, by eliminating the cowl through development of an advanced turboprop system. In these cases, adequate development and validation periods will become increasingly important and may require unreasonable investments.

Introduction of the turbojet engines caused a deterioration in the environment around airports. Since those early installations, progress in noise reduction has been continuous as shown by the trends on figure 8.

New engines are basically quieter than turbojets. Adding extensive acoustic treatment

Noise Reduction Trends

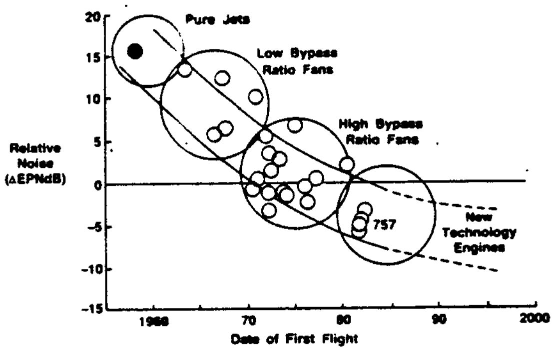


Figure 8

to the integrated nacelle-engine power package has further reduced community noise to the point where public outcries have largely subsided, or will as normal replacement occurs. This has been a painful problem to the aviation industry. Any further technology decisions that have environmental side effects must reckon with public opposition.

The progress in subsonic aerodynamic design over the past several decades has been significant but is more difficult to describe than the improvement in engine specific fuel consumption. This difficulty arises from the fact that airplanes are designed to unique market objectives, and mission requirements will emphasize high and low speed aerodynamic design capability and structural weight trades in differing proportions. This tends to obscure the significance of a specific technology advance such as improvements in high Mach number airfoil design. A good example of the hidden value of aerodynamic progress surfaces if one tries to compare the aerodynamic cruise efficiency of L/D of the 747 relative to that of the 707-320. The significant progress in aerodynamic design technology achieved in the twelve year interval between these programs is concealed by the differing design objectives, most notably the higher 747 cruise speed, and the relative difference in fuselage size. In fact, the L/D of the 320B is actually four to five percent higher than that of the 747 at respective cruise design points.

One way of illustrating the progress in wing aerodynamic design is to examine the trend with time of relative wing weight and streamwise thickness ratio for hypothetical wings designed to a fixed cruise Mach number and span loading.

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Figure 9 shows that progress in airfoil aerodynamic design has allowed a steady increase in wing thickness ratio which can be translated into significant wing weight savings.

Aerodynamic Progress
Constant Mach Cruise

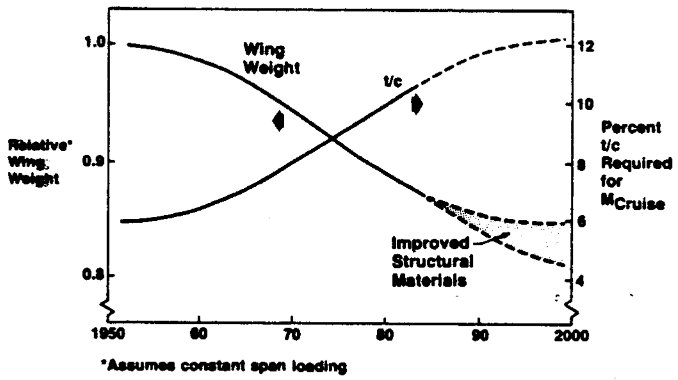

Figure 9

In actual practice there is a tendency to utilize the improvement in allowable thickness ratio to achieve a larger wing span for a given structural weight and to reduce wing sweep for improved low speed performance.

The next ten years offer exciting prospects for aircraft structural designers. New materials such as improved aluminum alloys and advanced composites are receiving widespread attention. A solid data base involving design standards and production techniques is being developed rapidly. If composite and aluminum-lithium structural materials are both successfully developed, a strong possibility exists for designing airplanes that would take maximum advantage of the properties of both. The potential for large structural weight savings is apparent in figure 10.

Future Structural Materials
Trend for Potential Weight Savings

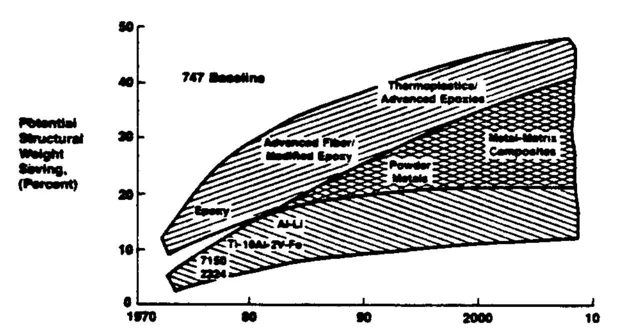

Figure 10

No technologies have advanced as fast as those associated with the electronic industry. In fact, the world expansion of electronic products is indeed revolutionary with no real end in sight.

In the aircraft industry these technologies have been introduced gradually in an evolutionary manner, although digital avionics did take one giant step with the 767/757 flight management systems. The development work to validate airborne applications takes time. For example, the present electronic flight deck displays were initially developed and tested for our SST back in the late 1960s.

Additional systems are becoming available that complement the work accomplished to date. Over the next ten years, for example, we will see increasing applications of fiber optics, flat panel displays, and electric controls, as shown in figure 11.

Avionic System Evolution

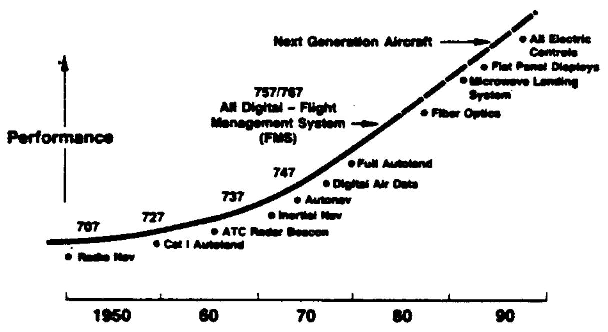

Figure 11

All must be carefully integrated into an efficient, high performance, low risk system. Premature introduction of a digital multiplex passenger accommodation system in the 747 created major situations of inconvenience and annoyance. The system simply wasn't ready. Fortunately, it was not a flight critical system, and safety was not impaired. It is of paramount importance that "flight critical" items such as electronic flight controls be technically ready when put into production.

Throughout commercial air transportation history, designer attention has focused on the "critical mass" of technology that is available for use. The critical mass is really a moving target, and its elements are usually evolutionary in their development and readiness. The next critical mass is now in formation with its roots

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incorporated as some of the improved efficiencies represented by our latest new airplanes. The relationships between these efficiencies and the aiming points of the next critical mass are illustrated in figure 12.

Aiming Points for New Technology Readiness

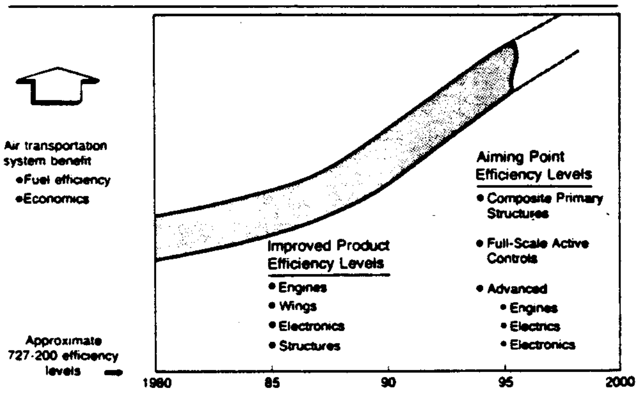

Figure 12

The 757 and 767, incorporating the improvements noted at the left of the figure, have a fuel mileage (seat miles per gallon) advantage over the 727-200 of over 50 percent, even after the effects of scale have been accounted for. The figure indicates that another major improvement of similar magnitude can occur during the decade of the 1990s. Beyond this are other major potentials such as boundary layer management and propfans.

As noted in figure 12, the aiming points will extend into the flight deck with a whole new relationship between the pilot and his aircraft. Control cables and the familiar yoke will disappear, and digitized voice technology can potentially reduce a large element of the communications workload. In their integrated product applications, the new efficiencies may represent a sizable advancement for aviation...perhaps the most significant that we've known since the marriage of the swept wing to the axial flow compressor. The timing of its eventual readiness will be influenced by the levels of effort applied.

Technology development passes through three phases that we sometimes refer to as Phases A, B, and C. Phase A is basic research. Phase B is the assembly of the body of technology until it can support actual use with acceptable risks. Phase C is application to a specific aircraft design. There are many words used to describe the three phases in the terminology of the Air Force, NASA, or others. Phase B tends to be the longest and most expensive. It generally includes a number of parallel actions over a number of years. For example, the increased use of composite applications in the newest airplanes is an evolutionary Phase B step which, along with other steps, will lead to Phase C, the actual use of a composite primary structure in a major commercial or military airplane. A part of the Phase B process is to develop the manufacturing technology required to commit a program to Phase C in this difficult affordability environment.

A good measure of overall technical progress in aviation is illustrated by figure 13.

Fuel Efficiency Trends

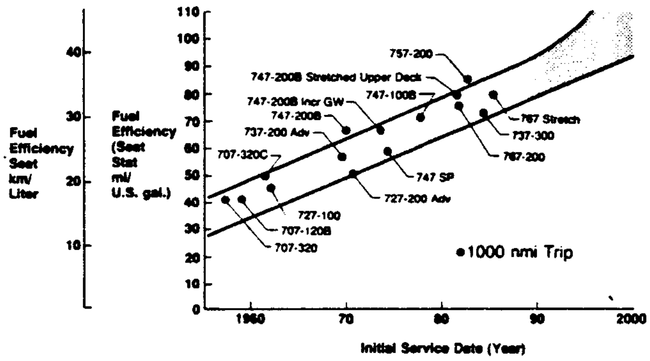

Figure 13

Airplane efficiency, in terms of seat miles per gallon, has increased almost *30 percent per decade* over the past 25 years. The individual technical trends just examined provide confidence that efficiency will continue to improve at this rate through the end of the century.

IV. Program Decisions Revisited

In our experience, we've seen that forces playing on aircraft decisions have frequently changed, often with little if any recognition or warning. Such circumstances can rapidly turn a seemingly sound program decision into a disaster...also, the reverse has happened.

Nonetheless, industry experience has shown a track of predictability through all this, and while not "golden," this track is useful when considering future decision environments that can be anticipated on the road ahead.

4.09

Much of this track, I believe, will be illustrated by the decisions and forces highlighted in the ten programs that we shall examine next. Of necessity, the coverage of each is brief and quite selective.

Boeing 377 Stratocruiser

The 377 Stratocruiser program was a post war commercial offshoot derived from the C/KC-97 series tanker/transports. Like the C-97 series, the 377 incorporated advanced systems that had been developed for the B-50 bomber. The lower portion of its double-bubble design was a B-29 circular cross section. The upper section was superimposed onto the lower as shown in figure 14.

Boeing 377 Cross Section

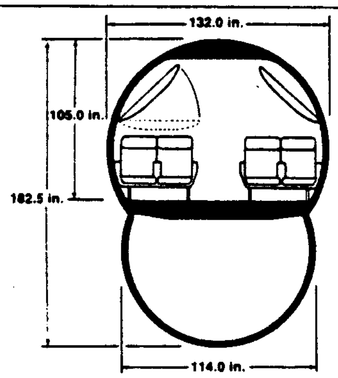

Figure 14

The illustrated design compromise yielded a narrow upper deck floor that proved to make five abreast seating virtually impossible. Similarly, it provided more height than necessary for the lower lobe. This in turn was used for the "lower deck lounge" extrapolation that amplified the airplane's luxury theme. The airplane in flight is shown in figure 15.

Boeing 377 Stratocruiser

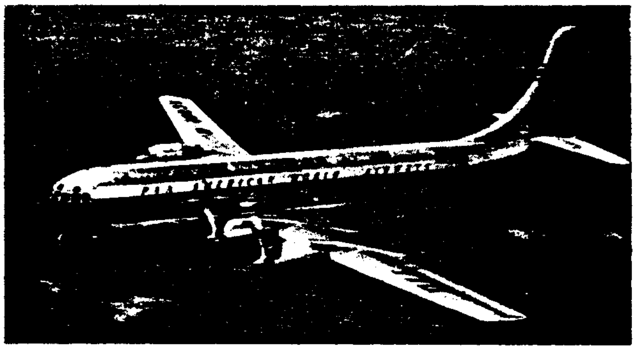

Figure 15

The Stratocruiser was a very large airplane with respect to its passenger capacities, which varied between 50 and 100 depending on route length and service class. Its R-4360 power plant represented the latest in piston engine technology, which at that time had been pushed to its limits with propeller combinations and super-octane fuels for added power gains.

The Pratt & Whitney R-4360 "corn cob" was a four row radial configuration of 28 cylinders. It had 112 spark plugs that were subject to very frequent fouling and change. In total it proved to be a complex and expensive engine, particularly during its introductory years.

The real problem, however, was not the engine but the propeller, particularly Hamilton Standard's. It incorporated new technology developments, which by today's standards, suffered from inadequate validation testing. The blades were constructed with an internal steel tube and a foam-filled external aerodynamic shell. Inspections to maintain airworthiness to commercial standards proved impossible, and most Stratocruiser accidents were due to magnesium housing fires or blade failures, with resulting imbalances that could tear off the engine. Pilots learned to fear the onset of any suspect vibration. In one such instance, a Northwest Airlines 377 was recovered after a water ditching in Puget Sound. Inspection revealed that in this case the propeller was blameless, and the feared vibration had been caused by cowl flaps left in full-open positions. Figure 16 summarizes the power plant difficulties.

377 Propulsion System Observations

- R 4360 engine an ambitious P&WA/Government program yielding higher thrust but lower reliability
  - No commercial experience
- Required major modification program to correct most engine performance and reliability deficiencies
- Hamilton Standard steel tube/foam core blade construction problems
  - Foam core delamination
  - Fatigue inspection difficult
- Implications of potential propeller blade fatigue were very serious
- Alternate propellers were available and were used in some cases

Figure 16

The Stratocruiser achieved a fine reputation for luxury but remained deficient in operating economics and power plant reliability. This led to an early production termination, and, of course, the program was a dismal financial experience for Boeing. I believe, of the many program decisions made, the following are of particular significance to this discussion.

1. The decision to proceed with a new airplane program with success expectations overly dependent upon luxury markets rather than operating economics. Thus, success was premised on premium fares and the higher-income travelers. The airplane could not stand up to competition of air fare reductions that were to become the real stimulants to U.S. travel growth.

2. In part, it could be said that the program's go ahead was justified as a means of holding a military design team together and also in providing that team commercial experience and presence in the post war era. However, it is doubtful that this could stand as a relevant consideration for a U.S. manufacturer today.

3. The Stratocruiser's power plant decisions suffered from use of technology that had not been sufficiently proven. More fundamental, however, was the fact that reciprocating engine technology had been extended *beyond* the limits of its operating efficiency to become overly expensive, complex, and unreliable. As such, this program describes a decision consideration that will remain highly relevant for decades.

B-52 Program

The B-52's concept was derived from the most significant advancement of post war aviation...the *revolutionary integration* of a swept wing with the axial flow compressor, achieved with the Boeing B-47. This development made the jet engine's potentials for high speed flight possible.

Thirty years after its initial flight, the B-52 remains the backbone of the nation's long-range bombardment capability. Almost 750 were produced, and the later models have been continuously updated since production ended in 1962. It is expected to remain a significant component of U.S. strategic forces, possibly into the next century.

The program was started as a large, straight-wing turboprop. It was to become the U.S. second generation long-range bomber, capable of carrying 10,000 pounds for 10,000 statute miles or, by Air Force rules, an operating radius of three-eighths of this (3,800 nautical miles) without refueling. By 1948, it became evident that the necessary engine and propeller for its mission were unavailable, and Boeing was hurriedly asked to provide the Air Force its concepts for a jet alternate. Figure 17 illustrates the evolution of the B-52 from that point into its jet-powered design.

Early Days of the B-52 Evolution

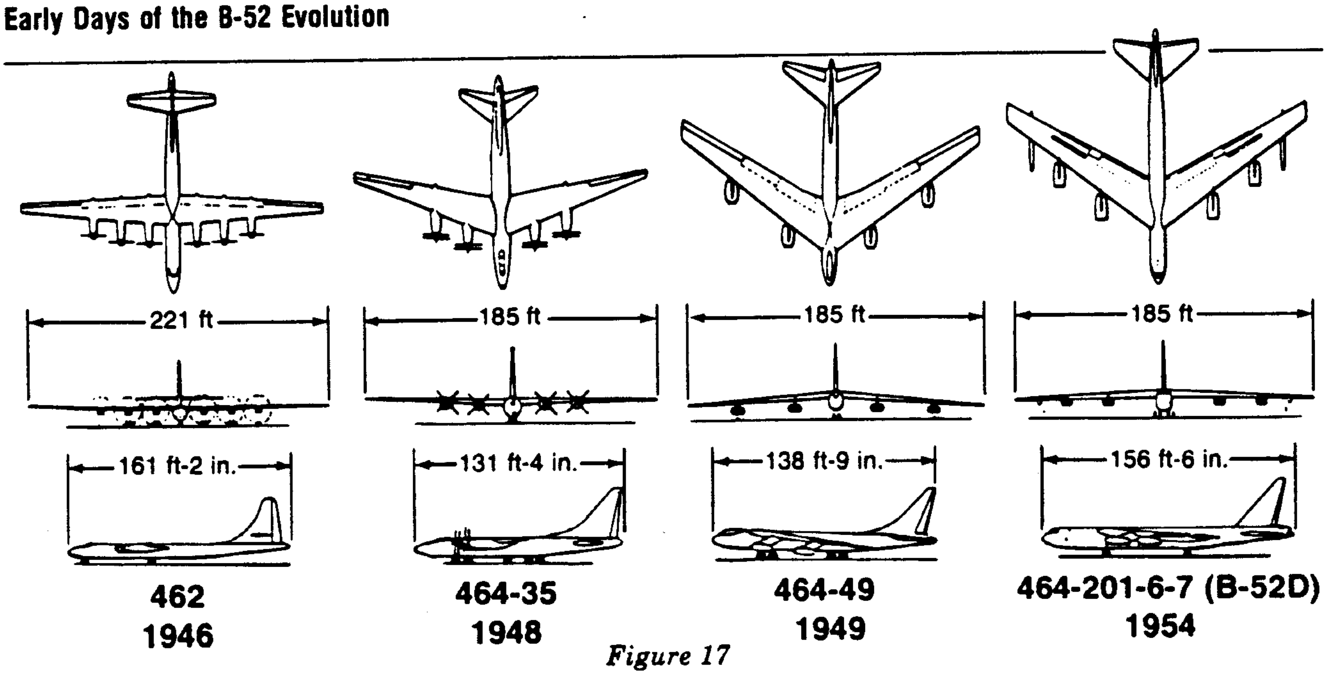

Figure 17

4.11

Boeing's successful work in developing the swept wing B-47 greatly encouraged the change, and really validated much of its concept. However, the B-52 involved other problems and the feasibility of proceeding with a jet configuration was also leveraged by the many rapid technological advancements made that were really unrelated to the B-52 objective, or for that matter, any specific design application objectives. One was the independent development of the J57 two-spool jet engine. The full significance of this to U.S. aviation is noted later. The main point, however, is that a new military engine was under development *before* its application was known. The timeliness of the earlier J57 start, of course, resolved the B-52's propulsion dilemma.

Similarly, aerodynamic advancements (unrelated to the B-52's needs) had occurred. Much of this technology readiness work was done by NACA, and much was Boeing's independent high speed airfoil work. Both were major contributors in providing the improved understanding of swept wing technology that was needed to support the jet bomber decision. Boeing's work, for example, had earlier revealed that the wing root could be grossly thickened without adversely affecting the high Mach number characteristics of the integrated airframe. This discovery, illustrated in figure 18, allowed the use of very long span wings on the B-52 without excessive wing weight.

B-52 Wing Thickness vs Span

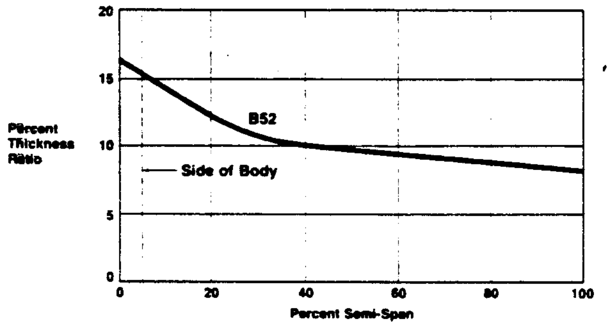

Figure 18

The program involved a prototype phase to validate the application of such advancements. There is much controversy today as to prototype cost-benefit relationships. Military acquisition methodologies of the time routinely included them, and the B-52 prototype cost in relationship with that of the total program is shown in figure 19.

B-52 Program History

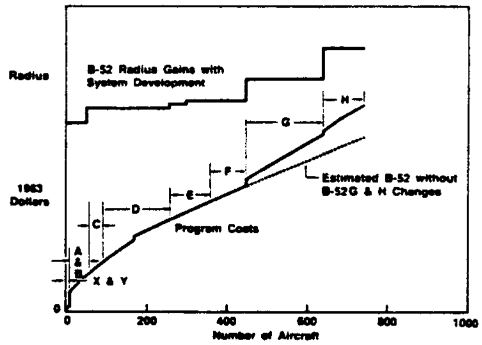

Figure 19

In terms of total program costs, the X and Y prototypes were insignificant. The B-52G and H series have been updated to incorporate a variety of offensive and defensive capabilities, including those shown in figure 20. In total, the airplane has exceeded its original design capabilities significantly and has been redesigned to perform missions for which it was never originally intended.

B-52 Program Phases

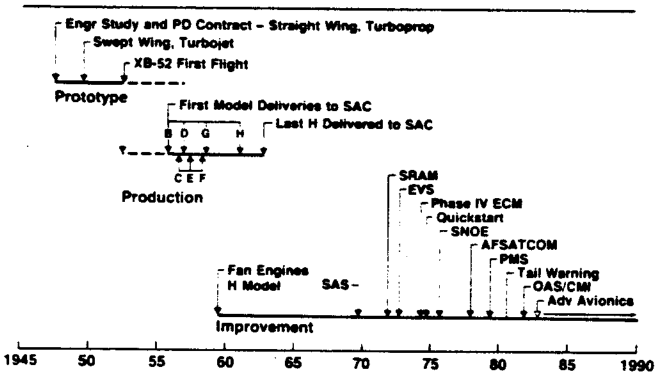

Figure 20

Improved capability costs have no doubt exceeded those of the initial development and prototypes several times over. Nonetheless, the total is probably far smaller than for production of an all-new airplane. I think the fundamental reason that the B-52 has remained in the Air Force's inventory plans for so many years is because of a fundamental change in the technological environment that was of its creation. With this in mind, the following decision points are illustrative.

4.12

1. The *then* acceptable government decisions that permitted funding of a new engine development with the intent of new technology readiness, but without specific applications defined. Similarly, the *then* acceptable decision environment that permitted the development of aerodynamic technology *and its validation* to the extent possible, again without the constraint of an audit track to a specific application.

2. The subsequent era of government decision making that has constrained the flow of aeronautical research to support the orderly and timely development of new advancement potentials. This started in the 1960s, at about the time the last B-52s were produced. Within a short span of years, NACA was recast into a different role, military research diminished and the nation has yet to achieve its third generation long-range bomber.

Lockheed Electra/Orion

The Electra program started as a four engine turboprop designed specifically for the medium and short range market beneath the 707 and DC-8. The go-ahead decision was made in 1955 in response to a requirement issued by American Airlines. First delivery occurred in October of 1958, the same month as the delivery of the first 707 commercial airplane. Although the Electra had some wing and power plant structural problems that bothered its early years, it developed into a technically successful commercial airplane. However, timing of the program was poor with respect to the emergence of the jets, and this would have been a financially disasterous program except that Lockheed cleverly exploited its broad product capabilities to use the airframe as an efficient naval surveillance platform. As the military Orion, the Electra has been in production since its first delivery in 1962. The combined program timings are shown in figure 21.

Lockheed has sold about 600 units, and the Orion is now also produced under license in Japan. The Orion military system in its various models was purchased by the U.S. Navy and also the military forces of Australia, Canada, Iran, the Netherlands, New Zealand, Norway, and Spain. The U.S. Navy version is shown in figure 22.

Lockheed Electra/Orion Programs

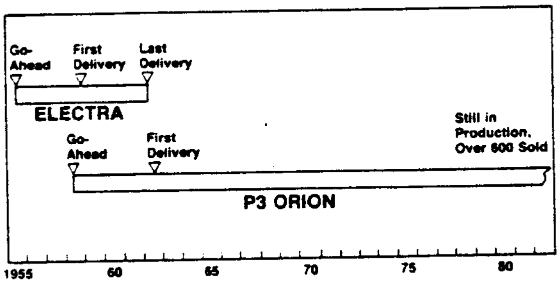

Figure 21

P-3 Orion

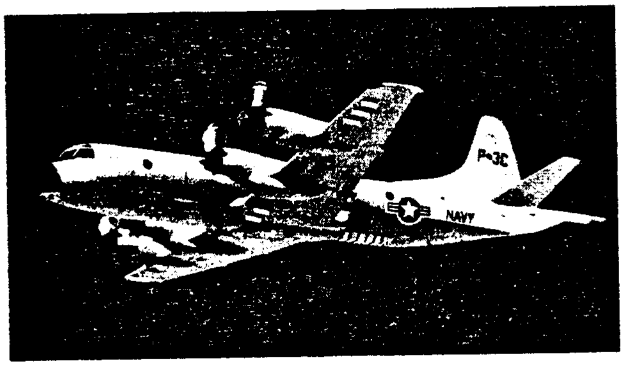

Figure 22

This program illustration exemplifies the hazards of new program starts with respect to market timing and technological obsolescence. The Electra came too late and with the wrong technology for its intended commercial market. It also illustrates a unique situation of extrapolating a failure into a remarkable success. As such the following decisions were key:

1. Failure: The decision to stay with better known technology and launch a four engine turboprop after the Caravelle (a twin engine jet) was launched for the same market. The power plant selected was the Allison 501. It should be noted that more efficient and proven commercial turboprop engines were available from the British industry. However, there was at that time a reluctance to become dependent upon an engine supply line extending across the Atlantic, and consequently a lesser experienced U.S. manufacturer was chosen. We'll discuss engine decisions later, but the point serves now to illustrate that supplier

decisions, if anything, have become more significant today, and programs can readily become win or loss situations by virtue of such decisions.

2. Success: The Orion's success, of course, represents another series of sound decisions that have tracked its long production life. The first of these was Lockheed's decision to market an airborne surveillance system designed around the Electra's obvious competitive advantages in this role. Thus, a competitive failure in terms of commercial requirements was reversed with its military mission.

707 Program

After its Stratocruiser experience, leaving propeller problems to others and proceeding with jet designs was not a difficult decision for Boeing to make. However, company efforts at convincing the military tanker people and commercial airlines that "jet was right" proved fruitless. Finally, after two frustrating years, Boeing's Chairman Bill Allen okayed the go ahead for a company-funded commercial prototype to demonstrate our conviction. This one-of-a-kind, the 367-80, was the beginning of the 707 and KC-135 programs. It was a big day when it rolled out, as shown in figure 23.

Dash 80 Roll Out

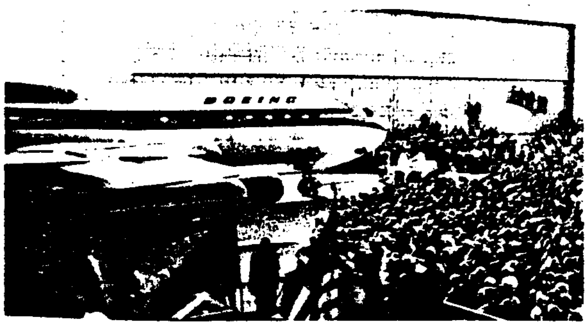

Figure 23

The Dash Eighty's first flight occurred on January 15, 1954. However, between go ahead and this event, the Air Force initiated a design competition for a tanker, not unlike the paper version that Boeing had tried to sell the Air Force earlier. Boeing lost the competition (partly due to Boeing's prototype knowledge) but Boeing delivery guarantees (with a prototype in hand) were irresistible, and by late 1954 Boeing was awarded the KC-135 production contract. The win was viewed as an opportunity to gain tooling that would have commonality for commercial production, and this became an influence in the increase of the cross section diameter from the prototype's 132 inches to the KC-135's 144 inches. Douglas came on to the market with a DC-8 which had a slightly wider body that was preferred by certain key airline customers.

The commonality decision was very right at a later time, but in this case it proved wrong for commercial competitiveness, and this forced a very costly redesign to a 148 inch cross section for Boeing. Figure 24 illustrates the three body widths involved.

Body Cross Section Evolution

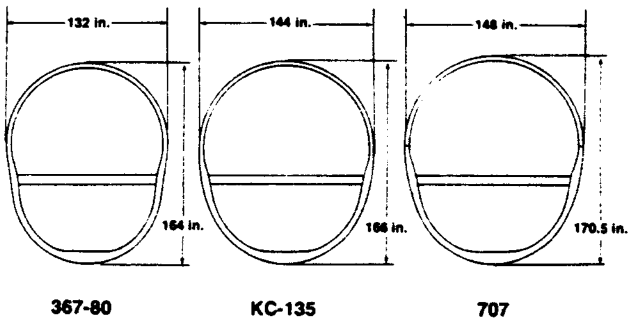

Figure 24

The DC-8 had also offered a larger and longer range wing, causing another major redesign headache and also invalidating what little was left in tooling commonality with the KC-135. The new wing change is illustrated in figure 25.

Wing Planform Comparison
707-120, 320

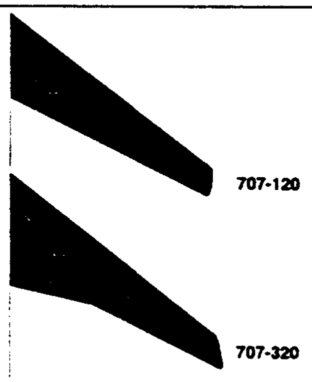

Figure 25

It was obvious from all this that gambling a quarter of the company's total net worth on building a prototype was only the start. Douglas was firmly entrenched in the commercial business and Boeing could either fold or increase its risks to obtain additional customers. Thus before delivery of the first 707, the production program involved two different wings, two body lengths, and two engines. By this time it also involved a commitment to build the 720, a lighter and shorter derivative. There were others as well, and a composite break-even situation developed as shown in figure 26.

Risk-Breakeven 707/720 Program

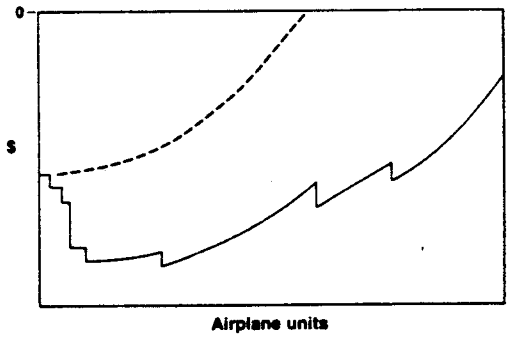

Figure 26

What actually happened is illustrated by the lower line of figure 26, as compared to the expectations which are represented by the upper line. Despite the turbulence of its launching, the program did evolve into a success and, as we've described, some of its more significant decisions were made in the program's early years as competitive pressures from the DC-8 exposed the 707's design vulnerabilities. I would consider the following decision points key to the 707's successful outcome.

1. The commitment (and collective decisions) that were to keep the 707 competitive. Much of this, as noted, was forced by specific competitive actions, and as such, Douglas really made some of the decisions for Boeing. However the commitment was fully a Boeing decision. It really preceded the 707 and has extended much deeper and far beyond the scope of this one program. *That commitment was to become a viable commercial competitor and to remain so.* It sustained the subsequent development of the Boeing jet transport family and over thirty years of continued product advancement. It has involved a continuity of discipline in design and production quality and the formation of a global customer support capability.

2. The second key decision was to proceed with the development of a company funded prototype after the unsuccessful and discouraging efforts in selling paper iterations to either the airlines or military. The prototype was invaluable, not only in paving the way for the 707's commercial acceptance, but for validation of the new technology integrations into its design. We may see this decision resurface in civil transport development within the next two decades.

Convair 880 Program

Convair emerged from World War II as a highly successful builder of the CV240, 340, and 440 series of transports...all for the short-range market. By the mid-1950s, Convair dominated this market segment, and knew its requirements better than the other U.S. manufacturers. Beyond the 440, the company was considering turboprops for additional short-range offerings, either as re-engines for its current designs or possibly as an all new airplane. With the exception of Sud's Caravelle (a tail-mounted twin jet), turboprops had become the primary short-range product focus on both sides of the Atlantic. However, despite its short-range market expertise, the company became enticed with a proposal from Howard Hughes to undertake the design of a big long-range jet transport for TWA. Working with Hughes was difficult, and by 1955 it became apparent that decision procrastinations had left Convair with no hope for a chance in the long-range market. Instead, the company lowered its sights onto the medium-range area with the CV880, a four engine design, thereby deferring its much better short-range market opportunities to other contenders. The timing of Convair's 880 decision with respect to this is illustrated in figure 27.

Go ahead was authorized on an order base of forty...ten from Delta and thirty from TWA. The CV880 was of sound technical design, but it persisted in a five abreast cross section, despite market objections. This made it vulnerable to the six abreast capability that Boeing countered with in offering the 720. The cross section comparison is shown in figure 28.

Competition

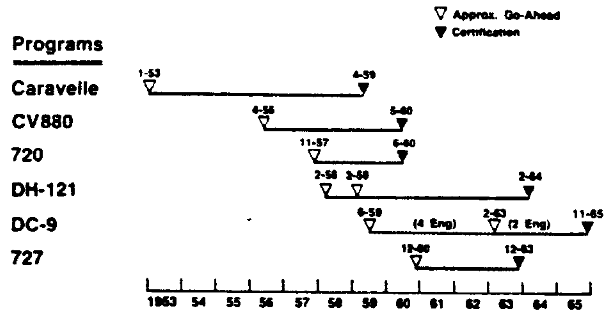

Figure 27

Cross Section Comparisons

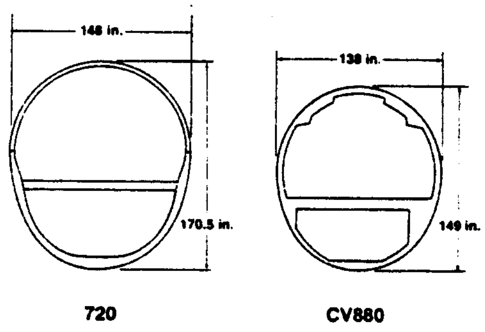

Figure 28

The noted difference in body diameter gave the 720 lower operating costs per passenger mile, and this, along with 707 fleet commonality advantages, virtually eliminated the CV880's market opportunities. Convair tried to recoup with the CV990, a larger version designed around an up-rated aft-fan version of the General Electric CJ805. The CV990 again was a good technical airplane, but continued disputes with Hughes over the CV880 and the high investment and concessions made to American in selling the CV990 were too much. Convair was, for practical purposes, through with the commercial aircraft business.

There were many decisions that affected Convair's fortunes with the CV880, but the following are particularly noteworthy with respect to the initial go ahead.

1. The decision to proceed with a four engine medium-range configuration when an intermediate-range twin-jet competitor was known to exist. Had Convair at this time moved directly into the short-range market, Boeing and Douglas could not have countered because of their other jet aircraft commitments, and the 720 could not have competed against a good short-range entry.

2. The decision to go ahead with the five abreast cross section when opposition from United, a key potential launch customer, was well known. Convair's firm position was not based on passengers; rather the five abreast decision was considered as an aerodynamic and performance solution to satisfy both transcontinental range and short field requirements.

3. The decision to proceed with a small order base and design that were both dominated by Howard Hughes. The pecularities of Hughes' business arrangements were well recognized at the time. In fact, Convair had suffered first-hand experience a few years earlier in an aborted piston transport sale.

deHavilland Trident Program

The Trident was a sound high technology configuration executed by a very competent team that started design work substantially in advance of the 727. Unfortunately, the program was delayed as design requirements became oriented toward the specific needs of British European Airways, a government-owned airline, and also because the British Government was restructuring the industry, which was creating uncertainty as to the future management of the project. Government policy of the day was forcing the domestic carrier to purchase British equipment. The BEA needs produced a tight body cross section, which the earlier 707 and DC-8 competitions had proven as unacceptable to the U.S. market. A comparison is shown in figure 29.

Cross Section Comparisons

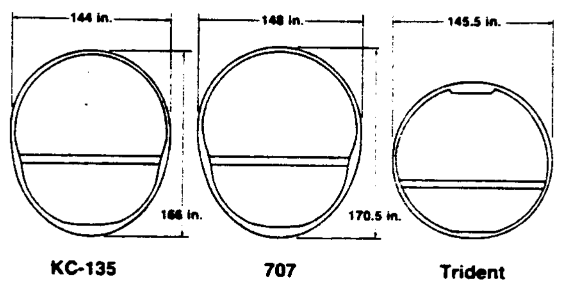

Figure 29

4.16

BEA's payload-range requirements and less demanding takeoff field length scaled down the Trident's original design, and at six abreast seating, this produced exceedingly tight shoulder clearances and very narrow swing-in entry doors that wasted floor space. This was all unacceptable to most airline customers. It also caused abandonment of the RB141 engine and start of a scaled-down engine, the RB163 Spey. Technologically, the airplane was very advanced, as noted in figure 30.

Trident Technical Features

- An integrated, clean-wing aft engine configuration having good characteristics in all flight regions
- A very advanced flight control system, combining triplicated power controls, an all-movable horizontal tail, and flight instrument innovations
- An improved bonded structure technology incorporating advanced fail-safe and safe-life techniques
- A new and efficient fan engine; the product of an orderly Rolls-Royce development program

Figure 30

Hawker Siddeley later produced stretch versions which sold at home and abroad. However, despite its technical soundness, only 117 Tridents in total were purchased by some nine airlines, and it was a financially unsuccessful program. It would be speculative to say just how much the British Government's policies flavored the program decisions that were made. Nonetheless, the following were key factors that affected the Trident's opportunity for success.

1. The decision (or decisions) that tailored the airplane's design to the needs of a single customer when only a cursory examination of the world market would have revealed differing requirements.

2. The decisions that cumulatively caused development to stretch into a six year program, thus allowing competitive aircraft to offer earlier deliveries.

727 Program

The 727 program began in May 1958 with a task force effort to identify the technology and configuration that would make a successful short- to medium-range commercial jet to complement the 707. The first designs produced a miniaturized 707, just as the initial "DC-9" was configured as a miniaturized DC-8. For economic reasons, we felt a two engine configuration would have better economics and our prime considerations in 1958 and 1959 were for two engine configurations having engines mounted under the wing. The torturous configuration path that covered the two-and-a-half year period is illustrated in figure 31.

727 Development History

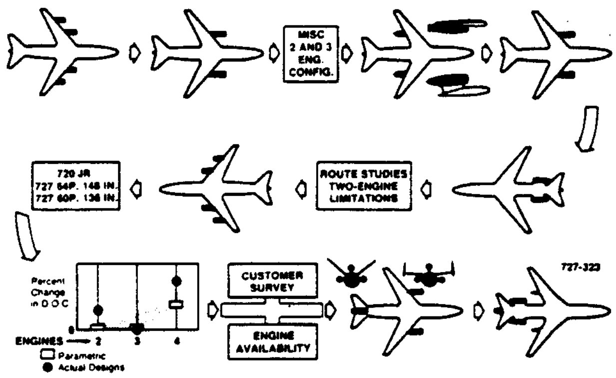

Figure 31

Conditions at that time precluded a European launch customer (due in part to the Trident start), and of the four major U.S. airlines, only United and Eastern were interested or financially capable. Eastern wanted to maximize economics with a two engine airplane, and United was seeking a four engine airplane because of their high altitude Denver requirements.

One must include a mention of the excruciating pain of trying to achieve a common denominator among varying airline requirements. All commercial programs go through a similar process and the engineers must work with a great many airlines, not just the few who are most likely to become program launch customers. It is a painstaking and iterative process, as illustrated in figure 32.

As this occurred on the 727, two other mainstream technical efforts were also proceeding. One was the development and wind tunnel verification of the many potential designs under consideration. The other was on-going technology staff developments that were independent of the program. The latter produced a triple-slotted flap which could yield a higher

One of Life's Frustrations

Figure 32

lift coefficient than any swept wing existing or contemplated. A three engine design was accepted as a compromise by both United and Eastern. The triple-slotted flap was incorporated to meet the short field requirement imposed by New York LaGuardia Airport runway 4-22.

The airplane really had two lives, and production rates have been highly variable as shown in figure 33.

727 Delivery History

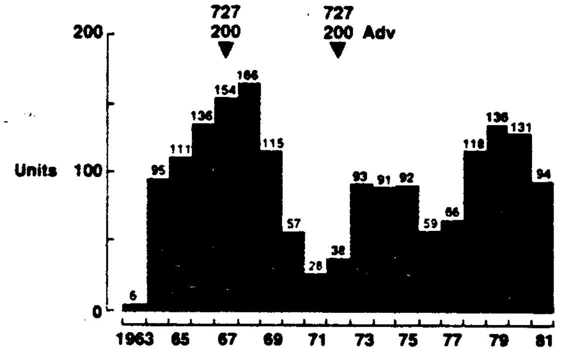

Figure 33

The first life included the original 727-100 and the early version of the 727-200, which made a fundamental mistake of adding body length at the expense of range, resulting in an airplane of limited performance and partially unusable economics. In late 1970 this was rectified by introduction of the "advanced" 727-200 having a higher gross weight, an upgraded engine, a new "wide body" interior and a variety of other improvements. Keeping the airplane competitive over its long production has entailed continuous nonrecurring investments, as illustrated in figure 34.

Cost of Nonrecurring Product Improvements as a Percent of Initial Nonrecurring Cost (Current Dollars)

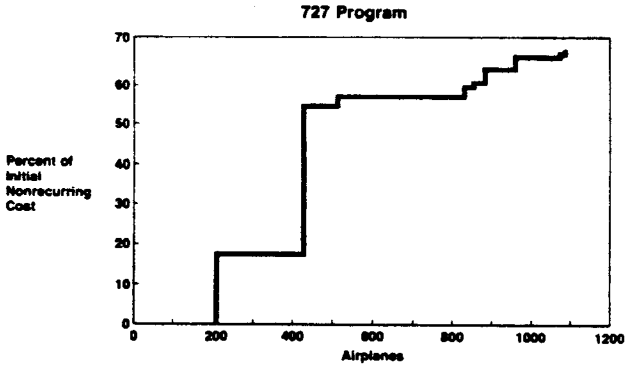

Figure 34

The cumulative investment for improvement has continued with production beyond that illustrated, and currently approaches 100 percent of the cost for initial development.

Evaluations of the 727's estimated market share were carefully made by Boeing's management before the initial production program was authorized. History has shown how wrong and how right these estimates were. This is illustrated in figure 35.

727 Market

Estimates vs Actuals

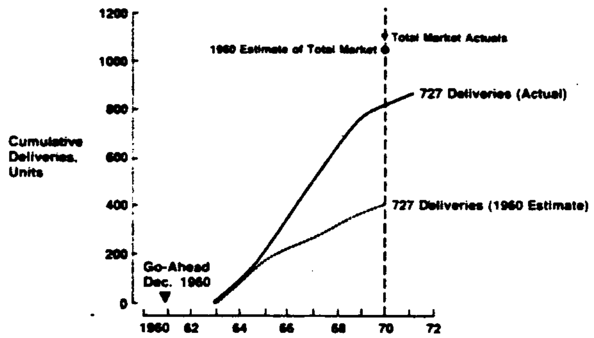

Figure 35

As noted, the 1960 predictions were that by 1970 the total market would stand at slightly over a thousand units, with the 727's share at about 400 units. These forecasts were made on the assumption that the 727 would face a U.S. competitor as well as Caravelle and Trident.

The 727's market position has held on through its "second life" with production approaching 2000 units. Some wrong decisions made by competitors and for the most part, some good

4.18

decisions by Boeing ultimately gave the 727 the enviable position of being unchallenged. The following decisions by Boeing at the time of go ahead were key. Given different circumstances, they could have become "wrong" decisions, since they represented considerable risk in terms of the company's total situation at the time.

1. The decision to seek a satisfactory (three engine) middle ground solution between conflicting major customer desires for a two engine and a four engine aircraft. (Despite Boeing preference for a two engine configuration).

2. The decision to push state of the art in order to achieve desired competitive performance objectives within the intended market.

3. The decision (and its timing) to commit a major production program for a new state of the art aircraft in a situation of turmoil and without a prototype.

DC-9 Program

Douglas launched the DC-9 in 1963 as a direct competitor to the BAC-111. The British program had a two year lead, and with flying prototypes, it successfully penetrated the U.S. market. The Douglas program was started without a prototype and with Delta as the only customer.

The initial DC-9 series 10 was aimed at the BAC-111, not at a potential Boeing entry. The possibility of Boeing's 737, to an extent, caught Douglas by surprise. Rather than risk defeat by unexpected competition, Douglas decided to "stay with the game." The pace of basic model development increased, and to stall Boeing, work on a *major* improvement derivative (the DC-9 Series 30 for Eastern) was authorized. Douglas successfully curbed the BAC-111 delivery advantage, but the Series 30 derivative failed to stop Boeing's initial 737 sale made in February 1965. Additional stretches were initiated, and Douglas maintained the order advantage. However, the DC-9 market share successes also incurred financial penalties which contributed to the Douglas take-over by McDonnell in 1967. Figure 36 illustrates the DC-9 program timing, and the ambitious pace of its developments before this happened.

DC-9 Program

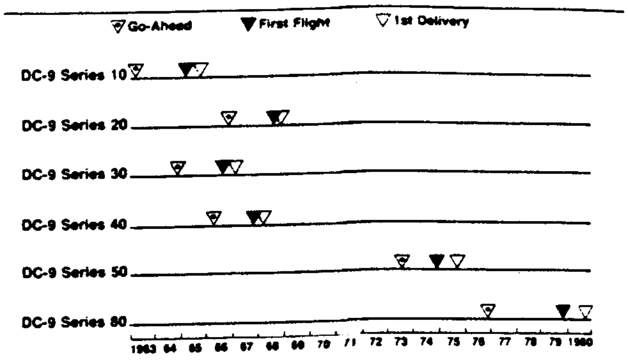

Figure 36

As noted, McDonnell Douglas has continued to stay with the game, and the DC-9's growth has nearly doubled passenger capacities over that of its initial model.

The DC-9 uses a cross section totally different from that of the DC-8, and this was probably a correct decision for the time of its launching. To an extent, however, its smaller dimensions opened the way for a 737 start with cross section and parts common with the 727. This plus the 737's conventional tail and wing-mounted engines produced a much shorter airplane with respect to passenger capacities and made growth versions easier. Figure 37 illustrates the cross section differences between the BAC-111, DC-9, and 737.

Cross Section Comparisons

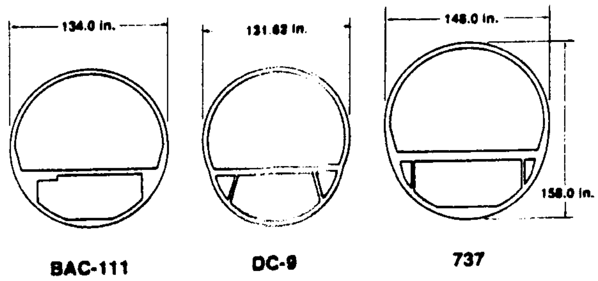

Figure 37

The DC-9 decision had much in common with the 707-320 decision made by Boeing in its earlier competition with the DC-8. It was a matter of staying in the game with additional major nonrecurring investments.

This unquestionably produced better airplanes by both. The real market loser in the DC-9 case was the initial market entrant, the BAC-111. With respect to this, the following DC-9 program decisions are noteworthy.

1. Douglas had correctly assessed its late-start market opportunities in U.S. market competition against the BAC-111. The Douglas airplane featured an upper deck cross section affording superior passenger appeal features (significant to this market), whereas the BAC-111 used the more restrictive circular body favored by the European industry.

2. Boeing probably would not have started the 737 had Douglas initiated the DC-9 program with an airplane more resembling its Series 30 derivative. In this matter, Douglas may have incorrectly assumed that the BAC-111 was the only competitor.

Supersonic Transport Programs

Even as the first subsonic jet transports were developed, the commercial potentials for supersonic flight came under serious study in the four nations that fostered their development. The costs for development were recognized to substantially exceed any civil aircraft program previously accomplished. As such, this caused British and French interests to merge into the Concorde. On this side of the Atlantic, funding required direct U.S. government sponsorship, with a series of competitions that selected Boeing as the airframe manufacturer. The Soviets operated in a manner conventional to their style, with the government assigning the SST task to Tupolev. However, the decisions surrounding the U.S. and European programs were unconventional, and the timing of the two is shown in figure 38.

Supersonic Transport Programs

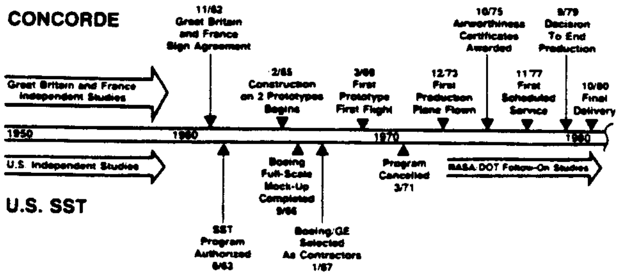

Figure 38

The Concorde, of course, is still in service, but the decision to terminate production was made several years ago as the market could no longer substantiate its economics. The earlier termination of America's SST, in my mind at least, has proven with hindsight to be a right decision but it was made for all the wrong reasons. Unfortunately, these wrong reasons have afflicted the pace of U.S. technology ever since, and the nation has suffered deeply because of this.

Both programs were conceived at a time when fuel prices (in constant dollars) were tracking a downward path. Both were known to be sensitive to fuel, since supersonic cruise requires more energy per unit of payload and range. The subsequent increase in fuel price made the U.S. decision "right" nearly two years after it was made. Nevertheless, the Concorde, shown in figure 39, has been providing safe and reliable Atlantic service since 1976.

Concorde

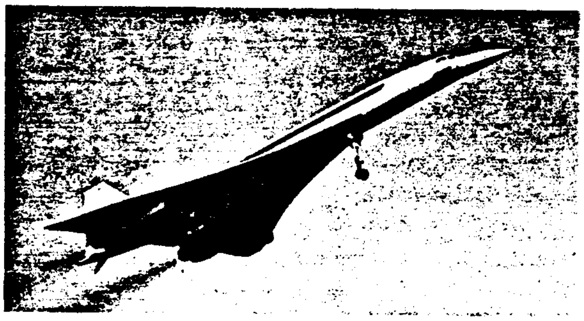

Figure 39

The funding nations, including the U.S., have reaped enormous legacies from their SST programs. A partial summary of the achievements derived from the U.S. program is noted in figure 40.

SST Spin-Off Contributions

- Modern flight deck technology, now being introduced on new generation commercial and military airplanes
- Large scale application of computers to aeronautical engineering problems
- Titanium alloy developments and new structural concepts
- Miscellaneous developments: lightweight seats, fuel tank sealants, noise reductions, guidance, hydraulic tubing, etc.
- Augmented flight control systems having current military and future commercial applications (relaxed static stability - active controls)

Figure 40

4.20

The European experience is similar, although of course, the actual SST production and service have enriched their development knowledge extensively. Both programs are graphic demonstrations that high technology efforts, regardless as to how they are sponsored, will find applications to generate values not thought of at the time.

However, the diverse decisions surrounding these two programs warrant two final observations with respect to their legacies:

1. The Concorde program proved that a large-scale international program could be made to work. Much was wrong, but it forced solutions that paved the way for internationalism that is becoming widespread today.

2. The U.S. decision, unfortunately, may have accomplished the opposite. It appears to have validated a growing trend of public and government opposition to technology that made the 1970s a decade of drought for U.S. research and development. We are reaping the bitter harvest of such decisions today. More specifically, the "wrong" aspect of the SST's cancellation was that, in the absence of a supersonic long-range bomber, it ended the idea of government supported high risk prototypes. Its completion would have made the B-1 and F-16 into better programs and the Space Shuttle a cheaper program. It also would have allowed the earlier introduction of many advancements in new subsonic airplanes.

747 Program

The 747 was conceived at a high point in world travel growth. Mass travel markets were in rapid expansion with the air system and major airports approaching capacity limits. The objective was to design a "super plane" that would capture high performance and low seat mile costs by its economy of scale. The airplane was intended to leap-frog the DC-8-63 and also to be oversized at introduction. It was intended that it become a "market fit" about four years after introduction. Such philosophy guided the DC-8 and 707 developments, and is illustrated in figure 41.

747 Sizing

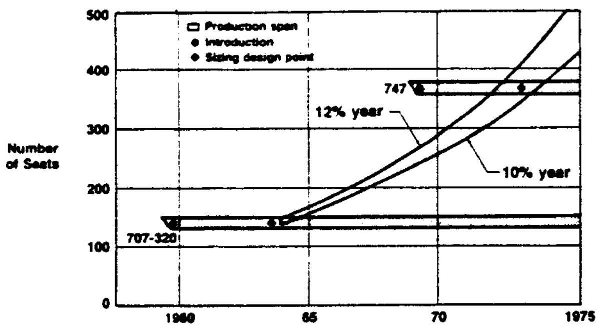

Figure 41

The risks were of a magnitude scale also, and most were recognized before go ahead. It required a new factory concept, and the 747's success would be dependent upon the development of the new and untried JT9D engine. Furthermore, all of the critical elements (factory construction, airplane design, and engine development) had to proceed concurrently to meet delivery schedules. On the plus side of this was the fact that the 747 was establishing a new size platform, one that competitors would be hesitant to challenge.

The expected emergence of the SST as a principal long-haul passenger transport was a significant consideration in both sizing and configuration. The body width had to be sized for freighter efficiency in the event this became a principal job as the SSTs took over passenger service. The resultant cross section, shown in figure 42, had little to do with passenger appeal in its selection, but was marketed as a great passenger comfort "breakthrough" by use of an innovative mockup and promotional campaign.

Body Cross Section Comparison

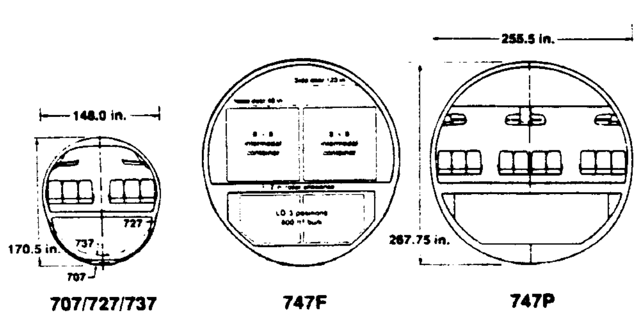

Figure 42

The airplane was designed for higher speed objectives to improve productivity without significant erosion to seat mile costs, which were to be substantially below that of any vehicle flying. Scale, of course, was a major factor, as figure 43 illustrates. However, the results exceeded those of scale effect alone.

Direct Operating Cost Comparisons

1979 U.S. Domestic Rules, 1979 Dollars, 1000 nmi Average Trip

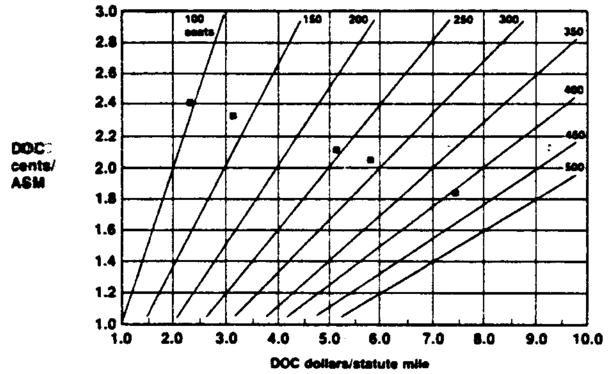

Figure 43

Bringing it all together on time was unquestionably the largest aircraft task (and the greatest risk) that Boeing had faced. The rollout is pictured in figure 44.

The Boeing 747

First Airplane Rollout

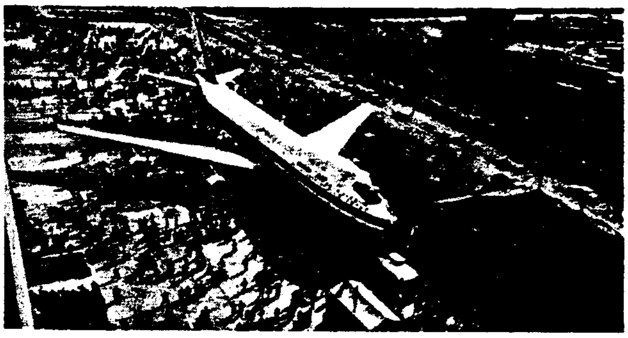

Figure 44

The airplane proved to have extremely good flight characteristics once initial bugs were corrected. Considering its uncontested "platform position," the program has gone on to contain an unusual array of product improvements and derivatives, including some 20 engine options. Figure 45 illustrates this, showing the derivative models available.

Current 747 Family

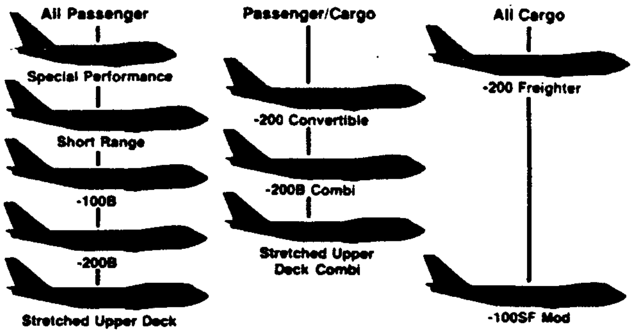

Figure 45

The additional nonrecurring costs for all this have fairly well tracked the 727 program experience noted earlier, by nearly matching initial development costs and following a similar time pattern as well. Critical decisions abounded on this program, in character with its size. However, the following are noted because of their significance to subsequent wide body programs.

1. The decision (before actual go ahead) to rebuild the entire production and management control systems of the company. The 747 really ushered in a new era of production management for Boeing, and without such, the later concurrency of the 767, 757, and 737-300 programs would have been unthinkable.

2. The decision that risked success on the concurrent start and development of both a new airframe and a new engine.

3. The decision to size the airplane to a relatively high assumption of market growth, with a relatively secure confidence that its scale would not be easily challenged.

Head-on Competitions

Many conclusions and contradictions can be drawn from the ten programs we've just covered. However, on the commercial side, decisions that surrounded head-on competitions have appeared particularly critical, and worthy of some special

4.22

observations. For such, I've selected the four sets of competitions illustrated in figure 46.

Head-on Competitions

CV240 M202	707 DC-8
DC-10 L-1011	767 A310

Figure 46

We noted Convair's successful piston experience earlier. But the first of its post war pistons, the CV240, appeared in the short-range market as a direct challenge to the slightly earlier Martin 202. The objective of both was to replace the DC-3 in a market that was fairly small but had good opportunity for expansion. In this case the M202 was underpressurized and had some structural deficiencies, faults that the CV240 avoided. The market was too small to carry both, or to tolerate fixes. Consequently the M202 was forced to terminate production at 31 units. Convair sold over 570 of the CV240s, and produced a total (including derivatives) of over 1,100 aircraft.

The 707 and DC-8 competed in a market that was sufficiently large for both. Boeing was first, but with some mistakes in size and range requirements that were *immediately* corrected. Had this not been done, the competition might have ended differently. However, it should be noted that the 707's ultimate success is due in part to Douglas' decision to terminate DC-8 production in favor of increased DC-10 sales. The 707 did not put the DC-8 out of production. It was a Douglas decision that favored Boeing.

The DC-10 and L-1011 competition involved a different market situation. These airplanes were caught, along with the A300 and 747, with a market expanding at a substantially slower rate than predicted at the time of their launch. Both were "too big", which depressed their sales while benefitting those of smaller aircraft such as the 727. Both are technically acceptable, but have suffered primarily because the market failed to develop sufficiently to support both or possibly even one.

The 767 and A310 competition must be regarded as still in its infancy. However, they are of a size, a timing and a technology that should support both in the market. The two unknowns at this time are the implications of affordability and internationalism on the market and its decisions. We'll cover this in more detail later.

These cases have pointed out the significance of decisions leading to head-on competitions. Mistakes may be tolerated, but only under circumstances of rapid correction. One can't correct the market need however, and head-on competitions in the face of insufficient market size means that one or both competitors may be unsuccessful.

Military vs Commercial Decisions

The purpose of this section is to briefly explain the fundamental differences between the military and commercial environments in which decisions are made. Commercial practices seem to be more streamlined than military practices and the Department of Defense is spending considerable time studying them. However, one must recognize the basic differences between the two environments. These are overviewed in figure 47.

Commercial vs Military Program Relationships

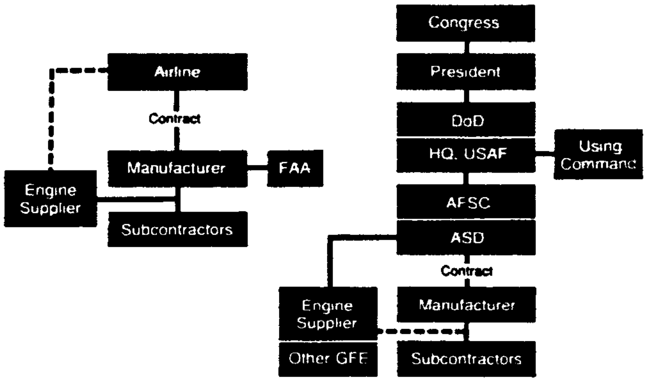

Figure 47

Commercial communication lines are short, including those involving decisions. After the product is agreed upon, manufacturer initiatives dominate. The number of people involved for the airline and for the manufacturer in administrating all aspects of the contract, including technical, is perhaps a dozen or so on each side. The FAA issues type and airworthiness certificates. They also issue a

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contractors production certificate. Their interface with the manufacturer in terms of numbers (not influence) is heavily dominated by Designated Engineering Representatives (DERs) and Designated Manufacturing Inspection Representatives (DMIRs) certified for the duty from the manufacturer's ranks.

The illustrated Air Force program is immersed in a far different environment, one which is not easily changed. The buyer is the Air Force Systems Command and its designated Aeronautical System Division as noted. The relationships are much more complex and formal, and Congressional oversight is maintained on a line item approval basis where major aircraft programs are concerned.

The Air Force must be prepared to publicly defend any decision it makes, which is not a requirement in commercial business *(public accountability vs private accountability)*. Similarly, the Air Force must be prepared for a formal protest on its procurement decisions and has procedures for this purpose. The method for commercial protest to be heard is the loss of future business. The only established process is through the courts, and this is generally avoided. The commercial product relies on a fixed price based on an end item specification, performance guarantees, service life policies, and warranties. The military system relies upon a complex interface in which every decision must be extensively reviewed and documented. Military logistics and spares requirements tend to prevent in-line product improvement except at rare intervals. Commercial practices assume that such improvement is normal, and no approval is needed so long as performance guarantees, price, and delivery are unaffected.

The military system is much more formal and derives advantages and disadvantages from this situation. Because of this formality, I have chosen to illustrate a common civil/military decision situation with a military chart as shown in figure 48.

The situation is oversimplified but is applicable to either the civil or the military case. For convenience, it uses the military definitions which, of course, have their civil counterparts. It omits the military "milestones". The objective of the system is to reduce risk...hopefully to zero when *production* is finally entered.

Risk and Decision Cost Profile

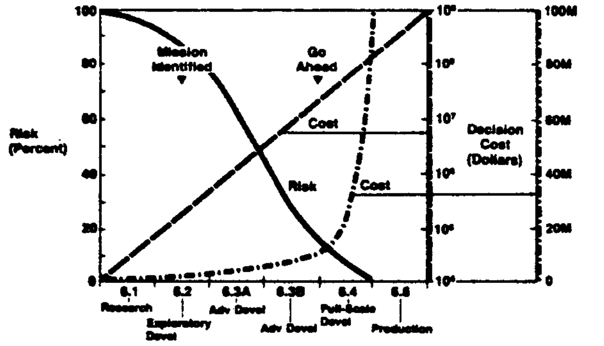

Figure 48

The cost of a major decision is shown on two scales. The diagonal straight line is plotted on a logarithmic scale; the curved line, similarly identified, is plotted on a normal scale. The message (applicable to civil or military) is that pre go-ahead decisions are cheap. But the decisions made after go ahead increase rapidly in cost until they become essentially prohibitive.

Since commercial programs are based on a market price (without relationship to cost), there is a very large incentive to apply large amounts of early capital (before go ahead) to facilitize and train for productivity and low recurring costs. Military programs, on the other hand, have difficulty authorizing large amounts of "up front" money and disincentivize large productivity investment by the contractor. Military program's profit is limited, and, when nonallowables, potential renegotiation, and other contractual burdens are considered, the profit may prove unattractive to some suppliers.

The type of technical and manufacturing decisions to be made are similar. The manner in which they are made is very different and will remain so.

The technology base for military and commercial aircraft is generally the same. At one time the flow was from military into commercial. Today much of the flow is reversed. In addition, the "audit trail" requirements that now force DoD to link research and technology to identified future weapon systems tend to constrain the military input to the base. This tends to increase the military reliance on civil and NASA basic research. Such "compartmentalization" is generally not practiced by other nations in the western world today.

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V. Engine Decisions

The success path of an airplane program, whether commercial or military, is heavily dependent upon the success of the propulsion system. From an aircraft manufacturer's standpoint, engine selection is critical and is sometimes more complex than decisions made on the airframe itself because three parties are involved (the airplane manufacturer, the customer, and the engine manufacturer) and because engines take longer to develop than airplanes. The purpose of this section is to discuss a few of the pivotal engine developments and to identify some of the more significant engine decisions involved. Thirty years ago it was commonplace for the U.S. Government to fund development of engines before the airplanes on which they were to be fitted were configured. While the policy has tended to disappear, the facts that supported it have not.

JT3/J57 Program

General Electric and Westinghouse, with sizable experience in turbine and supercharger technologies, were selected as principals to develop the first U.S. turbine engines.

Nonetheless, Pratt & Whitney recognized the probable future of the newer technology and initiated its independent design work on two engines. The larger of these, (the JT3), was a two-spool 10,000 pound thrust design. Its concept looked so promising that the Air Force joined to fund further development as the military J57, the first U.S. engine in the 10,000 pound thrust class, and forerunner to the JT3C and JT3D. The J57 is illustrated in figure 49.

JT3/J57 Engine

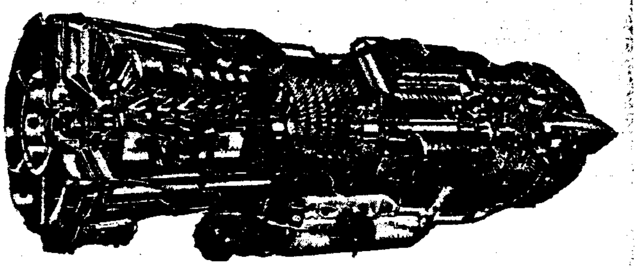


Figure 49

Development was not without difficulty. The original JT3 was designed with a constant external diameter and only two were built. The remainder incorporated a "wasp-waist" constant internal diameter (with higher efficiency, better sealing, and lower weight) and an engine accessory arrangement that reduced its frontal area. As noted earlier, the J57 military version became the propulsive foundation for the B-52. The critical decisions were:

1. The decision by Pratt & Whitney and the Air Force to develop a 10,000 pound thrust jet engine with no identified application.

2. The Pratt & Whitney decision that a two-spool axial flow configuration was correct and that they should work toward an opportunity to develop it and stick with that development.

3. The Air Force decision to change the B-52 from a turboprop to a jet bomber powered by eight J57s.

4. The Boeing decision to back commercial and military programs based on four JT3/J57 engines, which culminated in the decision to build the prototype.

Pratt & Whitney refined the JT3's technology and developed a larger, higher thrust engine called the JT4 (J75). This was the original engine used on long-range versions of DC-8 and 707 aircraft.

Bypass Fan Engine Development

The first bypass engine was a Rolls-Royce Conway with a bypass ratio of 0.3. The U.S. fan-engine developments tracked competitions between two technical paths: the single-spool aft-fan technology followed by General Electric and the two-spool front fan technology pursued by Pratt & Whitney.

Their development and timing were occasioned by Convair's abortive attempt to recoup its CV880 market failure by offering a larger CV990, designed around the CJ805-23. This was a superior aft-fan version of the engine that Convair had used on the CV880. The CV990's success was to depend largely on its being the sole market entry with the much more fuel efficient and higher thrust fan engine. Convair, General Electric, and American Airlines (the CV990 launch customer) were confident that

4.25

Pratt & Whitney couldn't counter with a competitive fan engine for Boeing's 720.

Pratt & Whitney was obviously in a spot; however, the company had previously initiated experimental test work on both rear and front fan engines, noting the potentials of the earlier Rolls-Royce Conway. Pratt simply built a front-fan engine by quickly configuring a JT3 demonstrator two-spool engine from which the first three stages were removed, and two stages from the larger diameter J75C were bolted in their place. This demonstrator provided a bypass ratio competitive to that of the CJ805-23. It evolved into the JT3D, and to this day, the engine has no third stage.

The JT3D was quickly adapted to the 720, giving the airplane a 41 percent increase in power and a decisive advantage over the CV990. It eventually powered the 707, DC-8, B-52, and some KC-135 aircraft. A comparison of the General Electric CJ805 aft-fan with that of Pratt & Whitney's JT3D is shown in figure 50.

Aft Fan vs Forward Fan

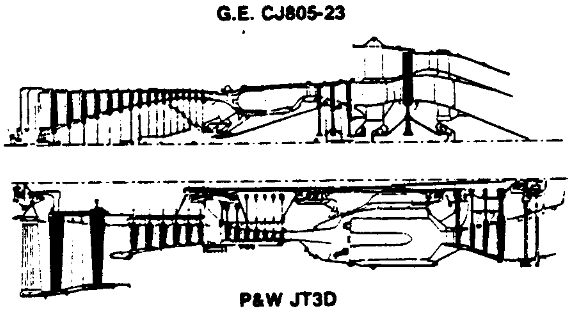


Figure 50

The key decisions appear to have been:

1. Decisions by Convair and American Airlines to launch the CV990 program based on the advanced General Electric aft-fan engine.

2. Pratt & Whitney's decision to immediately counter with a competitive fan engine (based, of course, on earlier experimental work).

<u>The JT8D</u>

Commercial competition of a different type initiated the Pratt & Whitney JT8D program. In this case, Boeing had selected the Rolls-Royce AR963 engine for the 727. The engine was to be assembled in the U.S. by Allison under a Rolls' license. However, Eastern Air Lines, one of the 727's launching customers, insisted that the engine must be totally of U.S. origin to assure that adequate engineering and critical parts support were provided. Boeing attempted to overcome Eastern's objections by persuading Rolls to establish a factory (with adequate technical staff) in this country. Rolls elected not to do so.

This provided Pratt & Whitney the opportunity to offer the JT8D, a fan version of its J52. This engine change was acceptable to the launching airlines, and last minute changes to the airplane were hurriedly made.

The development period was short, and engine failures during the initial 727 flight tests were frequent. However, Pratt corrected the deficiencies with a major redesign and the engine went on to power the DC-9 and 737 as well as all 727s. A "refan" version was later developed, stimulated primarily by noise compliance needs. The circumstances are noted in figure 51.

JT8D Engine

- A rapidly devised, fanned version of the Navy J52 core
- Lack of de-bugging resulted in initial flight hardware that destructed with engine surge (rotor/stator contact)
- Finally resulted in reliable and fuel efficient engine powering half the world's transports. The biggest peace-time engine program ever
- U.S. Government sponsored a re-fan to increase bypass ratio, and improve community noise and specific fuel consumption. The re-fanned engine now powers DC-9 Series 80 aircraft

Figure 51

The most important decisions were:

1. The Eastern Air Lines decision that it could not accept the 727 based on an engine with transatlantic technical and logistics support...coupled with the Rolls-Royce decision not to build a factory in the United States.

2. The Pratt & Whitney decision to rapidly offer an alternative solution.

The RJ500 is an arrangement between Rolls-Royce and Japan Aero Engines Ltd. to design and build an engine aimed at the 150 passenger airplane market. The arrangements may be extended to include other companies, perhaps even Pratt & Whitney. The circumstances for the joint program are still in their early stages and the final outcome is yet to be determined.

Advanced Turboprops

Today, there is a strong possibility that the right way to go for certain civil applications and certain military applications is with an advanced multi-bladed turboprop as shown in figure 56.

Propfan

Figure 56

This may or may not require an all-new gas generator, but it will require all-new gearbox and propeller developments. Although renewed interest in the turboprop was launched under NASA's sponsorship, the major effort and the major funding to support it lies ahead. Assuming that such a development offers fuel efficiency, perhaps 15 percent better than that available with a turbofan, it would appear that the following decisions need to be made:

1. How will the propeller development be funded? Development time appears to be longer than engine and gearbox development time.

2. How will an engine and gearbox development be funded?

3. Will the resulting program be multi-national?

4. What is the correct size for the initial development system?

These questions are based on the probability that such a propulsive system will most assuredly have a development time longer than that of an airframe and that it is probably not feasible to wait until the aircraft requirements are established before such a propulsive system development is committed.

VI. Decisions Forming the New Fleet

Fleet selections for future 1980 decade deliveries will be made from equipment options that are already known, and most of this equipment will incorporate one of the latest fuel efficient high bypass ratio engines that were just discussed. The majority of new commercial airplanes delivered will be selected from the range and size options presented in figure 57.

Commercial Airplanes for the 1980s

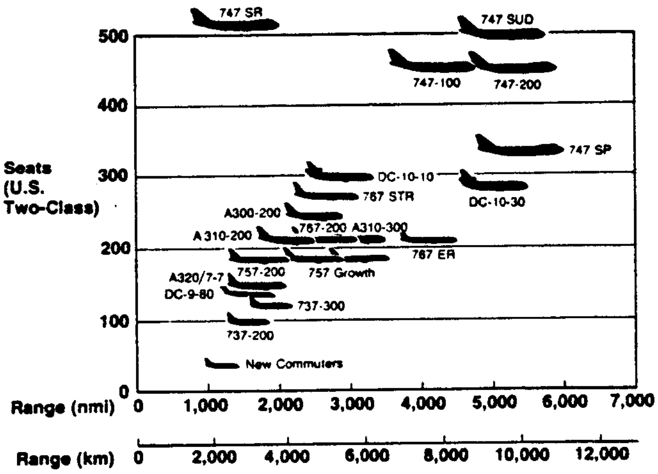


Figure 57

For purposes of discussion, figure 57 has included the A320, 7-7, and 767 stretch, although none have been firmly committed by their manufacturers. We shall next examine decisions leading to the Airbus equipment (A300/A310) and to the 767 and 757.

Airbus

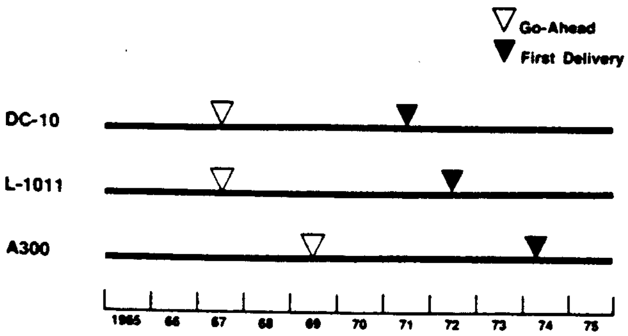
The same high air travel growth rates that stimulated the 747's wide body start for the international market pointed to the need for smaller-sized wide bodies configured for domestic service in Europe and the United States.

The Europeans jointly explored their domestic alternatives for developing a short-haul, high capacity airliner for several years, starting in 1965. However, the less constrained U.S. domestic market acted first, expressing a variety of airline range, airport,and route requirements, thus further tempering the course of the European program's definition.

On the U.S. side, some airlines favored a program committed to transcontinental ranges, and others would accept a shorter initial range with potentials for transcontinental growth. Supporting the latter was the 1966 specification issued by Frank Kolk (American Airlines) for a jumbo twin (high bypass fan engines) carrying 250 passengers (one class) with operating capabilities between La Guardia airport in New York and Chicago and, of course, Chicago to the West Coast. The Denver requirement (which forced the 727 to have three engines) and several other considerations also entered the equation.

However, the fundamental decisions changing to a transcontinental tri-jet and afterwards selecting the DC-10 were made by C. R. Smith, Chairman of American Airlines, who overruled Kolk's specification and demanded an initial transcontinental capability. *Rarely have decisions been so critical as these made by Mr. Smith*. Lockheed, the loser in this competition, decided to proceed in a head-on competition with its L-1011, and the big twin slot was open on both sides of the Atlantic. The timing of the three programs is illustrated in figure 58.

Wide Body Programs
(Post 747)



Note 747 Go-ahead 4-66, First Delivery 12-69

Figure 58

The emergence of the larger-sized U.S. airplanes caused the A300 to be scaled down by about 50 seats and, so sized, the Airbus consortium program was officially sanctioned for go ahead by the participating governments in May 1969. The key forces motivating the decision are summarized in figure 59.

A300 Program Influences

- Provided base for European industry expansion
- Market need recognized by Europe's airlines
- No similar two-engine aircraft competing
- Suitable engine available
- Government support

Figure 59

The timing was right. There appeared to be a definite market need, and it could be years before another opportunity would develop. Furthermore, the program had been carefully planned, recognizing the many difficulties that had surfaced in earlier joint European efforts. The biggest problem was to create an authority that would make binding decisions in the presence of conflicting partnership views...technical, financial, or political.

Affordability, as we have already noted, has different meanings to differing political or societal structures. Nowhere is this more aptly demonstrated than by the European value judgments in the decisions that initially funded the Airbus development and then provided the "staying power" to sustain Airbus through its first six years with less than 30 orders booked and only 13 deliveries.

The decision values considered that the industry was a pacesetter for Europe's technical progress, and as such, its employment must be maintained. It was stated that the Airbus base could provide stability, and, furthermore, it could be sustained by airline profits and the money market, thus reducing the European taxpayer burdens by an equivalent amount. If it could be sold abroad...so much the better.

The A300's start represented far more than the go ahead of a specific airplane. Its decision values stated a cohesive linkage between industry and government objectives, which has no parallel within the U.S. decision-making environment.

As traffic growth improved, the A300 was more closely sized for a number of world markets than was the DC-10 or L-1011, thus forcing the latter out of production. The European partners committed sizable reinvestments for A300 production needs and also to expand this initial product into the product tree illustrated in figure 60.

Airbus Products

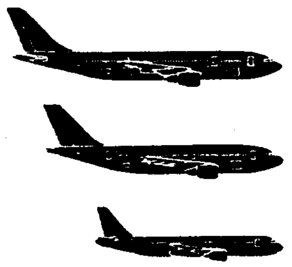


Figure 60

Although the A320 is not firmly committed as yet, we can assume that its go ahead (and also that of the A310) will require levels of competitive reinvestments over time, similar to those experienced by the U.S. industry...essentially doubling the costs of their initial development.

There were significant decisions leading to the Airbus start...we have chosen not to enumerate them. However, the most noteworthy point is the result...a new premise of internationalism and affordability to affect world decisions in commercial aircraft.

The 767 Program

The 767's origins were also very carefully considered, and over a time period that went back to a 1971 alliance with Aeritalia. Thereafter, the Boeing-Aeritalia team gradually recognized a sizable market and product opportunity below the A300 and above the 727-200. This involved replacement of the various 707 and DC-8 airplanes used in this range-size area. These aircraft had excessive range capabilities for the market and were also of an older and less efficient technology, thus of diminishing credibility with respect to market needs. The efforts in the general 1973 time period (7X7 program) were oriented toward the medium-range market, but with transcontinental growth. A Japanese industrial team also joined the program as an additional risk-sharing design and production associate.

There ensued a long period of study during which a three engine configuration appeared best, particularly with the transcontinental potential in mind. This consideration spawned Pratt & Whitney's development of the JT10D, sized for a 200 passenger 7X7 tri-jet configuration with transcontinental capability. However, it later became apparent that a twin engine configuration could be designed for the job with better efficiencies, and the program was so reoriented.

One of the most agonizing decisions was that of cross section. Extensive studies and work with the airlines throughout the 7X7 program identified the lower lobe configuration requirements and also the need for inclusive tour provisions. What was to emerge from this work was confirmed in the 767's larger cross section, which provides a standard seven abreast seating with a dual tight eight abreast inclusive tour provision. The cross section also provides a container and cargo capability that features an eight-foot wide flat floor.

The program's long development history was a plus in many respects, and this is summarized in figure 61.

767 Development

- Unusually long preparation period prior to go-ahead (about six years) allowed exceptional opportunity for orderly program
  - Preparation stressed product and cost definition
- An international collaborative effort from start with Italians and Japanese acting as risk sharing, design and production associates
- May be the most smoothly executed commercial development program in history

Figure 61

The 767 is also the first new aircraft to incorporate the full digital flight management system that was generated from the U.S. SST, then on through the NASA 737 demonstrator to become the new standard, which will eventually be applied to all western world major aircraft. The 767's long development period has also provided for the most thorough pre-production planning that has ever been accomplished. The principal decisions in the 767 program were:

1. The decision on the 7X7 program to carefully prepare for a medium-size, medium-range airplane deliberately placed below the DC-10/L-1011/A300 and above the 727-200.

2. The decision to include Italy and Japan as risk-sharing design and production associates.

3. The decision to develop an all new cross section specifically designed for fuel economy in the specific market intended and offering improved passenger accommodations with an inclusive tour backup.

4. The decision to orient the program specifically to a two engine configuration.

The 757 Program

The need for a successor to the 727 was recognized in the early 1970s. At first it was felt that a wide body version of the 727 with CFM56 engines could be generated, and considerable work was done on a configuration, the 727XX. Later a less ambitious but still expensive derivative (727-300) came close to go ahead but was rejected by United Airlines at the last minute. Fuel prices had escalated rapidly and it became quite apparent that a new wing would be required. The 757 was originally configured as a smaller airplane, but with a new wing, and with a two engine, under-wing configuration, much as it remains today. The cockpit was initially a two crew member modification of the 727 cockpit, but later the new 767 cockpit was adopted along with a great deal of 767 commonality. The design is now a well integrated, *all-new airplane*.

The present 757 size came from the beliefs of the two starting airlines,British Airways and Eastern Air Lines, that the traffic would support the size, and that fuel burn per seat mile would be improved by a size increase. The size was also somewhat influenced by the engines (General Electric and Rolls), both derivatives of larger engines that were down-sized to fit the 757. The smaller size JT10D, that had been developed to fit the three engine 767, was not a candidate when the 757 was originally committed. This situation dramatically changed when Pratt & Whitney offered to completely resize their engine for the 757 as the PW2037. The initial 757s will be delivered with the Rolls engines. The Pratt engine will not be delivered until one-and-a-half years later. The 757 has the *same* direct operating cost in dollars per mile as the 727-200 but has 50 additional seats. The basic 757 decisions were:

1. To optimize the airplane as an all-new design with 727 airplane operating cost but with much lower seat mile and fuel costs.

2. The Pratt & Whitney (and Boeing) decision to build an all-new engine for the 757 (the PW2037).

3. The Rolls-Royce decision to compete by offering a major derivative engine, making the aircraft competively available with either Rolls or Pratt engines.

The 757 and 767 are key members of the new airplane family as is shown in figure 62.

The Boeing New Airplane Family

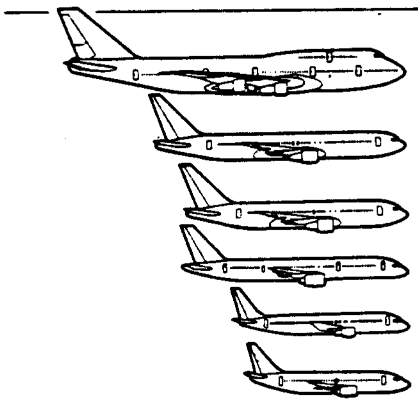


Figure 62

Ultimately, the airplanes noted will incorporate as much commonality as feasible.

Airplane Sizing Considerations

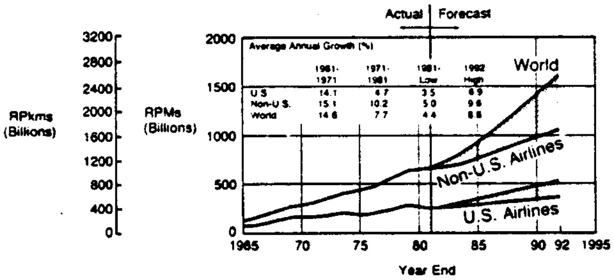
The basic fundamentals of airplane size, of course, include growth in revenue passenger miles. This must be modified by considerations of airline service proliferation and competition, which would drive the size below percentage increase in RPMs, and by airport and air space constraints, such as airport terminal slots, and peak period ATC capability, which would tend to

4.30

keep the size up. All of the original wide body airplanes (747, DC-10, L-1011, and A300) were conceived at a time when RPM traffic growth throughout the world was close to 15 percent per year. It was fundamentally from this 15 percent number that the 11 percent shown in figure 41 was selected for the basic sizing of the 747.

In the decade that followed, traffic growth severely diminished. This and anticipated growth rates for the future are shown in figure 63.

World Revenue Passenger Traffic
All Services

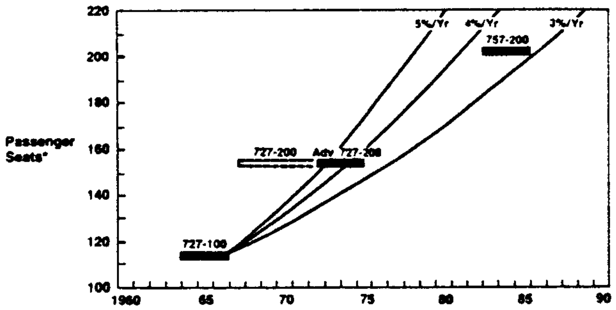


Note: Excludes U.S.S.R. and non-ICAO nations, but includes Taiwan and all-charter carriers.

Figure 63

In addition to the lower traffic growth, U.S. domestic competition proliferated by reason of airline deregulation enacted in late 1978. On a somewhat similar basis, international routes increased, due in part to more liberal bilaterals allowing more competition on both long and short range international segments. We are currently in a most unusual situation which finds the world's traffic growth stabilized and the U.S. growth negative. This has not happened before, and all forecasters, I believe, predict future growth within range of the bands shown. We tend to swing in cycles and to design our airplanes oversized when the growth cycle is high and undersized when the growth cycle is low. Since successful airplane programs must be of long duration, it is not appropriate to design new airplanes such that they will be a perfect market "fit" on the year of introduction. As noted in the discussion relative to figure 41, it is proper to design for a market fit three or four years after initial introduction. Figure 64 carries this philosophy to the relationship between the 727 and the 757.

757 Sizing



* Seating Capacities on 10/90, 36/32, Modern Stowage Bin Basis

Figure 64

Considering *all* the factors we've noted, figure 64 shows that the 757 is based on a "vehicle size for similar job" growth assumption of about three percent per year. This is far enough below the predicted growth values (highs and lows) of the next decade to account for deregulation and other causes of route proliferation. Unlimited proliferation is not feasible by reason of airport and airway constraints. The PATCO strike has given us a preliminary view of constraints that would have otherwise surfaced several years from now.

Keeping in mind the factors noted, and adding considerations for DOC and fuel burned per airplane mile and per passenger mile, one must conclude that the 757 is on the low side of the replacement spectrum for most of the world's 727 aircraft. The 727's replacement will be by the 757, 767, and A310 aircraft. The latter two depend on a growth from the 727 of about four percent a year instead of three.

The current fixation on a "150 passenger" airplane can only be based on the requirement at the bottom of the trunk system replacing the DC-9 and 737. It most assuredly has nothing to do with replacement of most of the 727s, and the market for such an airplane should be expected to be smaller by far than the combined markets for the 757, 767, and A310. Decisions in sizing must include the following:

1. The overall long-term airplane size for a given job has increased and will continue to do so. The only question is how much.

4.31

2. Past decisions which have resulted in improperly sized aircraft indicate that one must moderate thinking between "highs" and "lows" of market growth and seek a middle ground.

<u>New Commuters</u>

There is a worldwide need for new commuter airplanes that are as efficient as possible. In the United States this need has been accelerated by deregulation, which has created a large number of new commuter companies to service short-range segments dropped by the major airlines. To meet the world need there are a number of new programs aimed at the market. Four of these are displayed in figure 65.

Four New Commuters

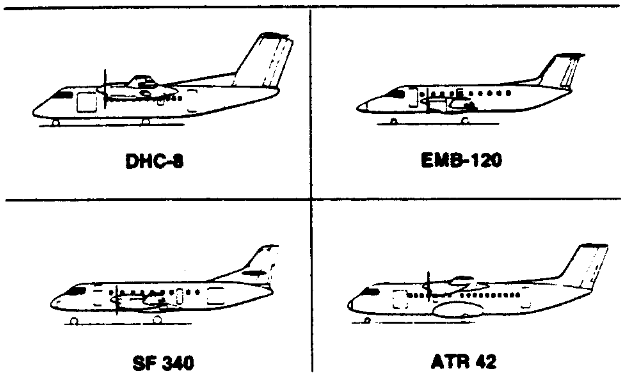


Figure 65

The DHC-8 is a new wing, two engine derivative of the existing four engine DHC-7. It has a four abreast body. It's development is funded by the Canadian Government. The SAAB-Fairchild 340 is an all-new airplane with development substantially funded by the Swedish Government. It involves a joint Swedish-American design accomplished at Fairchild with assembly in Sweden. The Embraer 120 Brasilia is a successor to the Bandeirante and is funded by the Brazilian Government. The ATR 42 is a joint effort between Aerospatiale and Aeritalia and is jointly funded by the French and Italian Governments. Three of the airplanes use a Pratt & Whitney engine manufactured in Canada, and one, the SF 340, uses the General Electric CT7 produced in the United States.

The commuter market has its own growth and constraint situation and growth is quite high. Financially, this market is difficult to design to since the aircraft must be complex enough to fit into the modern airline environment, yet the commuter airlines capital formation capabilities are limited.

This situation illustrates the affordability dilemma confronting the U.S. commuter industry. It is questionable that its risk environment would allow a U.S. company to participate, except in a partnership that would provide foreign risk capital. Also, government supported or not...our past experience would indicate a reasonable prediction of failure for one or more of the programs illustrated in figure 65. Despite the turbulence of the program decision environment, there is a predictable thread that emerges, which we'll examine next.

VII. Lessons Learned

The threads of predictability that we've picked from program decisions form a series of lessons learned. The "learned" appears open to question, since some persist as mistakes repeated. Some have been learned at huge expense...including failure of the company involved. A majority will still be applicable within the environment of internationalism and affordability on the road ahead. There will be new lessons ahead as well, and some of these may be emerging very soon. Nonetheless, the following are offered for consideration:

1. Engines take at least a year longer for development than do aircraft. Funding for a new engine independent of a specific airplane program is advisable unless interim engine use is contemplated or a large risk knowingly accepted.

2. When the market is ready, the successful manufacturer may have to *go*. The eventual prize sometimes goes to the company which is fast on its feet and strong enough...technically and financially.

3. A competitive loss sometimes motivates the loser more than it does the winner. Such possibilities should be examined by both, and in a timely manner.

4. Generally, the highest validated state of the art has produced the longest term, and therefore, the most rewarding program. The

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key word is *validated* and the potential for unanticipated costs and risks must be weighed and program actions to minimize them taken.

5. The good designer provides for growth without allowing the accumulation of design team security cushions, which may make the program unsuccessful and invalidate its need.

6. If one wishes to build commercial airplanes, one must be prepared for investments of about twice the original nonrecurring costs. These may cover a ten year period (as in the 727 and 747) or a three year period (as in the 707 and DC-9). A manufacturer must be prepared for either.

7. Commercial programs require "staying power" to carry them over market depressions or unexpected actions by competitors or governments. This is a requirement that will be accentuated in the future.

8. The use of a parallel design team or "red" team to examine alternatives and to play a major devil's advocate role may be a valuable procedure. If used, the team must have access to technical and financial resources and cannot be "throttled".

9. A good airplane requires a great deal of "up-front" money *before* go ahead if lowest recurring costs are to be achieved. This includes commitments for new technology machinery, training, and equipment. There is no provision in U.S. Government acquisitions procedures to adequately accommodate this need.

10. Government paper competitions that start production programs without prototypes *may* result in selection of the bravest and least informed winner, with uncorrectable consequences.

11. The readiness of technology is a long and expensive process and will frequently warrant funding stimulation. The payoff, when the need and technology readiness match, can be enormous.

12. Changes in a manufacturer's management team can change the company's responsiveness to problems and its ability to *use* basic company strengths. This does not mean that changes should be avoided...only that they should be well considered and deliberate.

13. Past decisions that have resulted in improperly sized aircraft indicate that one must moderate one's thinking to seek a middle ground between the "highs" and "lows" of market growth, competitor proliferation, and airport constraints.

14. A good airplane program requires at least two years of concentrated pre go-ahead study and planning. The penalty for omitting this phase is greater exposure to large changes late in the development program.

15. There is no substitute for understanding the real commercial or military market need and opportunity, especially the predictable change with time. The job of the airplane designer is to design the airplane the customer wants *while* still meeting or providing for the product criteria the customer will want five years after all the basic decisions are made.

16. Defining what constitutes "success" in the program is a necessary exercise. Designing two (or more) "head on" airplanes for the same market may result in financial failure (on normal profit criteria) of one unless the market is unusually large or segmented.

17. A good airplane must be designed to meet a broad spectrum of market requirements. Compromises are essential. In the commercial case, *designing* to the detailed requirements of one customer can result in an unsatisfactory program. *Starting* the right program with one customer while designing to a broad market is another matter and can be satisfactory.

18. Product support after delivery is just as necessary as a successful design. It is a commitment burden that grows with success.

19. In the world competition to acquire the most rewarding technology, the prize will go to the nation or manufacturer who runs the fastest. He who stops running will lose, regardless of lead or protection.

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20. The most constructive emphasis today or in the future is to build a superior product at the lowest manufacturing cost.

VIII. The Road Ahead

Early in this discussion, we noted that the future contained significant technological advancement potentials. Aeronautics has not reached its maturity and the efficiency gains of the past half century will be continued well into the next. Thus, new program starts prior to the end of the century could *potentially* supersede every airplane now flying, either civil or military. While the road ahead contains an abundance of technical opportunities, capital availability and other financial challenges accompany these opportunities. Furthermore, the long technology readiness process will be at least as difficult and at least as misunderstood.

The road ahead will also require that the manufacturer better understand the real market needs, and in cooperation with the customer, go through the agonizing process of compromise that has formed every successful aircraft program. The slogan "back to the drawing board...back to the bank" shown in the earlier cartoon is symbolic of the process, except now the drawing board has been replaced by a computer terminal, and the bank may be replaced by a trading company. The engine situation will be similar.

We have noted lessons learned and most will be applicable in some context to the future as well. So what will be different? The answer, from many aspects, is "everything."

National Policy and Planning

To an ever greater extent, aircraft have become an increasingly visible part of a much larger and complex high technology, international, commercial, and military scene. In this arena, national security implies a context beyond that of conventional arms balance...it implies a state of *national economic security*. In this field the United States finds itself the exception rather than the rule among the western democracies. For example, economically, the U.S. tends to formulate its domestic and foreign policy in terms of *process*; it sets rules and lets things turn out as they may. Elsewhere, such policy is more often defined in terms of desired economic outcomes; if the rules do not seem to be producing desired *results*, the rules will be changed. U.S. relationships between government and industry tend to be adversarial, while those elsewhere tend to be mutually supportive. The United States has great trouble forming and executing a long-range plan, while other countries, to a greater extent, tend to perform national planning in a more consistent manner. In most cases this has little to do with "democracy" and is more cultural or attitudinal in nature. Whether industries are privately or government owned, or a combination of the two, has some effect, but generally is not a deciding factor.

The U.S. is the major contributor to the western nations' umbrella against Soviet aggression. This means that a greater share of other western nations' GNP or national budgets can be allocated toward stimulation of industrial output, including aircraft.

What has this got to do with aircraft decisions? A lot. For example, if the countries support their aircraft industries by providing capital and by reducing risk through mechanisms of one type or another, or with low interest loans, then U.S. companies have little alternative but to seek alignments with foreign companies having access to such devices. U.S. companies are reluctant to align in mutual support on commercial programs, partially due to their historic competition...which in some cases no longer really exists. A much more forceful reason is fear that they will be exposed to anti-trust litigation. It's *not* the anti-trust law that is the problem, it is the history of an unpredictable and sometimes decade long process of interpreting the law. The volatility of aircraft programs makes exposure to such discontinuity unendurable, and one simply avoids the issue in favor of foreign teaming.

Military Considerations

The technology base for commercial and military aeronautics is the same...compartmentalization is generally not possible. Thus, such teaming exposes the U.S. to some amount of technology transfer which could go to third parties. However, since technology generally has a time value and Western Europe and Japan are running about as fast as the U.S., the transfer value is often more imagined than real, and reverse flow will be of greater and greater value.

Beneath both military and civil "name" manufacturers lies a vast network of suppliers. Programs dip into this network, and accomplishment would be impossible without it. The real "power" of American industry is here, and the reason this industry has been able to accomplish large national programs is because it exists. The start-stop-sputter character of U.S. military programs does far less to maintain the supplier base than do the more consistent commercial programs...so military preparedness, to a great extent, rests upon decisions affecting the health of the commercial industry. Erosion is not apparent until some form of national emergency arises, and it is then much too late to rebuild.

Elements of the base will be tied into foreign entities (or foreign owned entities) because the continuity and strength are improved and risks reduced by so doing.

U.S. military procurement tends to have two flawed characteristics, in addition to its burdensome procedures built to do business under rules of public accountability and potential protest. The real problems are: (a) unreliability of funding, and (b) lack of "up-front" money. Multi-year procurement, if it *really* became the norm and not subject to the whims of ensuing Administrations and Congresses could ameliorate the first, but the second is even more difficult, unless finalists in a government competition are authorized to implement machinery and training for productivity and low recurring costs. In such case, the loser would need to be protected from loss. As it is, commercial airplane manufacturing will be sounder and cheaper, and there is no foreseeable cure for the military problem. As a further consideration, U.S. military aircraft procurement has drifted into a combination of risk reduction focus and audit trail protection which, for many programs, means lower technology than will be available from foreign sources. A means of stabilizing military programs in any country is through foreign sales, and the trend toward a common international technological base is accentuating competition.

Commercial Market Considerations

The U.S. itself was, at one time, a large part of the total world commercial airplane market. Such is no longer true and there are now really three major markets of varying size: the U.S., Western Europe, and all the rest. The latter market is fragmented, but it includes a very important component...developing economies that will become major customers of the future.

Historically, the U.S. airline market has gone through immense cycles of feast and famine. When airlines make money, they buy aircraft. When they don't make money, they can't buy aircraft. U.S. airlines have been highly leveraged and, with predicted long term traffic growth, this will not change. Thus, access to the foreign markets is essential for a U.S. manufacturer if it wishes to reduce risk of financial failure in the cyclical domestic market slumps, and if it is to maintain the decisive economy of scale and program longevity. The governmental ownership of both manufacturers and airlines in Europe has largely closed this market to competitive U.S. products, which are thus very dependent upon the remainder of the world market for financial stability. However, much of the remainder is leveraged due to its growth, and is thus very subject to financing terms. European industry has government support for exports that is generally greater than EXIM and is available with more assurance of continuity.

Very simply, the cost of money has become difficult, and world airline deals may involve far more than aircraft and financing terms. In many cases they are decision drivers and negotiated as government-to-government deals involving other things the buying government wants...almost on a barter basis. Examples are technical assistance in nuclear or petrochemical fields, arranging markets for products or raw materials, military equipment, or bilateral treaties yielding landing rights or possibly military and civil advantages of some other kind.

The GATT offers rules governing a limited portion of these considerations, and some stability may be obtained through its administration. However, procedures are long and facts difficult to prove. U.S. companies must operate in a complex world linkage environment that involves U.S. government decisions. Private trading companies may be deficient in scope. U.S. government and industry, it would seem, are at a crossroads.

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Future Considerations

The correct U.S. answer is better aircraft at a lower cost. The role of manufacturing technology is in ascendancy, because it represents a very meaningful solution to affordability. This does not mean an era of simple, cheap products, since the marketplace will call for greater operating efficiency, but at affordable prices. These demands will still dictate technological superiority for the winner. However, complexity must be justified by cost-benefit considerations to an ever greater extent. The situation is also exacerbated by some additional circumstances. Our changed national priorities have gradually increased social obligation until all else is consequentially reduced, and the Federal budget squeeze appears to be with us for decades.

These, then, are the underlying reasons for our new era of "internationalism and affordability" into which the U.S. is already immersed...apparently without recognition by many, in its international context.

Internally the U.S. is still a "free market" for breakfast cereal...but airplanes, whether commercial or military, are in an international environment...one in which success may be dependent upon markets outside the U.S. and even outside Western Europe. Each decision now takes on a new dimension...the most complex dimension of them all.

The bottom line for the U.S. is unquestioned excellence at low manufacturing costs. It is a hard combination. It requires economies of scale, but scale requires an international market...and so on through the complex circumstances already described. The road ahead is surrounded by this environment.

IX. Conclusion

Aircraft programs, both commercial and military, differ, and their individual characteristics will affect the multi-billion dollar decisions made. It has not been my intention to say that any formula can be derived to assure success. There are, however, some trends and lessons that appear to override program individualism. If these have been largely identified, we have been successful.

Even the word *success* must be defined, since in our international world it will mean different things to the companies and governments involved. In many cases success will depend as much on decisions that are made outside the program as those made from within.

Many of the decisions we have noted are factored to market needs and their correct timing...the areas that historically have been the most immediate and critical. This has not really changed in the last 40 years, and I would expect that market needs and timing will also weigh heavily in future decisions.

Successful airplane programs are usually large and long term, almost by definition alone. Commercially, the phrase *bet the company* often applies, and military programs cannot be very constructive for either manufacturer or government if funding is on an unreliable start-stop basis. Advanced technology should be applied if a long program life is to be achieved. The importance of technology validation before application cannot be overemphasized. It requires substantial continuity of effort long before application is defined. Similarly, the fundamental base for manufacturing efficiency must be addressed before a program is started...not during it.

Beyond these influences are others that are exerting a change. *Internationalism* and *affordability* are the two that this discussion has particularly noted. By their nature, we cannot overlook a third...that of governments. Combined, these are powerful forces. They are changing the market's character and exerting a sizable stress on the manufacturing industry. What this implies for tomorrow is the question for today.

We live in a small world, and this nation is but one of the established players. All are not marching to the same drummer, and the rules of the competition are not constant. Our technology got us to the playing field, but will it get us to the goal? Can we adapt to this difficult situation? I believe the answer is yes. However, it will take understanding and judgment in the decisions we make. It is toward this understanding...in the field of aircraft...that this lecture is respectfully submitted.

Why the New Air Fleets Challenge the Designer

By J. M. SWIHART and J. I. MINNICK
Boeing Commercial Airplane Company

Because air transportation has become a mature industry, with both high promise and powerful constraints, its design engineers must exert their ingenuity on *all* elements of airplane-related operating costs

The dynamics of the U.S. air transportation system are legendary and volumes have been written about the reasons for the ups and downs of the industry. The Federal Government has had a major role in the development of all of this country's public transportation systems. It has often been caught in the middle of the field's issues, part of which have been caused by the lack of a definitive policy on transportation.[1] In October 1975, trying to set this right, the Department of Transportation (DOT) issued a document dealing directly with national transportation policy issues and, in effect, a "straw man" for all interested parties to criticize. The recent airline deregulation bill is a direct product of this document.

Government policy alone does not create or alleviate the problems of the industry. The pressures for airline growth and constraints of it likewise drive us. The major pressure for growth in the 1960s came primarily from the improved quality of air travel. The growth in the early part of that decade was due to the superior ride-quality, lesser trip times, and better economics of jet airplanes. On the other hand, shortages in jet equipment and a low consumer propensity for air travel constrained growth.

By the mid-Sixties, as more equipment became available, the public overcame its "fear of flying," fares dropped greatly, and air transportation experienced high growth. Together with a healthy world-wide economy and an increasing awareness of airport congestion problems, this growth motivated the airlines to order new wide-body transports. Most persons in the industry felt that these new transports would continue the trend in improvement in air-travel quality, would reduce (if not eliminate) airport congestion, and would permit lower fares owing to economies of scale. Lower fares, in turn, it was assumed, would stimulate traffic growth and thus fill the new wide-bodies.

But at the start of the 1970s the anticipated growth failed to materialize. The pressures and constraints of the Sixties had changed. The world-wide economy took a downturn; the Vietnam war was winding down; the space program had accomplished its man-on-the-Moon mission; and the field lacked the stimulus of large government programs. Improvements in air-travel quality had been incorporated in most airline markets and the cost improvements of the jets had, in large measure, been realized.

These constraints overshadowed the pressures and growth and, in conjunction with deliveries of the wide-bodied aircraft, caused overcapacity early in the Seventies.

Aside from the new wide-bodied transports, technology improvements were limited to fairly

AIRLINE TRENDS

Operation

- Traffic will double in the next decade.
- Departures, revenue aircraft-miles (RAMs), average route-segment distance, and trunk-airline fleets will grow relatively little.
- System-wide load factors will increase to 62-64%.
- Average number of seats per aircraft will continue to increase as fleet mix changes.

Revenues and expenses

- Domestic fuel price tripled in the last five years (quadruped in the last decade) and should increase over 9% per year for the next decade.
- Unit direct costs (c/ASM) are rising as labor and material inflation outruns productivity gains.
- Profits squeezed by downward pressure on yield (discount fares, charter plans) and operating-cost escalation.

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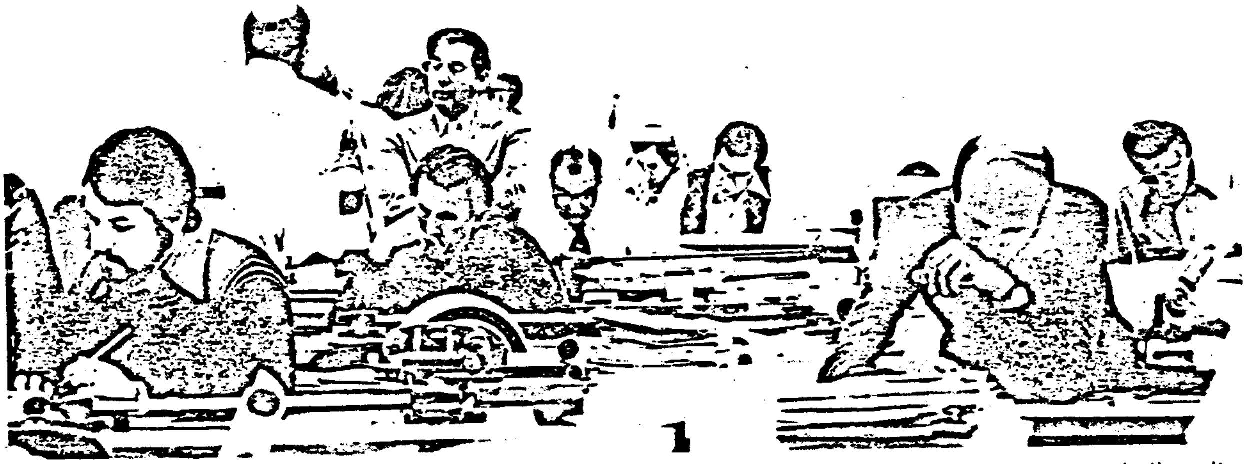
THE MEN BEHIND THE PLANES BEHIND THE AIRLINES



Sometimes we think that design engineers make up one of the most discriminated-against minority groups in this country, certainly not in the conventional sense of race, color, or sex, but more in the lack of appreciation of their impact on our society and our standard of living. Regardless of where or how a technological breakthrough occurs, the design engineer translates the idea into a producible end product. Unfortunately, in some industries, such as aircraft manufacturing, the individual designer seldom sees the results of his efforts as so many systems and subsystems compose the end product. But individual design efforts are paramount to the economic and technical success of the airplane.

It is safe to say that the more familiar the design engineer is with the application of the end product, the better will be his designs. In this article we present some historical data on the U.S. air transportation system, both on its operations and its economics. With this as a background, we then attempt to extrapolate into the future to see what it may hold. In this manner, we hope to give the design engineer a better appreciation of his contributions. There is a very fine line between profit and loss at our airlines, and the design engineer's efforts are critical to the airlines' economic success as they start to reequip with a new generation of transports.

small increments, leading many people to believe that this would be the "decade of derivatives."[2] Compounding these constraints was the 1973 fuel crisis and its impact on air transportation.

Thus, the first half of the Seventies brought a hiatus in sales of transports to U.S. customers. Moreover, the airlines were feeling the pressures of costs rising faster than productivity. Noise-legislation debates and the potential cost of noise-cutting steps such as reengineering also impeded the airlines' decision-making processes. The dilemma facing the carriers was the need to replace older jets with something more fuel-efficient which would meet the then undefined noise regulations.

For the first time builders saw that any successful new or derivative airplane program would have to meet the requirements of the marketplace in terms of size and range in a more economical manner than fleet mixes of current products. No longer could the marketplace accommodate just any new offering; it had to have *the* correct one.

Now the economy has strengthened. Overcapacity has been eliminated. Air travel has resumed its growth. The airlines' balance sheets are improving, and an underlying strength is returning to the U.S. commercial aircraft market.

In the light of recent history, we should ask, what pressures and constraints may the 1980s bring us? F-1 highlights the most critical items. Certainly the most powerful constraint will be *operating cost.* Inflation and environmental concerns raise costs at a faster rate than productivity can reduce them. Both the manufacturers and the airlines will be required to reduce inefficiencies wherever possible in order to minimize the apparently inevitable fare increases (which, if left unchecked, could hinder traffic growth).

A lull in the introduction of new-technology transports encourages stocktaking. By the time the first Boeing 767 airplane is delivered in 1982, a full decade will have passed since the introduction of the last U.S.-built airplane, the Lockheed L-1011. In contrast, the Sixties saw the introduction of at least six major U.S. airplane types (F-2).

Operational Trends: The airlines constitute a service-oriented industry whose product (supply) is available seat-miles (ASMs), the demand for which is measured by revenue passenger-miles (RPMs).

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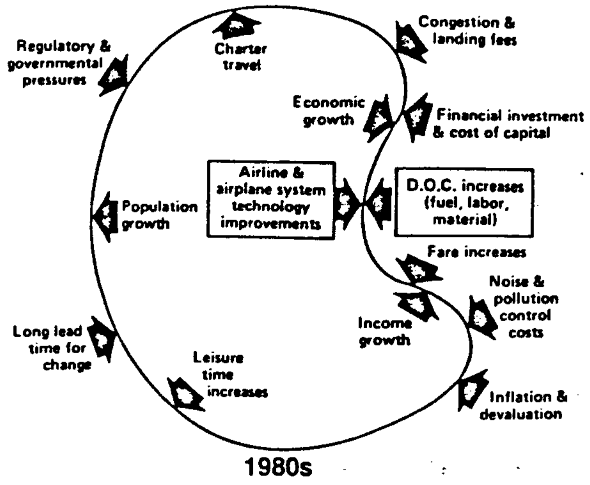
F-1 FUTURE PRESSURES AND CONSTRAINTS ON AIRLINE GROWTH



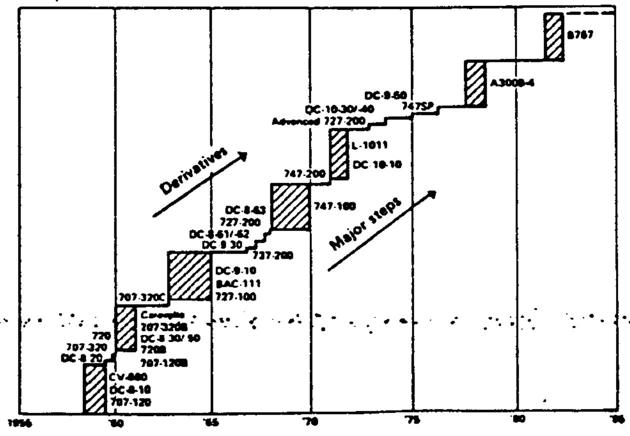
The ratio of RPMs to ASMs defines *load factor*. Obviously, the higher the average load factor, the better the chance of meeting operating costs and generating a profit.[3] The industry controls and measures its production of ASMs by the number of seats per airplane, the distance flown, and the number of departures offered. Competitive considerations, on the one hand, and prudent economics, on the other, decide the degree to which an individual airline exercises this control.

With these points as background, the following figures describe basic operational trends in the U.S. trunk carrier system. The data comes from information supplied to the Civil Aeronautics Board (CAB) by the carriers, as required by law. The various yearly values include all data for airlines classified as a trunk carrier in that year (thus all mergers are properly accounted for).

Revenue Passenger-Miles (RPM): F-3 shows the demand for U.S. air travel as RPMs flown by the trunk carriers in scheduled domestic North American service. Jet aircraft did not hit the domestic market fully until about 1964. Initial deliveries of B707 and DC-8 aircraft were used primarily for international flights and almost all transcontinental flights. Not until the smaller, three-engine 727-100 was delivered in early 1964 did the jet era really come to the domestic market. The demand for jet travel increased the average yearly growth rate between 1962 and 1972 by about a third over that for the previous decade, 1952-1962. Between 1972 and 1977, owing to various pressures and constraints, the average yearly growth rate dropped to about a third of the 1962-1972 value, but this rate applies to a traffic *level* much higher than before, so that the absolute value of RPMs still increased at a substantial pace.

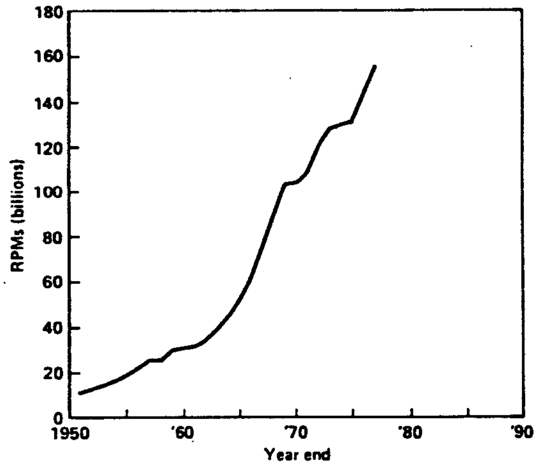
F-4 projects values of RPMs on a worldwide basis as well as for all U.S. carriers. It indicates nearly a doubling of traffic between now and 1990, even in a conservative scenario.

F-2 A pause has occurred in the rapid pace of new-technology airplane introductions into U.S. airlines.



F-3 REVENUE PASSENGER-MILES FOR U.S. DOMESTIC TRUNKS

For scheduled North American revenue passenger flights performed, except BN, NW, and PA data which contain U.S. domestic portion only. Includes all mergers. Data source: *CAB Handbook of Airline Statistics* (1951-1968), and *Air Market* (1968 and on).



Departures: The trend in departures corresponds to the growth in demand, as shown in F-5. Before 1959, departures were increasing at an average rate of 4% per year, as the airlines attempted to meet demand by increasing departures. Strikes caused the drop in departures in 1958. The initial use of jets on transcontinental flights did have an impact on domestic departures between 1959 and 1960. The nearly fourfold increase in productivity represented by the jets, combined with their longer range (which eliminated a lot of fuel stops), allowed trunks to reduce departures while still meeting demand.

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Then, from 1962 to 1969, departures showed a dramatic increase of nearly 7% per year. Contrary to the popular belief that this increase resulted from new point-to-point service, nearly 80% of it occurred between city pairs that had nonstop service in both 1962 and 1968! In 1969 airport congestion was one of the factors which prompted the airlines to order wide-body transports.

The concurrent downturn in the economy plus the introduction of the wide-bodies lowered departures in the early 1970s. The 1973 fuel embargo also tempered departures. For two years—1974 and 1975—the airlines had fuel quotas based on their 1972 total usage. Since 1974, departures have increased in response to demand, but at a rate much lower than in 1962-1969. Departures are currently approaching the 1969 highs, and there has been enough airport congestion again to raise it as a threat to air-traffic growth.

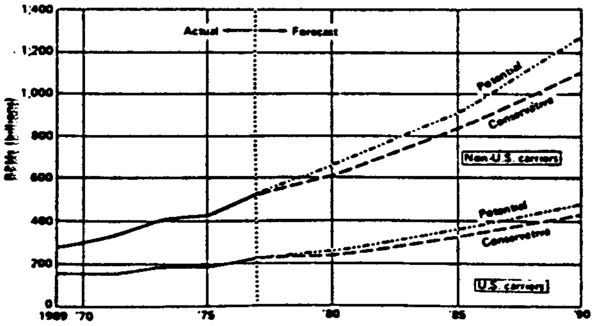
Revenue Aircraft-Miles (RAMs): F-6 shows total aircraft miles flown in scheduled revenue passenger service. The growth parallels departures, and the same constraints and pressures caused yearly variations. Growth in RAMs will probably be limited as the industry matures. Growth in RPMs will be primarily provided by an increase in average number of seats per aircraft plus some increase in departures.

Flight-Segment Distance: F-7 describes another indication of the maturity of the industry—the rise and then leveling of average flight-segment distance flown by the trunk carriers. Segment length increased at an average rate of 4.9% per year between 1951 and 1962. Between 1962 and 1970, the average yearly rate rose to 6.7% as the trunk carriers introduced jet equipment into all of their markets to meet the demand for jet travel. Since 1970 the average segment distance has held nearly constant at 600 statute miles.

F-4 WORLD REVENUE PASSENGER-MILES TOTAL SERVICES

Average yearly growth rate (percent)

	1960-1965	1965-1970	1970-1975	1975-1977	1977-1987 (Conservative)	1977-1987 (Potential)	1987-1990 (Conservative)	1987-1990 (Potential)
World	13.0	16.2	8.0	9.2	6.0	7.1	5.2	6.1
Total non-U.S.	14.0	16.4	13.2	9.5	6.6	7.8	5.8	6.8
Total U.S.	12.2	16.0	2.9	8.9	5.2	6.1	4.2	5.2

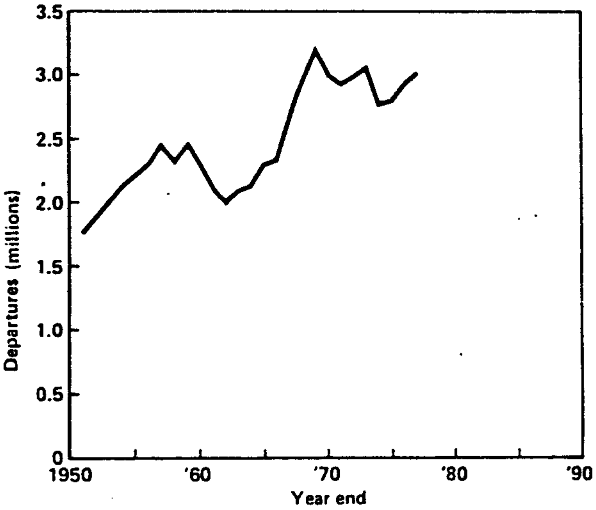


Note that in F-7 the average segment distance is defined as the total RAMs flown divided by the total departures. The constant value over the last seven years strongly indicates that city-pair markets are now being served by the correct mix of equipment and departures and that if all city-pair markets were to exhibit the same growth rates in the future, this average segment distance should remain fairly constant.

System Load-Factor: A steady downward trend in system load factor occurred between 1951 and 1971, as shown in F-8. In the early 1950s load factor approached 70% primarily due to the lack of equipment.

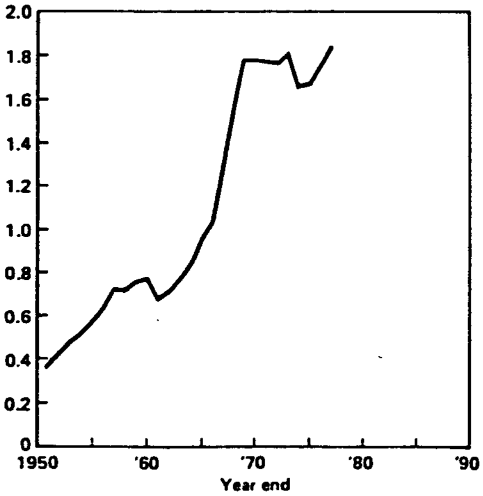
F-5 YEARLY DEPARTURES FOR U.S. DOMESTIC TRUNKS

Passenger departures calculated from total aircraft revenue departures performed by subtracting all-cargo services in proportion with all-cargo revenue-miles to overall aircraft revenue-miles. See notes for F-3.



F-6 REVENUE AIRCRAFT MILES FOR U.S. DOMESTIC TRUNKS

See notes for F-3.



As larger transports and more of them came into service, the load factor headed down. This decline continued even after the introduction of the jets due to a large increase in competitive frequencies and, to some extent, the attempt to develop new markets where the equipment was too large to meet the demand economically. The introduction of wide-bodies (more seats) in the late Sixties and early Seventies exacerbated the problem and before long greatly reduced profits.

To counteract this, the CAB launched the Domestic Passenger Fare Investigation (DPFI). The investigation led to higher fares, a fare structure based on distance, and elimination of most discount fares. In addition, the CAB selected a load factor of 55% as one of the major criteria the airlines would have to use in seeking fare increases.

The DPFI together with capacity agreements, route-award moratoriums, route swaps, and an easing of overcapacity reversed the downward trend in load factor.

Now, looking ahead, on a yearly and system-wide basis, load factor appears likely to rise to between 62 and 64%, as shown in F-9. Time-of-day, day-of-week, and seasonality effects will prevent attaining load factors much above this.[4]

Average Number of Seats Per Aircraft: As F-10 shows, the average number of seats per aircraft (ratio of available seat-miles to revenue airplane-miles flown) has grown steadily for over three decades at a rate of 4.3 seats per year. It is now 156 for the fleets of the domestic trunk carriers.

The only times the growth varied from this value to any large extent came between 1954 and 1958 (on the low side) and between 1958 and 1962 (on the high side). The 1954-1958 period primarily involved DC-6s, DC-7s, and the L-1049 Super Constellations. In a one-class configuration the DC-6s carried 64 first-class passengers, the DC-7 carried 68, and the "Connie" carried 71. Thus the growth rate between 1954-58 reflected an asymptotic approach to the largest available aircraft.

The jets increased growth rate through greater productivity and because they saw initial use in the domestic market on transcontinental routes, where they produced a disproportionate amount of ASMs. As smaller jets were introduced, the growth rate returned to its long-term value. Even the introduction of wide-body transports did not change this growth pattern, nor will the introduction of the Airbus A310 or the Boeing 767, according to Boeing studies.

Trunk Airline Fleets: Given these operational trends and assuming normal retirement schedules, you can predict, in broad categories, the composition of the trunk airline fleets in future years. T-1 presents one such prediction, made by Boeing's marketing organization. The two-engine standard-body category covers DC-9s and 737s. The three-engine standard-body category, the 727-100s and 727-200s. The four-engine standard-body, the DC-8 and 707 series of aircraft. The wide-bodies, 747s, DC-10s, L-1011s, and A300s. The new programs include 757s, 767s, and potential derivatives of the current wide-body airplanes.

This table makes two major points concerning future fleet mix. Standard-body aircraft compose more than 80% of the fleet today, but their share will drop to about 30% by 1987. The new airplane

THE AUTHORS

JOHN M. SWIHART (AF), at right, is Director of Product Development, Sales and Marketing. After joining Boeing in 1962 he played a large role in developing its design for the U.S. supersonic transport and became chief engineer for production SST airplane development in 1969. In 1971 he moved over to the Commercial Airplane Div., where he became deputy director for international sales in 1974. He took over his present post in 1976. Swihart received a bachelor's degree in physics from Bowling Green State Univ. and one in AE from Georgia Tech, then joined NASA-Langley in 1949. There he conducted research on a wide variety of early supersonic fighter and transport designs. For AIAA Swihart has chaired the Aircraft Design Committee and served as general chairman of the 1977 Aircraft Systems and Technology Meeting. JACK I. MINNICK since 1974 has been Manager of Market Requirements in Sales and Marketing where he has helped develop design requirements for the 767 and 757. He joined Boeing in 1954 and worked through various assignments to preliminary design of aerospace vehicles. Designing horizontal takeoff and landing booster systems for them led to being selected for the preliminary design team for the production version of the U.S. supersonic transport. He went to the Commercial Airplane Division's Marketing Organization in 1971. Minnick received a bachelor's degree in AE from the Univ. of Washington in 1954 and a master's in 1960.

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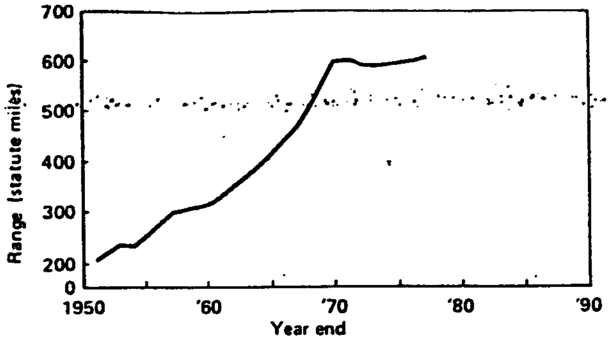
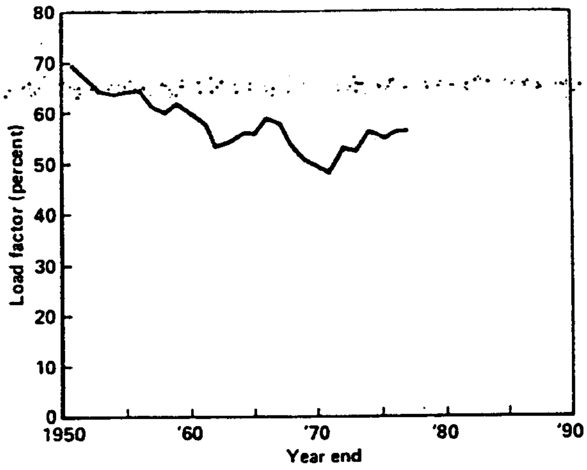
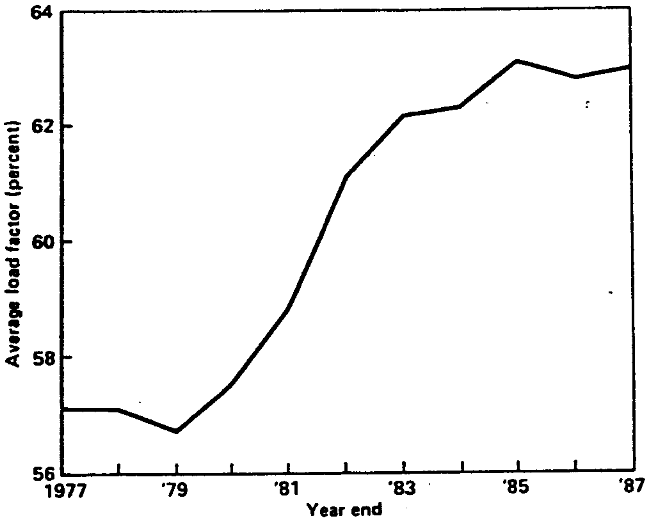
F-7 AVERAGE RANGE FOR U.S. DOMESTIC TRUNKS
Range calculated by revenue aircraft-miles divided by departures. See notes for F-3.



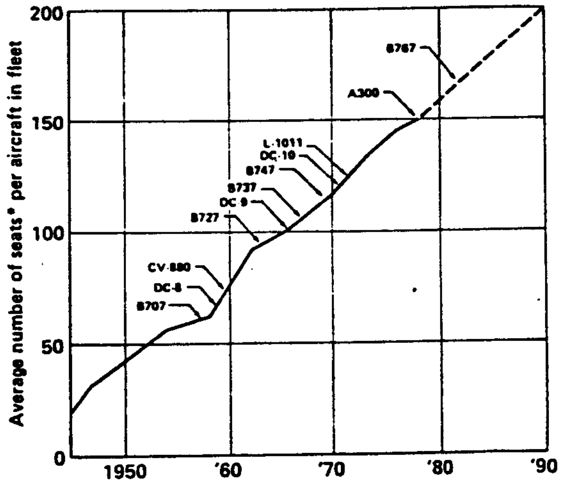
F-8 AVERAGE LOAD FACTOR FOR U.S. DOMESTIC TRUNKS
Load factor calculated by revenue passenger-miles divided by available seat-miles. See notes for F-3.



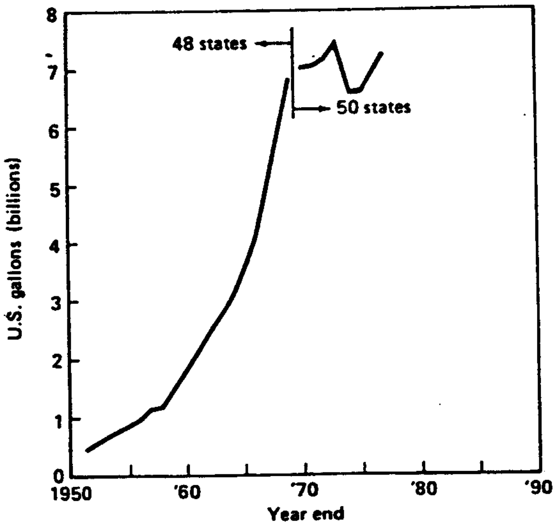
F-9 LOAD FACTOR GROWTH FOR U.S. TRUNKS AND PAN AM



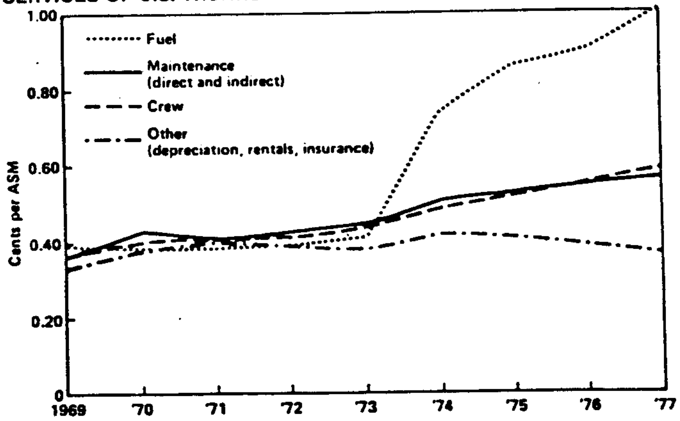
F-10 GROWTH IN AVERAGE NUMBER OF SEATS FOR U.S. TRUNK DOMESTIC OPERATIONS
Average seats calculated by dividing available seat-miles by revenue airplane-miles. Callouts indicate introductory dates.



F-11 FUEL CONSUMPTION IN U.S. DOMESTIC TRUNK OPERATIONS FOR ALL SERVICES
Data source: *CAB Handbook of Airline Statistics* (1951-1969) and *AAIMS* (1970 and on).

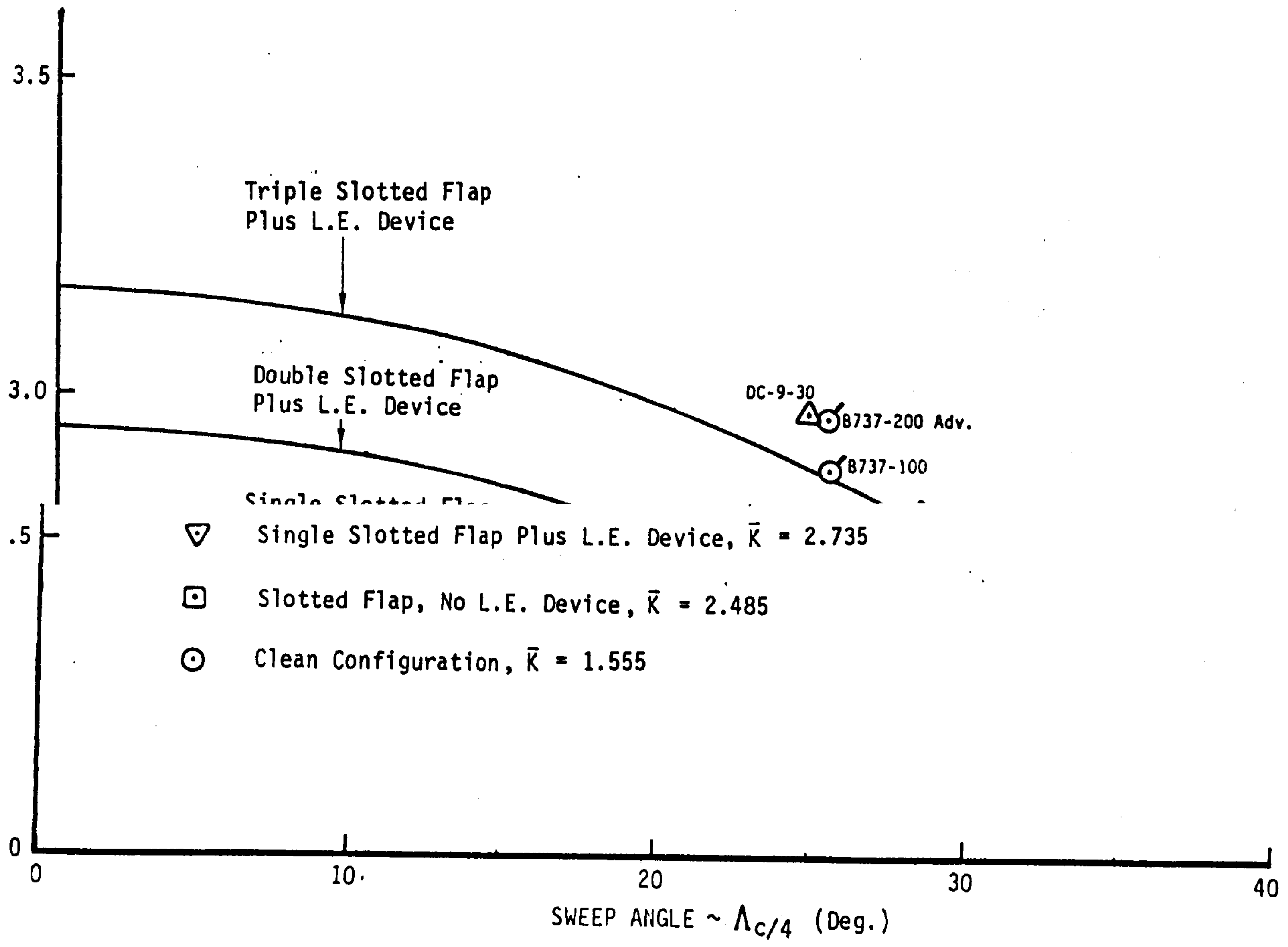


F-12 DIRECT OPERATING COST ELEMENTS FOR ALL SERVICES OF U.S. TRUNKS



T-1 TRUNK AIRLINE FLEETS

Aircraft	1976	1977	1978	1979	1980	1981	1982	1983	1984	1985	1986	1987
Two-engine standard body	231	222	220	220	220	220	227	228	203	151	85	52
Three-engine standard body	790	812	859	922	957	975	936	867	770	630	565	501
Four-engine standard body	408	367	333	296	277	264	247	221	145	61	30	10
Current wide-body	299	298	318	342	364	394	435	464	490	529	550	570
New programs							38	109	254	458	622	734
Total	1728	1699	1730	1780	1818	1853	1883	1889	1862	1829	1852	1867



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CHECK			Mc M.	9-18-78
APPD				
APPD				

programs will fill about 75% of the new and replacement requirements—a result of the marketplace requiring a quiet and economical airplane between the 727-200 and DC-10/L-1011 sizes.

Fuel consumption: Consumption of fuel by the trunk carriers paralleled the growth in air travel during the 1960s (F-11).

Entering the 1970s, a combination of reduced traffic growth, introduction of the efficient high-bypass-ratio fan engines on the wide-body aircraft, airport congestion (and its impact on operations), and the maturing of the industry brought a leveling of the increase in fuel consumption.

The oil embargo of 1973 and the resulting quotas (in effect during 1974 and 1975) caused the airlines to modify their operations toward a hub-and-spoke system in order to use the excess capacity of the wide-body aircraft more effectively. The result was a decline in fuel consumption in 1974 and 1975.

The years 1976 and 1977 saw the resumption of growth in fuel usage but at a much reduced rate compared to the 1960s.

Our analysis of the trunk carriers, using the future fleet composition discussed above, points to a rate of growth in fuel consumption on the order of 2% per year over the next decade. Almost half of the growth will be due to an increase in the size of the fleet. The remainder will be due to the growth in average number of seats per aircraft (replacement of smaller aircraft with larger ones more fuel efficient per seat).

Revenue and Expense Trends: The need for a universal method to evaluate the relative merits of aircraft on a consistent basis has plagued the airline industry almost from its inception. We say "plagued" because the field still lacks an entirely satisfactory one.

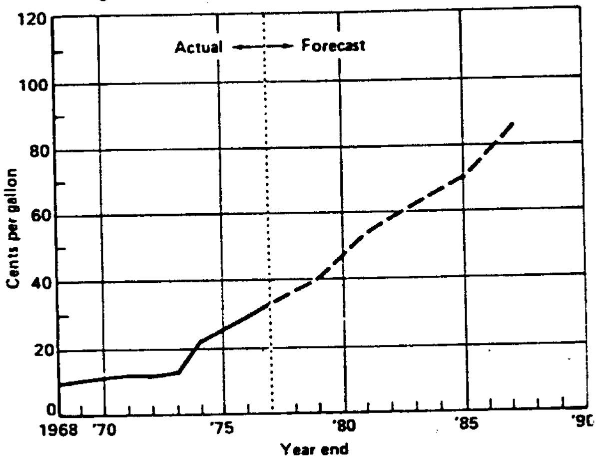
Mentzes and Nourse made the first attempt to develop a standard method of estimating the direct operating costs, and published it in their paper, "Some Economic Aspects of Transport Airplanes," in the *Journal of Aeronautical Sciences* of April and May, 1940—a work generally recognized as the forefather of all costing methods used in the industry.

Regulations governing the U.S. airline industry produce a wealth of public information that serves the analyst's needs. Financial and operating statistics for the U.S. certificated air carriers must be reported to the CAB's Bureau of Accounts and Statistics in a document called Form 41. The CAB prescribes the form, content, and frequency of submittal of this report in its "Uniform System Accounts and Reports for Certificated Air Carriers in Accordance with Section 407 of the Federal Aviation Act." Form 41 contains approximately 50 pages, each page a "schedule" calling for various types of data.

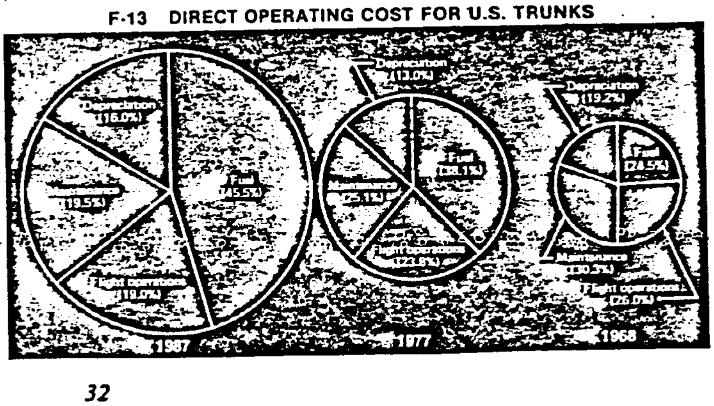
Virtually all of the cost and operational information published in periodicals, such as *Aviation Week, Airline Marketing and Management, Air Transport World,* the annual *Air Transport Association Facts and Figures,* and the CAB *Handbook of Airline Statistics,* are based on Form 41 contents. Some companies that specialize in computer services have compiled these data and sell it as direct readouts from their data banks. From such sources has come the following information on revenue and expense trends.

Operating Expenses: Operating costs of aircraft are broadly classified into two categories. Direct operating cost (DOC) includes items associated with ownership and operation of a given unit of actual ownership and operation of a given unit of equipment (e.g., flight crew, fuel, maintenance, depreciation, and insurance). The remaining expenses, related to management and the handling of passengers, describe indirect operating costs (IOC). IOCs include cabin-attendant pay and expenses

F-14 DOMESTIC KERSOSENE JET FUEL PRICE
CAB average for total domestic trunk and local service carriers.



F-13 DIRECT OPERATING COST FOR U.S. TRUNKS



4.43

4.44

passenger food, passenger liability insurance, control and communication, aircraft servicing and landing fees, passenger and baggage handling, passenger reservations, commissions (such as to travel agents), advertising, general and administrative costs, amortization, and ground-equipment depreciation and maintenance.

In general, aircraft design engineers have their greatest impact on DOC, because IOC does not depend on a particular aircraft and directly represents corporate business operations. Thus the following sections discuss DOC in greater detail than IOC.

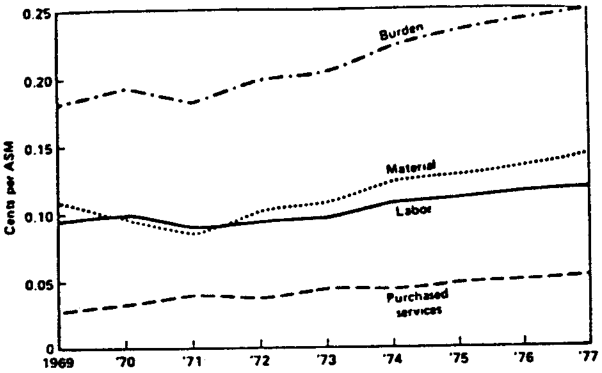
Direct Operating Cost (DOC): The unit cost of producing an ASM is of prime interest, since the airline industry produces and markets ASM. F-12 shows the major DOC elements with values expressed in cents per ASM. Before 1974, each of the four elements contributed approximately the same amount to the total DOC. The 1973 oil embargo and subsequent increases in fuel price dramatically changed this relationship. In addition to the obvious increase in the cost of fuel consumed in generating an ASM, a secondary effect of higher energy charges has been to increase the rate of inflation to the point where productivity increases no longer offset rising prices.

Thus we see an increase not only in fuel costs, but also in the costs of labor-intensive areas of flight crew and maintenance. Relative declines in the depreciation, rental, and insurance reflect the fact that the carriers did not add substantially to their fleets from 1973 to 1977. The aircraft being depreciated were bought, in general, before 1974 so depreciation now represents a quasi-fixed cost. Improved utilization of the current fleets has generated sufficient additional ASMs to keep depreciation nearly constant.

F-13 describes the changing magnitude and proportion of DOCs. The pie-chart areas represent (in current dollars) the magnitude of the DOC for each year. This data makes clear what the increase in fuel prices has done to change the proportions of the DOC elements. Note that the depreciation percentage dropped from 19% in 1968 to 13% in 1977 (reflecting the lack of new equipment orders in the 1970-1975 period). It will increase to 16% in 1987 as the new fleets (with their relatively higher initial investment) enter service.

Design engineers will need to expend considerable effort to insure that any new or derivative airplane program incorporates all possible technology which, in a balanced manner, reduces fuel consumption—such as increased wing span, advanced airfoil sections, high-bypass-ratio engines,

F-15 MAINTENANCE EXPENSE ELEMENTS FOR U.S. TRUNKS TOTAL SERVICES



better materials, advanced electronics, or attention to OEW.

The domestic price of kerosene jet fuel, as shown in F-14, has tripled in the last five years. Today's fuel contracts average over 40¢ per gallon (for 1978 the average price was somewhat less). We estimate that fuel prices will double sometime between now and 1985 to 1990.

Maintenance expense must also be reduced through design. F-15 describes elements of maintenance expense. "Purchased services" means items performed outside the airline's maintenance shops. The cost of these seems to be increasing at a pace similar to that of the airline labor rate. Part of the labor element of maintenance, burden, has been growing at the fastest rate. The last few years have seen labor contracts increasingly emphasize fringe benefits, such as retirement pay, more vacations and holidays, and better medical and dental insurance. Airframe and engine maintenance labor exhibits a characteristic similar to a production-line learning curve, and it has helped to keep maintenance cost consistent with or below inflation. The airline industry has moved well down the maintenance learning curve, however, with its fleet. But any new or derivative airplane program should include design-to-cost trades in the maintenance area. Good example: the wing design on the Boeing 767 airplane. We have increased the wing area to incorporate a somewhat simpler flap system needing less maintenance, but able to maintain approach speeds comparable to those of the 727.

On flight-crew cost the design engineer has little, if any, impact. The one exception could be to continue design efforts to reduce cockpit work load and thus help justify a two-man crew. Historically, crew pay has been a function of an airplane's productivity in terms of seat-miles per hour. With all subsonic jet transports having approximately the same speed, productivity is directly related to plane size (seats or gross weight). The 747 flight crews receive the highest pay. A disturbing trend, particularly with smaller airplanes, such as the DC-9 and 727, is toward a more uniform pay scale. This could cause a disproportionate increase in flight-crew costs for the smaller transports.

F-16 gives the total direct cost of producing an ASM. Before 1974 the cost was fairly constant at about 1.8¢ per ASM. Since 1974 costs have risen due to inflationary pressures and the high fuel prices. We anticipate that, with the rate of inflation exceeding the rate of productivity increases, the cost of producing an ASM will rise to about 4.5¢ in current dollars by 1990.

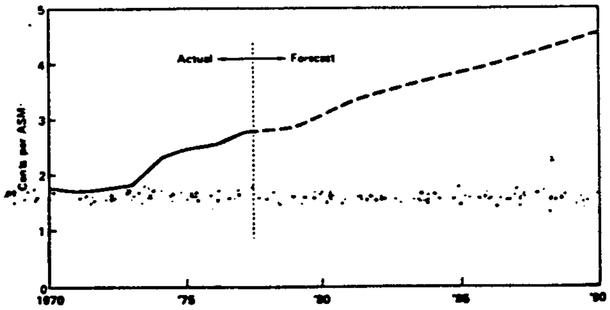
Indirect Operating Costs (IOC): IOC elements for the most part do not depend on airplane characteristics. By developing revenue-related cost-coefficients (based on historical data) for indirect passenger-related costs, we can estimate total IOCs for future years. This requires both RPM (F-4) and yield forecasts.

Total Operating Costs: F-17 presents total operating costs for the U.S. trunk carriers. The size of a circle indicates relative magnitude of total costs. Two items of interest appear on this figure. One, DOC and IOC contribute nearly equally to total operating costs, both now and in the future forecast. Two, the indirect cost elements remain nearly constant in terms of percentage value of the total IOC.

Operating Revenue: F-18 shows average revenue per RPM. Obtained by dividing the total transport-related revenues by the total RPM, the values include the impact of the various discount fares.

Although not shown in F-17, the average yield for the eight years preceding 1968 was 6.0¢ per RPM (in current dollars). Between 1968 and 1973 the average yield per RPM increased at a rate roughly equal to the differences between inflation and equipment productivity (for the reasons discussed previously).

F-16 DIRECT OPERATING COST FOR ALL SERVICES OF U.S. TRUNK AIRLINES



As a result of the large increase in fuel price, 1974 saw a substantial increase in fares followed by an increase in yields. From 1974 to 1977 the rate of increase in yield flattened.

The past year may well turn out to be an anomaly in yield. The rash of discount fares brought about by the unknowns of deregulation will undoubtedly reduce average yield for 1978.

Our studies indicate that the average yield must increase for the carriers to remain financially sound. We believe the increase should approximately equal the inflation rate.

Net Income: A financial forecasting method has been developed at Boeing that depicts an airline's income and cash-flow statements as well as balance sheets. It also provides financial sums for airline groups. Inputs include actual and forecast data on traffic, yield, airplane fleets and prices, inflation and interest rates, fuel prices, and other costs. Within the program, financial statements and various types of ratio analyses are used as a basis for judging the solution. Input factors are adjusted iteratively to give a reasonable financial solution.

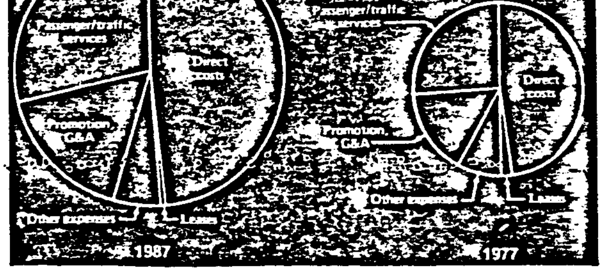
The Boeing method, together with the operating data and costs described in this article, has been used to develop the net income after taxes for the U.S. trunk carriers, as shown in F-19.

F-19 pictures a healthy future for air transportation in the U.S. in spite of inflationary and environmental pressures, but only if airline managements and aircraft manufacturers meet an enormous challenge.

Airlines will have the constraints on growth mentioned here, and their management must struggle with them. So does technology have constraints. F-20 presents some of the items which affect the manufacturer in his quest for improved technology and its byproduct, productivity.

Given that unforeseen technological advances do

F-17 TOTAL OPERATING COST FOR UNITED STATES TRUNKS TODAY AND IN A DECADE



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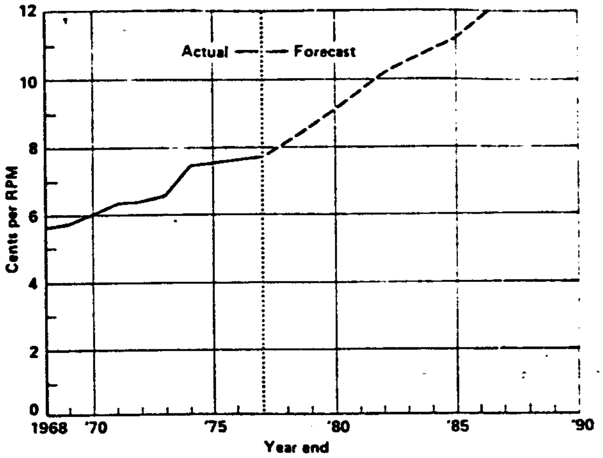
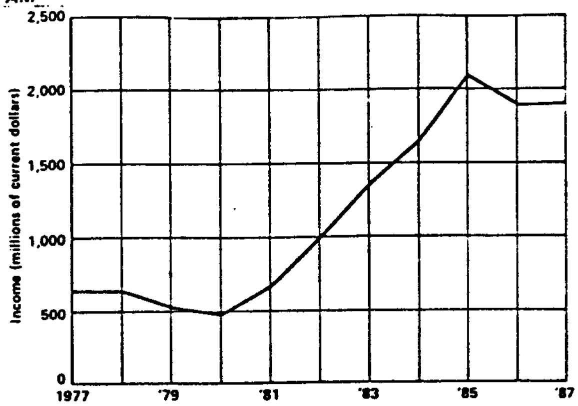
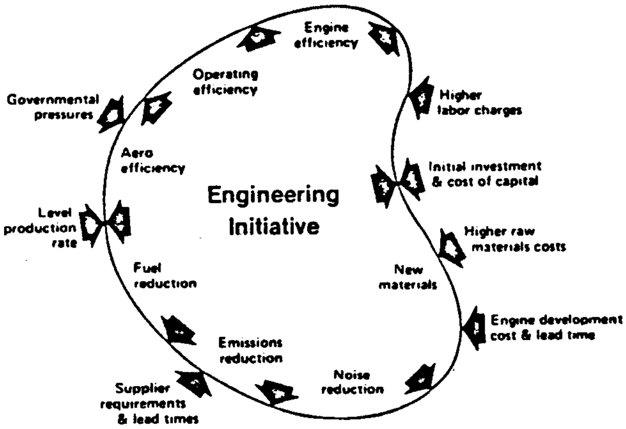
F-18 AVERAGE REVENUE PER RPM (YIELD) FOR U.S. DOMESTIC AIRLINES



F-19 NET INCOME AFTER TAX FOR U.S. TRUNKS AND PAN AM



F-20 PRESSURES AND CONSTRAINTS FOR TECHNOLOGY



not occur in the next few years (yielding an increase in productivity similar to the introduction of jets), the rapid historical pace of new-technology airplane introductions has unquestionably paused. Moreover, even given a breakthrough in technology in the near future, the R&D time required to exploit the advance plus the time to incorporate it into a certificated transport program would prevent the introduction of this "new" airplane until about 1990. Thus the design engineer at this point finds himself restricted to relatively minor improvements.

Nevertheless, the designer must press the technology, particularly in view of anticipated increases in fuel price. He must strive for efficiencies in aerodynamics and propulsion to reduce fuel consumption. Potential fuel rationing compounds the pressure on him to do so. He must also reduce engine emissions and minimize noise; laws now mandate this. Acting against the designer, moving him toward compromise, are the increasing costs of initial investment and the cost of capital. Proposed improvements through technology must be made on a cost-effective basis to be attractive to the airlines.

In addition, the commercial field cannot expect technology advancements to flow into it from the military sector as much as in the past. Commercial aviation must provide its own technology improvements. Charles F. Kettering said, "Research is something that if you don't do it 'till you have to, it's too late." The greatest pressure on the designer may be to seek out the needed advances and then apply them effectively without government support. This will require a higher proportion of R&D costs to be underwritten by the commercial industry than on any previous program.

Conclusions: U.S. air-transportation, now a maturing industry, has seen the historically rapid pace of new-technology airplane introductions pause. Twelve years will have passed between the 747 and the 767. Airplane productivity increases will be difficult to achieve in any large steps. Design engineers must tackle those systems and subsystems which influence airplane-related operation costs—fuel consumption, maintenance, initial capital costs, etc. At a fixed revenue level any reduction in operating costs translates directly, dollar-for-dollar, into before tax profit. The "bottom line" at the airlines is and will continue to be the name of the game in the years to come in air transportation.

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1. Simpson, R. W., "Get Ready for the Great Debate on Transportation," Feb 1976 *Astronautics & Aeronautics* (*A/A*), p. 38.

2. Hopps, R. H., and Sim, G., "Commercial Transports Decade of Derivatives," Feb 1975 *A/A*, p. 24.

3. Veron, T. H., "Airlines and the Elusive B-E Point," May-June 1969 *Financial Analysts Journal*.

4. Oral Argument Materials of Eastern Air Lines, Inc., before the Civil Aeronautics Board Dockets 31290 and 20891 of June 9, 1978.

Transport aircraft sales and deliveries, January-June 1978

Type	Firm orders at 30.6.78	Sales Jan-Jun 1978	Deliveries at 30.6.78	Deliveries Jan-Jun 1978
Propeller-driven				
Britten-Norman Islander . . .	N/A	N/A	734	—
Britten-Norman Trislander . .	N/A	N/A	54	1
de Havilland Canada DHC-6	626	38	589	29
de Havilland Canada Dash-7	14	8	3	2
Fokker-VFW F.27	674	9	659	1
Hawker Siddeley HS.748 . .	322	3	320	5
Short Skyvan	122	4	118	—
Short SD3-30	24	12	14	3
Pure-jet				
Aérospatiale/BAC Concorde	9	—	9	—
Airbus Industrie A.300	97	44	53	11
BAC.111	223	1	223	1
Boeing 707/720	934	—	923	6
Boeing 727	1524	33	1357	49
Boeing 737	582	42	521	15
Boeing 747	406	49	327	12
Fokker-VFW F.28	133	3	125	4
Hawker Siddeley Trident . . .	117	—	115	3
Lockheed L-1011	185	18	151	2
McDonnell Douglas DC-9 .	954	28	877	6
McDonnell Douglas DC-10	298	23	253	8
VFW-Fokker VFW 614	16	—	10	—

Notes: N/A indicates that no data was received from the manufacturer; a dash (—) indicates no sales or deliveries during the period; all information supplied by manufacturers.

U.S. Domestic Airlines ⟶ 2100 aircraft ⟶ 340,000 seats

wide bodies 110,000 seats

DC-9, B727, B737 120,000 seats

DC-8, B707, old B727 110,000 seats

Growth rate (1970-1977) ~ 6 %/year

World Airlines ⟶ 5000 aircraft ⟶ 750,000 seats

Growth rate (1970-1977) ~ 8.3 %/year

∴ Need 100,000 seats/year new & replacement worldwid.

Model	Present Price ($)	Cost/Seat
B727-200	13×10^6	$90,000
B737	9×10^6	90,000
B747	55×10^6	145,000
B757	18×10^6	105,000
B767	24×10^6	120,000

- NUMBER OF PASSENGERS IN AN ALL-TOURIST LAYOUT (SEAT PITCH 34 IN., .87 M): 180 OR MORE. CORRESPONDING DESIGN PAYLOAD: 20,000 KG (44,100 LB). AN UNDERFLOOR FREIGHTHOLD VOLUME OF AT LEAST 50 M^3 (1,762 CU.FT) WILL BE REQUIRED. STANDARD SIZE BELLY CONTAINERS ARE PREFERRED.

- RANGE, WITH ABOVE MENTIONED PAYLOAD: 2,200 KM (1,200 NM) IN A HIGH-SPEED CRUISE, ATA DOMESTIC RESERVES. MAXIMUM RANGE (REDUCED PAYLOAD): 3,200 KM (1,726 NM) AT LONG-RANGE CRUISE TECHNIQUE.

- MAX. CRUISING SPEED AT 9,150 M (30,000 FT) ALTITUDE: M = .82. DESIGN LIMITS: M_{MO} = .85, V_{MO} = 704 KMH (380 KTS) EAS.

- FIELD LENGTH REQUIRED FOR TAKEOFF AND LANDING, ACCORDING TO AIRWORTHINESS RULES: 1,800 M (5,900 FT) AT SEA LEVEL, ISA + 20 °C (95 °F), AT MAXIMUM (CERTIFICATED) TAKEOFF WEIGHT. RUNWAY LOADING: LCN = 30, RIGID PAVEMENT, 18 CM (7 IN.) THICKNESS.

- REGULATIONS: FAR PARTS 25, 36 AND 121. THE NOISE CHARACTERISTICS MUST SHOW AN IMPROVEMENT RELATIVE TO THE 1969 VERSION OF FAR PART 36 OF 1[illegible] EPNdB.

Fig. 1-5. Initial specification of a hypothetical short-haul airliner for introduction into service around 1980

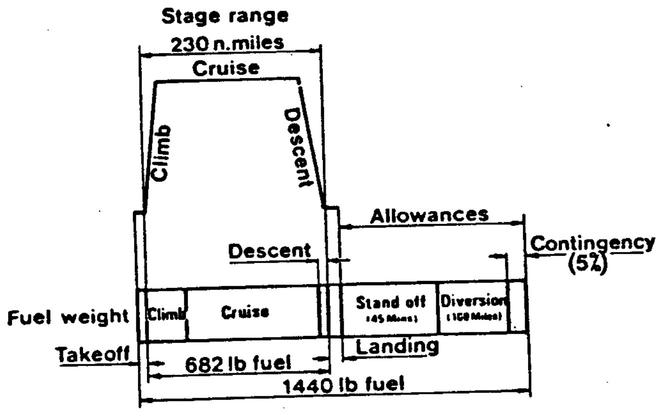
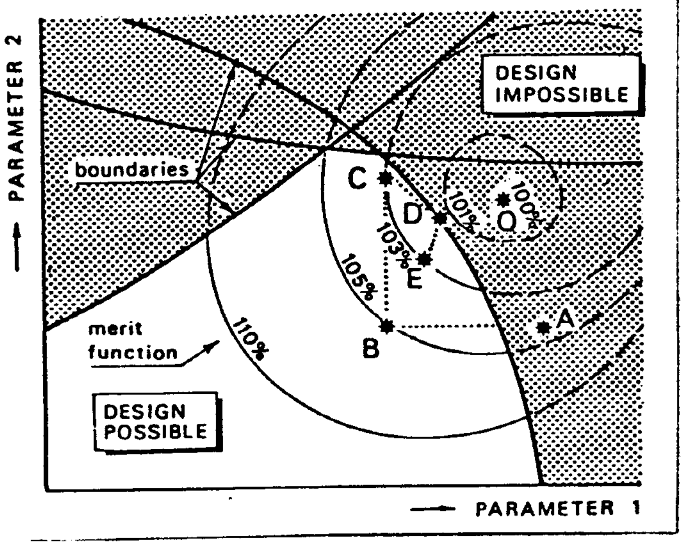
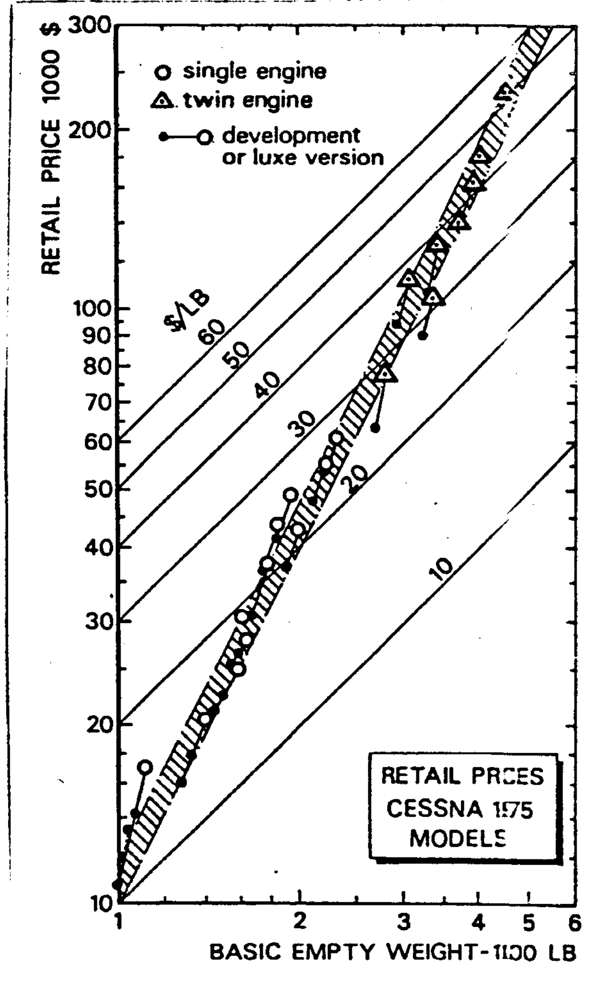


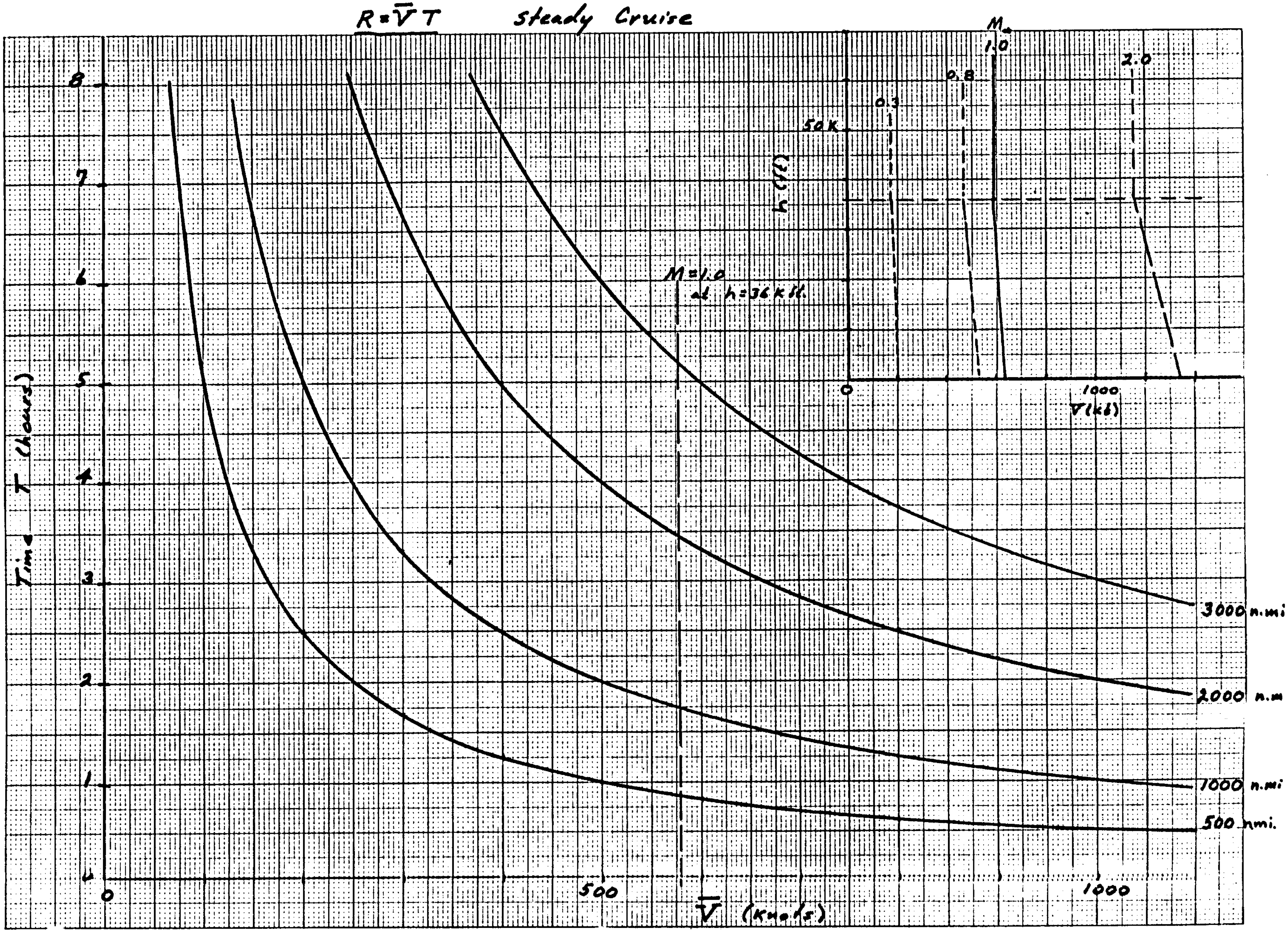
Fig. 11-7. Typical fuel breakdown for an executive aircraft (Ref. SAWE Paper No. 996)





a. Retail prices of light aircraft with reciprocating engines

4.49



R = $\bar{V}$ T   Steady Cruise
Time T (hours)
$\bar{V}$ (knots)
0
500
1000
1
2
3
4
5
6
7
8
3000 n.mi
2000 n.m
1000 n.mi
500 nmi.
M=1.0 at h=36 k ft.
h (ft)
50K
V (kt)
1000
M
0.3
0.8
1.0
2.0

Some Technical Extrapolations

by JOHN H. McMASTERS
and JAMES L. NASH-WEBBER

Periodically one hears that a new plateau has been reached in high-performance sailplane design; that some sort of practical limit has been reached in aerodynamic and structural design. Indeed, after looking at the extraordinary performance of modern Unlimited Class machines like the AS-W 17 and *Nimbus II* and comparing this performance with that of individual components (e.g. low-speed airfoils), it is difficult to see where one can now make the quantum jumps that Wortmann-type airfoils, fiberglass construction, and general drag cleanup have made possible in the recent past. Dr. Wortmann himself warned[1] as early as 1972 that we cannot expect airfoil performance to improve much beyond that presently attainable with his most recent sections, and that only minor improvements can be hoped for by tailoring a specific section to a specific design — a capability which is now at hand. These improvements may, however, be slight compared with the leap between early NASA (NACA) laminar-flow sections and the FX series of 1961-63 vintage. How valid then is the assessment that we have reached a practical limit and where can we look for further improvements if this judgment is not valid?

A good departure point is provided by Bikle's article[2] in which he shows the performance potential of a good selection of 1971 vintage Standard and Open Class machines. It can be seen that between 1955 and 1970, Standard Class maximum L/D has increased from approximately 30 to 36, and the corresponding Open Class increase has been from 40 to 45. Corresponding increases are also shown for speeds at both L/D_{max} and sink rates of 2 m/s (accompanied, of course, by substantial increases in wing loading). A conservative look at the problem might indicate a further steady increase in an L/D_{max} of around 50 for the Unlimited Class competition machines, with wing loadings eventually reaching 480 N/m² (10 lb/ft²) in strong conditions and a gradual improvement in high speed performance (i.e. a general flattening of the polar towards sink rates of 2 m/s at 100 kt. without loss of low speed capability). This is truly incredible progress to those who date their initiation into soaring from the 1930's, but hardly the kind of quantum leaps that those younger, more arrogant designers from the late 1950's and 60's are accustomed to.

Obviously L/D_{max} is only one simple index for assessing sailplane performance and sometimes it is not even the most important one (provided it is above some certain threshold). To put the next 25 years of soaring in perspective, we must backtrack here and look at the whole spectrum of "soaring" machines (some of which do not now exist) and look at prospects for improvements in each category over the next quarter century. Competition is not the only aspect of soaring.

- Sport Class
  - 1-26 Category
  - 13-Meter Class
- Competition Class
  - Unlimited Class
  - 15-Meter Class
  - Standard Class
- Ultralights
  - Hang Gliders
  - Ultralight Sailplanes
- Self-Launched Sailplanes
  - Motorgliders (conventional "powered" sailplanes)
  - Powered Ultralights
  - Man-powered Aircraft
- Miscellaneous
  - Commercial Gliders
  - Solar-Powered Airplane
  - Research Sailplanes

Given this fascinating range of "motorless" flying devices, we can identify tentative (and always incomplete) areas of possible improvements based on the usual major technical/economic areas (e.g. aerodynamics, structures) in aeronautical engineering. Application of a little ingenuity and a lot of hard work should result in very substantial gains being made in the next twenty-five years in these areas. Here is where some of these gains may come from.

Improved Design and Analysis Techniques

The introduction of large-scale computers has caused a mild revolution in aerospace vehicle design — the full di-

Table 1. Comparison of Current and Future Sailplanes.
Performance is shown in Figure 3.

Type	Span m (ft)	Area m² (ft²)	AR	Weight N (lb)	Wing load N/m² (lb/ft²)	L/D_{max}	at V (kt)
1. Rigid wing hang glider	10.0 (32.8)	14.0 (150)	7.15	1160 (260)	83 (1.75)	10.5	26.3
2. Self-launching UL sailplane (Archaeopteryx w/ ¼ BHP)	18.0 (59)	18.6 (200)	17.4	1670 (375)	89.6 (1.875)	42 (w/power)	25
3. SGS 1-26	12.2 (40)	14.9 (160)	10	2640 (593)	177 (3.7)	21.5	42
4. Std. Cirrus	15.0 (49.2)	10.0 (107.5)	22.5	3240 (734)	324 (6.82)	35.2	51
5. AS-W 12	18.3 (60)	13.0 (140)	26	4040 (909)	310 (6.5)	43	48
6. Nimbus 2	20.3 (66.6)	14.4 (154.9)	28.6	4150 (933)	288 (6.03)	47.2	52
7. VVG sailplane	18.6 (61)	10.25 (110)	33.8	4900 (1100)	478 (10)	43	59
8. Laminar sailplane	18.3 (60)	11.6 (125)	28.8	4450 (1000)	383 (8.0)	92.8	70
9. Laminar sailplane configuration (without boundary-layer control)	18.3 (60)	11.6 (125)	28.8	4450 (1000)	383 (8.0)	49.3	58

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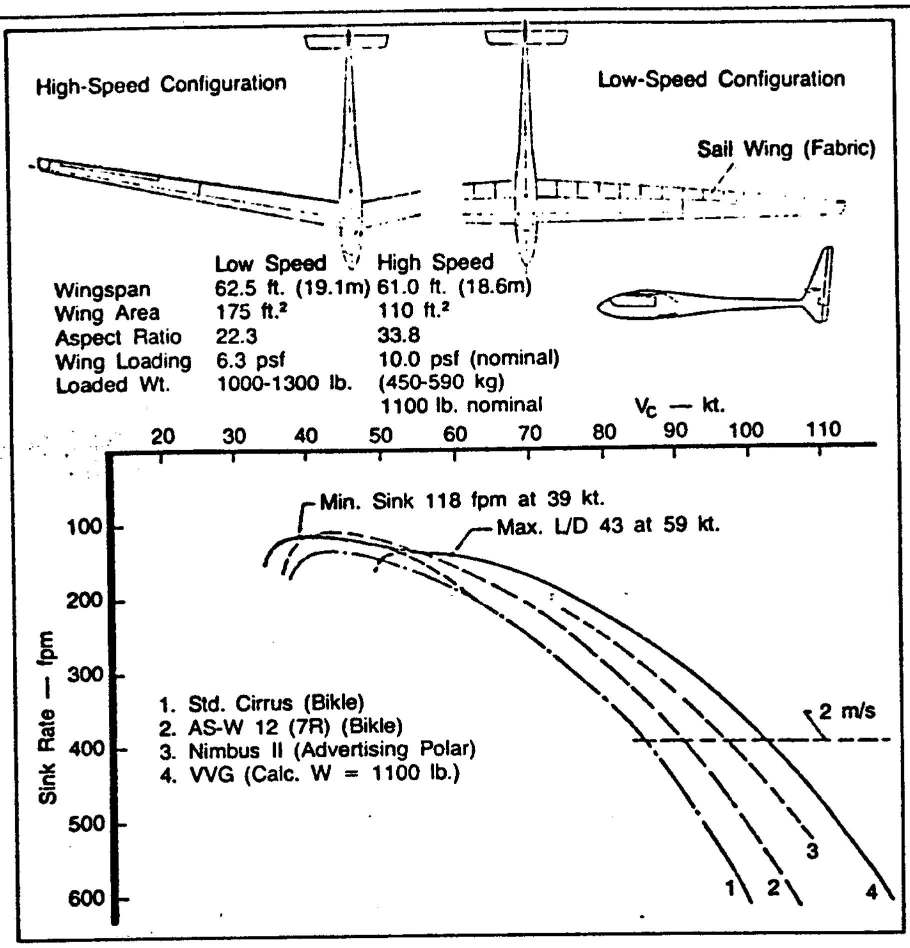


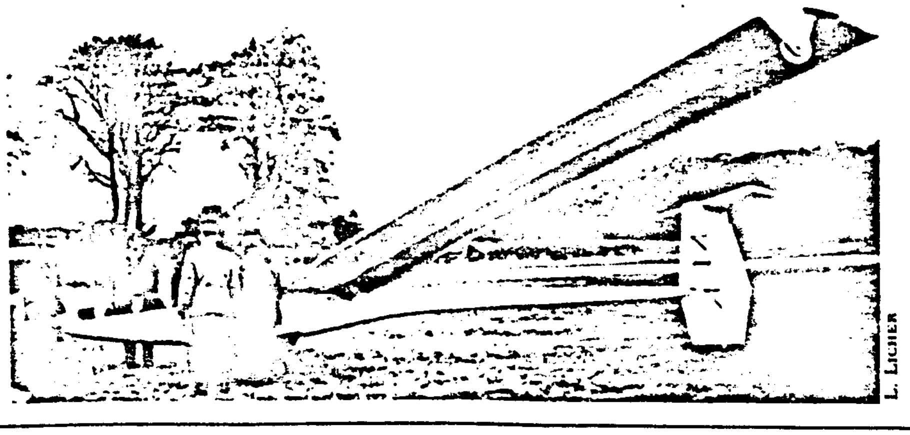
Figure 1. A Very Variable Geometry Sailplane.
This design results from McMasters' 1972 study for a "Sigma" category sailplane using Wortmann's fabric sailwing concept to provide full-span camber-changing "Fowler-type" flaps giving area changes of over 50%. The rather undesirable variable sweep wing is used to tension the trailing edge of the fabric flap when extended and to minimize trim changes when changing area and camber, thus also minimizing horizontal tail size.

mensions of which are not yet clear. Many elaborate design programs covering both the definition of an overall system and individual components are in widespread use by the major aircraft companies. The mathematical complexity, long run times, high user cost, and the general hassle of using such programs puts their application outside the reach of most manufacturers of General Aviation aircraft, not to mention the amateur and semi-professional sailplane designer. Simplified programs and analysis/optimization schemes need to be developed for the small-scale designer and can be within the next 25 years.

Variable Geometry Techniques

A strong argument can be made that a near limit has now been reached in achievable performance with a sailplane of fixed geometry. This is mirrored in the pressures exerted to modify the recently existing Standard Class rules. There are many ways to vary the geometry of a sailplane in flight with varying degrees of performance increase possible. Some of these are embodied in the design shown in Figure 1. Among the more obvious possibilities are:

- Variable camber: This is the usual case of employing camber changing flaps on the wing. This technique is already in widespread use, however recent advances in analytic techniques indicate[3] that the performance limits for multi-element, slotted flap systems have not yet been reached.
- Variable area: The desirability of changing the wing loading of a sailplane between climb and cruise flight conditions is well-known, and the obvious way to accomplish this is to vary the area of the wing in flight. Poor results with the BJ-3/4 partial-span Fowler flap systems, except in very strong lift conditions, led to the more exotic schemes for the *Sigma* and Wortmann's idea[4,5,6] for, 30-100% chord increase with fabric "sailwing" flaps. A combination of the analytic advances in variable camber flap technology, advanced structural techniques and full-span Fowler flaps appears to hold great promise for performance increase.
- Variable span: Varying wingspan to both alter aspect ratio and wing area in flight has fascinated designers for decades. However, the quantitative nature of performance increase with this technique has seldom been addressed and the mechanical difficulties of how such span increase is to be achieved make the technique unpromising. In 1972 Goodhart[7] presented an analysis, based on the *Sigma* computer program, which showed variable span to have little advantage. However, the Stuttgart Akaflieg people don't think so and the new SF-29[8] is now a reality. Actual data from flight test data will finally be available to assess the overall merit of the scheme.
- Variable incidence: Any component (e.g. the fuselage) of an aircraft has drag values which change with angle of attack. Thus, if one could alter the lift coefficient of the wing while operating the fuselage at its nominal optimum angle of attack, a



small saving in parasite drag could result. Thus the idea of a variable incidence wing emerges. The magnitude of drag reduction by this technique is probably small, however, and must be weighed against structural complexity and weight penalties.

- Variable thickness/chord: Contemplate the new Beatty B-5[9] wherein the camber changes as the thickness/chord ratio changes.
- Variable center-of-gravity: Use of highly cambered flaps/variable area wings can result in severe trim changes and hence large tail sizes and/or trim drag values. Thus, one might consider variable cg or translating the wing fore and aft; or even variable-sweep wings.

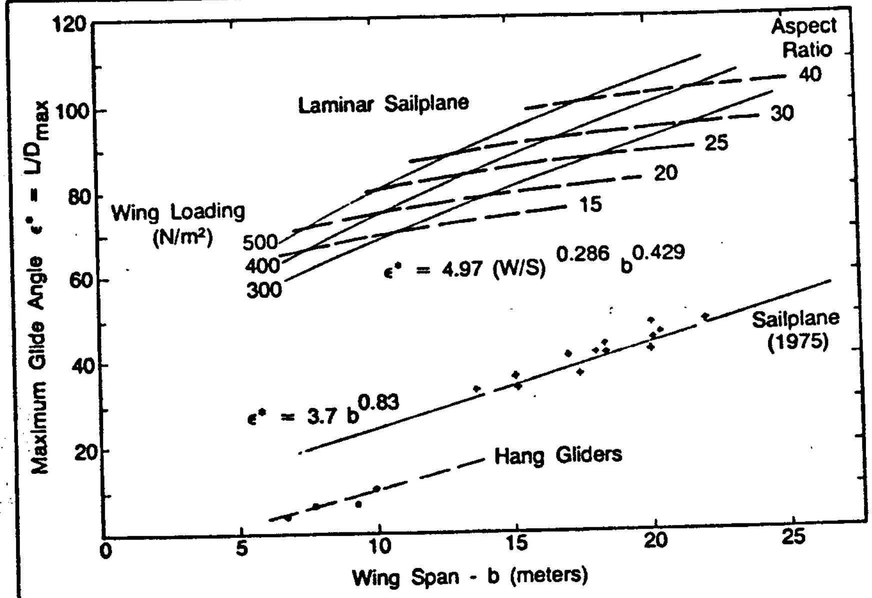


Figure 2. Effect of Wing Span on Maximum Lift/Drag Ratio for Several Types of Soaring Machines.

A Box of Equations for Figure 2.

Current Technology Sailplane

$$C_D = C_{D_O} + \frac{K C_L^2}{\pi AR}$$

$K = 0.0038\pi AR + 1.04$ (good wing/fuselage junction and Wortmann airfoil)

$$C_{D_O} = 0.003\left(\frac{S_{wet}}{S}\right)$$

$$\epsilon^* = \left.\frac{L}{D}\right|_{max} = \frac{1}{2}\left[\frac{\pi AR}{K C_{D_O}}\right]^{1/2}$$

(No Rn scale effects accounted for explicitly.)

$$\epsilon^* = 3.7\ b^{0.83}$$

(based on statistical data)

Laminar Sailplane

$$C_D = C_{D_O} + \frac{K C_L^2}{\pi AR}$$

$K = 1.00$ (ideal wing)

$$C_{D_O} = 1.328\left(\frac{S_{wet}}{S}\right) Rn^{-1/2}$$

$$R_n = \frac{V\bar{C}}{\upsilon} = \left[\frac{2}{\rho\upsilon^2}\right]^{1/2}\left[\frac{W}{C_L AR}\right]^{1/2}$$

(Reynolds number based on average wing chord)

$$\epsilon^* = \left[0.7\right]\left(\frac{S_{wet}}{S}\right)^{-4/7}\left(\frac{\rho\upsilon^2}{2}\right)^{1/7} W^{1/7}\ AR^{2/7}$$

if $W = 218b$:

$$\epsilon^* = 4.97\left(\frac{W}{S}\right)^{2/7} b^{3/7}$$

(Std. Sea Level with $\frac{S_{wet}}{S} = 2.5$)

W = gross weight (N)
b = wing span (m)
S = wing area (m²)
AR = aspect ratio $= \frac{b^2}{S} = \frac{W}{S} \bullet \frac{b^2}{W}$
$\bar{C}$ = average chord = S/b
1 lb. = 4.45 N
1 kg$_f$ = 9.8 N

Std. Sea Level
$\rho = 1.22$ kg/m³
$\upsilon = 1.46 \times 10^{-5}$ m²/S

$$\frac{S_{wet}}{S} = \begin{cases} 3.1 \text{ conventional sailplane} \\ 2.5 \text{ flying wing} \end{cases}$$

High Lift Devices

"High Lift Devices" refer to the whole range of mechanisms for selectively increasing the lifting capability of a basic wing of given size. This range covers simple flaps, slats, slots, etc., some of the more elaborate variable geometry approaches discussed previously, and, finally, much more complex boundary layer control and circulation augmentation techniques. The whole topic is too elaborate to detail here, but will loom large in the soaring literature from now on. An excellent modern overview of the topic is provided in the impressive paper by A.M.O. Smith.[3] Herein probably lie the next major gains in sailplane aerodynamic performance.

Boundary Layer Control

The ultimate advance in drag reduction and aerodynamic performance gain could come from mechanical boundary layer control. This would take the form of sucking the low energy air in a boundary layer from *all* surfaces of the aircraft. The mechanical complexities of doing this are somewhat mind-boggling, however. An additional consideration of major proportions to the soaring purist is: "Where does the power to drive the requisite pumps come from?" Certainly not from an engine in the sailplane! However, at least two alternative sources of energy to drive an electric motor exist: a) solar cells on the wing surface and b) air-driven generators in the tip vortices. Perhaps by the year 2000 . . . .

Such a scheme is of at least academic interest since it demonstrates the actual upper limit that sailplane performance can hope to reach. Even if that performance can never be attained, it's fun to know what the theoretical boundary might be. Figure 2 shows the results of a very simple-minded analysis of the performance potential of a range of "laminar sailplanes." The variation in L/D $_{max}$ with wingspan, aspect ratio, and wing loading for the laminar sailplane are shown in comparison with statistical data for L/D $_{max}$ versus wingspan for modern fiberglass ships and current hang gliders. Figure 2 shows clearly that talk of having reached an ultimate aerodynamic limit is largely nonsense when we view the problem from the plane of the theoretical. To further demonstrate this point some polars for a variety of soaring devices are compared with a typical laminar sailplane in Figure 3 and the possible configuration of one version, based on a combination of several of the advances discussed in this article, is shown in Figure 4.

Wing Tips

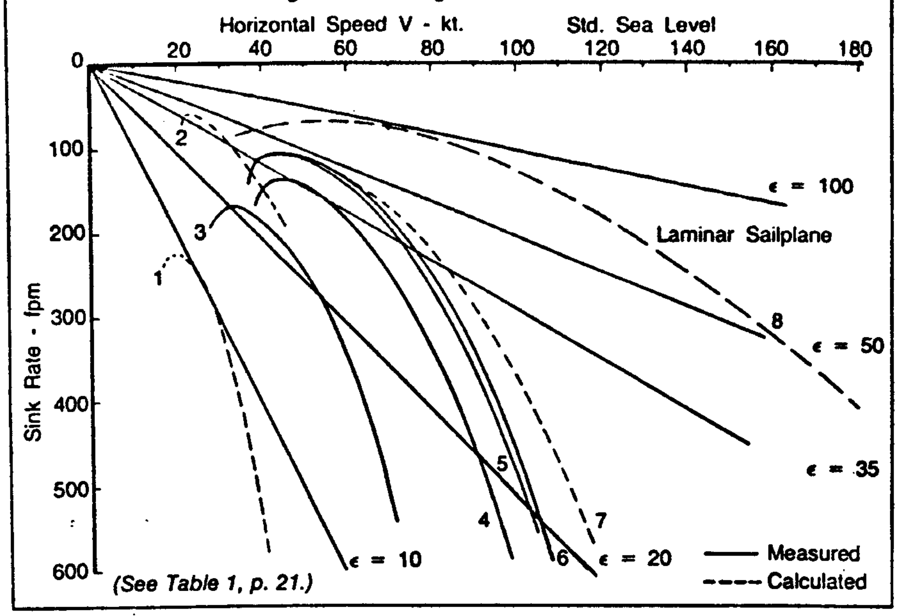
Nearly optimum overall wing planforms are a commonplace item in sailplane design, however a lot of room still exists for second order advances such as improving the flow in the wing/fuselage junction region and in the area of the wing tips.

- Tip vortex energy extraction: NASA has recently spent a lot of money to develop what it calls "winglets"[10] — little wings placed at large dihedral angles at the tips of conventional wings. The idea here is that these winglets are immersed in the local flow field of the tip vortex. If the winglet is shaped and oriented properly, so that they develop lift relative to the *local* flow direction, a part of this lift appears as a *thrust* (i.e. negative drag) in the aircraft's flight direction. We then see an apparent reduction in induced drag. A neat little trick and it apparently works. The gains aren't huge but could be significant on a sailplane while thermaling.
- "Vulture" tips: There is apparently some connection between winglets and branched, slotted wing tips of the vulture. No one seems to have a complete understanding, even at this late date, of all the tricks these devices play. It seems probable that mechanical equivalents (*not* necessarily direct copies) of land soaring bird wing tips could help improve sailplane thermaling efficiency and delay tip stall — particularly in ultralights.

Annular Biplanes

And then there is the so-called "annular" or "ring" wing. This device has almost magical induced drag characteristics since it "apparently" has no wing tips, and because one can usually build a biplane which weighs less than an equivalent area monoplane, there may be some mileage in such schemes, particularly for ultralights. Going a little further, if one squashes the "ring" into an ellipse (when viewed from the front) and staggers the upper and lower halves of the ring, it might be

Figure 3. Gliding Performance. ε = L/D



(See Table 1, p. 21.)

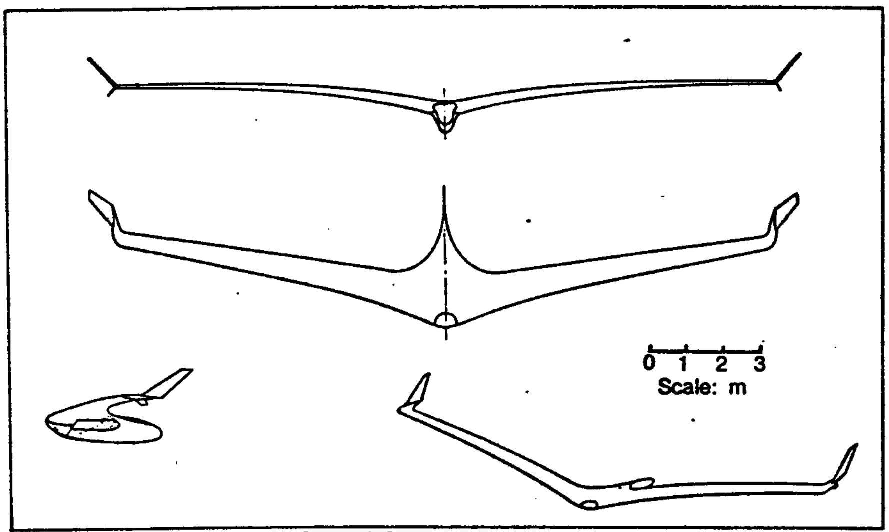


Figure 4. The "Diomedae" Control Configured Fully-Laminar Flying Wing Sailplane.

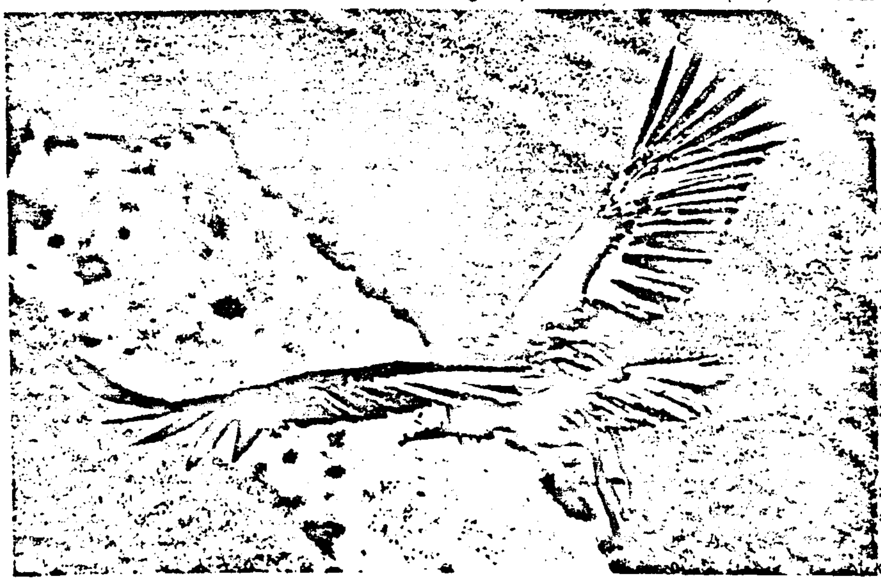
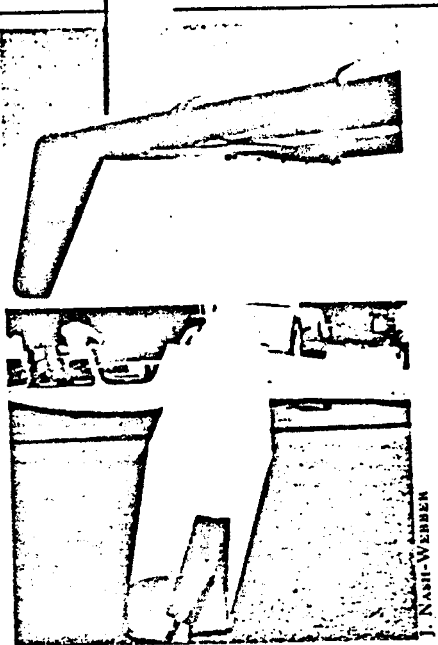
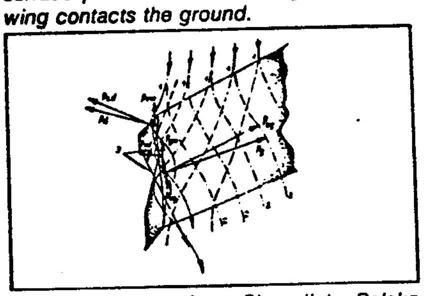


Figure 5. The Ultimate In Winglets: *A California Condor in a low-speed glide with glide-path control devices (feet) extended.*

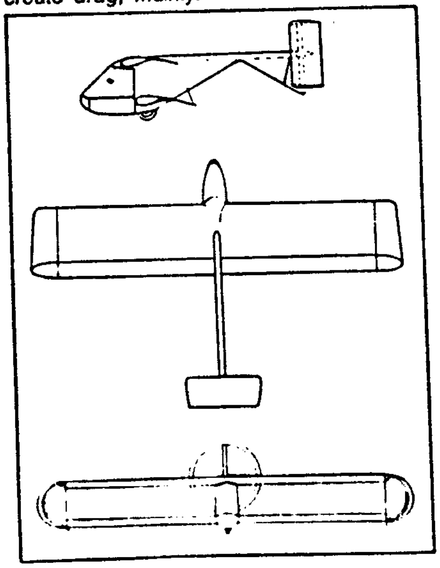


Darmstadt University's Akaflieg has made an experimental installation of NASA winglets on its D-37 prototype. Note how the lower surface pivots to avoid damage when the wing contacts the ground.



Vector diagrams from Skrzydlata Polska show how winglets utilize tip vortices to produce thrust.

Figure 6. A Hypothetical "Ring Wing" (Annular Biplane) Ultralight Glider Configuration.
Span 7m (23 ft.), area 14m² (150 ft.²), empty weight 335N (75 lb.). The curved wingtips must lift (inward) to be effective. Simple streamwise non-lifting tip plates create drag, mainly!



possible to construct a very light wing, without tips, which behaves as a giant slotted flap with low drag.

Variable Asymmetry Wings

An elliptic planform wing is "optimum" in rectilinear cruise as everyone knows. There is also, however, a corresponding optimum *asymmetric* planform for each turn condition (bank angle) which thus reduces trim drag. If one could alter the symmetry of the wing in flight by variable span and/or variable area techniques, it might also be possible to eliminate conventional ailerons, thus making full span flaps more feasible, to reduce wing torsion loads, and to alleviate flutter problems. There are half a dozen possibilities here, such as *inflatable* cuffs on the wing tips which either telescope in and out, or simply roll up. Telescopic rigid wings are clumsy by comparison.

Control Configured Vehicle (CCV) Technology

The notion of "stability *and* control" in aircraft design should actually be read as "stability *or* control." In general, stability and maneuverability (controllability) are conflicting requirements. In the case of natural flying devices (birds, bats, etc.) the evolutionary trend has almost universally been from an initially highly stable configuration towards one of high maneuverability at the expense of stability. This has been possible due to the co-evolved highly efficient sensing system which is directly connected to the control devices of the animal. In the case of human piloted aircraft, there is substantial lag between the time that a control input requirement can be sensed and then acted upon, thus the machine must possess a relatively high degree of inherent stability for the humble human to "keep on top of it." Thus, most aircraft have tended to be "stability configured." The upshot of this is that modern conventional aircraft (including sailplanes) require a rather large empennage and other major design compromises whose sole function is to bring the stability/control of the machine within the limited range of the weakest link in the system — the pilot.

Great advances have been made in control system technology during the last decade, particularly in the areas of autopilots and "fly-by-wire" systems which now allow the designer to circumvent the time constant limits of a human pilot. The concurrent incredible advances in microelectronics hold promise for highly sophisticated fly-by-wire systems within the economic and reliability range of interest to sailplane designers. Since the empennage and fuselage on conventional sailplanes represent major sources of both drag and weight, their size reduction or outright elimination and replacement by an electro-mechanical autopilot system capable of artificially stabilizing the sailplane could result in very substantial performance gains. Such a system may eventually make feasible a truly high-performance flying-wing sailplane such as shown in Figure 4, a goal which has tantalized yet eluded designers for decades.

Advances in Sailplane Structures

General Jimmy Doolittle's maxim: "Simplificate and add lightness" has always been the watchword of the soaring fraternity. Although the trend towards ballasted ever-higher wing loadings in racing machines might cause us to think that weight-saving is no longer important, we should always remember that on the weakest days the lightest ships may be the only ones to make it home, and thus win the contest. However, these ships must also be strong enough to carry huge amounts of ballast to deal with strong days.

A study[11] performed for the Penultimate Sailplane Group at the Massachusetts Institute of Technology shows that a go-for-broke no-cost-limit 19-meter ship using a combination of the most modern advanced-composite materials might be built at only 51% of the weight of a conventional glass machine, while still able to carry enough ballast to equal the highest wing loadings now permitted. Such a ship would only have an advantage on the weakest of days, and its $100,000-plus price tag would likely deter most of us, but the trend is clear: Already we see the HP-18 available with a graphite spar, and the AS-W-17 with a graphite fuselage, in each case at a considerable increase in cost, but with impressive weight savings. Fortunately, the cost of some exotic materials is falling steadily as they find their way into mass-market goods (e.g. Kevlar in tires, graphite in golf clubs). In particular, graphite fiber may soon be available at prices which compare well with aircraft-grade aluminum.

We can safely predict that these

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trends will continue, and that the use of high-strength, high-stiffness fibers will slowly expand in the next few decades, spreading out from two quite disparate centers: The super-racers, whose constituency will continue to bear almost any cost, and the high-performance ultralights, such as man-powered aircraft, in which a few pounds saved is the difference in performance, and the number of pounds of material to be bought is not totally prohibitive. Will the 1-26X of 2001 be made of a superplastic? We predict that as Schweizer engineers pore over the accounts, they will conclude that it makes sound economic sense — and so it *will* be done.

On the simplification front, the advent of cheap computer-driven tools will lead to cheaper moulds and production tooling for plastic sailplanes, and we can look forward to computer-directed final finishing processes at once cheaper and much more accurate than the current manpower-intensive ones. The laziest among us will enjoy a ship as accurate as A. J.'s (but he'll still win of course). The whole question of how to simplify the structure of *all* types of soaring machines, without degrading strength and performance, merits substantial future work. Your checkbook will measure this progress.

Advances in Instruments and Control

It was recently remarked in *Soaring* that the poorest-performing sailplanes, which need help most, are generally cursed with the most wretched instrumentation. Indeed, were every current sailplane brought up to the instrument standard of the most sophisticated, we should be well off, if considerably out-of-pocket! The present slow trend towards better varios having really accurate compensation, filtering, and netto features will continue for the fleet as a whole, and the racers will mostly carry some form of speed-command gadgetry on board, ranging from "big leaks" to the costliest of air-data computers.

Advances on these fronts will be aided by the explosive growth in the minicomputer/microprocessor industry, with a continued steady cost decrease likely. Mechanical instruments will continue to increase in relative and absolute cost. Thus your 1-26X of 2001 may well feature a totally electronic panel duplicating all the visual and audio information of today's finest panels, at a smaller cost, relative to the airframe cost, than is found today.

We predict, however, the natural death of the climb-rate averager, as more people finally appreciate the near uselessness of historical information when trying to set a MacCready ring or its equivalent on the basis of estimated future conditions.

As the microcomputer becomes an accepted integral part of most aircraft, including sailplanes, the way will be open for almost any desired degree of automation of the flying. One may argue whether it is at all desirable, but the fact remains that it will be possible cheaply to automate optimal (in the MacCready sense) inter-thermal *and* dolphin-style cruising, with perfect control of structural loads and pilot comfort/discomfort in turbulence. It is even possible that thermaling will be automated, but we have a gut feeling that the little gray computer inside your skull will always do a better job in this area. We are guaranteed, however, to see a proliferation of schemes for automatic flap and/or automatic pitch control. These already range from the almost passive Schuemann scheme,[12] which depends on accurate control of center-of-gravity position, to the complex hydraulically-actuated flaps used on the Akaflieg Aachen's experimental FK-3, *Nimbus*, and PIK-20 flying testbeds. The more exotic schemes will take aeroelastic deformations of the wing structure into account explicitly. Kesselyak[13] has shown that it is possible to use these deformations beneficially to extend the low-drag speed range. These matters are treated in depth in this year's volume of *Technical Soaring* (Vol. IV). In the end, the best method will win out, and become ubiquitous, since the cost/benefit ratio is already clearly advantageous.

We are also likely to see more careful consideration of optimal maneuvering strategies in the future. Already, top pilots are more than careful to avoid overly-large control deflections, but still change speed and bank quickly. We can look for *g*-meter inputs to the micro-computers of 2001's racers, which will generate audio signals to tell the perspiring pilot how to maneuver optimally to minimize losses.

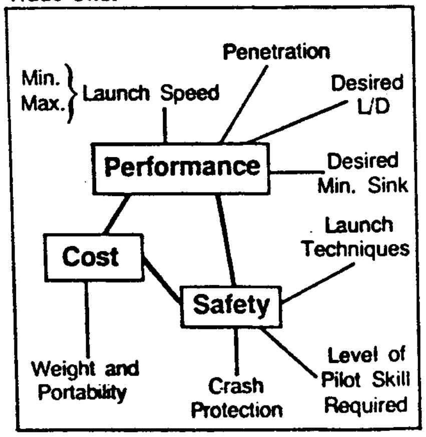
Finally, thermal finders: We'll give you odds that the black box you'll buy for good money will prove a great letdown, in every way, but that is a 1976 prediction!

Ultralights

Ultralight "gliders" are a rather controversial topic within the soaring community — perhaps rightfully so based on bad PR during the 30's and the recent poor accident record in hang gliding. But for many grass-roots soaring types who can only look on enviously while their more affluent colleagues fly magnificent fiberglass superships, and for a large constituency of young people with little, if any, previous flight experience, the main virtues of the basic hang glider (simplicity, low cost, easy transportability) outweigh the safety and performance limitations of the type. With the rapid recent rise in hang gliding activity, there has been concern and pressure for the SSA to become involved in hang gliding — or to ignore it altogether. The situation has been partially resolved by the steady growth of an independent organization, the U.S. Hang Gliding Association, which reflects the fact that, while there are major areas of overlap in interests and participation, the hang glider people are largely a different, younger group than the bulk of soaring enthusiasts. There are, however, many possible types of ultralight motorless flight vehicles, and while the SSA can breathe a little easier not having to take any major responsibility in shepherding the emerging hang glider movement, other ultralight types remain of major potential interest to traditional soaring types.

Most promising immediately is the so-called *ultralight sailplane* (with a foot-launching option) which fits into the performance/price gap between high-performance hang gliders (e.g. *Icarus V*, *Quicksilver*) and the Schweizer 1-26. The desirable characteristics of such a machine have yet to be fully defined. However it is clear that

Figure 7. Ultralight Sailplane Design Trade-Offs.



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some sort of balance must be achieved between three basic factors: performance, cost, and safety. This is shown diagramatically in Figure 7.

The advent of a true ultralight sailplane with adequate performance (including penetration capability), low initial and operating cost, and adequate safety could be a major boon to a wide segment of the SSA membership and justifies much technical work over the next 25 years. We haven't begun to scratch the surface in this area yet. As an aside, the ultralight sailplane represents a project whose magnitude can be handled in many universities. A current construction project along this line at Purdue is stimulating enormous interest among aeronautical engineering students — most of whom would not have contemplated full-scale construction of a true sailplane of even 1-26 size, cost, and performance. In this connection, Stan Hall's *Vector* is a nice idea,[14] but like other current foot launched sailplanes (e.g. VJ-23/24), its construction seems overly complex. There is a lot of room for improvement here, too.

Beyond the simple ultralight sailplane and advanced versions of hang gliders (perhaps employing Princeton type sailwings[15] as a compromise between performance and transportability), a number of ultralight types can be envisioned. First among these are the array of man-powered aircraft. The range here includes:

- Kremer Competition MPA's (fine museum pieces)
- Sport MPA's (solely man-powered) or man-powered augmented ultralight sailplanes
- Sport MPA's augmented with non-internal combustion engine stored energy devices

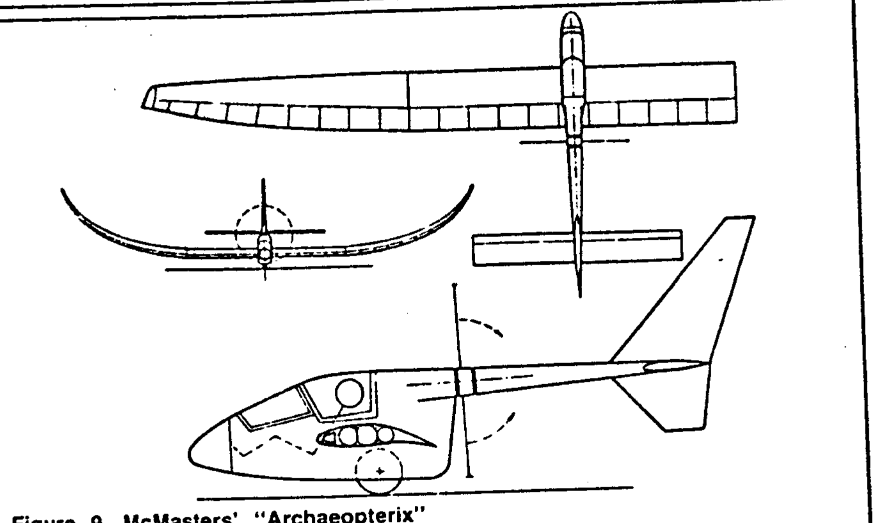


Figure 9. McMasters' "Archaeopterix" Self-Launching, Stored-Energy, Man-powered Sailplane.

- Powered ultralight gliders (the ultralight equivalent of the motor-glider)
- Solar powered ultralights

All of these types are envisioned as more or less conventional looking aircraft with geometry distorted in varying degrees (i.e. large wing spans, large wing areas) to match the very minimal values of available power and the consequent very low flying speeds. Ultimately, the question of penetration capability looms large in the design of such vehicles, and the overall design problem of striking a good balance between low power required, low weight, relatively large size, adequate structural strength and stiffness, sufficient speed range, and cost, is an extremely complex one. Whole new ranges of competition can emerge in these areas, and the ultimate direction these developments will take simply cannot be predicted with the knowledge we have at hand.

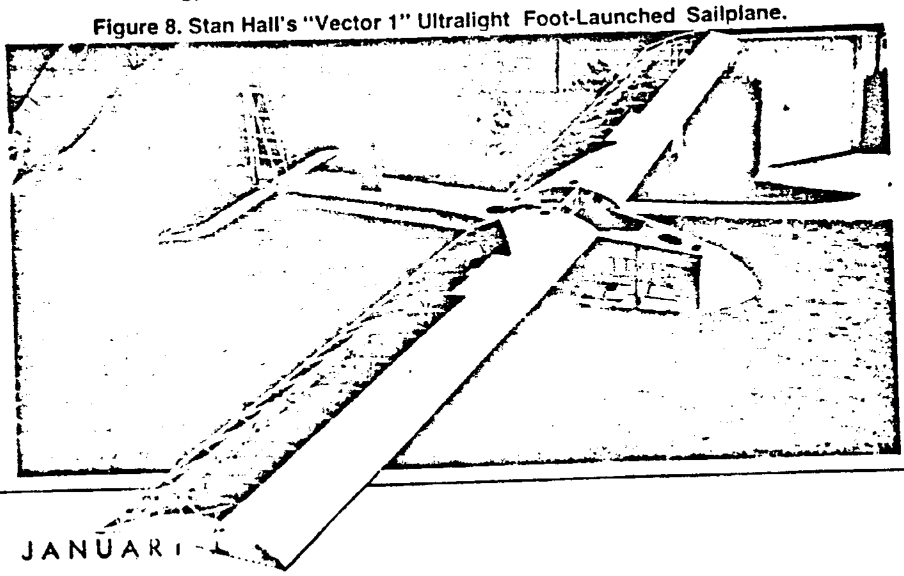
Finally, one can contemplate the emergence of ornithopters, powered by the pilot or by very low power stored energy devices in a quest to truly fly like a bird. The technology in both MPF and ornithopters is still quite primitive and offers a rich field for further development. However, one can safely predict that a man-powered, unaugmented ornithopter movement will not be giving the FAA of 2001 any gray hairs.

Miscellaneous

In addition to all the advances described above, there are numerous bits and pieces of progress to be made in almost every other technical and operational aspect of soaring. For example, we haven't really mentioned motor-gliders. Somewhere along the line someone will come up with a good configuration for a real *self-launching* sailplane rather than the present crop of powered aircraft that look "sort of like" a glider — or sailplanes that have an ugly growth at some undesirable place on (in) the fuselage. Then there is the prospect for low-energy tow techniques. Owners of superlative *Nimbus* 9's and AS-?-32's will probably still rely on aero tows (if they can stand $25 a crack in 1976 bucks!) but we'd rather see a cheaper alternative. Somehow kites or balloons may figure in these schemes, or maybe even solar-powered electric winches with flywheels.

Flight testing will continue to increase in importance and sophistication, particularly if good instrumentation in small, cheap packages becomes available. We would love to get our hands on some Bikle-style polars of a few modern ships in various turn attitudes, and a few Joe Average super-

Figure 8. Stan Hall's "Vector 1" Ultralight Foot-Launched Sailplane.



ship owners might like to have accurate custom polars for their particular machine courtesy of a vendor at the local sailport.

As it now stands, the soaring pilot foots the whole bill on new generation sailplanes for R&D that isn't paid by dedicated volunteers, Akafliegs, etc. Sport soaring might thus benefit if, as Wortmann and others have suggested, commercial and government agency uses could be identified for "glider" type vehicles. Oran Nicks at NASA has proposed the development of large glider trains (with an optimized tow vehicle) for low-cost cargo transport.[16] One can also contemplate lots of uses for self-launched sailplanes in atmospheric research, resource surveillance, etc. There must be many more possibilities besides U-2's and quiet airplanes using Schweizer 2-32 airframes.

Finally there is the question of who is going to do all the fascinating work (and more) that we have outlined above. Given that the commercial potential for racing sailplanes will remain limited and consequently that the number of Schweizers, Laisters, et al must remain few; who in the U.S. will generate the kind of sophisticated stuff that the German Akafliegs continually produce? Aeronautical engineering departments at MIT, Stanford, Cal Poly, etc. are, of course, one possibility. But who pays for this? Does it require a lot of money? We don't know the answers to these questions, but think one area of real progress in technical soaring between now and 2000 will be in motivating students (and faculty) in various universities to do soaring-related engineering. Herein lies the great virtue of the ultralight — they are relatively cheap, simple, and embody most of the aspects of full-blown sailplane design. They have the glamour to attract student interest, and yet are viable student-level projects which don't cost many thousands of unavailable dollars. If an American ultralight Akaflieg movement can be established now, then it could eventually evolve into a full sailplane-oriented technical community. We think the encouragement of such university (or even major city) based organizations should be considered an SSA priority in the near future. Get-togethers like the past two MIT Symposia can be great spurs to this sort of organizing, and have huge value in reporting and publicizing soaring progress. These conferences should continue to be given the same level of material and moral support as any national soaring contest.

Concluding Comments

Crystal ball gazing is a tenuous business and probably meaningless in the long run. The ideas presented in this article represent the current best guesses of only two members of the soaring community. We can't begin to claim we have foreseen a fraction of the technical possibilities which will emerge over the next 25 years. Further, many of the possibilities which are clearly technically feasible may become economically or socially (although hopefully never aesthetically) absurd. It is our principal goal in presenting this survey to tease the imagination of those who will contribute to the major advances which will unquestionably occur between now and the year 2000. It is further hoped that this article has demonstrated that soaring and sailplane design has in no sense reached its ultimate potential. While the next quantum leap in competition sailplane aerodynamic performance may be a while in coming, this provides a little breathing space for the instrumentation/electronic gadget/computer revolution which is almost upon us to develop. Then there is the vast ultralight spectrum which is just now emerging. Everything in good time and the sky's the limit!

John McMasters is chairman of the Ultralight Committee of the SSA Technical Board and Associate Editor of *Technical Soaring*. His interest in soaring and ultralights dates to the late '60's and he has contributed five articles to *Soaring* since 1971, including two co-authored with Curtis Cole and Paul MacCready, Jr. in the two MIT Symposia on Low-Speed and Motorless Flight. When this article was written, he was on the faculty of the School of Aeronautics and Astronautics at Purdue, but he is currently working on low-speed aerodynamic problems for Boeing. His primary interests are in the design and philosophical aspects of low-speed and natural flying devices — particularly sailplanes, ultralight gliders, and man-powered aircraft.

T. Guy Spencer, Jr.

Jim Nash-Webber is Chairman of the SSA Technical Board and Associate Editor of *Technical Soaring*. He progressed to soaring through an early interest in competitive aeromodeling (South African Junior Champion, 1957), and is currently President of the MIT Soaring Association, an SSA Instructor, and a fledgling competition pilot. Trained as an Aeronautical and Mechanical Engineer, he does experimental R&D on magnetohydrodynamics and other fluid mechanics problems at MIT, while maintaining low-speed flight problems as an obsessive avocation.

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ADVANCED CONCEPTS IN VARIABLE GEOMETRY SAILPLANES

by JOHN H. McMASTERS *Chairman, SSA Design and Configurations Technical Committee*

"Surprisingly, the seventies was not a time of much technical development in sailplane design. As the decade started, the Nimbus I and Glasflügel 604 were already producing the 48:1 glide ratios which were not to be significantly exceeded except by Dick Butler's ultimately-tuned 604."

— George Moffat
***SOARING*, Jan. 1980, p. 38**

PART I

The Unlimited Class is dead, long live the Unlimited Class!

We have heard this refrain off and on for a number of years and regardless of the merits or faults of a class of machines which can be afforded by very few of us, they remain the pinnacle of sailplane development. Here is the ultimate frontier of conventional racing sailplane development — the technical and aesthetic excellence which is perhaps without peer in any branch of aviation. Even for those who may never aspire to own such a monster, the design challenge it represents holds a fascination which cannot easily be denied.

For almost a decade Unlimited Class competition has been dominated by the magnificent triumvirate: The *Nimbus II*, AS-W 17, and Glasflügel 604. Each of these machines can be characterized as: excellent, beautiful, damned expensive, and, by modern standards, conventional. To a technically bent but casual observer, these machines with maximum glide ratios of near fifty (plus or minus three) must represent some sort of near practical limit in achievable aerodynamic performance. And if we take the narrow view of sailplane performance, wherein maximum glide ratio is the sole measure of the goodness of the beast, then yes!, we have indeed reached a near theoretical limit in performance — short of resorting to mechanical boundary layer control to "laminarize" the entire surface of the machine.

Of course our casual observer is a fool to believe that the *Nimbus* or AS-W 17 is the ultimate sailplane — or more correctly, the ultimate soaring racer. The plateau (rather

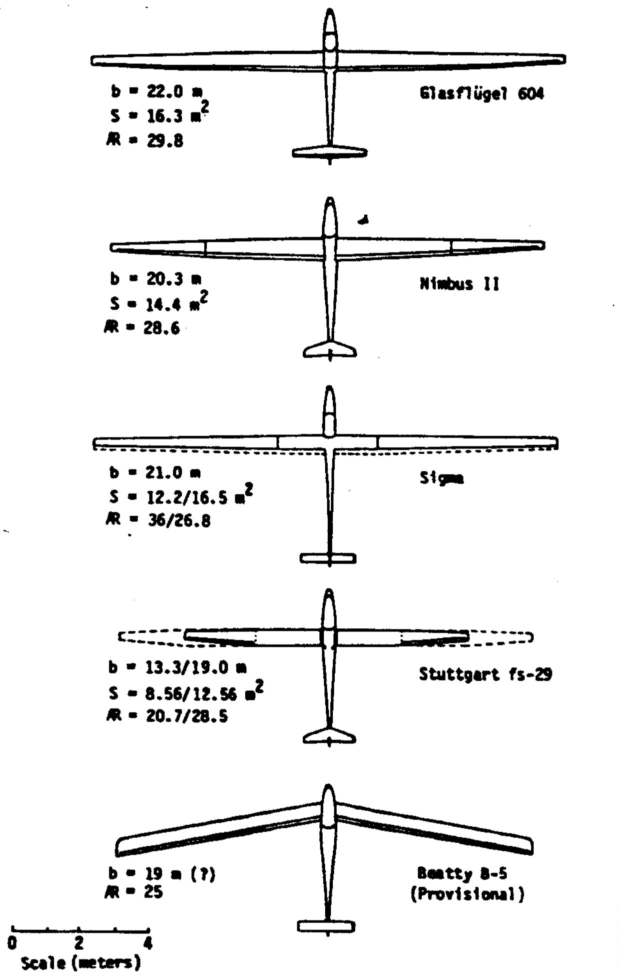


Figure 1. Planview Comparison of Several Open Class Sailplanes

than pinnacle) we have presently reached in sailplane design is that we have a more complete realization of all the factors which must be combined in careful balance to produce a machine which can climb and glide in a fashion which will maximize cross-country speed under a very wide variety of meteorological conditions. Insofar as maximum glide ratio, and the speed at which it is achieved, is a measure of this performance, then its value has merit in comparing various competing sailplanes. By itself, however, glide ratio tells a very limited story.

Of course the *Nimbus* or the 604 in the hands of a Butler, Johnson, Smith, et al., continues to win contests, but it is also possible to identify a whole range of technological advances which will make these machines as obsolescent as they in turn had excelled beyond their predecessors. Jim Nash-Webber and I (Reference 1) attempted to spell out some areas where these improvements might be made, and others (References 2, 3, 4) have done the same. But more importantly, while the big three continue to win the contests, a fair number of more advanced prototypes in this category have been built — each intended in its way to go beyond the present limit. However, while the existence of the bulk of these prototypes is known to many of us, few details of their actual performance (successful or otherwise) have appeared since the cessation of work on the original British *Sigma* project.

While the general trend in competition sailplane development has been toward use of advanced composite materials (Kevlar, graphite, etc.) and higher wing loadings, the rather obvious direction of the next purely aerodynamic advance has clearly been toward *variable geometry wings* — wings which can alter their shape at the discretion of the pilot to be more nearly optimum for a given flight condition. In modern military parlance these would be referred to as "mission-adaptive wings." Fitting a fixed wing with a simple hinged cruise flap is the easiest way to tailor it to a specified flight condition. Better (but more difficult) would be to alter the wing's overall camber, area, and/or span. Birds do this to great advantage, of course, and with better materials now at hand it is clearly only a matter of time (and money) before we too can accomplish much the same feat with the same facility with which we presently retract a landing gear. A representative sampling of existing advanced variable geometry sailplanes is shown, drawn to the same scale, in Figure 1.

As will be discussed presently, each of the schemes shown in Figure 1 has merit, and each approach has been seriously flawed in practice. The best route is presently still a bit unclear, but the variable geometry idea is going to stay with us for some time to come — until proven either useless or infeasible, or the proper approach is finally settled on. By understanding the problems encountered so far in this line of development, it can be hoped that those few brave souls who decide, against all good advice, to pursue super-span madness will be able to contribute something more than another failure to our presently meager base of experience.

Background

The idea for the present article came to mind as a result of the Third International Symposium on Low-Speed and Motorless Flight held at NASA's Langley Research Center (References 5, 6), Dick Butler's splendid use of the 604 in last summer's Nationals, and George Moffat's column in a recent issue of *Soaring* (Reference 7). How does one follow an act like a Butler or Smith/604 or Horvath/*Nimbus* combination? To my mild surprise, *none* of the twenty-eight

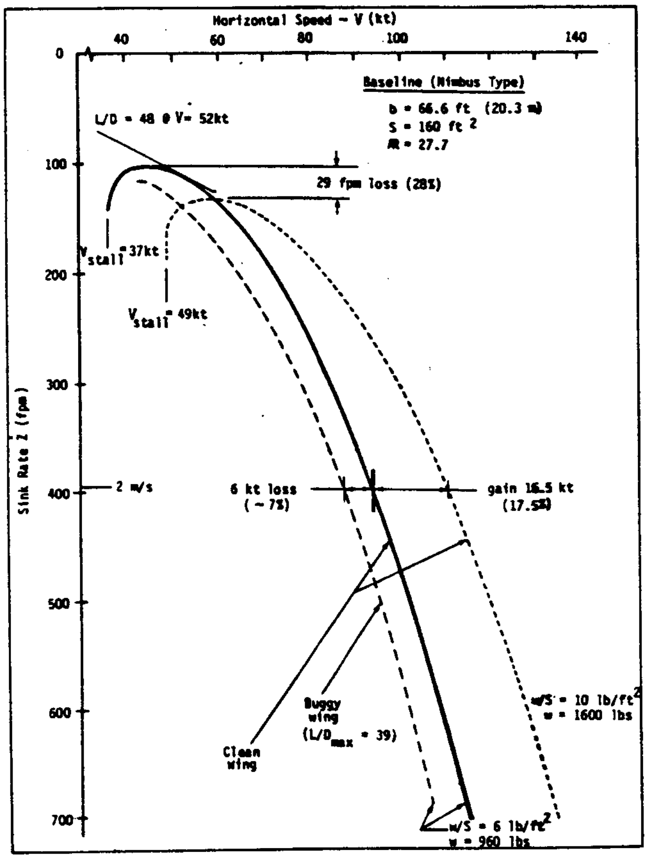
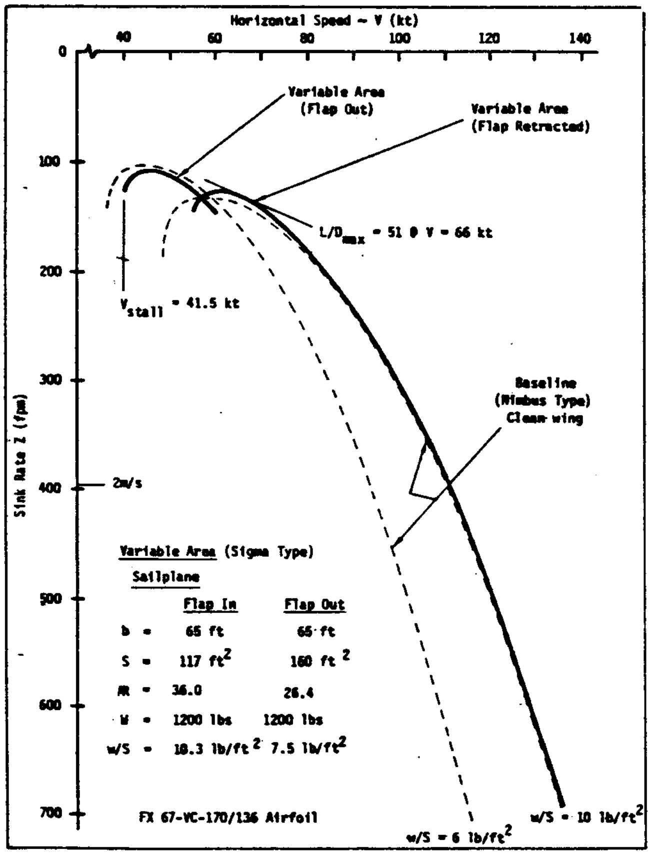


Figure 2. Nimbus Type Sailplane Performance



SHEET

Figure 3. Predicted Performance of a Sailplane with a Fowler Flap

papers presented at Langley in March '79 gave any new clues. People like Mike Teter told us rather cheerfully how to fly smarter in a contest, but nowhere was the next generation of racers discussed except very peripherally (e.g., Dieter Muser on "Advanced Composites in Sailplane Structures," Reference 6).

Well, one seldom goes to a conference only to listen to people present papers. Those are just as well read at leisure from the book of proceedings of the conference. Rather, one goes to talk to the people who attend such conferences. A couple of hundred people showed up at Langley last year, and among them (not presenting anything formally) were Pat Beatty and David Marsden. Pat has completed and flown his new B-5, and David Marsden (who should be far better known to all of us than he is) was on the verge of having his modification of the *Sigma* ready to go with a slotted flap replacing the Wortmann/Fowler flap of the original. Both of these are potentially show-stopping *new* Unlimited machines — especially if the rumored performance Marsden has achieved with his adaptation of the *Sigma* proves true.

In order to aid a discussion of the specifics of the various variable geometry schemes presently envisioned, a bit of background information on general sailplane design principles has been arranged in Boxes A and B. The more technically literate reader may skip these, but I've included the information here for completeness. Box A discusses basic racing sailplane design objectives. Box B discusses the characteristics of appropriate airfoil sections, which ultimately form the basis of successful sailplane-type wing design.

Variable Geometry Wings

Figure 1 shows the planforms of several *existing* sailplanes employing various basic types of advanced variable geometry wings presently envisioned. These schemes can be classified as:

- A wing of fixed span fitted with an *area* and *camber* changing (Fowler) flap (e.g. *Sigma* Braunschweig SB-11).
- A wing of variable span and area (e.g. Stuttgart Akaflieg fs-29).
- A wing of fixed span with an airfoil section of variable *thickness* and *camber* (e.g. Pat Beatty's B-5).
- Any combination of the above plus variable sweep and movable ballast.

The reasons which drive each designer to the choice of a particular type of variable geometry scheme are sophisticated and complex. Good rationalizations for each quite dissimilar choice can be cited amidst much arm waving. Since no measured performance data on any of the above machines have been published, however, it is difficult to assess the actual performance gains to be had from any of them and which may gain more performance than another. To put the specific discussion of the pros and cons of each on a quantitative basis, I have made some estimates of the hypothetical performance to be had from representative applications of each approach to an Open Class racer. To further tie down the comparisons, I have replotted the data from Dick Johnson's flight tests of a *Nimbus II* (Reference 8). This data, shown in Figure 2, is for Johnson's test at a wing loading (W/S) of 6.13 lbs./ft.² with wings smooth and with bug simulation together with my calculation of the

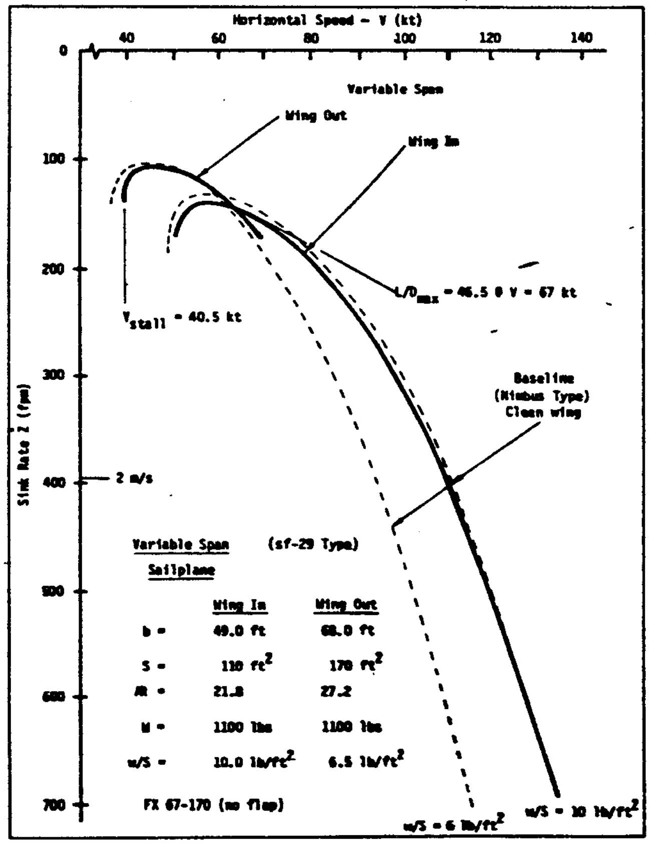


Figure 4. Predicted Performance of a Variable Span Sailplane

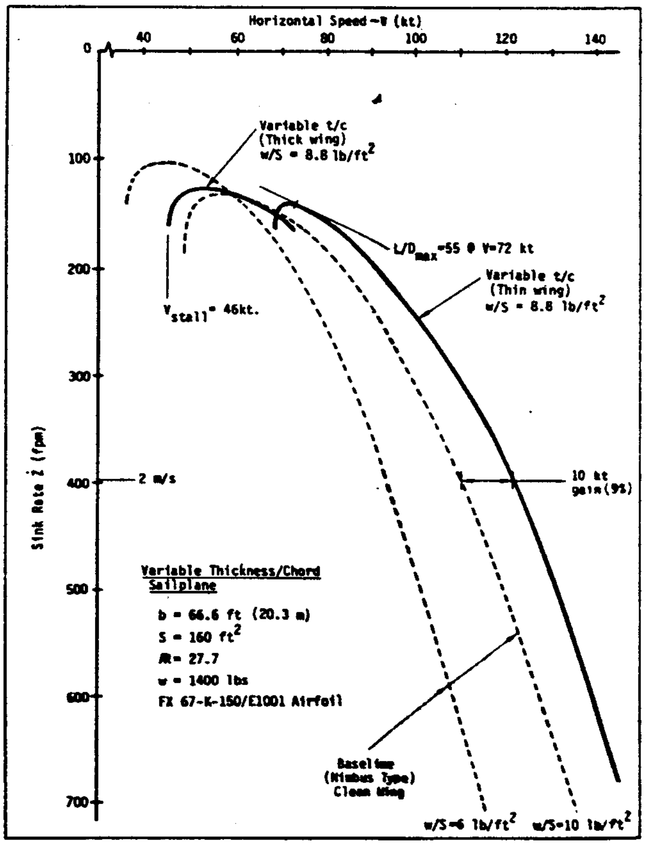


Figure 5. Predicted Performance of a Sailplane with a Variable Thickness Wing.

performance of the smooth wing machine at a wing loading of 10 lbs./ft.[2]. While the absolute values of Johnson's test results may be controversial, I believe the polars shown in Figure 2 are representative of a current generation Open Class racer. Thus I have labeled them "*Nimbus* type" polars (assuming a machine of *Nimbus* dimensions and airfoil sections) and have used them as a baseline against which the predicted performance of the various variable geometry sailplanes can be directly compared.

The characteristics of the *Nimbus* and the subsequent variable geometry sailplanes I assumed in my calculations are shown in the appropriate figures (2 through 5). The airfoil drag characteristics assumed in each example are those of the actual similar type machines as shown in Figure 6. All examples assume smooth wings, the drag values are adjusted for the Reynolds number appropriate to a given wing loading and lift coefficient value, and fuselage drag coefficients are adjusted to the appropriate reference wing area value. Each of my assumed sailplanes "looks" like its counterpart in Figure 1, with the exception of the B-5 type machine, which I assume to be similar in shape to an ordinary *Nimbus*. With these notes in hand, it is now possible to get down to (theoretical) brass tacks on which way one might profitably go in improving the already astonishingly good performance of a *Nimbus*-type machine (after reviewing Boxes A and B as necessary).

★*Next month, in Part II, John McMasters will present the results of some detailed performance estimates of representative sailplanes employing the various types of variable geometry wings discussed in general terms here.*

BOX A

A REVIEW OF FIRST PRINCIPLES

In order to discuss the specifics of individual variable geometry sailplanes, it is worthwhile to set down a few basic principles of design to guide the discussion. Basically we presently have a machine of almost standardized shape. It is characterized by a long slender fuselage of minimum cross-sectional (frontal) area; a narrow, high-aspect-ratio wing of superb smoothness and devious contour; and an aft-mounted empennage designed as a careful balance between minimum size, adequate stability and control, and good taste. At this level, *all* sailplanes look alike and are white. This recipe (except for whiteness) was settled on once and for all over forty years ago, and no one has yet demonstrated a better one.

Beyond this point we have a myriad of differences of opinion on almost every detail of the design. On the basis of hard-won experience over the years we have pretty well settled on the values of some basic parameters, however. I doubt that George Moffat or any competent aeroelastician would advocate wings on *competition* machines with spans much in excess of around 25 meters or aspect ratios much beyond 30-32 (although *Sigma* exceeds 36 by a hair). Use of carbon fibers, for example, puts these levels within reach structurally, but lateral control and weight (which increases more rapidly than span) considerations confine us to the values specified as a practical matter (according to me — and I'm prepared to debate the matter hotly).

Accepting these values as the bounds of span and aspect ratio, we have said nothing about the "best" values of wing area, weight (ballasted or empty), and consequently wing loading (weight per unit of wing area). To determine these values, and hence define the size of our sailplane, we must resort to some fundamental formulas of airplane design in general.

(Box A continued next page)

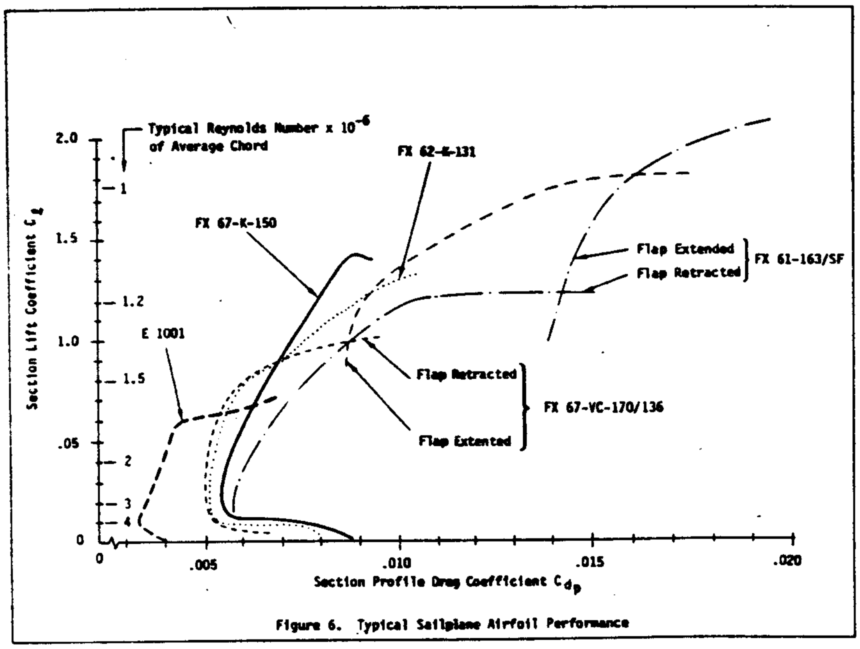


Figure 6. Typical Sailplane Airfoil Performance

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(Box A continued)

Reduced to the problem of a machine in a constant velocity glide, we may write:

$$\text{Glide Angle} = \theta = \dot{Z}/V \cong D/L$$

$$\text{Weight} = W \cong \text{Lift} = L = \tfrac{1}{2}\rho V^2 C_L S$$

$$\text{Drag} = D = \tfrac{1}{2}\rho V^2 C_D S$$

Where: $\dot{Z}$ = Sink Rate

V = Flight Speed

b = Wing Span

S = Wing Area

AR = Aspect Ratio

ρ = Air Mass Density (Mass Per Unit Volume)

C_L = Lift Coefficient

C_D = Drag Coefficient $= C_{D_v} + C_{D_i}$

C_{D_i} = Induced Drag Coefficient

C_{D_v} = Viscous Drag Coefficient

W/S = Wing Loading

These simple formulas can be rewritten and expanded in several forms (Reference 9). For the present purpose we may select the following:

$$\text{Flight Speed} = V = \sqrt{\frac{2\,W/S}{\rho\, C_L}} \quad ; \quad \underset{\substack{\updownarrow \\ V_{dive}}}{C_{L_{red\ line}}} \le C_L \le \underset{\substack{\updownarrow \\ V_{stall}}}{C_{L_{max}}}$$

$$V_{dive} \ge V \ge V_{stall}$$

$$\text{Glide Angle} = \theta \cong \frac{1}{L/D} \cong \frac{\dot{Z}}{V} = \frac{D}{W}$$

$$\text{Drag Coefficient} = C_D = C_{D_v} + \frac{k_w C_L^2}{\pi AR} \quad ; \quad AR = b^2/S$$

$$\text{Drag Force} = D = \frac{\rho}{2} C_{D_v} S V^2 + \frac{2k_w}{\pi\rho}\left(\frac{W}{b}\right)^2 \cdot \frac{1}{V^2}$$

$$\text{Sink Rate} = \dot{Z} = \frac{\rho}{2} C_{D_v}\left(\frac{S}{W}\right)V^3 + \frac{2k_w}{\pi\rho}\cdot\left(\frac{W}{b^2}\right)\cdot\frac{1}{V}$$

In addition, several other factors must be kept in mind:

1. Viscous drag and maximum lift coefficients vary with Reynolds number. The usual trends for sailplanes are for drag coefficient to significantly decrease and maximum lift coefficient to slightly increase as Reynolds number increases over the range of values in the sailplane problem. A simple relation for calculating the *average* Reynolds number on a wing is:

$$\text{Average Reynolds Number} = \overline{Rn} = (1.845 \times 10^5)\sqrt{\frac{W\ (\text{lbs})}{AR\, C_L}}$$

2. Wing weight increases more rapidly than wingspan (i.e., doubling the span would result in a wing far more than twice as heavy if both are constructed of the same materials), and any time one adds a gizmo or cuts into what could have been primary structure, a weight penalty is incurred.
3. Sailplanes frequently bank and turn. This foolish requirement introduces a whole range of additional design problems. For the immediate discussion, it is sufficient to note that:
   a. Turn Radius $= R = \dfrac{V^2}{g \tan\phi}$; ϕ = Bank Angle
   b. You must generate more lift by flying either faster at a given altitude, or by flying at a higher lift coefficient when turning than when flying level.
   c. Trim drag in turn increases rapidly as bank angle increases or, when flying at a given bank angle, the wingspan increases.

From the designer's point of view what all these equations and cautions say can be summarized as follows:

1. To fly slowly, we need a low wing loading and a high lift coefficient. For a machine of given weight this means we need a large wing area, and to do even better we could profitably use some means of increasing camber.
2. To fly fast we want a high wing loading and small wing area. In addition, the value of the lift coefficient will be low, and thus the value of the induced drag will be low. Thus, large wingspan is of diminishing value, and low viscous drag is all important. [Note: In the drag formula, viscous drag is increasing as the square of the velocity, and the sink rate formula shows that sink rate thus increases as nearly the cube of the speed. Induced drag is never negligible, but becomes increasingly small as speed increases.]
3. To minimize the sink rate of the machine, it is first important to fly slowly. This also reduces the turn radius at a given bank angle. At the condition of minimum sink rate, it turns out that the induced drag is between 60 percent and 70 percent of the total drag, and thus the next dominant term is the span loading (W/b). Thus, we want a large wingspan (b) and a lightweight (W) machine. This is nearly the reverse of our high-speed recipe.
4. At the condition of minimum glide angle (maximum lift-drag ratio) the viscous and induced drag terms are roughly equal and both must be minimized. As a result, a small wing area, small viscous drag coefficient, and large wingspan are all desirable. To first order, the value of L/D maximum is independent of the vehicle weight. Thus by ballasting the machine, we do not alter the glide angle appreciably, but only increase in proportion the forward speed and rate of sink at which the minimum glide angle occurs. [To second order, however, watch your Reynolds number as chord gets smaller!]
5. At any flight condition, low values of viscous drag are essential in order to achieve high performance.
6. The "best" sailplane will ulitmately be the one which can perform well under a wide variety of atmospheric conditions. Thus we want a machine which can be flown with a wide range of weights (or wing loading) and with aerodynamics to allow both excellent minimum sink rate performance *and* high-speed capability.

BOX B

LAMINAR FLOW AIRFOIL PERFORMANCE

The text in Box A described the importance of viscous drag minimization in maximizing sailplane performance at all flight conditions. It has been a cardinal rule in sailplane development for decades that to achieve low values of viscous drag, long runs of laminar flow on all surfaces of the machine are essential, and the adverse effects of wing/fuselage junctions, etc. must be minimized. Once the wing sizing has been accomplished and the problem of induced drag thus dealt with, improvement in flow quality on *all* surfaces of the machine is the final area in which the aerodynamicist can exert any major influence on performance.

On a machine like a high-performance sailplane, something like 60 to 75 percent of the total *viscous* drag is that attributable to the profile drag of the wing. Thus we pay great attention to the design of airfoils which at one stroke provide near minimum viscous drag over a wide range of lift coefficients, high maximum lift coefficient capability, reasonable stall characteristics, and sufficient thickness within which to build a structure of adequate strength and stiffness. As pointed out in several references (10,11) achieving all this in a single fixed-geometry airfoil is a near impossibility.

As the airfoil design problem sorts out, it becomes clear that achieving low drag at low-lift coefficients and high lift with low drag are two quite different problems, and if one is confined to a single airfoil section, the necessary compromises in performance can be quite dramatic. Some typical modern examples of airfoils important to this present discussion are shown together with their *ideal* performance in Figures 6 and 7.

At the present state-of-the-art in laminar flow airfoil development, it is now far easier to design airfoils with improved high-lift capabilities than it is to design ones which will produce lower drag values (at low lift coefficients) than was achieved by the NACA some thirty-five years ago. And the worst of the low-drag problems is that these sections require near perfect conditions of surface quality to perform as advertised. As we try to squeeze the last bit of low-drag performance from laminar sections, they become increasingly sensitive to any surface contamination (e.g. bugs). To see this effect dramatically, one need only refer to Dick Johnson's flight test articles to see the results of bug simulation on a *good* sailplane wing. Whether Johnson's simulation is representative of real bugs or not, the contamination effect is clearly displayed, and some of the more advanced sailplane types under discussion here demand high-speed airfoils which are even more sensitive than the friendly old Wortmann sections on the *Nimbus* and AS-W 17.

As general guidelines to the feasible in airfoil design, we may offer the following:

1. Very low profile drag at low-lift coefficient values can be achieved on the computer (and occasionally in the wind tunnel) if the extent of required low-drag bucket (lift coefficient range over which low drag coefficients are achieved) is small. Here a perfectly smooth and accurate surface is required, particularly in the leading edge region, and the thinner the section, the lower the drag value.

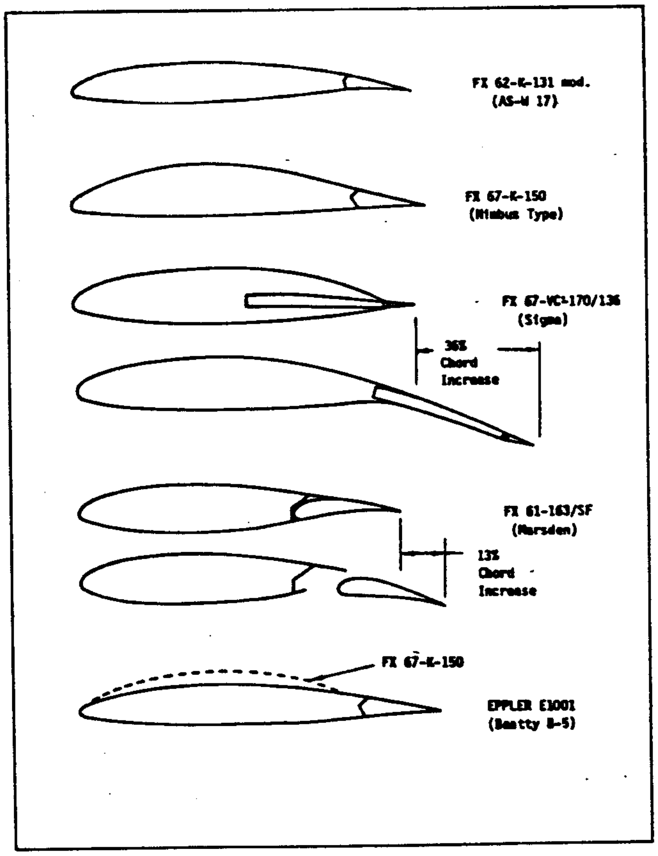


Figure 7. Modern Sailplane Airfoils

2. Increased camber, coupled with careful tailoring of the airfoil contour, can produce airfoils with dramatically high maximum lift coefficients. They will, however, be miserable at low-lift levels (high-speed conditions) because they will frequently separate (stall) on their *under* surfaces, at low angles of attack.
3. The extent of the low-drag bucket can be extended while retaining high maximum-lift capability by carefully tailoring both the upper and lower surfaces. The result is usually a section in which thickness increases in rough proportion to the width of the bucket with minimum drag increasing in proportion to increasing thickness.
4. Camber-changing, simple-hinged, "cruise" flaps give improved performance compared to sections without them — but the gains to be had are limited.
5. Slotted flaps are more powerful than simple-hinged flaps commonly used on most flapped sailplanes, but they also produce substantially higher profile-drag values.
6. The alternative (to slotted flaps) of a camber *and* area changing (Fowler) flap as used on *Sigma* and the SB-11 is attractive *aerodynamically* on a wing in Unlimited Class racer dimensions.

A VARIABLE-GEOMETRY GALLERY

The three aircraft pictured below on these pages illustrate two variable-geometry concepts — variable span and variable chord — discussed by author John McMasters in this series. It is noteworthy that these high-technology aircraft are the product of student academic flight groups (Akafliegs) in West Germany's technical universities.

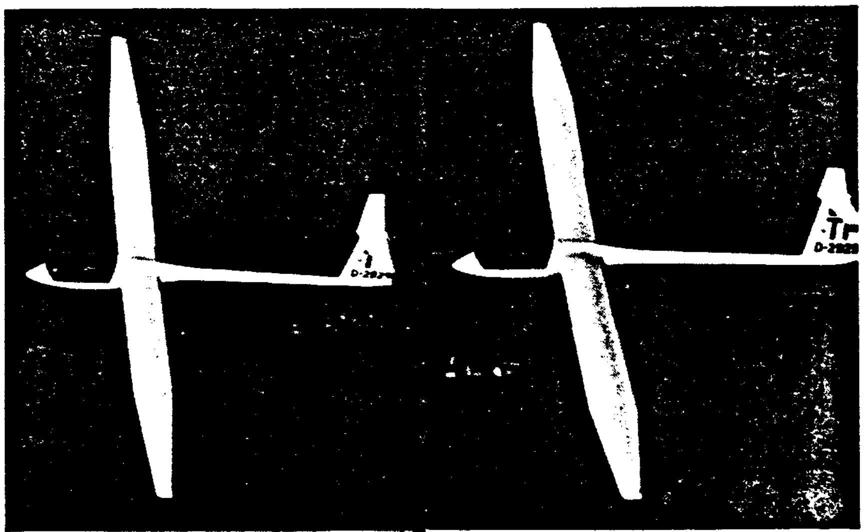
Variable span is represented by Akaflieg Stuttgart's fs-29 (left). The pilot of this remarkable craft can extend its span with twelve strokes on a ratcheted lever, increasing its stretch from 13.3 to 19 meters. The aspect ratio changes from 20.7 to 28.5, and the area enlarges by 47 percent.

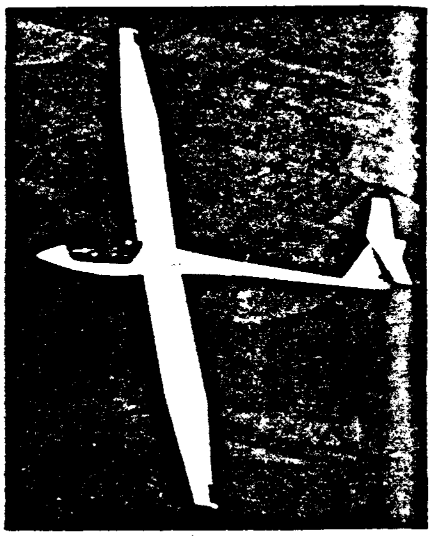
Akaflieg Braunschweig's SB-11 (center) and Akaflieg Munich's Mü-27 (right) employ variable-chord flaps to boost area — the SB-11 by 25 percent, the Mü-27 by 36 percent. The Mü-27 is the largest of the three, weighing 1800 lbs. when fully loaded; it is the only sailplane to use on-board power to extend and retract the wing elements. A button on the control stick activates electric motors to drive the flap mechanism. The bottom photos detail articulation and camber of the variable surfaces in extended and retracted position.

PART II

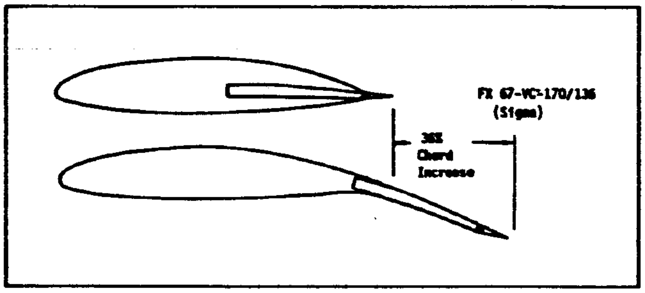
ADVANCED CONCEPTS IN VARIABLE GEOMETRY SAILPLANES

by JOHN H. McMASTERS





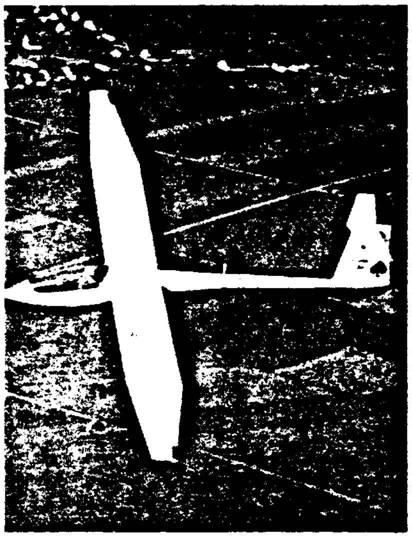
In Part I, some general aspects of the design of advanced racing sailplanes with variable geometry wings were discussed. With these general considerations in mind, we can now go on and explore the actual performance gains which might be expected from the use of each of the variable geometry approaches previously discussed. As specified in Part I, the performance predictions made are the author's best estimates and are not necessarily the values actually achieved by the particular sailplanes of each type which now exist. — J. M.

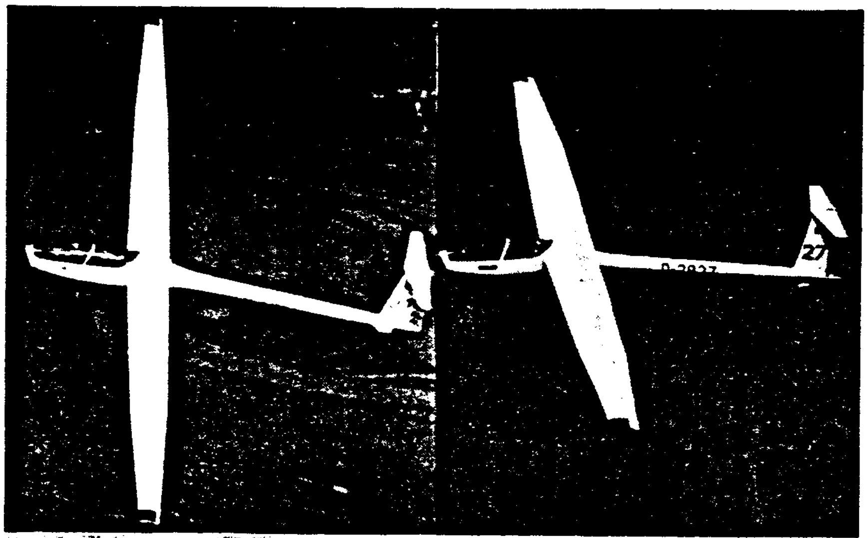


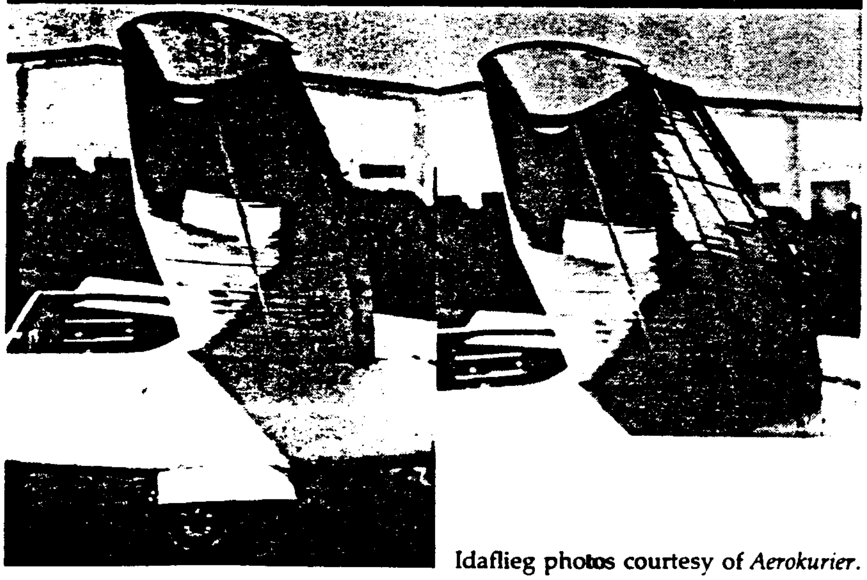
Sailplanes With Fowler Type Flaps

While the British *Sigma* (Reference 12) is the now classic example of extreme Fowler (area and camber-changing) flap application, this machine owes much to the previous experience and writing (Reference 13) of Pat Beatty and Fritz Johl of BJ-3 and BJ-4 fame. What Beatty and Johl proved with the BJ-3/4 was that unless the Fowler flaps ran the *full* span of the wing, the extra weight and complexity was very marginally worthwhile. (A *partial span* Fowler flap when extended creates a large "step" in the trailing edge of the wing, which is aerodynamically like another pair of wingtips producing a consequent increase in induced drag.) Further, the spanwise structural break on the wing undersurface (a critical area in high-speed flight) between the main airfoil and the leading edge of the large-chord flap when the flap is retracted, resulted in a performance penalty due to a partial loss of the potential full run of laminar flow on the wing.

The upshot, according to Goodhart et al., was the opinion that while the performance advantages of the Fowler flap were correctly described by Beatty and Johl, the BJ's hadn't gone far enough by half. Thus was born the *Sigma* with its specially designed Wortmann airfoil (c.f. Figure 7 of Part I). The *Sigma* was and is a beautifully complex airplane. According to Frank Irving, the level of craftsmanship demanded to accomplish the necessary sealing of the flaps in the extended position and to make the whole system work was beyond the project's capabilities at that time. The *Sigma*







Idaflieg photos courtesy of *Aerokurier.*

did work in its fashion with flaps either retracted or extended, but cycling the flaps during flight proved impossible on a reliable basis. After much work and soul searching, the *Sigma* was finally offered to serious investigators to pursue further work and research. Over a period of time inquiries were received from twelve qualified applicants with the *Sigma* eventually being entrusted to David Marsden in Edmonton, Alberta, Canada. Marsden (designer of the variable geometry *Gemini*) has now laid his skilled hands on the machine and fitted it with a sophisticated slotted flap. We have not heard the end of *Sigma* yet. Nor have we seen the end of the splendid Wortmann airfoil that went with it. Helmut Reichmann, armed with the SB-11 at Chateauroux, showed that whatever the mechanical shortcomings of the *Sigma*, a little more time, care, money, and carbon fiber could make the basic scheme work more than well-enough.

Whether the *Sigma*, or even the SB-11, used the Fowler flap to its full advantage is a subject of possible controversy. More experience will be required to tell, but the basic objectives of its use are easy enough to explain. Fundamentally what we want to do is construct a high-speed wing which is compromised for low-speed performance requirements *as little as possible*. This wing would have an airfoil optimized for very low drag at relatively low lift-coefficient values and would have moderate span, low camber, and minimum surface area. Now to fly this wing as slowly and as efficiently as possible, we need to dramatically increase its area (thus reducing wing loading) and at the same time change its camber to increase its maximum lift coefficient capabilitites. In this process we want to keep the *total* drag (both viscous and induced) to a minimum. And to minimize induced drag for a machine of given weight, we also want to create an optimum planform wing of large span.

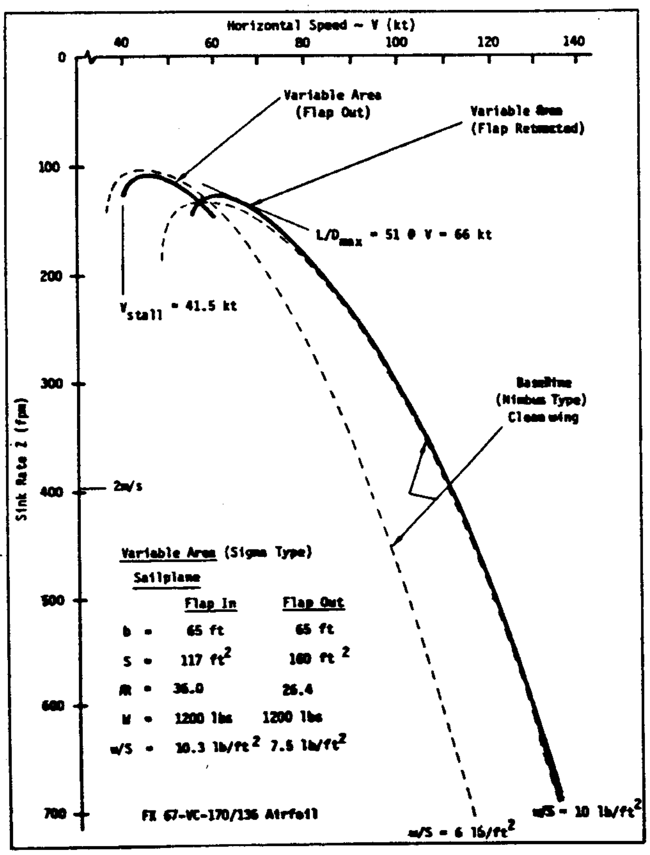


Figure 3. Predicted Performance of a Sailplane with a Fowler Flap

Well, we can't do all these things simultaneously with ease (although the bird does rather well), but we can arrive at a reasonable approximation as the performance shown in Figure 3 indicates. Here my assumed wing has the same 36 percent area-increasing flap as that on the *Sigma*, and one can see that in this case, while the high-speed performance is little better than a "heavy" *Nimbus*, the low-speed performance is nearly as good as a "light" one. Thus, what has been gained is performance *flexibility* with a machine of fixed weight and span. Even larger gains might be had if one pursues Wortmann's later suggestion (Reference 14) for a fabric "sailwing" Fowler flap where area could be increased by up to twice that of the high-speed wing.

The main liability, other than cost and mechanical complexity, of the use of Fowler flaps is that they are largely limited to wings of almost fixed span. Thus, with flap retracted and span set by low-speed, induced drag (i.e., span loading) requirements, the high-speed wing becomes one of often extreme high aspect ratio. This makes the problem of providing an adequate margin of structural strength, stiffness, flutter, and control reversal within reasonable weight limits extremely difficult. However, ingenuity will likely triumph, and the overall scheme has obvious attraction.

Variable-Span Wings

The problem of the Fowler flapped wing having *too* much aspect ratio for optimum high-speed flight, coupled with the same general sizing problems — both high and low-speed — which any sailplane faces, is solved conceptually in a very easy fashion: Telescope the wings so that span and area change simultaneously, with both values changing in the "right" direction for both optimum high and low-speed performance. That's of course easy to say but hard to do in practice — although there are several powered and unpowered examples historically, the most recent being Stuttgart's fs-29 (Reference 15).

While the variable span-and-area approach appears logical, albeit mechanically awful, it runs into more problems than simple mechanisms. First, one does not have *carte blanche* in changing span and area. The geometry of the basic wing sets definite limits on the ways these values can actually change, and the result turns out to be not exactly what one wants. Specifically, while telescoping the wing *in* (á la the fs-29), we do get a beneficial reduction on area, but if we reduce it as much as we really want to, we lose *too much* span. While induced drag decreases as speed increases, it is never negligible within the range of sailplane performance limits, and the 13-meter "wing-in" span of the fs-29 is *not* sufficient for an Open Class racer at any weight until you get to very near redline speeds.

The second problem with variable span is that for really efficient low-speed flight we want an airfoil with quite a bit more camber than is ideal for high-speed flight. Now, once one has decided to telescope the wing, how does one fit an efficient camber-changing flap to a significant portion of the wing? Headache upon headache. Well, again, I tried to work the problem several ways, and finally came up with the performance shown in Figure 4. Compared to the previous Fowler flap result (Figure 3), the results don't seem promising, given the mechanical problems to be confronted. To my not very great surprise, these results agree well with those past (well-done) results obtained by other wiser heads (e.g. Goodhart, Reference 3). Variable span works well for the birds, but only when coupled carefully with a mechanical replica of bird feathers and spars, *and* area and camber-change capabilities. Score a major triumph for the bird which does not demand long runs of laminar flow to survive and fly quite satisfactorily.

So much for the aerodynamic problems. Have our ingenious German friends blown smoke in our faces and solved the problems anyway with the fs-29? My latest information claims not. Having done a marvelous job and coming up with an excellent end run on the mechanical aspects of the scheme (i.e., gloving the outer wing panels *over* the inner carbon fiber stubs), they still face the problem of quickly moving all that wing in and out at the instant of perceived need. Well, the fs-29 system (which runs on Teflon bearings, etc.) still has a *lot* of friction I'm told and apparently tends to bind inconveniently if the wings flex wrong. And there may be problems of asymmetric extension if there is slop in the extension mechanism. (Variable asymmetry can be a good thing, but only if done in a properly controlled fashion.) Headaches upon headaches.

Variable Thickness Wings

Variable span and Fowler flaps, as concepts, have been with us for over forty years. But leave it to Pat Beatty to come up with a non-obvious alternative as incorporated in his new B-5 (Reference 16). Having introduced *big* Fowler flaps into racing sailplane design, he has now turned to a scheme which still seems implausible after the fact. But think! Drag (ugly viscous drag) is most of the problem at high speed. And thin, low-cambered airfoils minimize wing profile drag. Suppose we pitch out any requirement for "high-lift" capability and go for *absolute* minimum profile drag at *low* lift coefficient values. Suppose we combine the fruits of Dr. Eppler's computer (or analytic wind tunnel) with the wing size of a *Nimbus*, and we get — Figure 5. Or at least the high-speed part of it (where *stall* occurs at a glorious 70 kts).

This last problem is "easily" solved (at least with the same mechanical ease as fitting the wing with a Fowler flap) by physically arching the upper wing skin to a new position, equivalent to a good low-speed Wortmann section for thermaling. Yes, the wing changes from an Eppler 1001 airfoil (of a mere 11 percent thickness) for "cruise," to a good old Wortmann FX 67-K-150 (15 percent thick) for climb. And it works. Mechanically it works. Aerodynamically, matters are not so clear.

The Eppler airfoil on the B-5 was generated theoretically according to good theory. And an independent analysis we made last summer indicates that *if* the wing surface is perfectly smooth, wave free, and true to contour, the very low drag values are probably achievable. To get these values, about 70 percent of both surfaces of the wing are covered by laminar flow, the bulk of it is *marginally stable*. Now if you put a few bugs on that wing or get a little surface imperfection introduced by the surface contour changing scheme, you may lose a great deal of this laminar run, and the fancy wing deteriorates in performance to a heavy (and complex) "dirty *Nimbus*" level. We shall await Pat Beatty's flight test results. The *concept* is attractive; its consummation in *practice* must be considered an *extremely* risky business!

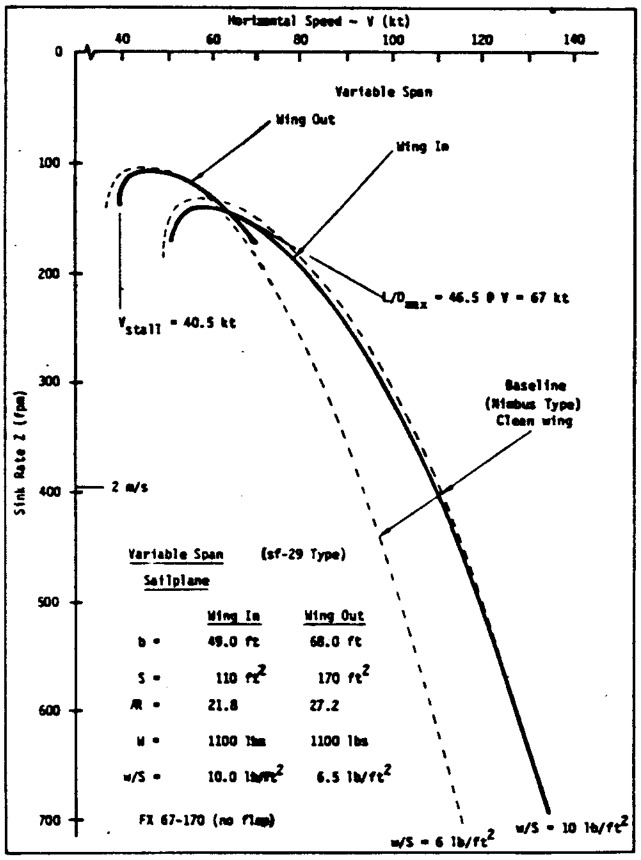


Figure 4. Predicted Performance of a Variable Span Sailplane

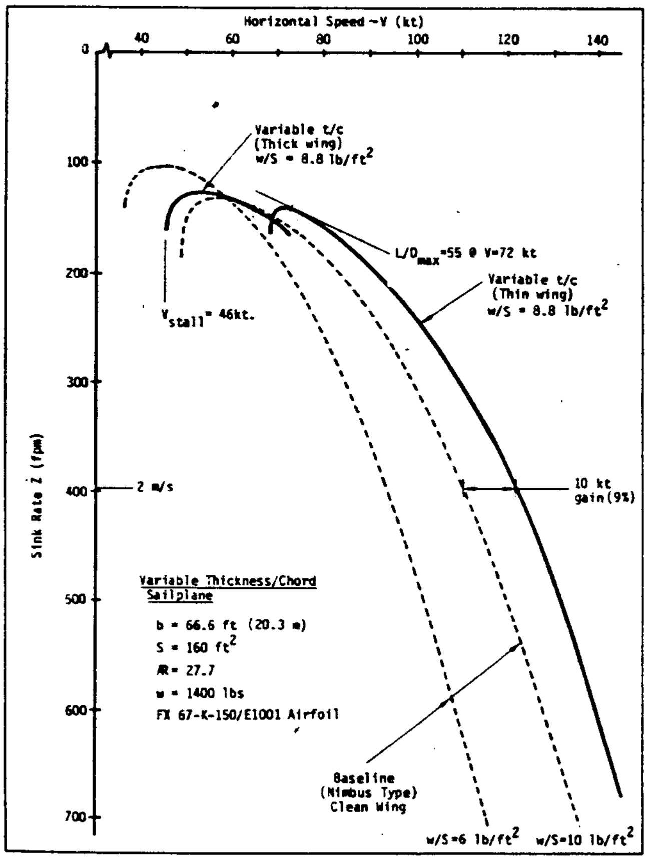
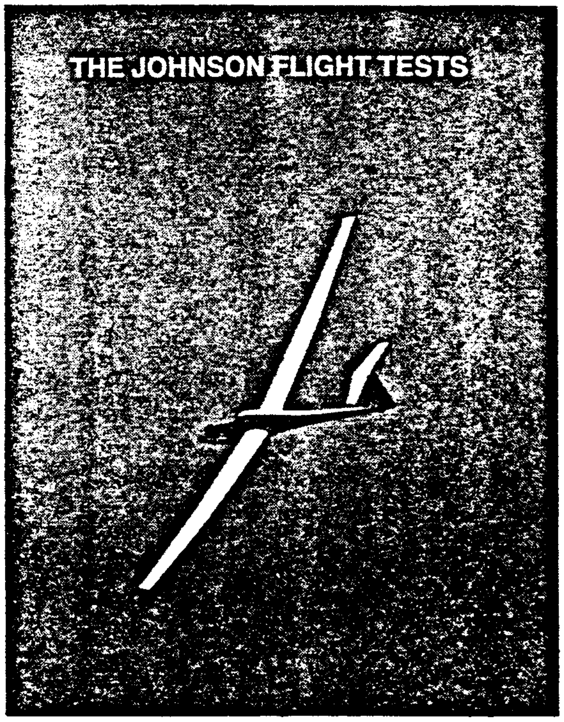


Figure 5. Predicted Performance of a Sailplane with a Variable Thickness Wing.



Now What?

As the examples previously described show, it is now possible to contemplate a next generation of high-performance racing sailplanes with a good deal more performance (in an overall sense) than current models. Each of the three basic variable-geometry schemes which might be employed to give these performance increases has now been tried with, so far, highly mixed results. Which scheme, if any, is "best" cannot now be identified with any surety, but through the dedicated efforts of a few visionaries among us, some lessons are beginning to emerge. In general we may say that a *good* variable geometry scheme would have the following characteristics:

- The particular aerodynamic approach taken would not result in any serious deterioration of handling characteristics of the machine over its entire flight envelope.
- Aerodynamic gains would be carefully evaluated against weight and complexity increases which might destroy the gain in pure aerodynamic performance.
- Any mechanical system for varying geometry must be actuatable quickly and with very little more effort than one now expends actuating current racer cruise flaps.

With all these considerations in mind, my own favorite among the schemes proposed so far is the big Fowler flap. In addition, I can readily envision this flap combined with a scheme for mild span extension. An inflatable cuff or extensible tongue at each wingtip which would increase overall span by only 1 meter (out of perhaps 20), coupled with a 30 percent area-changing Fowler flap and a lot of ballast, should be interesting. At least until some real genius develops a scheme for causing rain to fall inside a ballast tank at the discretion of the pilot.

As a parting note, it should be observed again that variable geometry wings are *not* peculiar to Open Class racing sailplanes. We have already seen the application of *Sigma*-type Fowler flaps to the highly successful SB-11 15-Meter racer. The forthcoming 15-Meter Darmstadt D-40 will also have a Fowler flap of the same sort but simpler and less extreme. Until Pat Beatty sorts out the B-5, we still have to wait for the possible extension of his scheme to a 15-Meter ship. Either way, only the variable span wing appears improbable for 15-Meter applications. On the other extreme, there is a *very* simple variable thickness/chord scheme afoot for application to an ultralight sailplane. No class of sailplane can profit more from variable geometry (if accomplished simply and with low weight) than the ultralight. But more on that another day.

Parts I and II of this series were written prior to the SSA Convention in Seattle where author John McMasters has participated in various formal and informal discussions with U.S. and overseas designers. The result is a previously-unplanned Part III for next month which discusses the "add-a-tip" variable geometry of the new AS-W 22, as well as such exciting ideas as how the emerging "straight-ahead" (little or no thermaling) cross-country techniques may reopen the dominance of super-span sailplanes, how space-age "fly-by-wire" control of unstable but ultra-high performance designs can mean still greater achievements, and how a laminar flow flying wing can get an L/D of 65:1 and even 92 or 98:1 using LFC — laminar flow control! — Ed.

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PART III

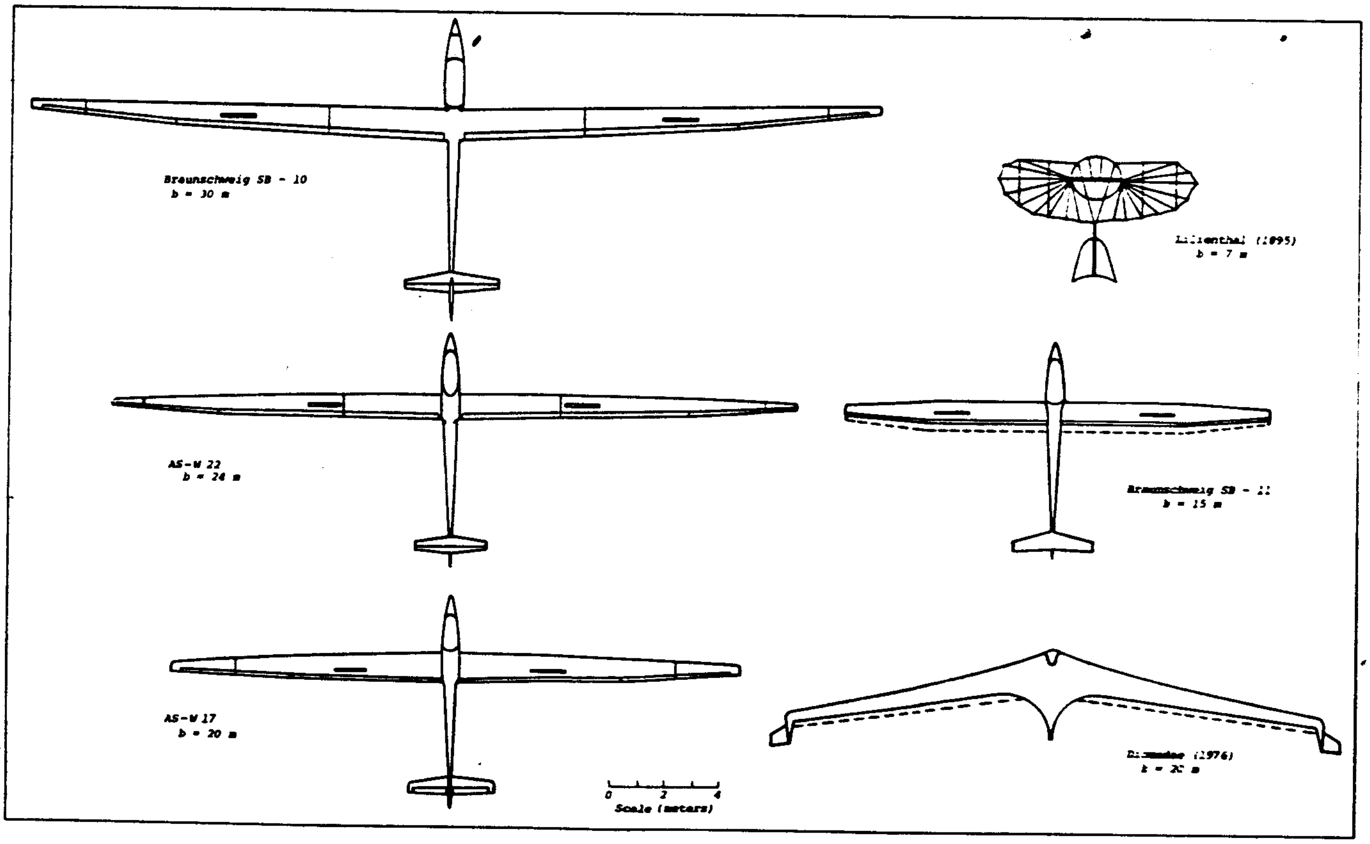
ADVANCED CONCEPTS IN VARIABLE GEOMETRY SAILPLANES

by JOHN H. McMASTERS

In the previous two installments of this article a prognostication was made that variable geometry techniques for racing sailplanes would be with us for a long time to come. I still believe this, but it is also apparent that there are other ways to design advances for very expensive racers. This was most recently dramatized to me by discussions with Gerhard Waibel and others at the winter SSA Convention in Seattle. The question thus arises as to how and where the more "conventional" new AS-W 22 (Reference 17, fits into the performance picture for the various variable geometry schemes previously discussed (*Soaring*, April & May '80). My merry pocket calculator cried out to be allowed to make a comparable performance assessment of a machine of AS-W 22 dimensions relative to the "baseline" *Nimbus* used in the earlier discussion. Not to be denied, here is its (and my) diagnosis.

Shown in Figure 8 is a planview comparison (again to the same scale) of the present AS-W 17, the new AS-W 22, and *Diomedae*, a little fantasy beasty of my own which will be discussed later. My! Isn't the AS-W 22 with its six-piece, all-carbon-fiber, 24-meter wing a monster? (The AS-W 22 *is* a variable geometry sailplane. The span can be altered — on the ground — by adding and/or subtracting wing panels, thus varying span from 21 to 24 meters. It also has a cruise flap.) I must say at the outset that I hope to remain on speaking terms with Herr Waibel, and must thus point out that all my numbers are approximate and theoretical

Figure 8. Some Advanced Sailplanes



and may not represent exactly what Gerhard had in mind when he designed his new beauty. I'd be more than pleased to hear his side of the story when he's prepared to tell it in the pages of *Soaring*.

Before we play with some numbers on an AS-W 22 type machine, a few items not adequately discussed earlier should be elaborated:

- We are rapidly entering the next-generation era of composite materials (e.g. carbon fiber, Kevlar). What these materials allow us to do (if intelligently used) is to build either lighter structures of given stiffness and size than was possible with fiberglass or build bigger structures — in this case longer span, higher aspect ratio, and maybe thinner wings of adequate stiffness and weight (provided we can afford the cost of these new materials). Read both Waibel's and Holighaus's papers on these matters in the *Proceedings of the 1980 SSA Convention*. Good stuff.
- Competition flying techniques have changed with advances in sailplane performance. The thermal-soaring racer, optimized to fly in the classic MacCready sense, is not necessarily the best "dolphin-style" configuration. If less time is "wasted" in circling, then the race can be won by a machine with very high glide ratio lumbering along at slightly less than breakneck speed between regions of lift. Such an approach favors wings of great span. Read the new gospel according to Reichmann.
- Bugs, and the reduction of their adverse effects on performance, will loom larger and larger as we soar into the eighties. We know a great deal about the aerodynamics of racing sailplanes in their pristine bug-free configurations. We also know a lot about airplanes with thoroughly turbulent boundary layers. But in between? We have a lot to learn here, and as an alternative to developing bug-proof wings, we might in the meantime reduce total drag by working at the *induced* drag problem once more. And that (coupled with new materials) means notching the wingspan up to some new limit. Whether that new limit is 24 meters, 30 meters, or whatever remains to be demonstrated.
- There are whole branches of aeronautical technology being developed for airplanes other than soaring racers which haven't been exploited by sailplane designers yet. Of most immediate potential interest is the progress being made in microprocessors and developments in reliable stability-augmentation systems. What these new "fly-by-wire" systems mean for us is that the reflex limits of the puny human brain (which evolved for walking, not flying) can be partially circumvented, and we may yet see the emergence of the *ultimate* racer in the form of the long-sought pure flying wing. More on that later.

Superspan Madness

With the previous comments in hand, we can now attempt to assess the goodness of the AS-W 22 recipe as the next generation Open-Class racer. I don't have the data on the specific airfoil Waibel has used in his new machine, but with the data previously presented in this series, a few minor adjustments can be made to the basic *Nimbus*/AS-W 17 configuration to see what the increase in span and aspect ratio of an AS-W 22 type wing might do to the straightaway glide polars of the baseline *Nimbus* shown in the previous Figure 4.

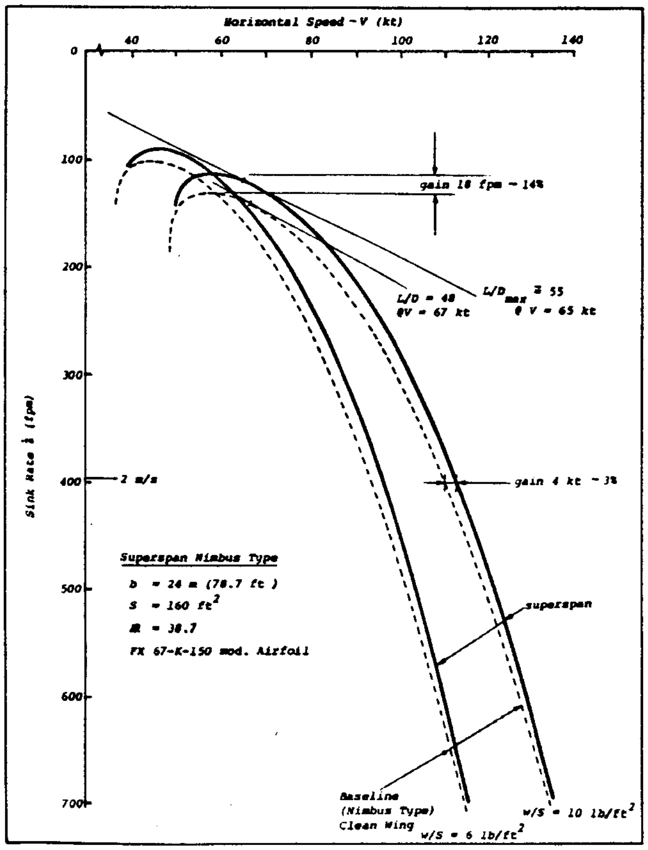
In the process of this comparison we will close an incredulous eye on several factors. Namely:

- The idealized induced drag calculated by my analytic wind tunnel assumes that there is little torsional deflection of the wing over the entire speed range. Such local deflections on such a huge wing might distort the span loading to the extent that the full benefits of the span increase might not be achievable over the full range of speeds. That's one of Gerhard's structural problems, however.
- We shall take it on faith that the airfoils on the ten-inch chord wingtips work as required in a consequent seriously depressed Reynolds-Number range. Note also that because the wing span has increased dramatically and the area hasn't, the aspect ratio has increased considerably compared to the *Nimbus* baseline. Thus the distribution of chord across the span has been reduced resulting in all wing sections operating at a lower Reynolds Number at each speed when compared to the more modest-span machine.
- We shall ignore altogether the problems of performance and lateral control when the new monster wing is forced to turn and bank as it inevitably must on occasion.

With all this said, and the values assumed for our superspan machine listed on the graph, we estimate the performance shown in Figure 9. Here the comparison is made for clean wings at 6 lbs./ft.^{2} (a value representative of an AS-W 22 with a sixty-pound pilot) and 10 lbs./ft.^{2} wing loadings.

Well. The clean-airplane performance increase is not truly spectacular, but if achieved in practice, it is likely good enough to win some contests. To see whether the increased span helps the bug problem, we can look at the comparison

Figure 9. Predicted Performance of a Superspan Sailplane



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shown in Figure 10 for the dirty airplane at 6 lbs./ft². The answer is yes and no, depending on which end of the polar we look at. What these results show is merely an amplification of the general design recipes spelled out in Box A of Part I (*Soaring*, April '80) of this series of articles.

Over the low-speed region of the polar (including the points for minimum sink rate and maximum L/D) where induced drag is large in proportion to the total drag of the machine, span increase helps performance quite a bit. By simply changing span, however, we have done little to change the components which influence viscous drag (i.e., the airfoil section, the total surface area of the machine, or the shape of the fuselage, etc.). And it is largely viscous drag which influences the high-speed end of the polar. We've already seen where an assault on the viscous drag part of the problem led Pat Beatty with his B-5. Waibel's approach is likely the more reasonable, especially in buggy conditions, particularly when a fellow as clever as Gerhard makes the other necessary minor adjustments for viscous drag sources necessary to translate from AS-W 17 to AS-W 22. We shall see soon enough.

Whether my performance estimates look encouraging for an AS-W 22 type machine or not, I hardly expect anyone to transfer his order for an AS-W 22 to a B-5 or fs-29 on the basis of my analytic wind tunnel results. I should, however, point out a few additional aspects of my estimation technique:

- None of the machines I've analyzed is the exact counterpart of the actual versions built. My numbers are, however, reasonably representative of the performance *potential* of each real airplane.
- The analysis methods I've used are representative of current transport aircraft industry prediction methodology. Account is taken of a number of "second order" factors such as variations in viscous drag and maximum lift with Reynolds Number (or flight speed and lift coefficient), wing span efficiency factor variations with lift coefficient and aspect ratio in the induced drag calculations, and so on. (No parabolic drag polars are used.) Such methods usually make rotton absolute predictions, but are pretty good at predicting the effects of configuration *changes* on a good set of baseline data.
- Only my analytic wind tunnel (computer) is clever enough to calculate maximum lift/drag ratios like 49.32876 and draw a pencil fine line to comply with such "accurate" values. All real *experimental* data — including *all* wind tunnel data — show some variations about some mean line. (This has something to do with the character of turbulence, experimental measurement equipment inaccuracies, and Heisenberg's Uncertainty Principle). When presenting any experimental data as a single fine line, the wary reader should always remember that at best this line represents the statistical mean of a band of data, all scattered points within which some physical reality is represented. Thus, in making performance comparisons between real airplanes, we should be overlaying polars which look like fuzzy bands rather than fine lines. Dick Johnson knows this very well; many less sophisticated sailplane buyers don't seem to, however. Alas.

Epilogue

We have now evaluated four distinct recipes for advanced Open Class racing sailplanes and I have about run out of calculator budget. But before leaving the field once more for the greener pastures of ultralight sailplanes, some final questions come to mind. Where does racing sailplane performance increase end? What is the limit? Is there a limit?

To get some possible answers, I went back to the article Jim Nash-Webber and I wrote three years ago on soaring in the year 2000 (Reference 1). Therein is the *Diomedae* flying wing shown in its fully laminarized form and redrawn here without the laminar flow control (LFC) in Figure 8. It still looks good with a maximum glide ratio of between 92 and 98 with the LFC system and around 60 without it. The flat, flat polar looks very good, but the DM 700,000 price tag (in DM 1980 with LFC) may be a little formidable. Fly-by-wire control and solar-powered LFC systems don't come cheap. On the other side of the coin, however, what brave soul in 1970 would have projected a market for DM 100,000 AS-W 22's? We clearly have a long way to go before we can even predict limits for the Open Class.

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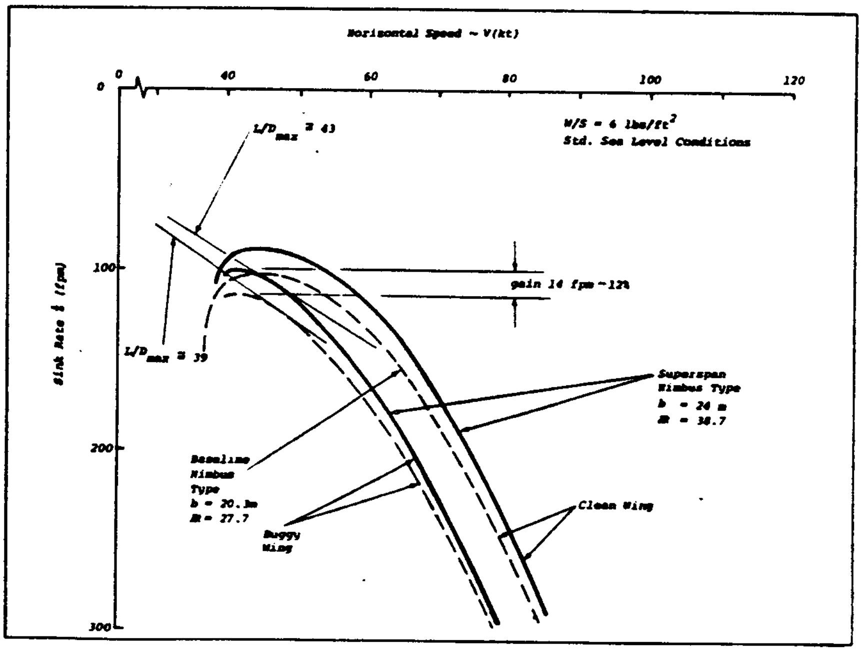
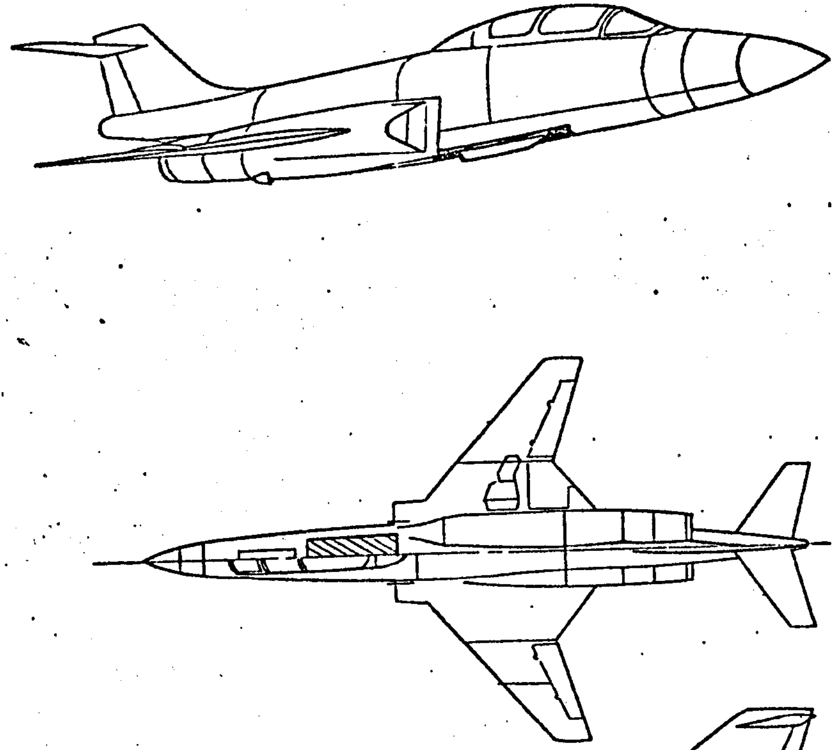
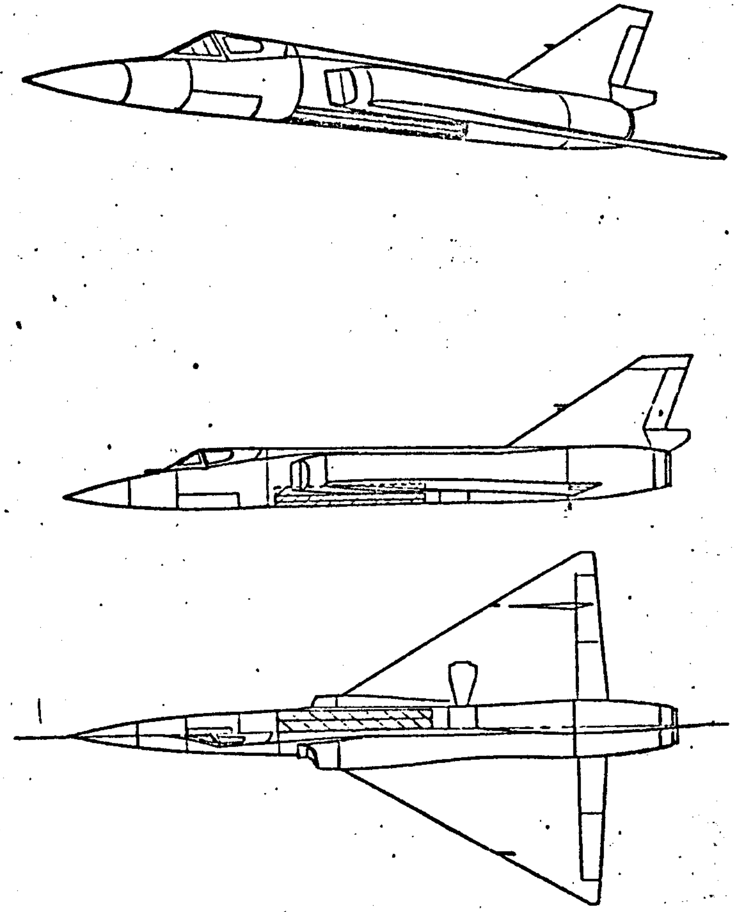


Figure 10. Effects of Bug Contamination on a Superspan Sailplane



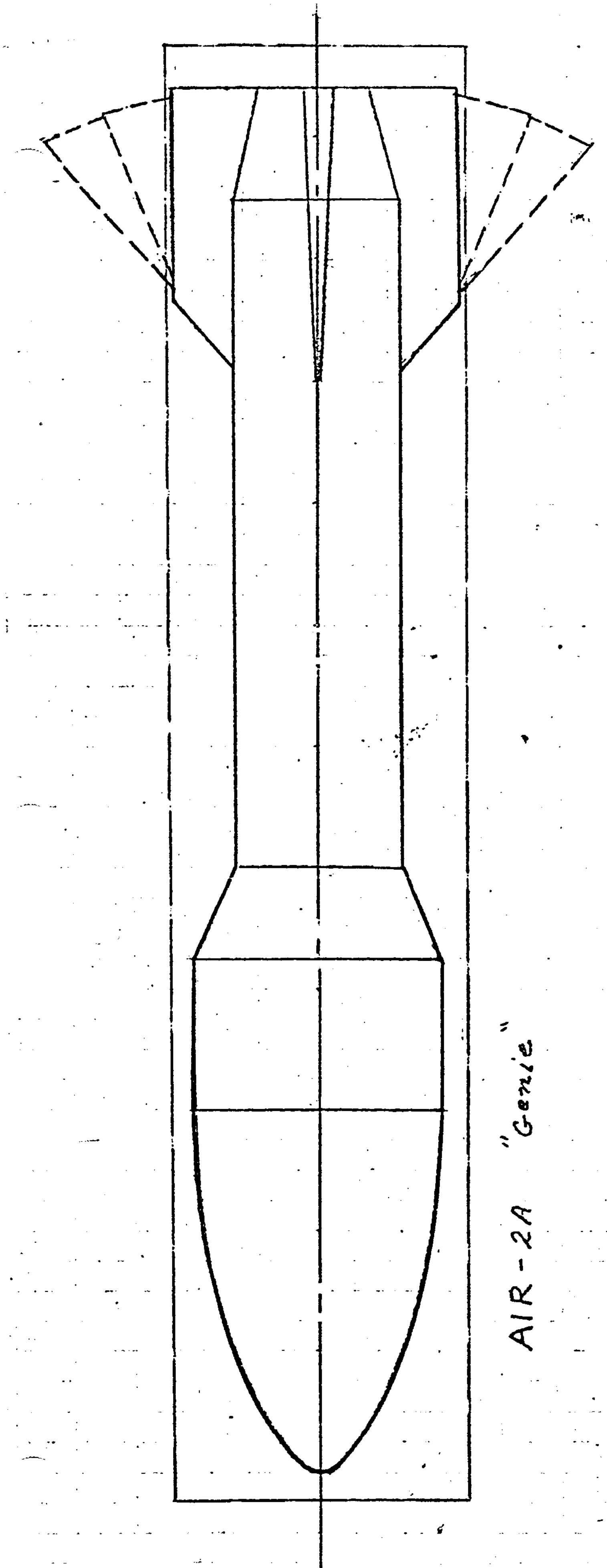
McDonnell F-101 B



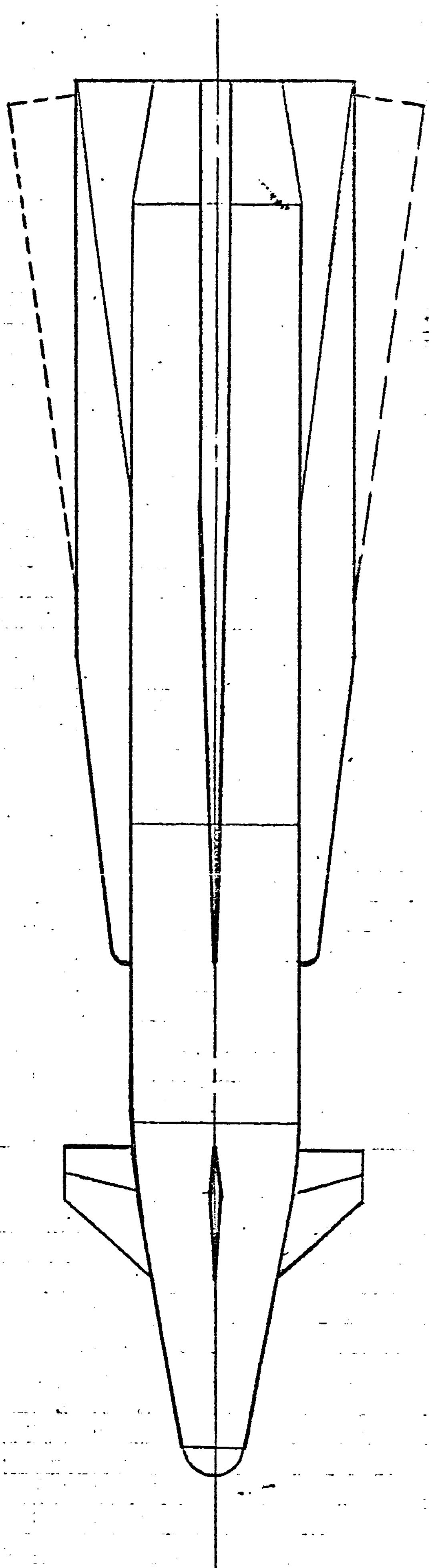
Convair F-106 A

D0 6000 2165 ORIG. 4/71

	F-101 B	F-106 A
Span (ft)	39.7	38.3
Wing Area (ft^2)	368	661.5
Length (ft)	67.4	70.7
Aspect Ratio	4.28	2.22
Gross Wt. (lbs)	40,000	35,000
Max. Thrust (lbs)	29,960	24,500
Max. Speed (M)	1.7	2.3
Max. Wing Loading (lbs/ft^2)	108.7	52.9
Max. Thrust Loading (T/W ~ lb/lb)	0.749	0.70



AIR-2A "Genie"



ZAIM-68A "Quetzalcoatl"

LARGE WINGED SURFACE EFFECT VEHICLES

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Richard R. Greer

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JANE'S SURFACE SKIMMERS

HOVERCRAFT AND HYDROFOILS

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LARGE WINGED SURFACE EFFECT VEHICLES

by

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and

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Richard R. Greer is Chief Executive of Water Research Company, Phoenix, Arizona and holds several U.S. and foreign patents on various aspects of hydrofoil and SEV systems. He received his B.S. degree from UCLA, and is actively engaged in practice as a Certified Public Accountant as well as instructing at Phoenix College and Arizona State University. He is a member of the American Institute of Certified Public Accountants, the Arizona Society of Certified Public Accountants, the Navy League and the American Society of Naval Engineers.

ABSTRACT

This article describes the results of a feasibility study of a class of large winged surface effect vehicles (WSEVs) intended for naval use. The objective of the study was to define a basic system which would: (1) possess speed and operational capabilities substantially greater than existing ACV and CAB surface effect vessels, (2) circumvent size limitations inherent in conventional aircraft designs, and (3) offer an economically and operationally viable alternative to existing water-based and/or airborne naval systems for a variety of military and commercial missions. The resulting proposed WSEVs are a family of low aspect ratio channel shaped vessels operating on the principle of an aerodynamic lifting wing in ground effect during cruise. The baseline configuration is shown to be suitable for a number of missions including low density cargo transport: highly mobile ASW, air defence and missile launch; and quick response, long range "carrier"/support for helicopters, coastal patrol craft, and SEV patrol and landing craft. Two of the benefits of the basic design are that the configuration is well suited to propulsion by non-hydrocarbon fuel systems (e.g. hydrogen) and use of extensive laminar boundary layer control.

INTRODUCTION

To a very large extent, the general trend in transportation system development has been towards ever increasing productivity—or simply increased speed. Nowhere is this trend more obvious than in aviation where cruise speeds for commercial transport aircraft have increased almost exponentially from the start of regular passenger service in the 1920s, through the advent of supersonic designs and culminating in hypersonic transport during the past decade. All this has tended to ignore the finite supply of conventional hydrocarbon fuel—a resource which now appears to be dwindling at a frighteningly rapid rate.

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The "energy crisis" can be confronted in at least two ways by the transport system designer: 1) he can seek to improve the efficiency of conventionally fuelled systems, perhaps even at the expense of degrading productivity, and 2) he can seek alternative fuel sources. Both approaches appear viable and are probably complementary. The reasonable near-term approach is to increase the economy of conventional fuelled devices, a longer range goal being to gradually move to alternative energy sources as the technology and experience with the use of, say, liquid hydrogen becomes better established.

The basic purpose of this article is to describe a specific type of vehicle (the winged surface effect vehicle or WSEV) intended for both commercial and military naval use which is both economical and has the growth potential to utilise effectively non-hydrocarbon fuels. The discussion presented is an extension of the authors' previous study (1) of various surface effect vehicle systems intended to: 1) have substantially higher speed capability than existing hydrofoil and SES vehicles; 2) be simpler mechanically than existing SES vehicles, thus reducing both initial and maintenance costs; and 3) circumvent size and operational limitations of conventional aircraft, particularly those intended for naval or transoceanic cargo missions.

A number of methods for increasing the speed capability of waterborne vehicles have been the subject of very active research and development effort for the past decade. Principal among these are (1) hydrofoils, (2) air cushion vehicles (ACVs),(3) captured air bubble (CAB) systems, (4) ram wings, and (5) wing-in-ground-effect systems (WIGS).

The hydrofoil offers the possibility of substantial speed increases over conventional ships. However, it appears that the maximum feasible speed for such a vehicle is about 80 to 100 knots. Structural problems also place definite limits on the size and weight of such vehicles. The largest hydrofoil craft built to date is the USS Plainview (AGEH-1) with a weight of 320 tons. At the speed specified above, the specific power requirement (HP/ton) appears to be excessive.

Air cushion and other aerostatically supported vehicles appear to offer substantial increases in speed and efficiency compared with hydrofoil supported vehicles. However, they are mechanically complex and consequently expensive, both initially and from a maintenance point of view. In addition, for this type of vehicle to operate efficiently, it must remain in very close proximity to the water surface; this latter factor imposing fairly severe limits on speed and rough sea operations.

The captured air bubble (CAB) system seeks to improve the efficiency of an aerostatically supported vehicle by fitting what amounts to an ACV with rigid sidewalls and flexible seals at each end of the resulting channel to minimise lateral support cushion air spillage. The penalty here is the added hydrodynamic drag of the submerged side walls. The ram wing (2) seeks to reduce mechanical complexity of the ACV and CAB by generating the supporting cushion by aerodynamic means rather than by auxiliary powered "blowers". The major limitations of the ram wing are that it must again operate in very close proximity to the water surface and must have sufficient forward speed to establish a supporting air cushion.

To increase a vehicle's speed and operational capability much beyond that possible with an ACV or hydrofoil system, it appears necessary to physically remove the vehicle as far as possible from the water surface. This has led to the concept of the wing-in-ground-effect system (WIGS), based on the well-known aerodynamic principle that an airplane wing, when operated in close proximity to a solid surface, experiences a substantial reduction in drag due to lift (induced drag). Most pure WIG systems proposed so far have resembled conventional aircraft with fairly high aspect ratio wings and some sort of "boat" hull for flotation at rest and during take-off run. A typical WIG vehicle in the same size and weight category as the present C-5A jet transport is compared in Figure 1 to the 500-ton version of the winged surface effect vehicle (WSEV) proposed in this study. The use of "conventional" high aspect ratio (greater than 3 or 4) wings in close proximity (heights less than half the wing span) to a rough sea surface creates severe operational problems and the usual structural considerations in any aircraft design place upper bounds on the size of such vehicles, given the present state-of-the-art in material and structural technology. The largest aircraft built to date is the 382-ton C-5A, and at present it is difficult to contemplate conventional aircraft weighing in excess of 500 or 600 tons.

An extremely important alternative scheme to the "conventional" (i.e. high aspect ratio) WIG type vehicle is the *aerofoil boat* developed by Lippisch (3) and his associates. Lippisch's design is basically a very low aspect ratio WIG with heavy overtones of the ram wing. The great contribution of this extensive development and testing

TABLE 1.—VEHICLE CHARACTERISTICS

AIRCRAFT Type	Wing Span (ft.)	Wing Area (ft².)	Weights Empty (tons)	Weights Loaded (tons)	Power or Thrust BHP or LBs.	Max. Speed (kt.)
1. Martin JRM-1 flying boat	200	3686	37·5	72·5	9,000 BHP	180
2. Lockheed 89 piston transport.	189	3610	57	92	12,000 BHP	225
3. Convair C-99 piston transport.	230	4772	70	132·5	18,000 BHP	260
4. Boeing B-52G jet bomber	185	4000	85	244	110,000 Lbs.	550
5. Boeing 707-320 jet transport	154·7	2942	69	156	68,000 Lbs.	520
6. Lockheed C-5A jet transport	222·7	6200	160	382	164 000 Lbs.	500

HYDROFOILS: Type:	Length (ft)	Gross Wt. (tons)	Power (SHP)	Max. Speed (kts)
1. AGEH-1 Plainview	212	320	28,000	50
2. Typhoon (USSR)	103	65	3,500	43
3. Vikhr (USSR)	156	117·5	4,800	43
4. FHE-400 (Canada)	151	212	22,000	60

AIR CUSHION VEHICLES Type	Length (ft)	Gross Wt. (tons)	Power (SHP)	Max. Speed (kts)
1. SR.N 4 (U.K.)	130	180	17,000	70
2. Bell JEFF (B)	87	162·5	16,800	50
3. Sormovich (USSR)	96	30	2,500	75

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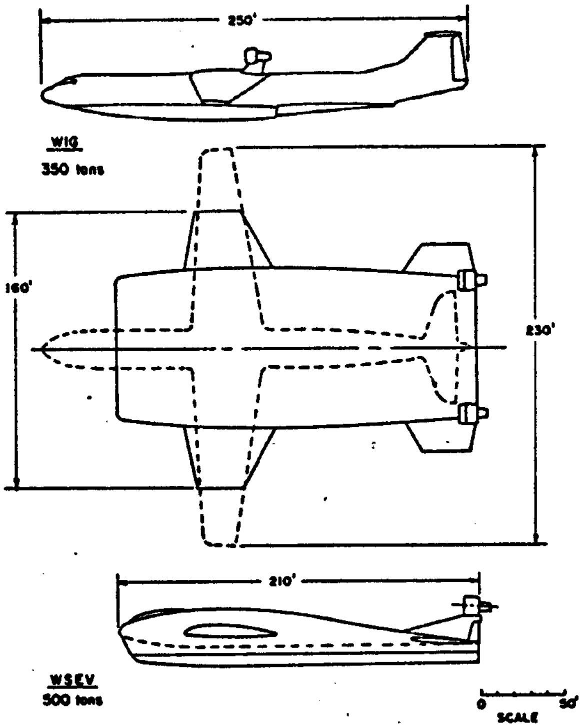


Fig 1. WIG-WSEV size and configuration comparison

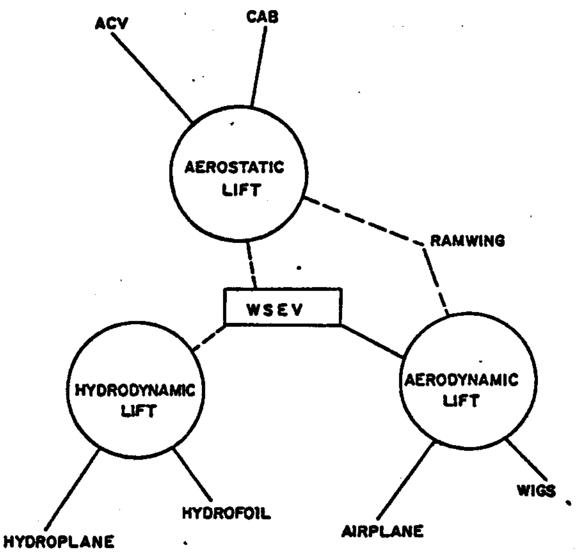


Fig. 2. System relations

effort is the apparent solution of the difficult stability and control problem inherent in operating a wing at various heights above the water—a difficulty also encountered in the basic ram wing and solved, in model form, by Gallington (2).

The characteristics of several existing hydrofoil boats, air cushion vehicles and a number of very large transport aircraft are listed in TABLE 1. It appears that on the basis of this data, and the previous discussion on the limitations of various types of surface effect vehicles, the most favourable means for producing very large vehicles with substantially higher speed capability is some variation on the wing-in-ground-effect scheme.

The goal of the work reported here was to investigate the feasibility of combining certain desirable features of the hydrofoil, surface effect, and WIG systems into a single type of vehicle suitable for a variety of potential naval missions. The inter-relationship between the types of vehicles suitable for these missions is summarised diagramatically in Figure 2. The resulting WSEV described in this article differs from the Lippisch scheme in both configuration and, more importantly, in anticipated size.

BASIC DESIGN FACTORS

Several fundamental design factors influence the selection of any specific vehicle configuration. Among these are power or thrust requirements, propulsive efficiency, and lift/drag characteristics Due to the unconventional nature of the proposed winged surface effect vehicle, a brief review of some design factors is in order.

POWER REQUIREMENTS AND PROPULSIVE EFFICIENCY

For a vehicle in steady (constant velocity) motion, the lift force (L) must equal the weight (W), and the thrust (T) must equal the resistance force or drag (D). The power required to propel the vehicle is:

$$\text{Engine Horsepower} = \text{SHP} = \frac{TV}{C\eta} = \frac{WV}{C\eta L/D} \qquad (1)$$

where

C = 550 if V in ft./sec.
326 if V in kt.
η = propulsive efficiency
V = velocity

and thrust required is:

$$\text{Thrust} = \text{Drag} = \frac{W}{(L/D)} \qquad (2)$$

Equation (1) may be rearranged to show the influence of aerodynamic efficiency (L/D) and speed on the specific thrust horsepower (SHP/W) required. Figure 3 illustrates the magnitude of this quantity for general values of L/D and for the specific case of current technology hydrofoil boats. Typical values of propulsive efficiency are 60% at 50 knots for a supercavitating propeller and 85% for a ducted fan from 150 to 300 knots. Figure 3 indicates that a vehicle with sufficiently high values of lift-to-drag ratio and propulsive efficiency could operate much more economically than a hydrofoil vehicle even at speeds in the order of 200 knots.

LIFT FORCES

The usual expression for the fluid-dynamic lift force on a vehicle is:

$$L = \text{lift force} = qC_LS \qquad (3)$$

where

C_L = lift coefficiency
q = $1/2\rho V^2$
ρ = mass density

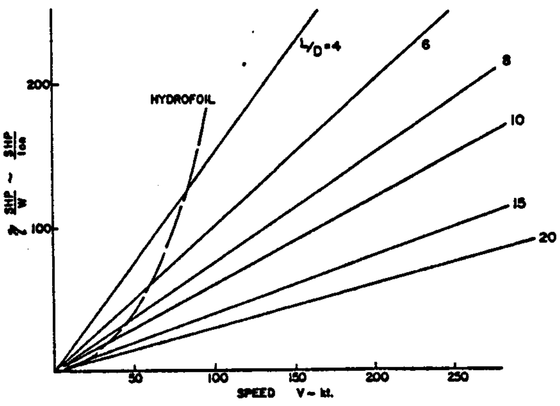


Fig. 3. Specific power required versus speed

The very large difference between the density of water and that of air is one of the key elements in the design problem. High values of "q" can be obtained in water at even moderate velocities, while the "q" in air at the same speed is about 840 times less. In consequence, if the vehicle is to be sustained largely by aerodynamic rather than hydrodynamic lift (e.g. hydrofoils), the lifting area and/or lift coefficient would have to be correspondingly higher than those required in a water supported vehicle. Values of the weight per unit area of "wing" required for support in water and air are shown in Figure 4 as a function of speed and C_L.

DRAG FORCES

The resistance (drag) force on a vehicle operating in a fluid (air or water) is usually written:

$$D = \text{drag force} = C_D q S \qquad (4)$$

where

C_D = total drag coefficient

The Drag Coefficient in Equation (4) for a vehicle like a hydrofoil boat or an airplane is generally a complicated function of vehicle shape and several scale factors (Reynolds, Froude, and Mach numbers).

In first order aerodynamic theory, it is customary to write:

$$C_D = C_{D_p} + C_{D_i} = C_{D_p} + \frac{K_w C_L^2}{\pi AR} \qquad (5)$$

where

$$AR = \frac{(\text{span})^2}{\text{area}} = \frac{b^2}{S}$$

K_w = Wing "span efficiency factor"

In Equation (5) C_{D_p} is the "parasite" Drag Coefficient of the total vehicle, and C_{D_i} is the wing induced Drag (drag due to lift) Coefficient. C_{D_p} may be a very complex function and is usually dependent on vehicle angle of attack (and thus C_L). However, for purposes of making preliminary performance estimates it is usually sufficient to write:

$$C_{D_p} = C_{D_o} + K_o C_L^2 \qquad (6)$$

where: C_{D_o} and K_o are approximately constant, and are simply curve fitting parameters. The quantity AR/K_w in Equation (5) is usually referred to as the effective aspect ratio of the lifting surface. In simple theory, only the factor K_w is modified when a wing operates in ground effect. The span efficiency factor K_w has a theoretical minimum value of unity for a wing with an elliptic lift distribution operating outside ground effect. It should be noted that the total "airplane" or "Oswald efficiency factor" K frequently used for preliminary design estimates can be expressed in terms of the above formulation as:

$$K = K_o \pi AR + K_w \qquad (7)$$

which then allows Equation (5) to be written in the simple form:

$$C_D = C_{D_o} + \frac{K C_L^2}{\pi AR} \qquad (8)$$

GROUND EFFECT

It is well known that the aerodynamic characteristics of a lifting wing are strongly altered when the wing is operated in very close proximity to a "solid" surface (ground effect). The main effects are:

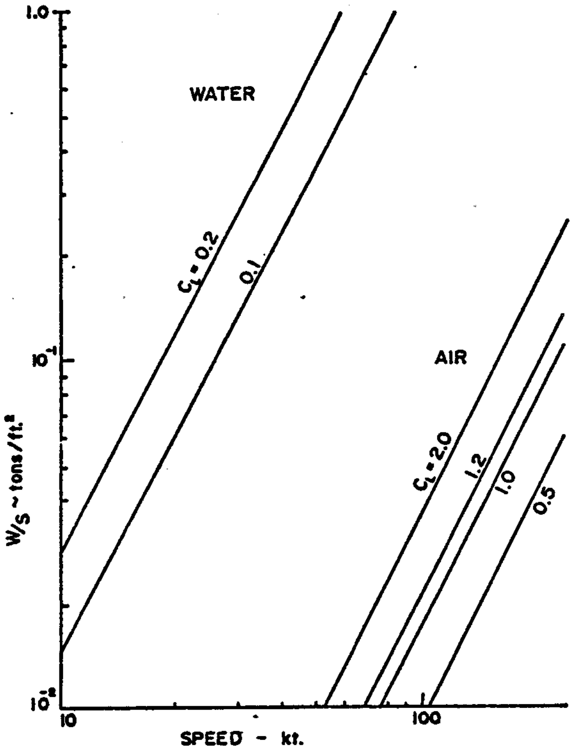


Fig. 4. Wing sizing relations

a) the induced drag of the wing is decreased with increasing proximity to the "ground plane" due to interference of the "ground" with the full development of the downwash flow field, b) the lift curve slope of the wing is increased with increasing ground proximity, and c) the aerodynamic centre of the wing may shift considerably, resulting in substantial changes in pitching moment with ground proximity. To a lesser extent, the lift force at a given angle of attack may increase slightly with decreasing ground clearance.

For a simple planar wing, first order aerodynamic theory predicts that the magnitude of the influence of ground effect is proportional to the ratio of the height of the aerodynamic centre of the wing above the ground (h) to the wing span (b). Thus the reduction in induced drag is directly proportional to the ratio h/b. If the wing is to operate at a fixed height above the surface, then increasing wing span will result in decreasing induced drag.

In the present application it is desired to limit wing span for structural and operational reasons; e.g., in turns, the wing tip of a high aspect ratio wing may come dangerously close to the water surface and when docking the vehicle for loading. Consequently, if a simple planar wing is to be used, there is a definite upper limit to possible performance improvement to be had from exploitation of ground effect. Ashill (4), however, has shown that fitting a wing with inverted "tip plates" can magnify significantly the possible reduction in induced drag of a wing, depending on plate geometry and ground plane proximity. Ashill's theoretical results have been partially verified by wind tunnel tests, and the general validity of the theory has been verified by both wind tunnel tests and actual flight test for the case of low aspect ratio wings (5,6). The results of Ashill's analysis are shown in Fig. 5. It should be noted that this analysis assumes a smooth, rigid ground plane. The influence of a more realistic wavy, flexible surface typical of even a relatively calm sea is very difficult to estimate, and should be the subject of additional research.

THE WING SURFACE EFFECT VEHICLE (WSEV)

The basic criteria for the selection of the WSEV configurations studied were:

1) A simple, aerodynamically clean shape was desired.

4.81

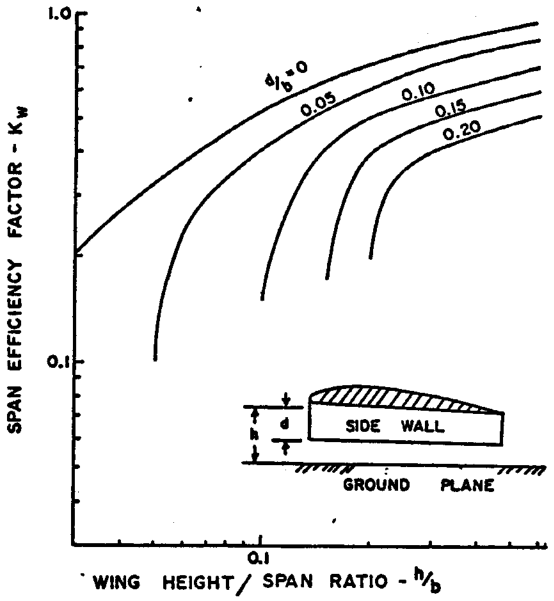


Fig. 5. Effect of wing height above ground on induced drag

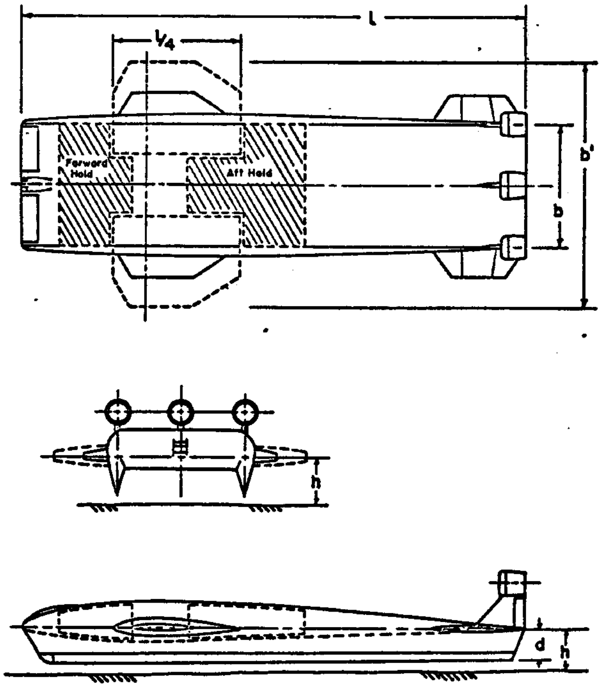


Fig. 6(a). Basic WSEV configuration

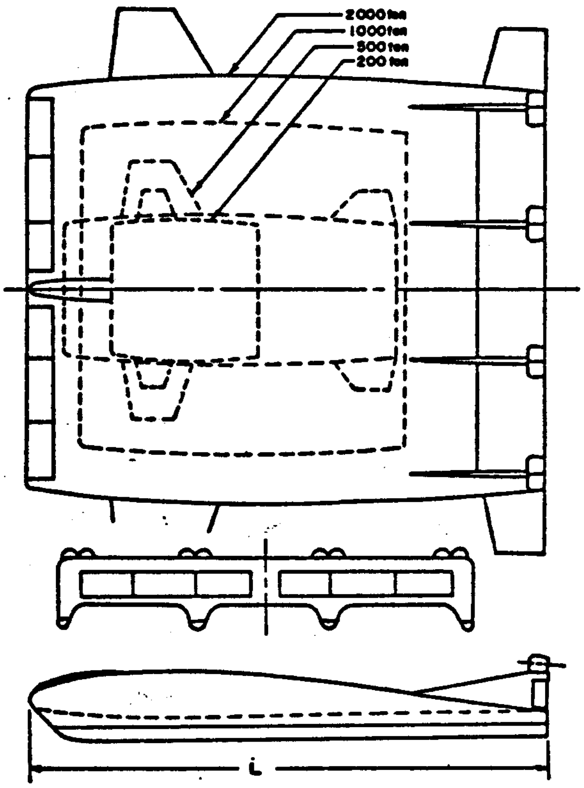


Fig. 6(b). WSEV size comparison

2) The overall system should be mechanically simple.
3) The viability of the system should not depend upon dramatic break-throughs in structural design, materials, or propulsion technology.
4) A shape which would provide maximum manoeuvrability in confined spaces was desired; and preferably one compatible with the Water Research Co. docking system (7).

In addition, the vehicle must possess adequate stability and flight handling characteristics throughout its entire operating envelope. As discussed thoroughly by Lippisch (3) and Gallington (2) this latter requirement is not so simple in the case of a vehicle with the capability of operating at wide variations in height about the ground plane.

The baseline WSEV configuration—Figure 6(a)—selected in a simple rectangular lifting surface with two or more (in the case of the larger vehicles studied) vertical sidewalls. The sidewalls produce the previously described increase in effective aspect ratio (decrease in drag due to lift) while the wing is in cruising flight, and provide flotation for the vehicle when it is at rest. WSEVs in four weight categories were analysed, and these are shown to the same scale in Figure 6(b). Their characteristics are summarised in TABLE 2.

The bridge is smoothly faired into the leading edge of the central wing, producing adequate visibility and minimum disruption of the airflow in this critical area. The ducted fans of the propulsion system are mounted on pylons at the rear of the vehicle so that: 1) the fans do not interfere with the airflow over the wing; 2) spray ingestion and flow distortion into the fan ducts is minimised; and 3) the fan ducts and pylons provide directional stability. The payload is carried in holds distributed around the vehicles' centre of gravity. For all the vehicles studied, even a very low centre-section airfoil thickness/chord ratio produced holds of very substantial depth.

TABLE 2.—SUMMARY OF BASELINE VEHICLE CHARACTERISTICS

	$V_{to} = 85$ knots	$C_{lto} = 2{\cdot}0$	$n = 0{\cdot}85$	
	W = 200 tons	W = 500 tons	W = 1,000 tons	W = 2,000 tons
Channel Width (b)	80 ft	80 ft	200 ft	250 ft
Wing Span (b)	120 ft	160 ft	250 ft	300 ft
Wing Area (S)	8,200 ft²	20,400 ft²	40,800 ft²	81,600 ft²
Length (l)	93 ft	210 ft	204 ft	326 ft
Side-wall Depth (d)	10 ft	10 ft	15 ft	15 ft
Aspect Ratio (b^2)	1·38	1·14	0·98	0·77
W empty (tons)	80	210	490	1,040

4.82

The major advantages of the proposed WSEV shape are: 1) the geometric aspect ratio (and hence wing span for a given lifting area) can be kept low enough to minimise stability problems during cruise, particularly over rough sea surfaces; 2) the width of the vehicle can be minimised to ease docking and loading.

The stability problems encountered while operating the simple channel shape at variable heights are resolved by fitting the basic centre section with two pairs of stub wings mounted fore and aft on the side walls. By proper sizing and incidence selection the pitching moments of the central channel can be altered to provide stability and/or control as the vehicle accelerates and cruises at the appropriate height above the water surface. It should be noted that the forward pair of stub wings act in much the same fashion as a canard empenage on more conventional aircraft. Tests of this scheme on a large radio controlled model shown in Figure 7 have yielded encouraging results. The stub wings also have the effect of increasing the *effective* aspect ratio of the machine in cruise.

The most difficult phase of WSEV operation is during transition from flotation to cruising flight. Substantial speed must be attained before a significant portion of the vehicle weight is supported by aerodynamic lift. Submerged surfaces operating at high speed lead to large resistance forces and excessive power (or thrust) requirements during acceleration to take-off. A realistic balance between wing size, power required, take-off speed and mechanical complexity can be achieved by proper selection of high lift devices. The baseline configuration assumes that the craft is fitted with a system of retractable leading edge slats and a simple single flap at the channel trailing edge. This system is simple, allows the basic wing to remain at a low geometric angle of attack even at high lift coefficients, and allows the vehicle to operate much like a ram wing during transition. A conservative value of CL = 2·0 at take-off was used for vehicle sizing assuming this high use lift system.

Two features of the proposed WSEV configuration make it particularly attractive:

1) The basic "airplane" nature of the vehicle allows it to fly out of ground effect for extended periods, thus allowing the WSEV to avoid obstacles and adverse weather or sea states. This capability cannot be matched by any hydrofoil or ACV system.
2) The basic channel shape of the vehicle is very favourable for the use of extensive laminar boundary layer control (BLC). While mechanically complex and suffering from problems of possible salt water corrosion of the BLC slots (unless a plastic outer wing skin is used), parasite drag reductions of at least 50% should be feasible. This drag reduction has not been applied in calculations for figures, tables or elsewhere in this article.

Structurally, the WSEV would most nearly resemble a large aircraft, although several factors tend to modify conventional aircraft design practice. Among these factors are:

1) The cargo area is within the wing itself, thus offering the possibility of distributing the load across the span. This results in a very significant relief in normal bending loads, with a consequent possible weight reduction.
2) The low aspect ratio inverted channel shape of the tip-plated (or sidewalled) wing is a much more favourable structural shape than that of a conventional aircraft, particularly in the case of vehicles with weights in excess of 500 tons.
3) The proposed WSEV operates in the speed range between 200 to 250 knots as compared with the 500 knot speed capability of jet transport aircraft. Thus, powerplant weight should be a smaller fraction of the total empty weight even accounting for the use of heavy duty marine turbine engines for propulsion.
4) There is no requirement for a pressurisation system or a wheeled landing gear.
5) The structural régime of the WSEV falls in the range classified as "ultra-low density" (8), wherein there is substantial advantage in the use of large composite material honeycomb panels for the skin and general use of "secondary spreading" of load carrying structural elements (e.g. spar caps are made hollow).

Several factors are important in selecting the propulsion system for the WSEV:

1) The anticipated speed of the vehicle allows the profitable use of ducted fans as the main thrust producer. Fans have substantially higher values of propulsive efficiency than hydrodynamic thrust devices such as supercavitating screw propellers and water jets.
2) The basic operating environment of the WSEV is at or near sea level, thus no compromises in engine design for altitude performance are required.
3) In cruise, all lift comes from aerodynamic forces, thus no power and associated ducting is required to maintain cushion pressure as in an ACV.

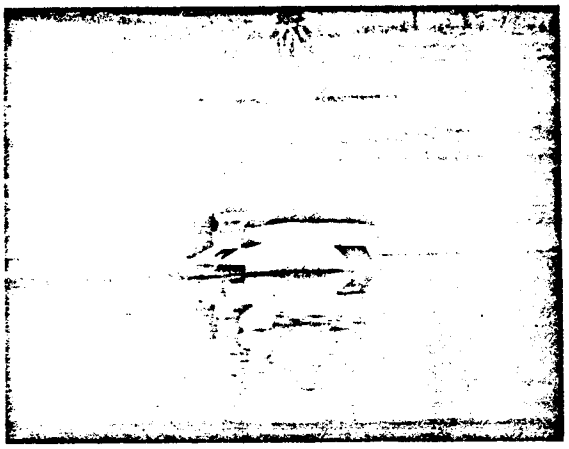


Fig. 7. WSEV radio control test model

Two WSEV powerplant schemes appear promising. The first possibility is use of high by-pass ratio (8-10:1) turbofans. The technology of these engines is well established although specific fuel consumption is somewhat higher than desirable. The alternative is use of marine shaft turbines mounted in the aft portion of the wing centre section, and connected to the fans by extension shaft. The second scheme is mechanically more complex than the first, particularly if the turbines must be cross-shafted. However, this layout may be preferred for two reasons:

1) The marine shaft turbines are more robust and have lower specific fuel consumption than the turbofans.
2) The arrangement leads naturally to consideration of propelling the vehicle by non-hydrocarbon fuelled engines. There is adequate space and weight allowance in the larger WSEVs for a nuclear powerplant and its associated shielding, and there is space for the tankage required to use liquid hydrogen in an optimised engine.

WSEV PERFORMANCE AND ECONOMICS

The performance estimates made to date on the proposed WSEV relate basically to the vehicle in rectilinear cruise. Specifically, estimates have been made on: 1) vehicle size/power relations, 2) maximum and cruise speeds and 3) payload range relations. Basically, vehicles in four gross weight categories have been considered: 200, 500, 1,000 and 2,000 short tons. Performance characteristics for these baseline vehicles are summarised in Table 3. Predicted range/payload envelopes are shown in Figure 8.

The viability of this new scheme depends in large measure on obtaining maximum benefit from drag reduction through ground effect. In theory (4), large benefits can be gained even with the proposed vehicle configuration operating with substantial clearance between the vertical sidewalls and the water surface. This conclusion is, however, at variance with the views of Chaplin and Masters (9) which indicate that the clearance between sidewall and water surface must be kept very small. Operating a WSEV type vessel at high speed in close proximity to a rough sea surface raises major questions regarding structural strength and vehicle stability due to wave impact. This entire matter requires very careful experimental study and should be the subject of substantial future research.

It should be noted, however, that in the WSEV configuration, the lifting surface is well clear of the water surface, and the basic low aspect ratio shape of the WSEV reduces yawing moments produced by one sidewall impacting a wave compared to a WIG wing tip impacting at the same speed.

4.83

TABLE 3.—WSEV PERFORMANCE ESTIMATES

		W=200 tons	W=500 tons	W=1,000 tons	W=2,000 tons
CRUISE:					
(in ground effect)	C_L*	0·35	0·40	0·32	0·27
	L/D*	16	16·5	15·5	14·0
	V*	200	195	215	230
	SHP req*	18,000	42,500	100,000	240,000
	T req*	25,000 lbs.	60,500 lbs.	130,000 lbs.	285,000 lbs.
CRUISE:					
(sea level, outside	L/D	10	9·5	9·0	8·4
ground effect,	V	230 kt.	235 kt.	250 kt.	260 kt.
90%W)	SHP req	30,000	80,000	180,000	400,000
	T req.	36,000 lbs.	95,000 lbs.	200,000 lbs.	430,000 lbs.
RANGE/PAYLOAD:					
1,000 n. mi.	+F (tons)	20	50	110	240
	U (tons)	100	240	400	720
3,000 n. mi.	F (tons)	55	140	290	640
	U (tons)	65	160	220	320

+F=Fuel load plus 10% reserves and U=Payload

A possible further means of reducing wave impact problems is to make the lower portion of the sidewalls flexible. A patent (10) for this kind of arrangement has recently been issued to the Water Research Company. The clearance problem and the influence of a "real" (i.e. flexible and wavy "ground" plane) sea surface on vehicle drag and stability would appear to be the major area of uncertainty in assessing the overall feasibility of the proposed system.

Due to the preliminary nature of the WSEV proposal, full economic analysis of the system has not been performed. Several simple indices for comparing the "direct operating economy" of transportation vehicles can, however, give a fair indication of relative merits of various schemes.

Three basic performance indices (figures of merit) have been used here. The first of these is simply:

$$Q_1 = \frac{\text{Payload weight (tons)}}{\text{gross weight (tons)}} = \frac{U}{W} \qquad (9)$$

The index Q_1 is meaningful if two competing vehicles are compared for trips of equal length, and it gives an indication of the percentage of the total vehicle weight which is "useful".

The second index used has been extensively studied by a number of authors, most notably GABRIELLI and VON KARMAN (11):

$$Q_2 = \frac{\text{Weight} \times \text{Distance travelled}}{\text{Unit of Energy Consumed}}$$

$$= \frac{\text{Weight} \times \text{Speed}}{\text{Power}} = \frac{WV}{P} \qquad (10)$$

The quantity Q_2 can be made dimensionless and can be defined in a variety of ways depending on whether the weight specified is the gross weight or the payload weight, the speed is maximum or cruise, etc. Q_2 is a meaningful comparison if the operating ranges used are equal.

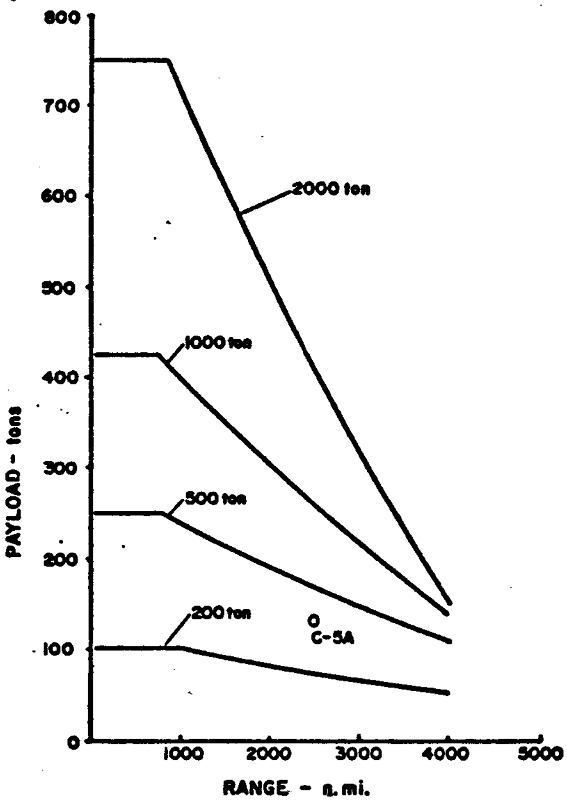
The third index used is:

$$Q_3 = \frac{\text{Payload} \times \text{Speed}}{\text{Gross Weight}} = \frac{UV}{W} \qquad (11)$$

This is frequently referred to as the "productivity index" of a vehicle.

Values of the three Q parameters for the C-5A aircraft, a typical ACV (12), and the four versions of the WSEV studied (W = 200, 500, 1,000 and 2,000 tons) are summarised in Table 4. This table, together with the range-payload diagrams shown in Figure 8, gives a reasonably clear picture of the direct operating economics of the WSEV when compared to large aircraft and ACVs. Because of the large differences in cruise speed, the average productivity (Q_3) of the WSEV is about half that of the C-5A, but several times that of the ACV. On the basis of energy consumed per ton-mile travelled (Q_2), however, the WSEV is very much superior to the C-5A and the ACV. Payload/gross weight fractions (Q_1) for the WSEVs are very favourable compared to the aircraft and the ACV.

The above results are, admittedly, a very limited comparison of the overall economics of the systems considered. However, for a number of military missions, the WSEV may be attractive, particularly when consideration is given to the additional, less quantifiable characteristics of the WSEV:



Assume 10% Fuel Reserves

Fig. 8. WSEV range/payload diagrams

1) The vehicle does not cruise in the water or at high altitude thus reducing vulnerability to both underwater and airborne detection and countermeasures.
2) The favourable geometric shape of the WSEV provides cargo holds of large size thus allowing bulky cargo to be carried more easily and with less penalty than in the basically round tubular fuselage of conventional aircraft.

TABLE 4.—ECONOMIC SUMMARY

	C-5A	ACV	WSEV A	WSEV B	WSEV C	WSEV D
W (tons)	382	300	200	500	1,000	2,000
Ver. (kt.)	457	60-80	200	195	215	230
$+Q_2 = \frac{WV}{P}$ (at cruise)	9·1	5·8	16	16·5	15·5	14·0
* 1,000 n. mi. Range:						
$Q_1 = \frac{U}{W}$	0·45	0·30-0·40++	0·51	0·49	0·41	0·37
$+Q_2 = \frac{UV}{P}$	4·1	1·5-3 ++	8·2	8·1	6·4	5·2
$Q_3 = \frac{UV}{W}$	205	20-30 ++	102	96	87	85
* 3,000 n. mi. Range:						
$Q_1 = \frac{U}{W}$	0·36	—	0·35	0·33	0·24	0·19
$+Q_2 = \frac{UV}{P}$	3·3	—	5·6	5·5	3·7	2·7
$Q_3 = \frac{UV}{W}$	164	—	70	65	52	44

$+Q_2(\text{dimensionless}) = \frac{7{\cdot}2\ W\ (\text{tons}) \times V\ (\text{kt})}{P\ (\text{SHP})} = \frac{W\ (\text{ton})}{T\ (\text{ton})}$

++ Range of 100 n. mi.

*No reserves, cruise values of V (kt.), P (SHP), W (tons)

WSEV Applications

The initial mission envisioned for the larger WSEVs was low-density cargo transport over ranges from 1,000 to 3,000 nautical miles. In this case the basic configuration of the vehicle, sized to provide sufficient lifting surface area to "take-off" at speed and power settings consistent with hydrodynamically imposed "hump speed" and aerodynamically imposed cruise speed limits, results in cargo spaces inside the lifting surface capable of accommodating a wide variety of bulky low-density cargo. Further, the payload weight/space capacity of the larger vehicles is consistent with typical military cargoes such as troops, fully assembled helicopters, etc. Despite the previously assessed decrease in "productivity" compared to large aircraft like the C-5A, the increased "economy" and the favourable geometry of the WSEV which allows it to circumvent conventional aircraft size limits, plus the favourable operational characteristics such as use of water rather than land bases and decreased vulnerability to detection and interception make the WSEV attractive for "light-weight" (compared with ships) naval logistic missions.

The same features of the WSEV which benefit the logistics mission may offer potential advantages in other naval military missions. Specifically, the large "flat" catamaran type platform configuration of the vehicle and the high speed (compared to ships and SES vehicles) cruise capability give the WSEV substantial mobility.

High Mobility Platforms

For naval missions which require a rapid deployment capability with "light" payloads (but beyond the limits of conventional aircraft) and which require the existence of a relatively stable platform while stationary, the basic WSEV configuration may be promising. Three missions for such a capability can be initially envisioned: (1) ASW, (2) early warning and air-defence missile launching, and (3) short-to-intermediate range ballistic missile launching. All of these missions might be equally well served by other SEV types (e.g. the CAB); however, none possesses the high speed/range capability of the WSEV. Future loss of overseas bases and/or the need to establish well defended ports in unforseen locations make the air defence platform attractive. Similarly the need for a rapidly deployable "shore bombardment" platform may increase in future military scenarios.

Rapid Response "Carrier"/Support Vessels

Given the size and range/payload/speed characteristics of the WSEV, an intriguing set of "carrier" vehicle missions can be contemplated. The recent experience in Vietnam has shown the need for a class of very nimble, high speed patrol craft of relatively small size (13). Unfortunately, manoeuvrability and range are not consistent design variables and to provide a limited number of these types of vessels with the range, let alone the required support (weapons replacement, maintenance, fuel) facilities, to cover the realistic operating areas is a very formidable task. Similarly, the same design conflict exists between range and nimbleness in helicopters, VTOL attack aircraft and ACV landing craft. Since all of these vehicles fall into the category of low-density cargoes when carried by a "mother" vessel, and none individually exceeds the size and payload capability of the WSEV, it is then natural to evaluate the possibility of configuring a WSEV as a carrier/support ship for each case. A typical WSEV, capable of carrying a pair of nominal 200 ton patrol hydrofoils together with required weapons support personnel and fuel, is shown in Figure 9. The catamaran type configuration of the basic WSEV is particularly well suited to the launch and retrieval of the patrol craft. The patrol craft would be carried in semi-submerged wells in the underside of the WSEV, there being adequate space inside the wing to completely enclose the above deck structure of the patrol boat and its retracted hydrofoil support struts, thus minimising the aerodynamic penalty to the WSEV in cruise flight. In the same way, ACV landing craft could be carried in underwing wells, with troops and their assault weapon housed inside the wing. For helicopter/VTOL missions, the large holds in the WSEV would serve as hangar space and an elevator would raise them to a take-off platform on the wing upper surface.

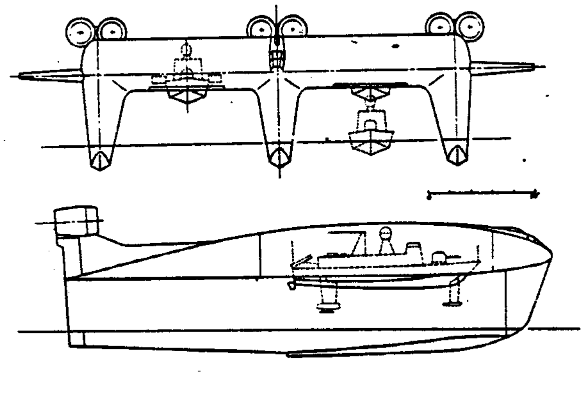


Fig. 9. WSEV—coastal patrol carrier/support configuration

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Hydrogen Fuel Depot Vessel

The crisis both in cost and ultimate availability of hydrocarbon fuels has lead naturally to extensive considerations of alternative fuel sources. Strong cases have been made (14, 15) for the desirability and feasibility of using liquid hydrogen (LH_2) as a prime fuel—particularly in aircraft. The three main disadvantages to the use of LH_2 in military applications are:

1) On a unit energy content per unit weight basis LH_2 has about one fourth the weight density (weight per unit volume) of conventional hydrocarbon fuels. This is a severe penalty in volume limited vehicles.
2) LH_2 requires cryogenic storage, with a consequent fuel tankage weight penalty.
3) Some type of energy source (nuclear, solar, etc.) is required to "manufacture" the LH_2.

Again the WSEV may serve as a promising basis for a vehicle designer to exploit use of hydrogen fuel. The configuration of the WSEV, sized to fly at relatively low speeds on aerodynamic lift, is less limited in volume than either hydrodynamically supported vessels or high speed aircraft. Thus it possesses the internal capacity to carry the tankage required for LH_2 directly. In addition, one can easily contemplate a *fleet* of WSEVs, all hydrogen fuelled, *one* of which is configured to carry a nuclear reactor and associated storage tanks to serve as a "tanker". Thus the requisite LH_2 could be manufactured on site, with the tanker possessing the same range/mobility characteristics as the other elements of the fleet. To reduce vulnerability, the tanker WSEV could be configured to be externally indistinguishable from the other fleet elements (landing craft carrier/support vehicles, etc.).

Concluding Comments

A preliminary design description has been presented on a type of naval surface effect vehicle which holds promise of circumventing weight and size limits in conventional aircraft design, and speed and complexity limitations on existing SES vehicles (e.g. ACVs and CABs). The proposed WSEV design is based on well established aerodynamic principles and required no major breakthrough in aerodynamic or propulsion technology. The performance of the WSEV is significantly enhanced, however, if advances can be made in ultra low-density composite structural technology, and the practicality of extensive boundary layer control and LH_2 fuel can be demonstrated. A favourable trade-off exists between productivity and economy for the transport of low-density cargoes, even without the use of advanced technology refinements.

The ability of the WSEV to carry large bulky cargo over significant distances at relatively high speeds compared to competing naval SES systems, and its ability to actually fly at substantial heights to avoid high sea states and other surface obstructions make the WSEV concept attractive for both military and commercial applications. In addition a number of interesting purely military missions have been briefly described which would indicate a wide future use for WSEV type vehicles.

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4.86

V. Conceptual Design I - Configuration Selection (3 hours)

A. Geometry Definitions

B. Basic Principles

1. Fuselage Layout
2. Weight and Balance
3. Wing Shape
4. Landing Gear
5. Engine Placement

C. Survey of Flying Configurations (Morphology)

D. Examples of Configuration Selection

1. GERMAN AIRCRAFT
2. CARAVELLE RADIANCE
3. DC-9/B737 COMPARISON
4. ~~[illegible]~~

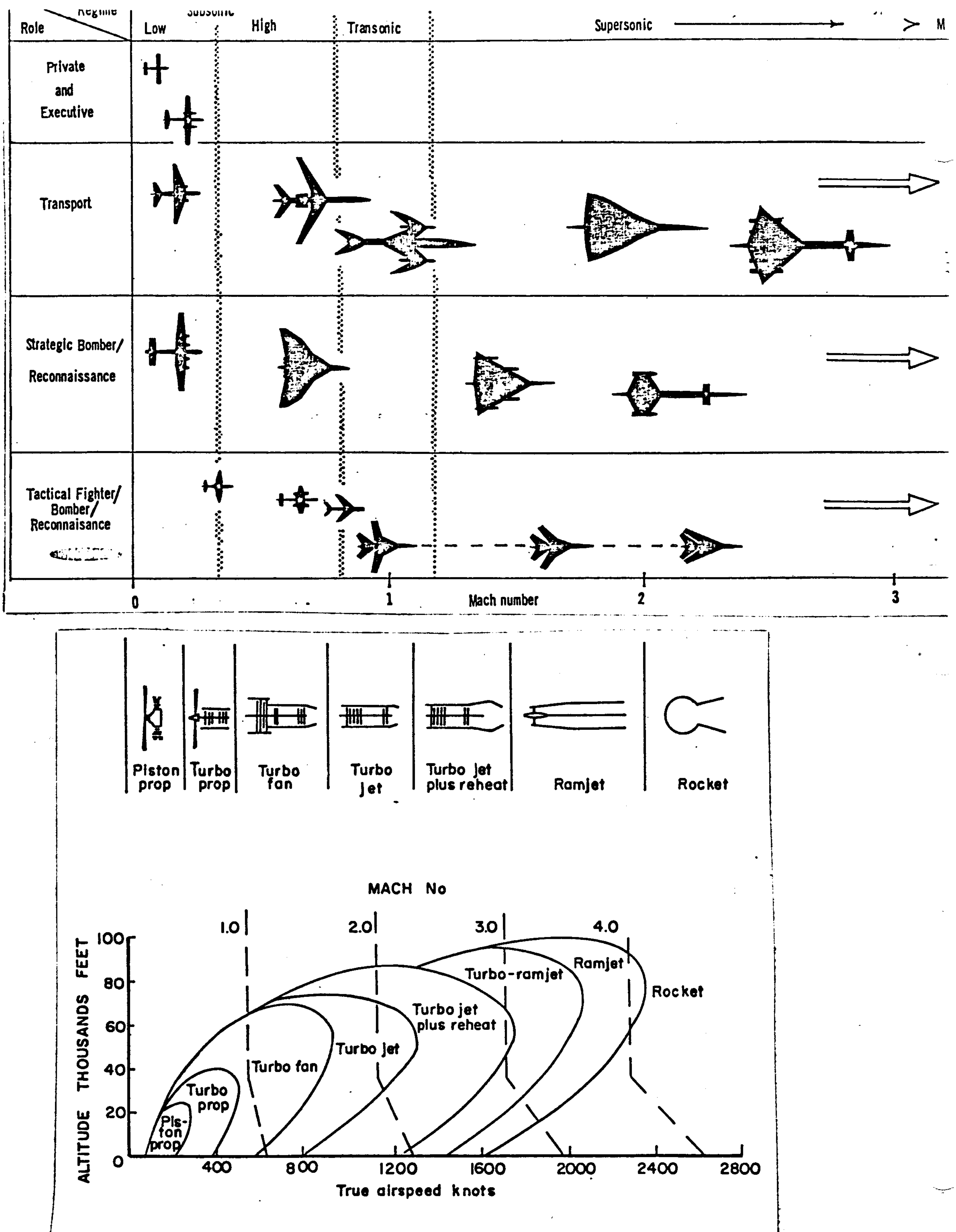


Fig. 4.6 The Level Flight Propulsion Picture (figure from Reference 16, first published by GT Foulis and Co. Ltd. of Sparkford, Yeovil Somerset, England in 1966)

5.02

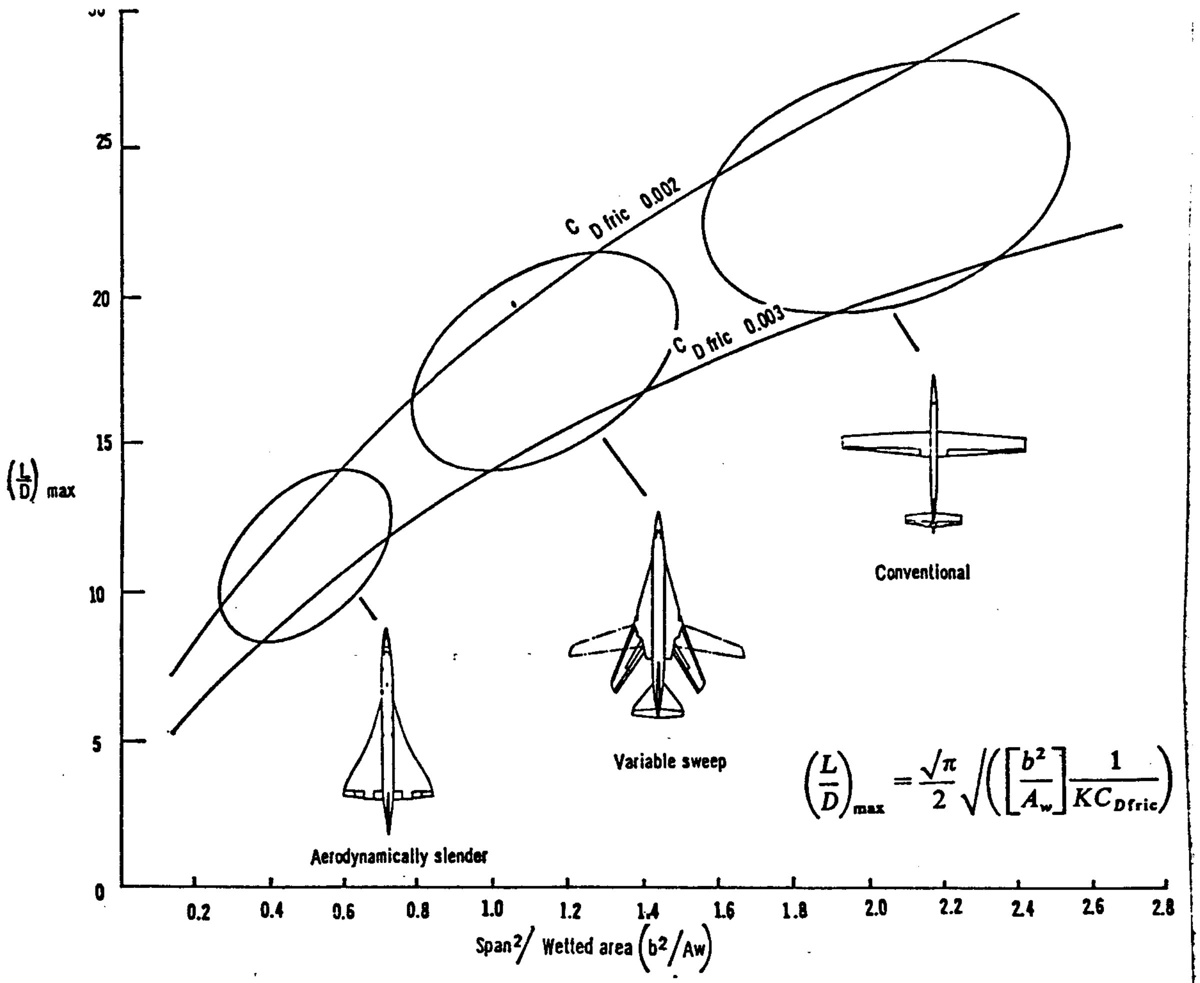


Fig. 6.23. Subsonic (subcritical) cruise relationship between span, wetted area and frictional drag coefficient

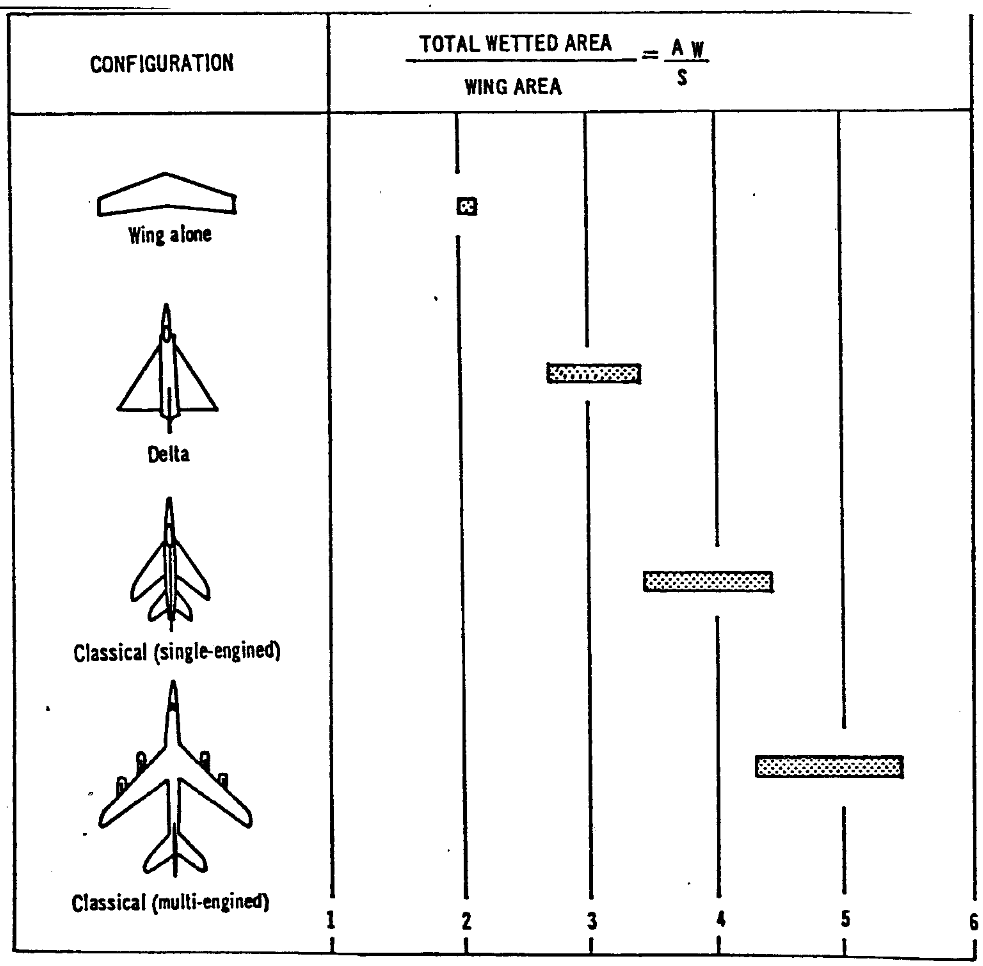


Fig. 6.2. Wetted area of airframe in terms of wing area for different configurations

C

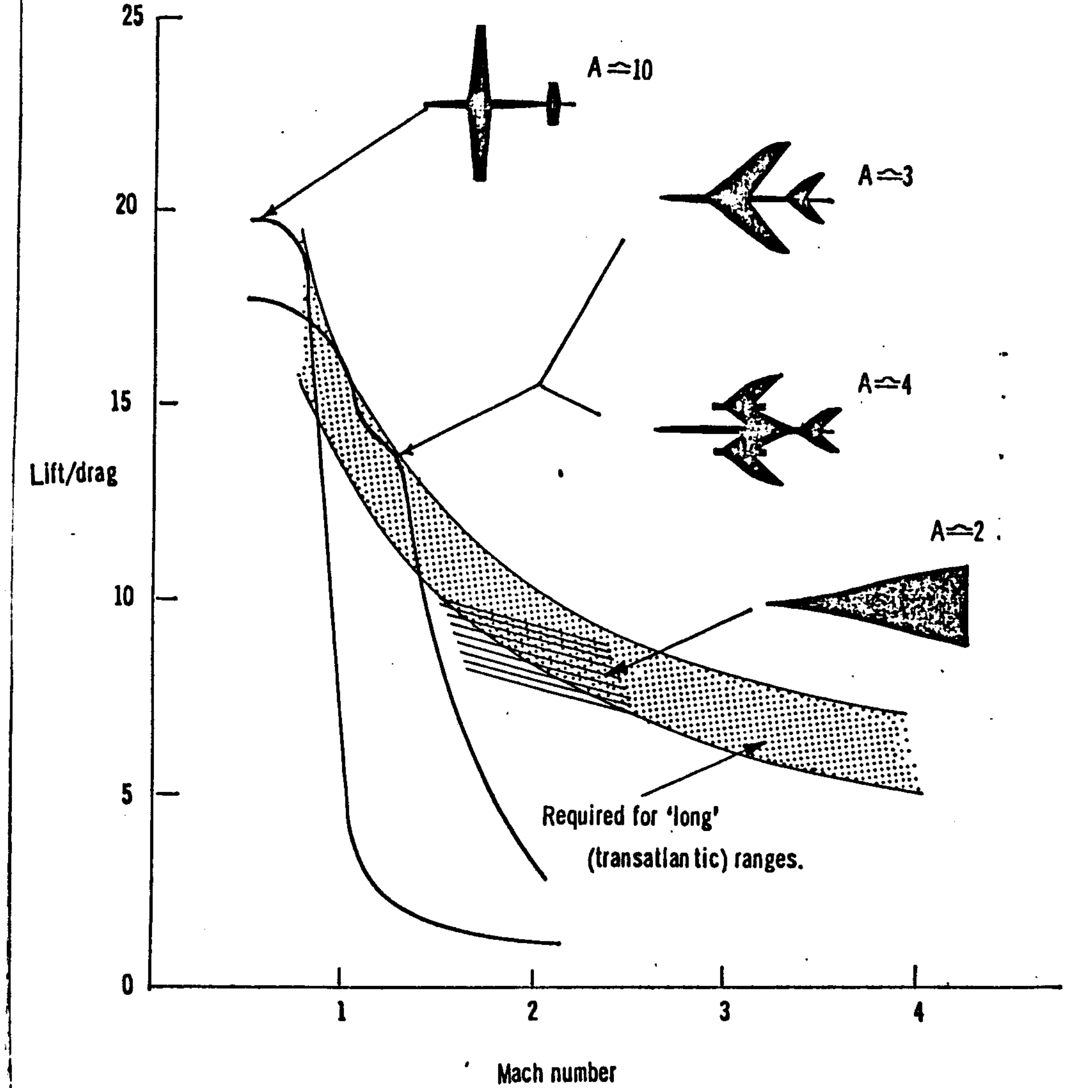
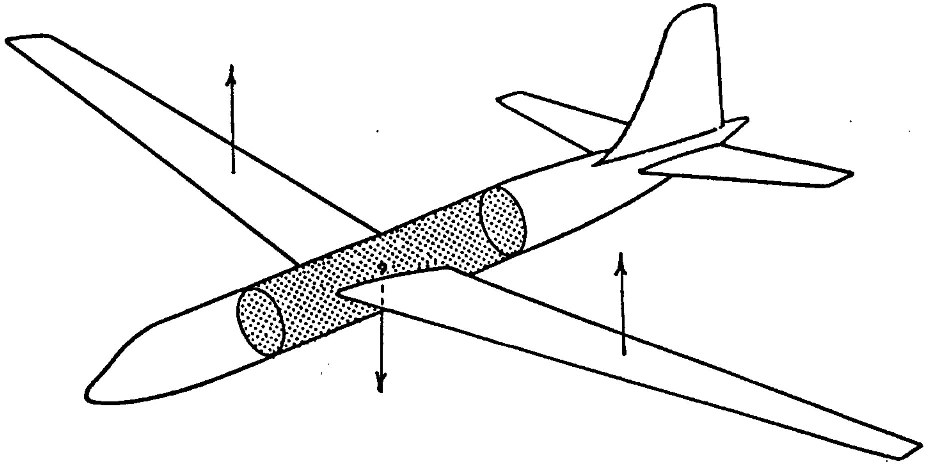
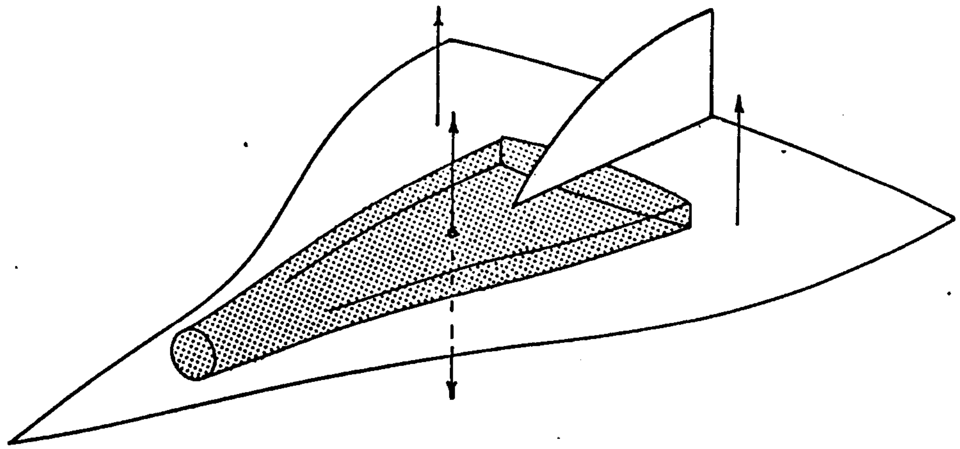


Fig. 6.21. Shape trends giving 'good' lift/drag ratios for typical long ranges of transatlantic order (3000 nml)



a. The 'classical' layout of an aeroplane (discrete lift)



b. An 'integrated' layout (distributed lift)

Fig. 3.3. The two basic families of aeroplanes

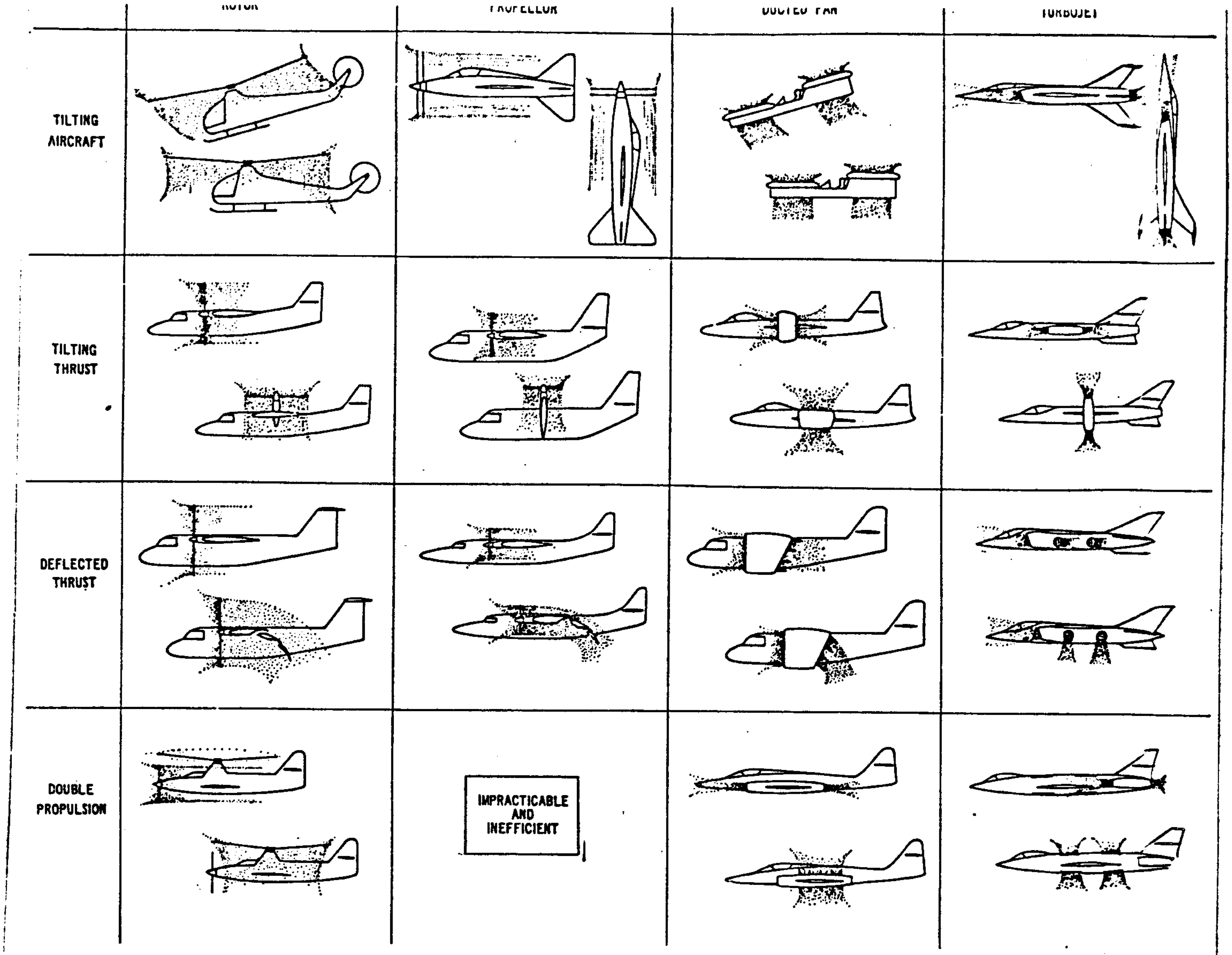
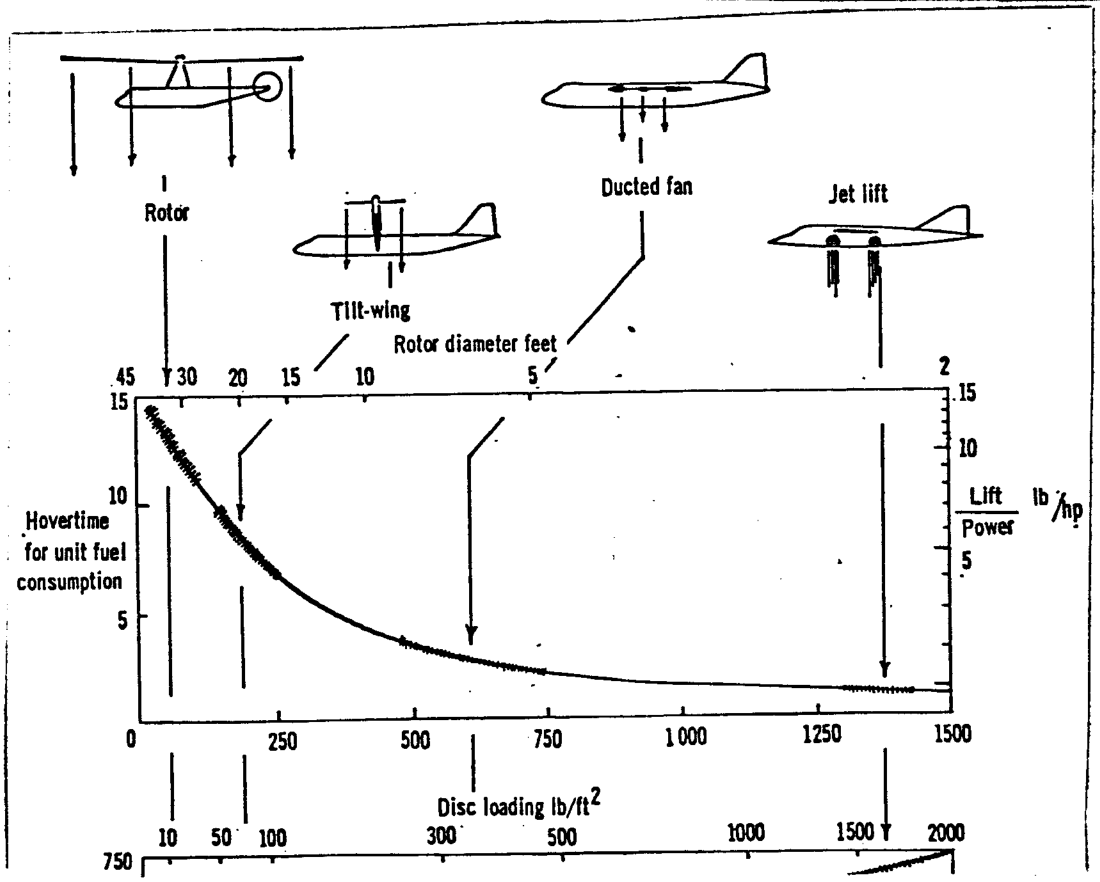


Fig. 7.14. The VTOL family of aircraft



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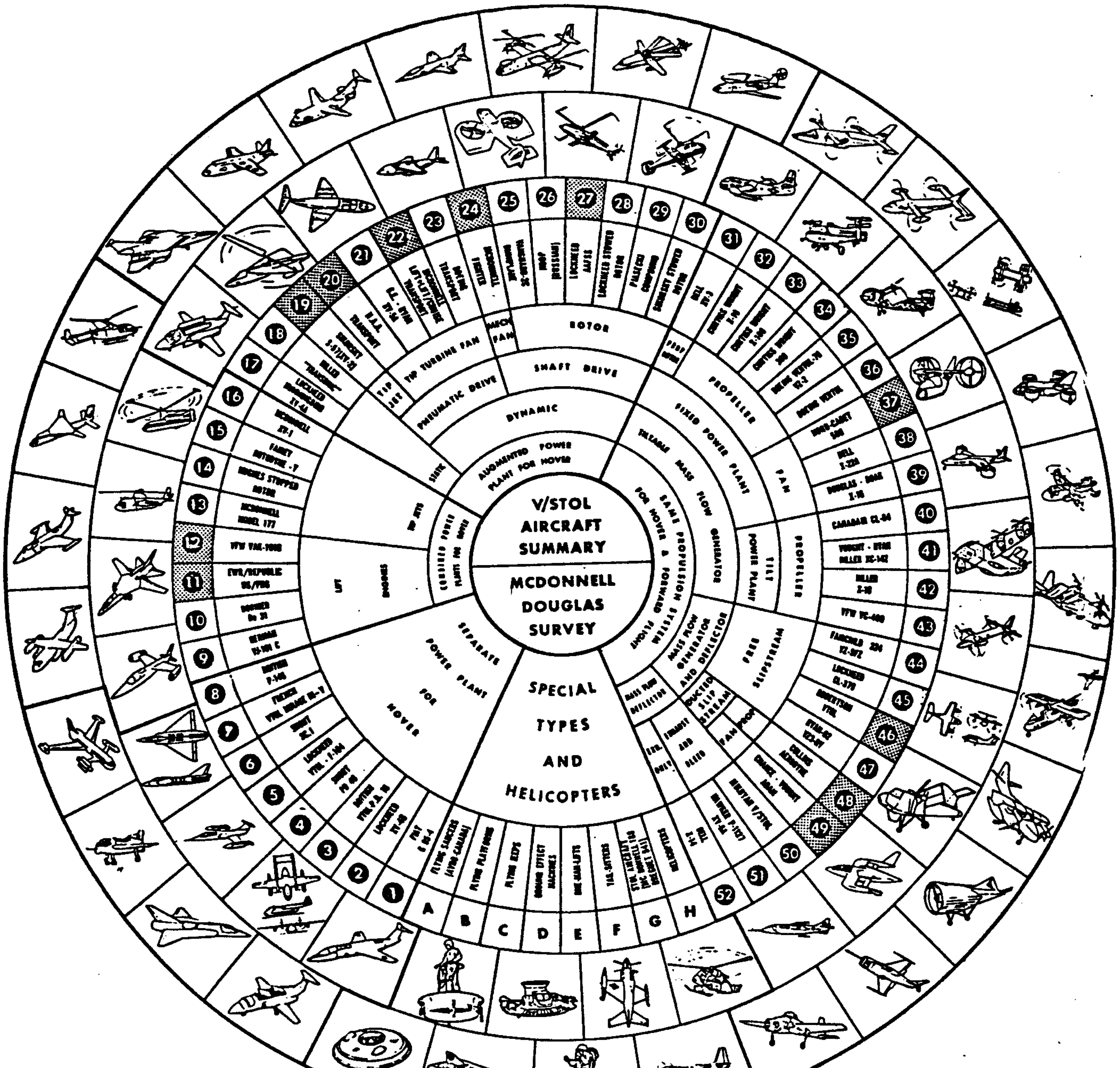
THE V/STOL MERRY-GO-ROUND

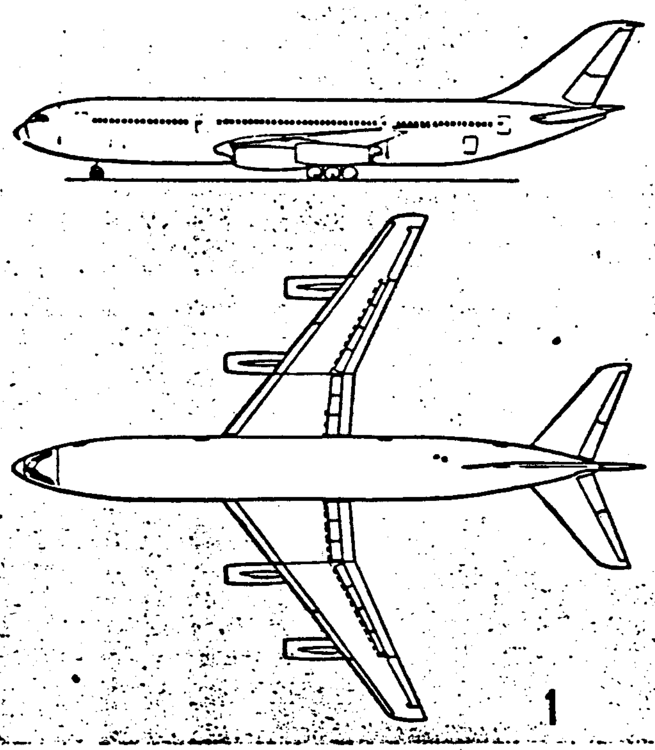


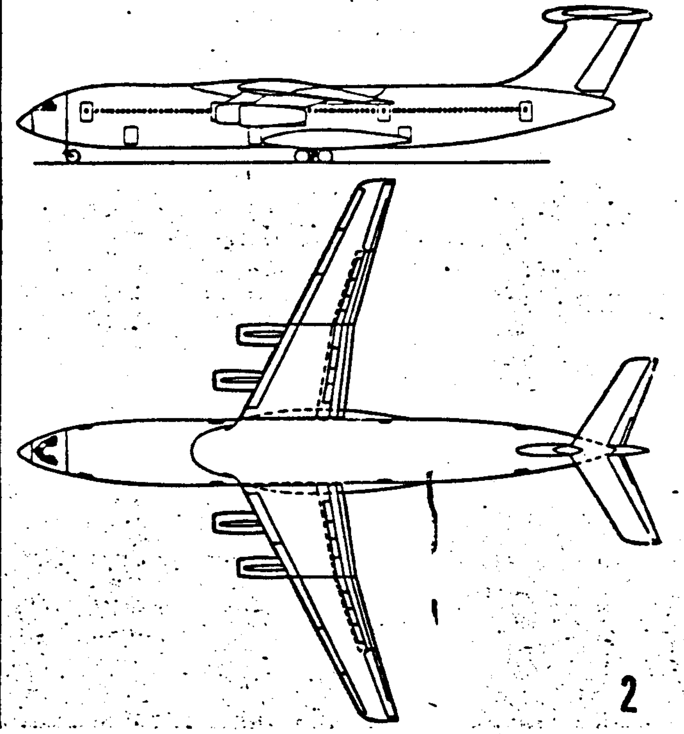
FLYING PROTOTYPE

UNDER CONSTRUCTION, WIND TUNNEL TESTS OR MOCK UP

DESIGN CONCEPT STUDY

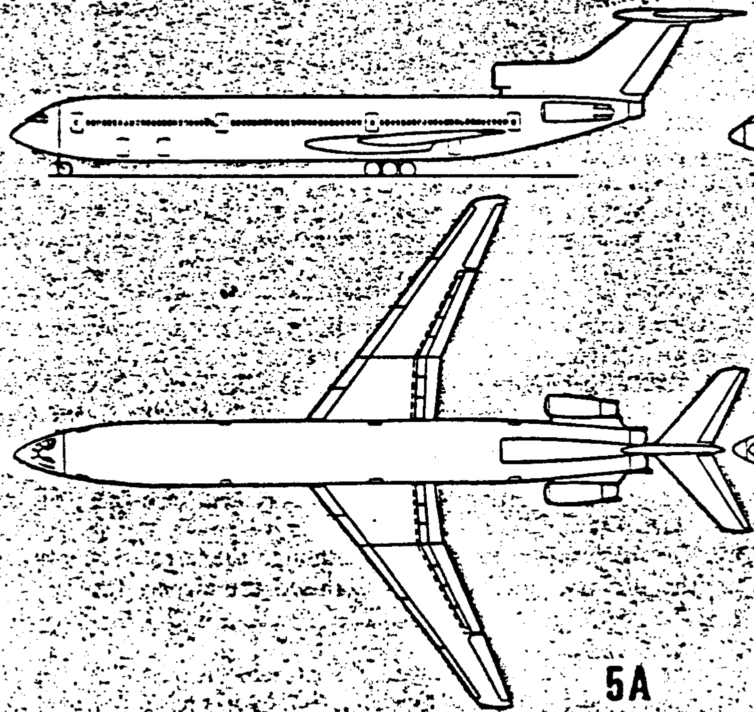
5.46

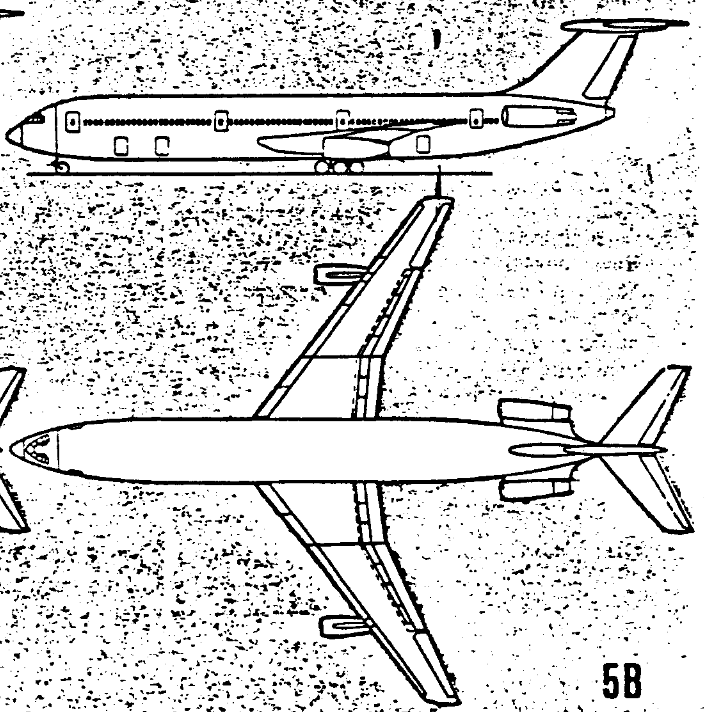




Soviet Drawings Show Design Studies for Il-86

Unusual detailed look at the iterations in Soviet aircraft design is provided in these drawings of configurations investigated by the Ilyushin Design Bureau before arriving at the final configuration of the Il-86 wide-body transport, panel No. 1. In No. 2, a high-wing configuration similar to the Il-76 freighter was examined. Configura-





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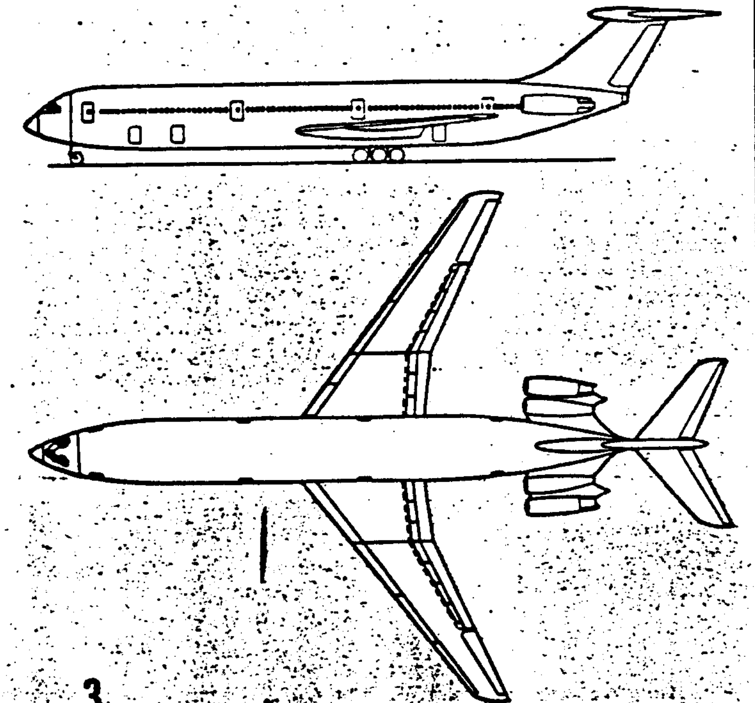

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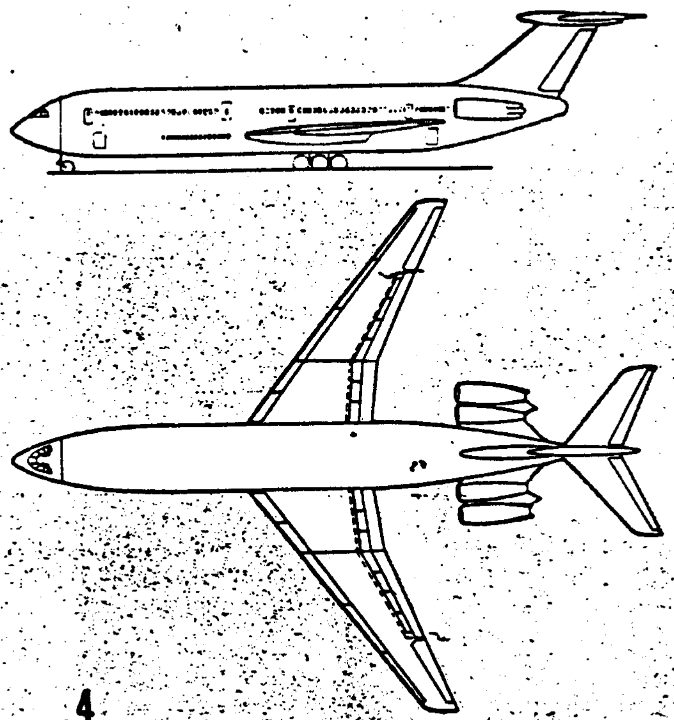
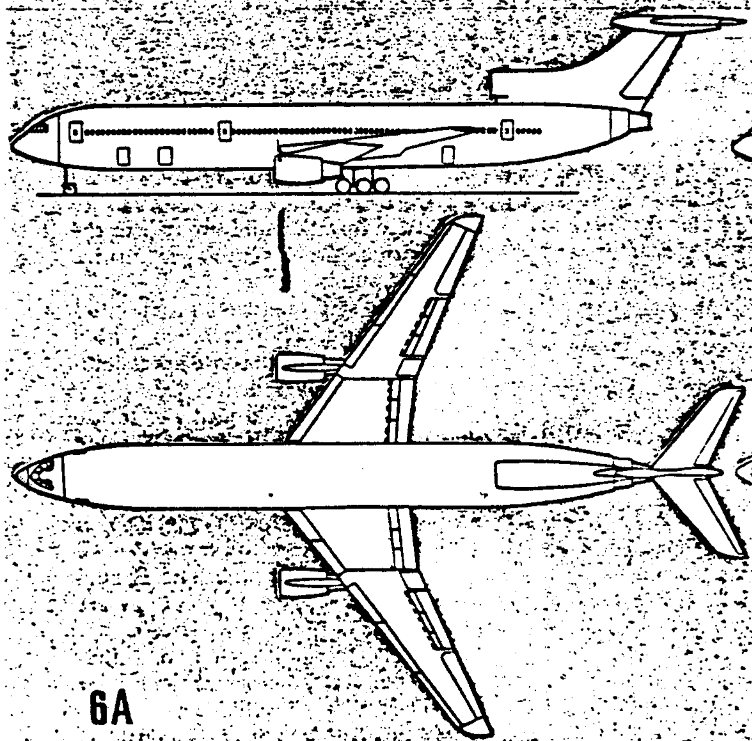


4

tion No. 3 was based on the Il-62 passenger transport, with four aft-mounted engines. No. 4 is a two-deck version of No. 3. No. 5A has a three-engine arrangement similar to the Boeing 727. No. 5B has two aft-mounted and two wing-mounted engines, a configuration never before produced anywhere. No. 6A is an arrangement with three high-bypass-ratio turbofans—which have never been produced in the Soviet Union—similar to the Lockheed L-1011 TriStar, while No. 6B also has three high-bypass turbofans arranged similarly to the Boeing 727. No twin-engine designs were considered, an indication of the Soviet lag in high-bypass-ratio technology.



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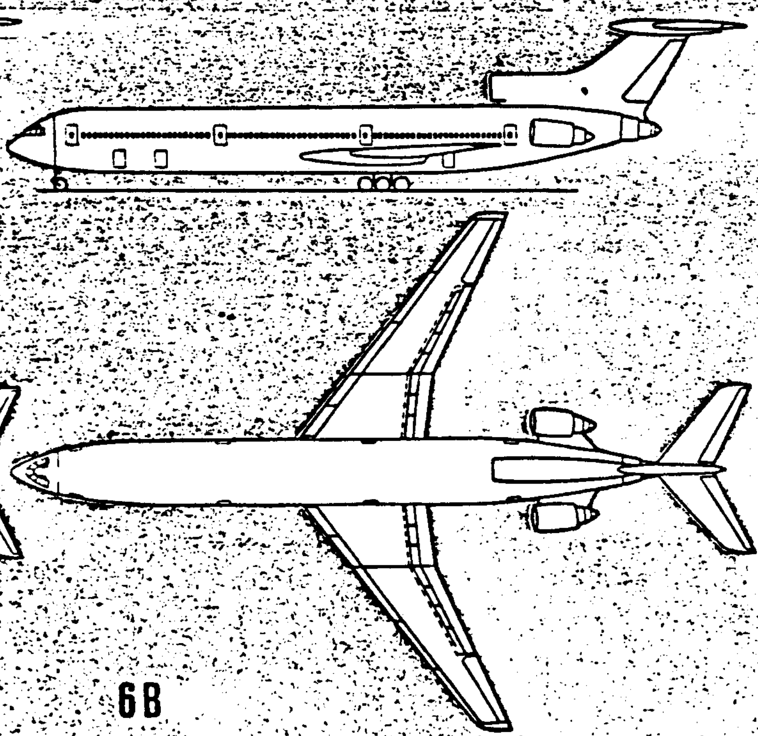
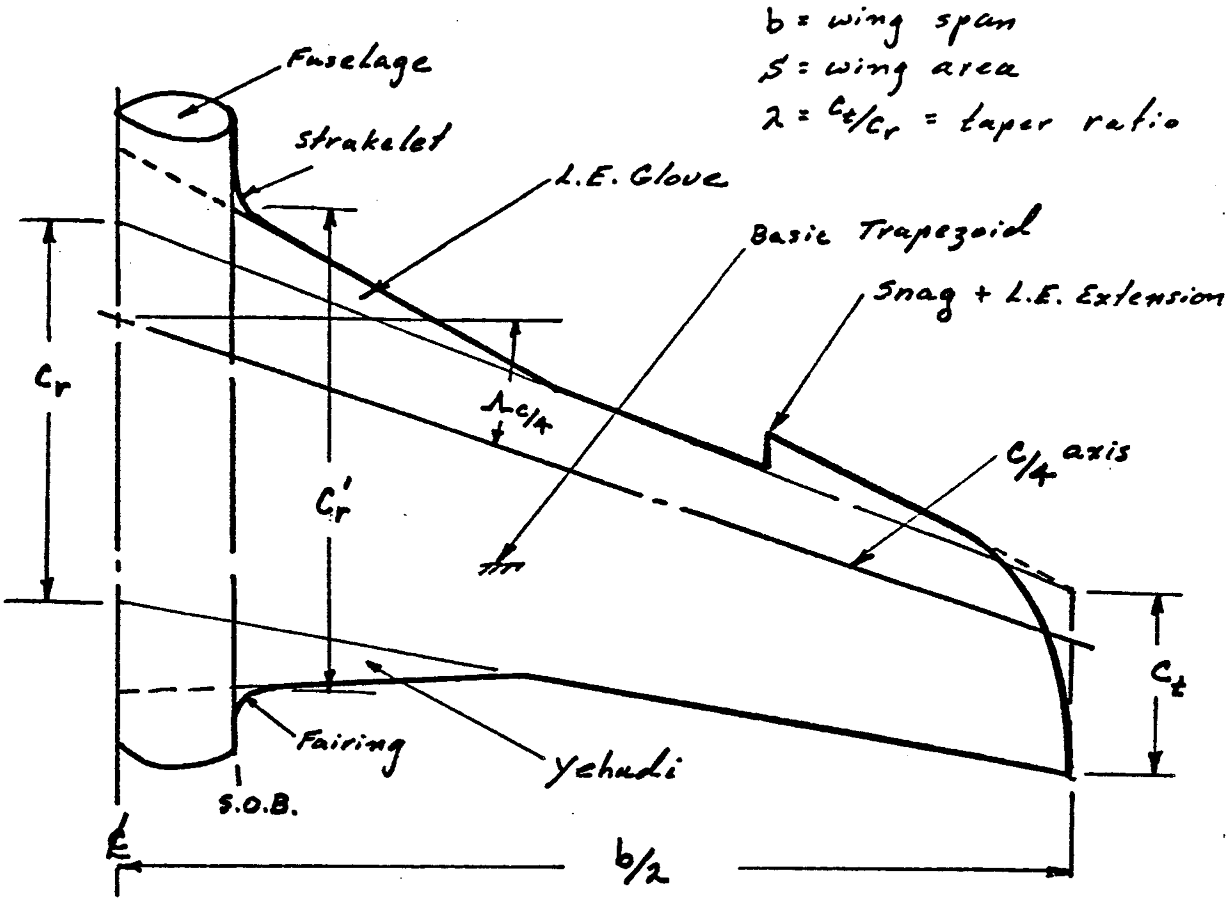


6B

3-08

Wing Ref. Area Definition



As it turns out, there are at least five values of "reference" wing area (high lift devices retracted) in common use throughout the international aerospace industry. These include:

1. Trapezoid area (extended to the plane of symmetry).

2. Gross area (total projected wing area extended to the plane of symmetry).

3. Exposed area (excludes the wing area inside the fuselage).

* 4. Weighted gross area (a typical example being the Boeing Wimpress area).

5. Aerodynamic reference area (defined at the discretion of the particular project group).

---

6. For bat wings, ogee planform triplanes and planes of ~~non~~ contiguous domain over the half-shell punt.

7. For all other cases:

$$S_{RR} = \text{Reseach Ref. Area} = \oint \sum_{i=1}^{N} \Delta S_i \, dl$$

where $\Delta S_N \Rightarrow A_N > A_c \geq c$

A_W = wrong answer

A_c = correct answer

WING AREA and Æ:

The reference wing area definition shown on page 3 is based on effective aerodynamic area recommended by J. K. Wimpress as head of Aerodynamics Staff in 1965. Reference wing area is the effective area in the wing reference plane projected through the wing dihedral angle to a "shadow plane." This projection is performed to more closely represent wing area normal to the lift direction. A similar projection is performed on span in the definition of effective aspect ratio. For the calculation of effective aerodynamic characteristics, span is defined as the outboard edge of the lifting surface as represented by the wing trailing edge. Therefore, raked tips are included in span calculations but normal or "working tips" are not included.

MEAN AERODYNAMIC CHORD AND LOCATION:

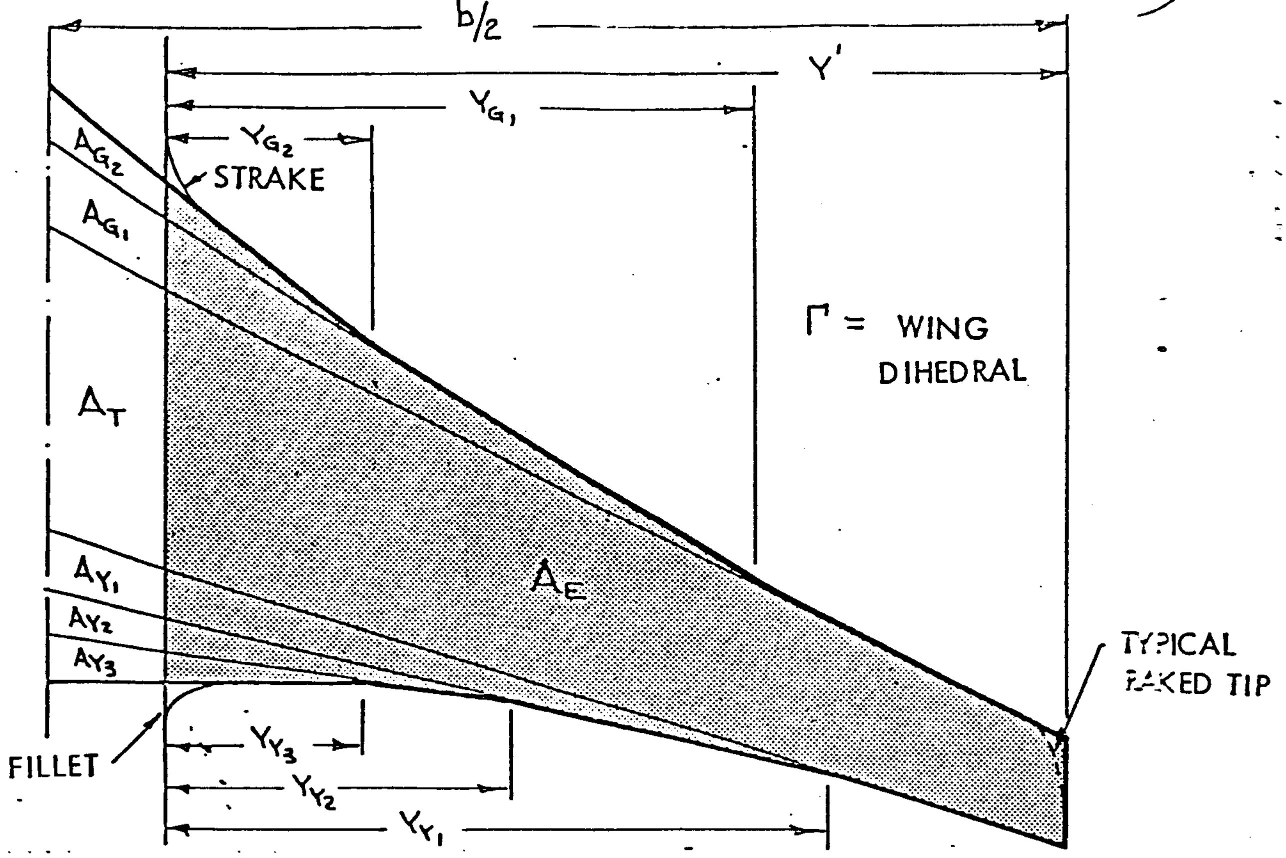
The general definition of Mean Aerodynamic Chord (MAC) is:

$$MAC = \frac{\int_o^b C^2\,dy}{\int_o^b C\,dy} = \frac{\int_o^b C^2 dy}{AREA}$$

In the case of a trapezoidal wing, the above simplifies to:

$$MAC = \frac{2}{3} C_R \left(\frac{\lambda^2 + \lambda + 1}{1 + \lambda}\right) \quad \text{where } \lambda = \frac{C_T}{C_R} = \frac{\text{TIP CHORD}}{\text{ROOT CHORD}}$$

This simplification is less than satisfactory for wings with gloves and yehudies. In current wing designs, gloves are loaded to carry lift similar to the basic wing leading edge. Therefore, neglecting gloves in MAC calculations ignores a potentially significant fraction of lift and pitching moment generating chord. Common design techniques on the inboard wing tend to maintain or increase the sweep of the maximum thickness and pressure recovery points to prevent the inboard shock from unsweeping. As a consequence, yehudies which end inboard of 50% semispan tend to be lightly loaded and do not significantly contribute to wing lift or pitching moments Yehudies which end outboard of 50% semispan, however, tend to be loaded similar to the basic wing trailing edge because they are used to delay the high speed wing breakdown which starts on the mid wing. A more representative simplification of the MAC calculations for wings with gloves and yehudies is to ignore yehudies which end inboard of 50% semispan but to include all other yehudies and gloves.



GENERAL NOTES

- PARAMETERS COMPUTED IN W.R.P. AND THEN PROJECTED THRU Γ
- IGNORE LE STRAKES AND TE FILLETS
- WING WITH RAKED TIP EVALUATED AS A SQUARE TIPPED WING WITH SPAN OF ACTUAL WING TE

EXPOSED WING AREA

$$S_{EXP} = A_E$$

AVERAGE EXPOSED CHORD

$$C_{EXP} = A_E / 2Y'$$

REFERENCE WING AREA

$$S_{REF} = \left(A_E + A_T + \frac{Y_{G_1}}{Y'} A_{G_1} + \frac{Y_{G_2}}{Y'} A_{G_2} + \cdots + \frac{Y_{Y_1}}{Y'} A_{Y_1} + \frac{Y_{Y_2}}{Y'} A_{Y_2} + \frac{Y_{Y_3}}{Y'} A_{Y_3} + \cdots \right) \cos\Gamma$$

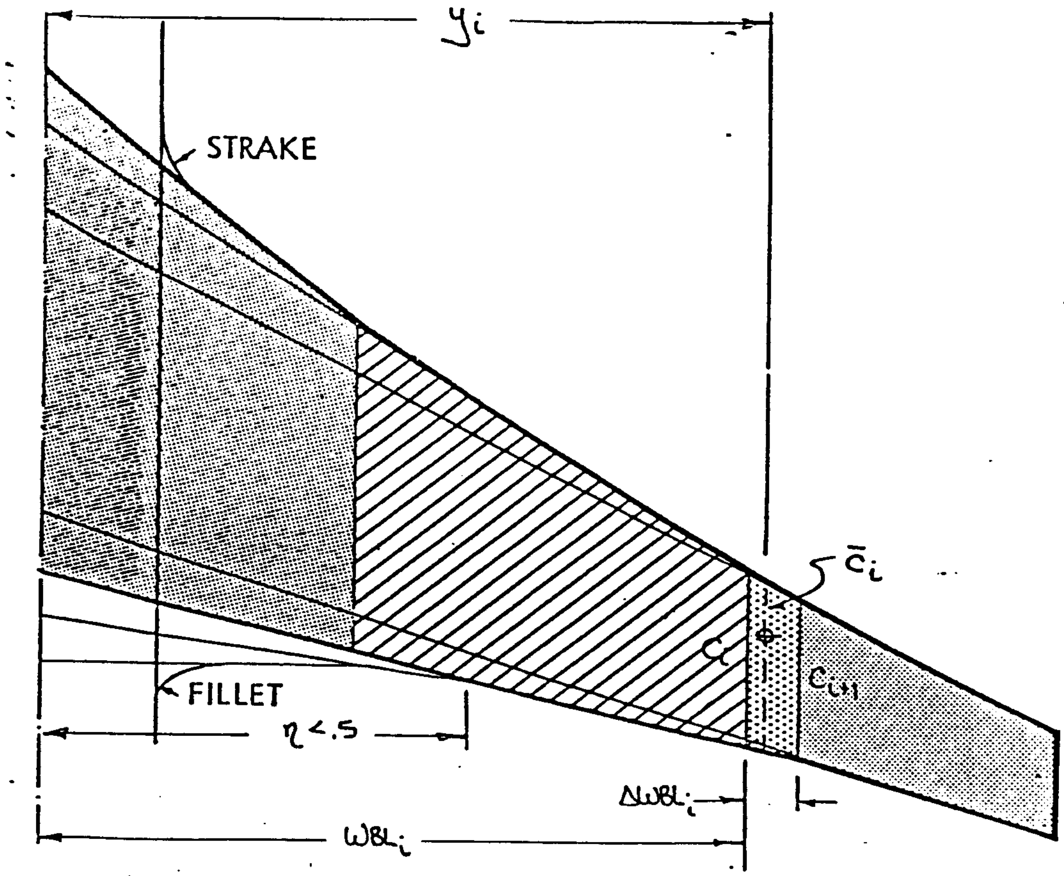
REFERENCE ASPECT RATIO

$$A\!\!R_{REF} = \frac{(b \cos\Gamma)^2}{S_{REF}}$$

AVERAGE CHORD

$$C_{AVG} = S_{REF} / b\cos\Gamma$$

"Wimpress Area" Definition



GENERAL NOTES

- IGNORE LE STRAKES AND ALL YEHUDIES WHICH END INBOARD OF 50% SEMISPAN.
- WING WITH RAKED TIP EVALUATED AS A SQUARE TIPPED WING WITH SPAN OF ACTUAL WING TE

EFFECTIVE MAC

$$MAC = \frac{\frac{2}{3}\sum \Delta WBL_i \left(c_i^2 + c_i c_{i+1} + c_{i+1}^2\right)}{\sum AREA_i}$$

WHERE $AREA_i = \Delta WBL_i \left(c_i + c_{i+1}\right)$

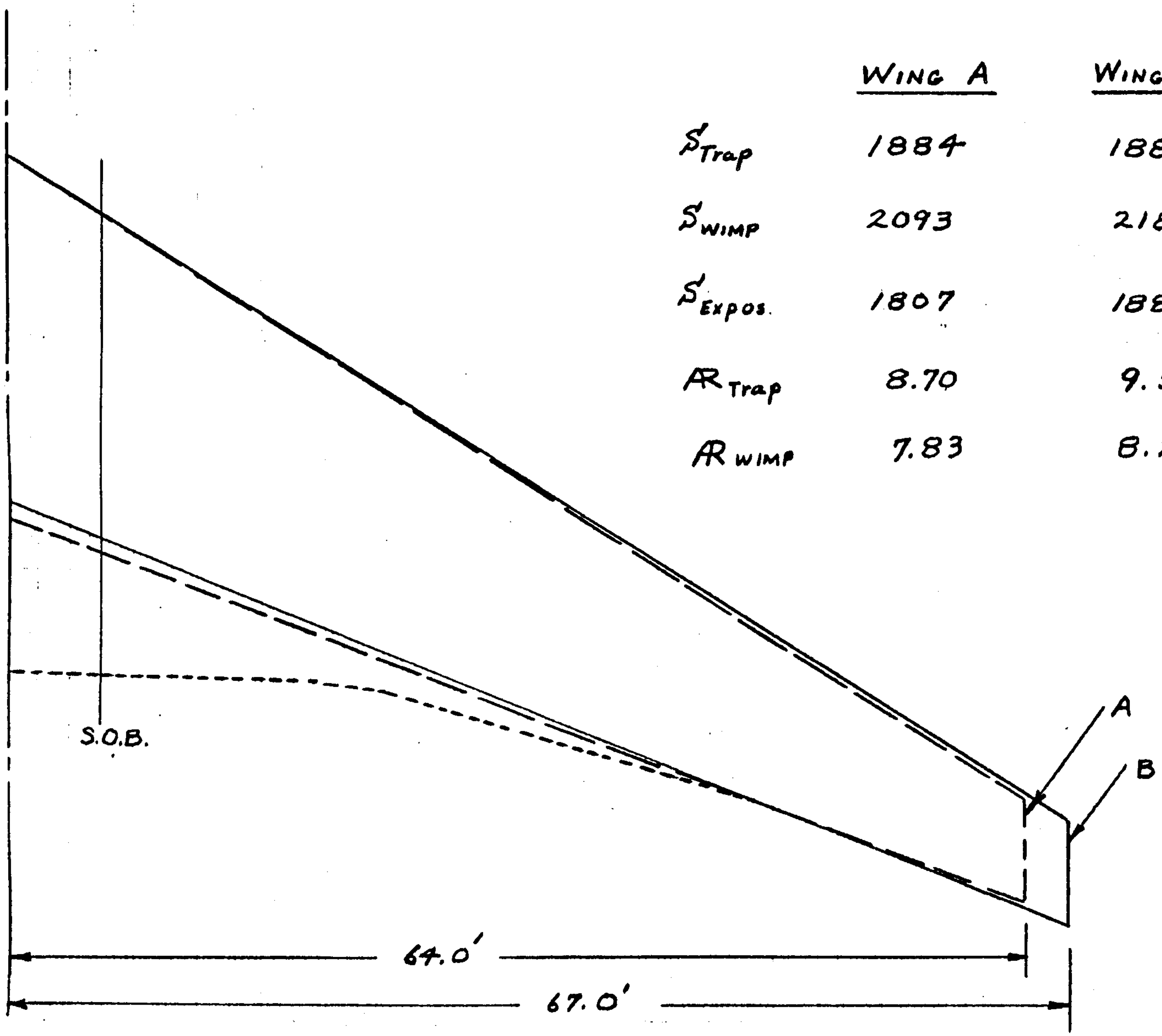
SPAN OF MAC

$$\bar{y} = \frac{\sum \bar{y}_i \, AREA_i}{\sum AREA_i} \qquad \bar{y}_i = WBL_i + \frac{\Delta WBL_i}{3}\left(\frac{c_i + 2c_{i+1}}{c_i + c_{i+1}}\right)$$

LOCATION OF MAC/4

BASED ON $\bar{y}$ & LOCUS OF LEADING EDGE

Mean Aerodynamic Chord Definition



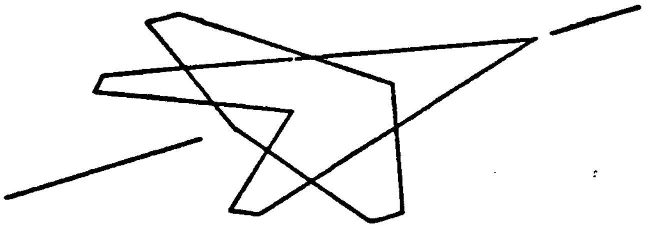
	Wing A	Wing B
S_{Trap}	1884	1884
S_{WIMP}	2093	2180
$S_{Expos.}$	1807	1888
AR_{Trap}	8.70	9.53
AR_{WIMP}	7.83	8.24

AIAA Paper 80-3031 (March 1980)

— WINGS —

JOHN CHUPRUN, JR. *
USAF/AERONAUTICAL SYSTEMS DIVISION
WRIGHT-PATTERSON AFB, OHIO

WINGS



INTRODUCTION

The aims of this "Brief" are three-fold:

- To show the wide variety of wing shapes that have found their way to application on aeronautical systems.

- To highlight the powerful leverage that wings possess in fulfilling their prime function, i.e. generation of lift.

- To provide some insight on the design considerations leading to selection of wing geometry for a particular application.

The overall objective is to convey some appreciation for and fundamental insight on how and why wings evolve to their geometric configurations.

WING SHAPES

In the beginning, mans' desire to fly provided the impetus for conceptualizing various schemes to create lift. To fly - like a bird, was a popular starting point.

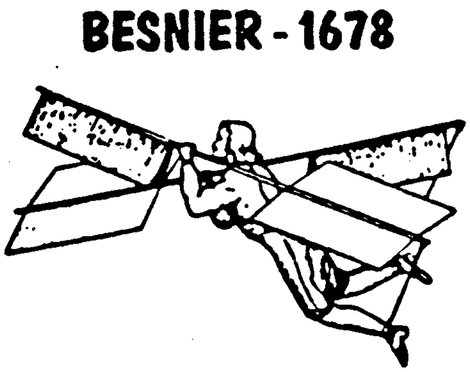


Figure 1. Emulate the Birds

* Director - Design Analysis
Deputy for Development Planning

It was natural then, that early concepts for wings were not unlike those found on Nature's creatures of flight. Shown here are the flying machines conceived by some of the early entrepeneurs of powered flight.

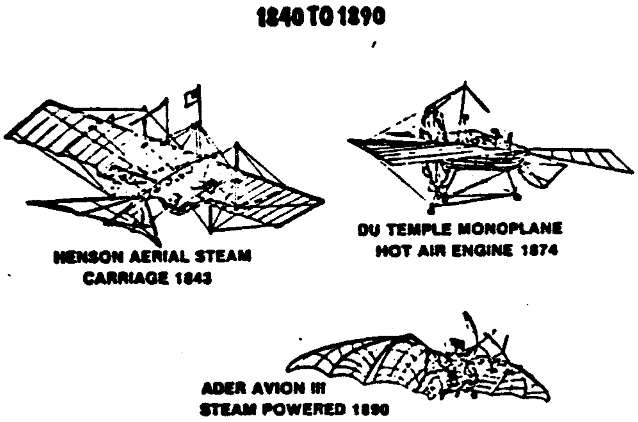


Figure 2. Early Wing Concepts - Powered Flight

Given the difficulties of a suitable powerplant, some turned their attention to Gliders - soar like a bird. Note, however, the wings on the Chanute and Wright Glider are taking a form less "bird-like" in character.

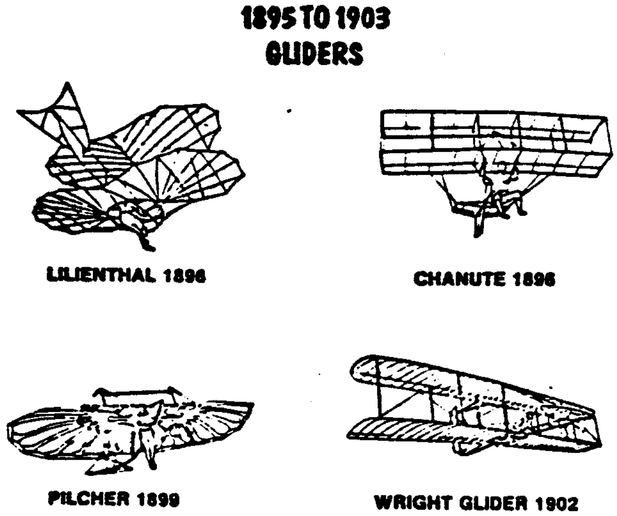


Figure 3. Early Wing Concepts - Gliders

From this sketch of the Wright Flyer, two prominent and important geometric properties of wings can be seen; their span and area. Later in this paper, the influences of wing span and wing area will be emphasized.

1903
WRIGHT FLYER

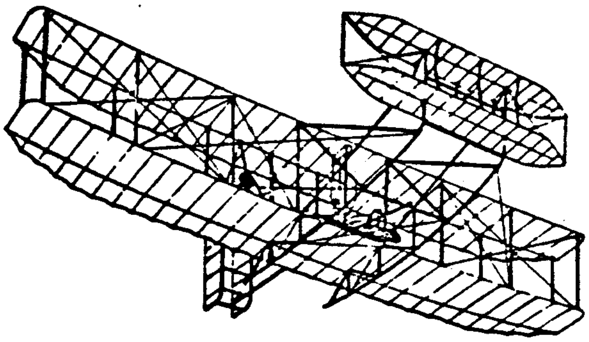


Figure 4. Wings of the Wright Flyer

Since the time of the Wright Brothers and their successful flying machines, the evolution of wings has brought forth a fantastic variety of wing shapes.

An historical review of wings will produce a long list of wing concepts and descriptors. Shown here, is a sample. They are arranged in alphabetical order, for lack of a better categorization.

WING CONCEPTS/DESCRIPTORS

ARROW	DIAMOND	MONOPLANE	SWEEP BACK
ARTICULATED	DOUBLE DELTA	MUTABLE	SWEEP FORWARD
AUGMENTER	ECLECTIC	OBLIQUE	SWING
BENT	ELLIPTICAL	OGEE	TAPERED
BURNELLI	END PLATED	POLYMORPHIC	TILT
BI PLANE	FIXED	RECTANGULAR	TRAPEZOIDAL
BLOWN	FLOATING	RING	TRIPLANE
BOX	FLYING	ROGALLO	TWISTED
CANTILEVER	GULL	ROTARY	UNDULATING
CASCADE	HONEYCOMB	SAUCER	VARIABLE AREA
CHANNEL	INFLATABLE	SHEARED	VARIABLE CAMBER
CIRCULAR	INVERSE TAPERED	SKEWED	VARIABLE CHORD
COMPOSITE	LAMINAR	STAGGER	VARIABLE INCIDENCE
CONICAL	LEX	STRAIGHT	VARIABLE SPAN
CRANKED	"M"	SUPERCRITICAL	VARIABLE SWEEP
DELTA	MANTA	SWEPT	"W"
			"X"
			"Y"

Figure 5. Multitude of Wings

But from this ordering, it can be seen that wing concepts have literally varied from A to Z. Only those starting with J, K, Q & Z are missing. You are challenged to fill in the blanks and to make additions to this list.

The infinity of geometric permutations possible in wing planforms are illustrated in the following four figures. Four categories are defined: straight, swept, delta/arrow and variable.

'STRAIGHT' WING PERMUTATIONS

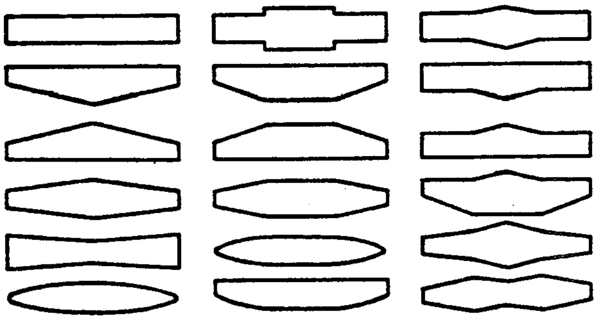


Figure 6. Straight Wing Permutations

Seemingly simple planar wings have been bent, cranked, twisted and warped into the shapes necessary to achieve specific objectives.

SWEPT WING PERMUTATIONS

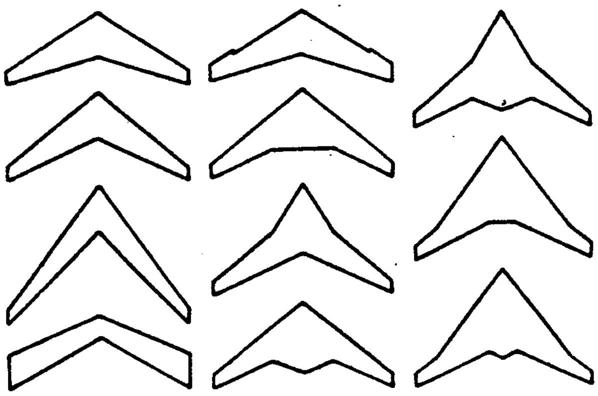


Figure 7. Swept Wing Permutations

The broad range of permutations in evidence have been driven by the integration of structural, aerodynamic, balance, fabrication, economic and functional considerations.

5.15

Delta, arrow, and double-delta permutations with various clipped-tip, trailing edge and leading edge treatments have been used.

DELTA-ARROW WING PERMUTATIONS

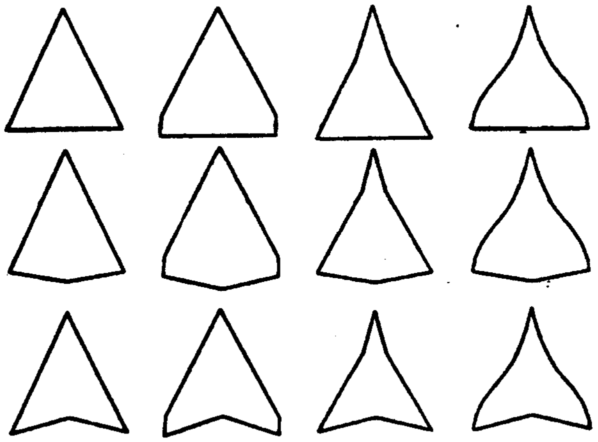


Figure 8. Delta/Arrow Wing Permutations

Variable wing geometry schemes that change sweep, span, incidence, area, camber and thickness ratio inflight, to suit special demands, have been developed.

VARIABLE WINGS

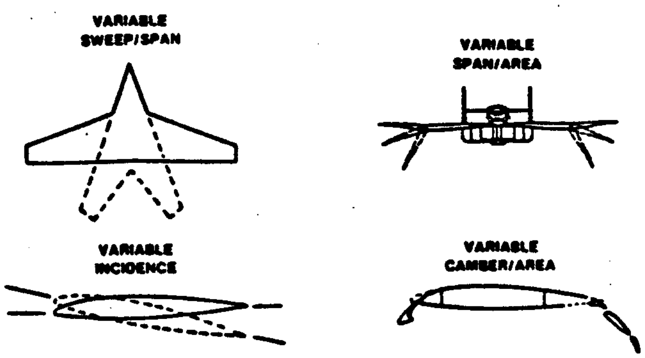


Figure 9. Variable Geometry Wings

Amazing as it may seem, all of the wing planforms illustrated have found their way to application on various aircraft. For some one of a kind varieties, the wing geometry may simply represent the artistic flair of the designer. But in the main, all have good reason for their selection. <u>Wings are unique</u>, they evolve to the configuration necessary to best meet the needs of a particular application. Some aircraft that exemplify variety and uniqueness of wing design will now pass in review.

Wings for subsonic long range cruise at high altitudes are characterized by their long wing spans and lots of area.

SUBSONIC/HIGH ALTITUDE/LONG RANGE

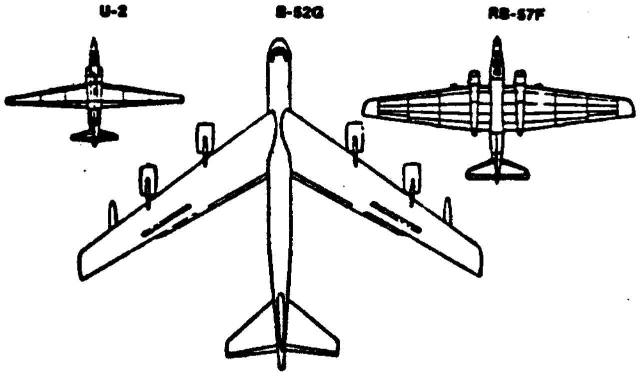


Figure 10. Subsonic Cruise Wings

In contrast, wings for supersonic cruise are more highly swept, retain large area, but distribute that area more chordwise than spanwise.

SUPERSONIC/HIGH ALTITUDE/LONG RANGE

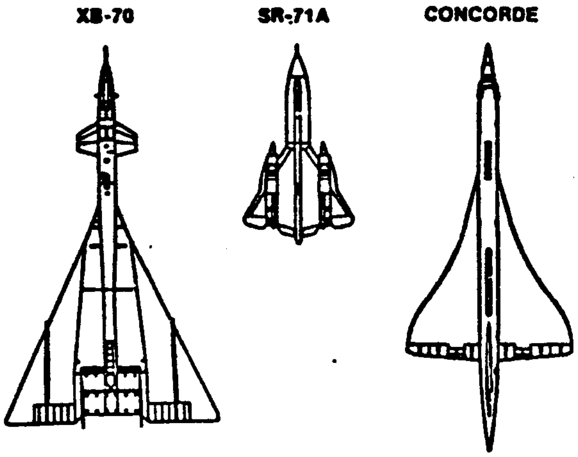


Figure 11. Supersonic Cruise Wings

Variable sweep wings are driven by the multi-modes of operation demanded in the specifications for these aircraft.

SWING WINGS

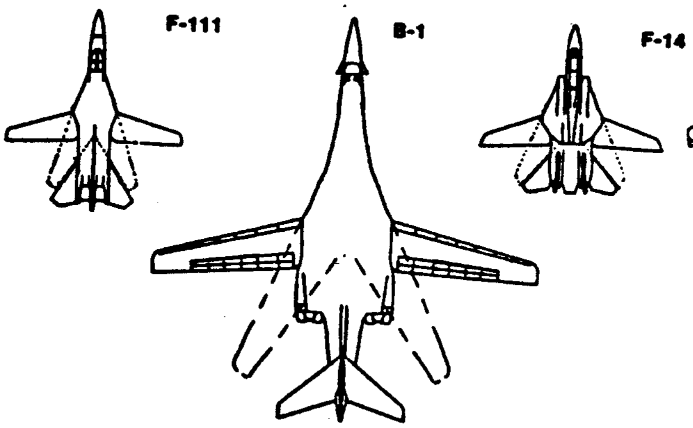


Figure 12. Variable Sweep Wings

Fighter wings must satisfy the demands for high maneuverability and fast transient response throughout the full spectrum of their mach-altitude flight envelope. Fighter wings present a significant challenge in their design.

FIGHTER WINGS

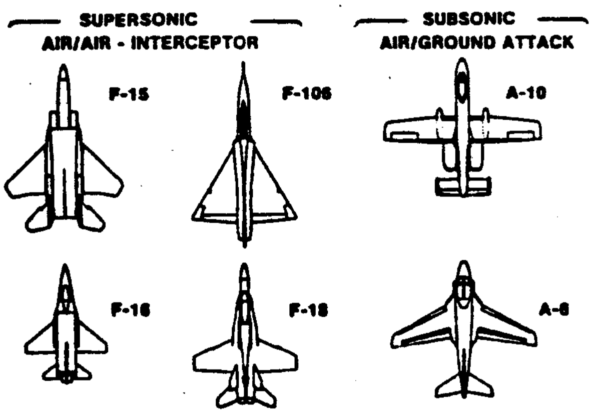


Figure 13. High Maneuver Wings, Supersonic/Subsonic

Transport wings are carefully tailored for cost-effective cruise considerations and fulfillment of take-off and landing field length requirements.

TRANSPORT WINGS

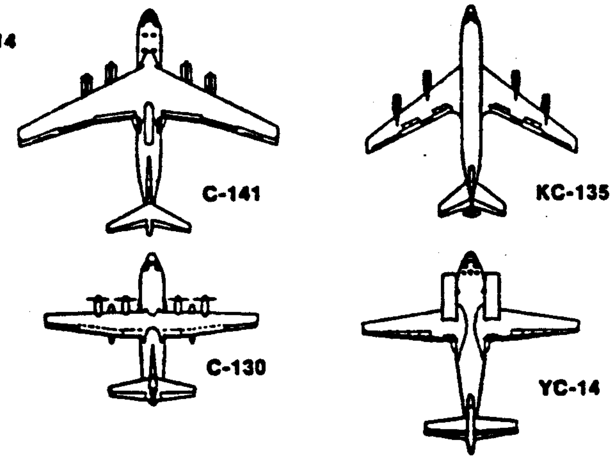


Figure 14. Cost-Effective Cruise Wings

TRANSPORT WINGS

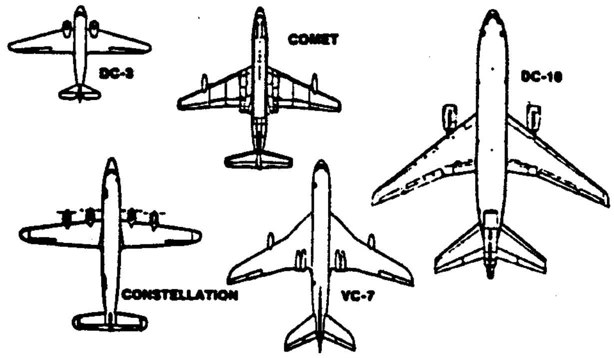


Figure 15. Cost-Effective Cruise Wings (Contd)

Examples of some unique wing planforms are shown here. The Vought "Flying Pancake" and "Cutlass" tended toward flying wings. The Republic "Thunderceptor" had an inverse tapered swept wing with variable incidence. It is apparent how the Douglas "Skyray" got its name. The Avro car was a true "Flying Saucer" configuration. The Ryan "VZ-11" was a fan-in-wing VTOL concept. The X-24 is usually referred to as a lifting body, but it might well be coined a "BWING" - short for a body wing.

UNIQUE WINGS

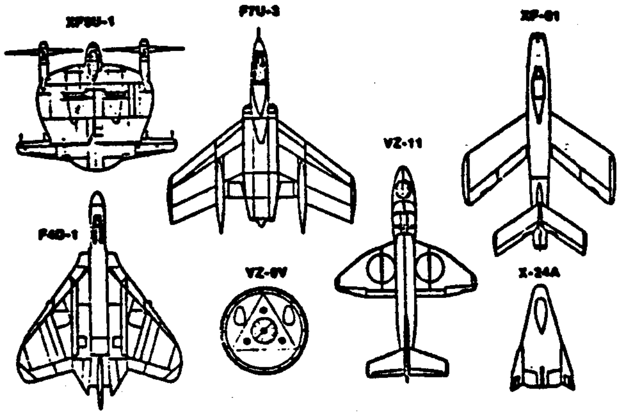


Figure 16. Unique Wing Solutions

UNIQUE WINGS

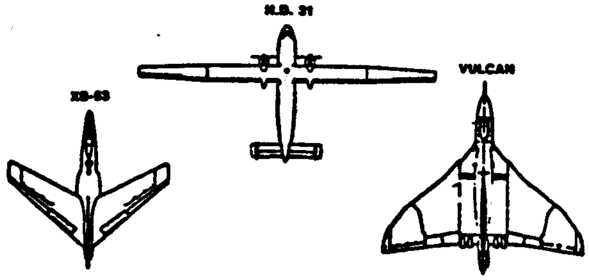


Figure 17. Unique Wing Solutions (Contd)

The Convair XB-53 only made it up to serious consideration for prototyping, but it was planned to have a 30° swept forward wing with variable incidence tips. The French Hurel-Dubois (H.D. 31) is impressive, because it had an aspect ratio of 20.2. The Vulcan Bomber represents a highly successful "Flying Wing" type with compound leading edge sweep break points.

Treading the light fantastic, here is a real beauty. It was coined the mutable wing, not to be confused with mutilated. This polymorphic wing concept portrays the extremes of variable geometry than can be forced - when designers attempt to satisfy difficult and conflicting performance demands in a single aircraft.

MUTABLE WING

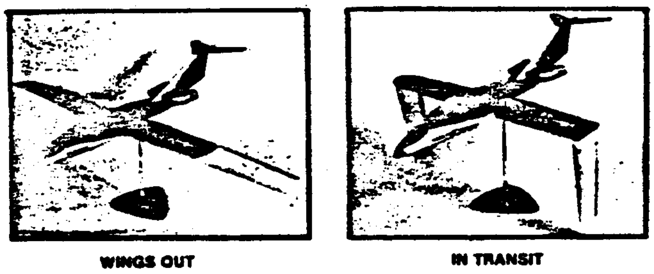


Figure 18. Polymorphic Wings

MUTABLE WING

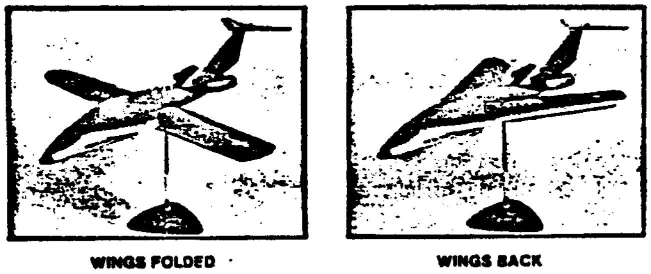


Figure 19. Polymorphic Wings (Contd)

The design was driven by a diverse set of mission performance objectives; short take-off and landing distance, long range and endurance, the ability to fly at high altitudes and at low level with a high speed dash capability. Attempting to fulfill all the objectives, it was necessary that the wing take on a form that is optimal for each phase of flight. This concept varies span, area, sweep and a few other parameters. Needless to say, this one didn't make it, because of the obvious complexity. But, neither did the design requirement get fulfilled. When unusual wings are presented, you can suspect they were driven by a set of performance specifications that could not be fulfilled by straightforward means.

Thus far, wing planforms have been emphasized. Focusing attention to other wing views, the next two figures show plan and frontal views of the wing tip region. A wide variety of wing tip closures are in evidence.

WING TIP SHAPES - PLAN VIEW

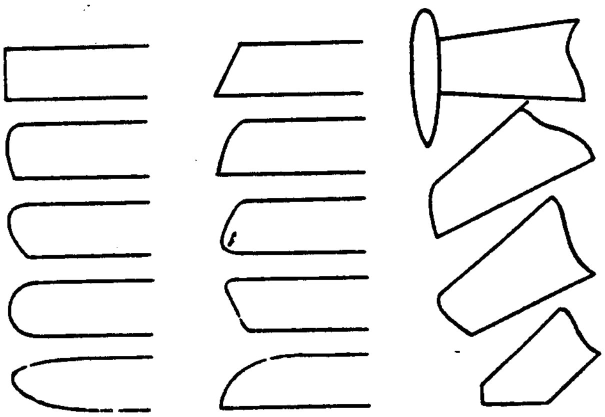


Figure 20. Wing Tip Shapes

WING TIP CLOSURES - FRONT VIEW

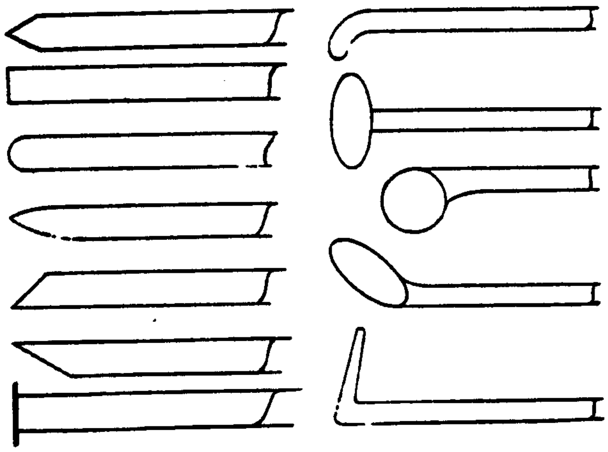


Figure 21. Wing Tip Closures

"Overall" front views of wings also present an interesting display of variations. There are high, low, mid, and pylon mounted wings. Wing - fuselage tie-ins were especially prolific in the bi-plane era. Wing cross-sections, i.e. airfoils are main subjects in themselves. They will be addressed in another paper by Mr. Abbott.

WINGS — OVERALL FRONT VIEWS

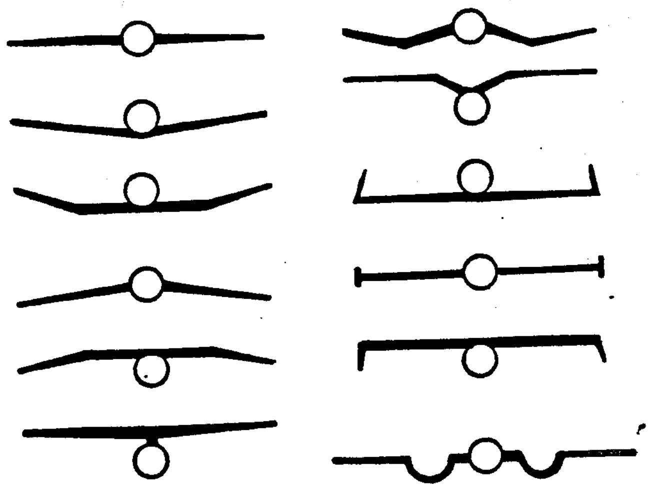


Figure 22. Monoplane Wing/Fuselage Interfaces

WINGS-OVERALL FRONT VIEWS

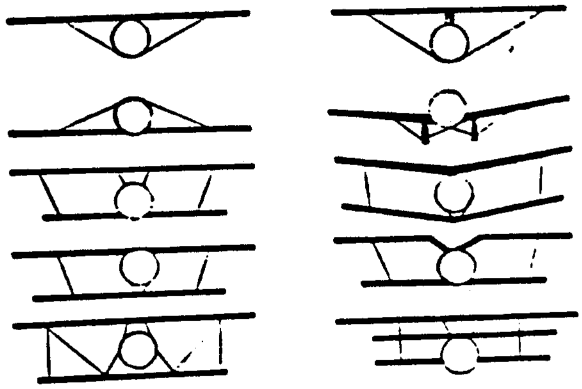


Figure 23. Multi Element Wing/Fuselage Tie-Ins

The variety of wing shapes are endless. Any attempt at completeness of presentation is a formidable task. The aim here has been to provide a representative overview.

Moving on to other aspects, we will now take a look at the power that wings have in producing lift.

WING LEVERAGE

Wings possess strong leverage. As everybody knows, wings provide lift much greater than their own weight and drag, but how much? These two figures give an idea. The data are representative of modern fighter and transport aircraft.

WING LIFT TO WEIGHT RATIO

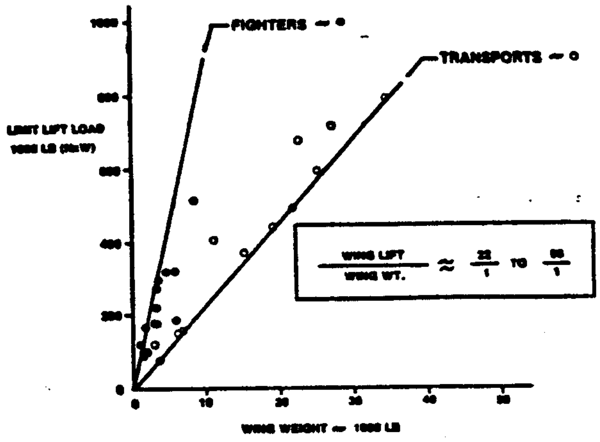


Figure 24. Lift to Weight Leverage

WING LIFT TO DRAG RATIO - SUBSONIC

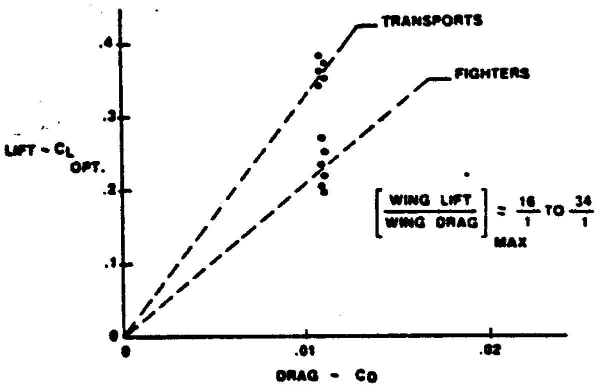


Figure 25. Lift To Drag Leverage

Observe the differences. Wings on fighters are allowed to lift about 90 times their own weight, but on transports the allowable lift to weight ratio is approximately 22 times. Still, 22 times their own weight is not bad! On the other hand, the maximum lift to drag ratio of transport wings are higher than those for fighters. The key wing parameters that contribute to this difference between fighters and transports are wing span and wing area, given the differences in design limit load factor. Fighters must generate high lift to maneuver. High maneuverability is not demanded in a transport, but cruise efficiency, which is a direct function of lift to drag ratio, is vital.

Looking further at this difference, it can be seen that maximum lift to drag is a strong function of wing aspect ratio.

LIFT TO DRAG RATIO SUBSONIC

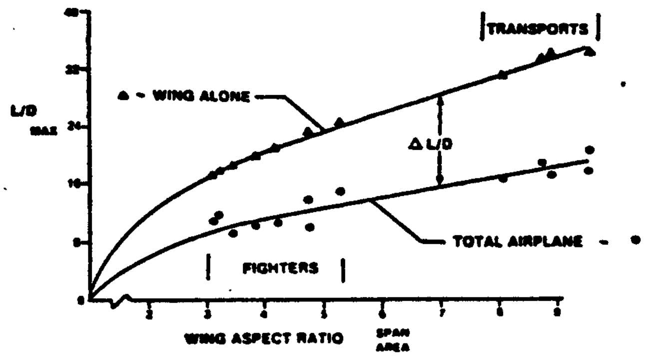


Figure 26. Statistical L/D Correlations

Aspect ratio is merely the ratio of $(\text{Span})^2$ to wing area. This correlation shows the strong influence aspect ratio has on (L/D) max. Since the wing is the primary contributor to lift, the lift to drag ratio is shown for the wing alone and for the total airplane, which includes the fuselage, empenage and propulsion elements. Note that the wing alone has a significantly higher L/D than the total airplane. From this, one can conclude that "The Wing Is The Thing", the other components of the aircraft are a drag. This fact also brings out the reason for the consideration of flying wings. The wing alone; unburdened by the drag of other airplane components can produce a machine with a high degree of aerodynamic efficiency. Recognizing that aspect ratio is a powerful wing parameter affecting airplane L/D, it would seem that a simple means to increase L/D would merely be to design wings with higher and higher aspect ratio. This is precisely true, however a conflict arises that prevents this. Wing span and wing area are also strong drivers on wing weight. The following two figures were derived from correlations of wing weight for fighters and transports.

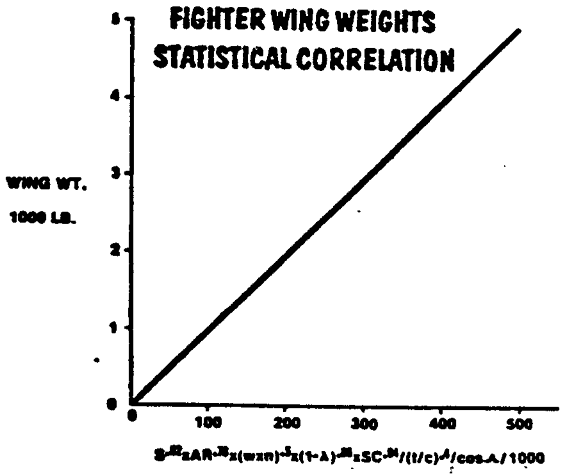


Figure 27. Wing Wt. Correlation Parameters

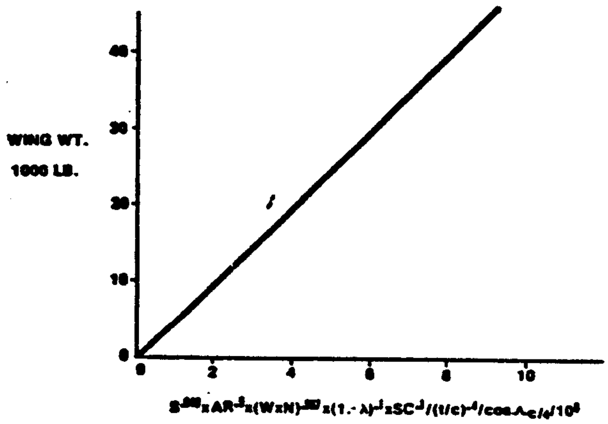


Figure 28. Wing Wt. Correlation Parameters

From these two charts a "first approximation" of the wing weight can be derived; given the characteristics of the wing, i.e. its area, aspect ratio, the gross wt. of the airplane, load factor, taper ratio, surface control area, thickness ratio, and sweep. The accuracy of the correlation lines are within ± 5% for the sample of aircraft considered in the correlation. Focus your attention on the first two parameters, wing area and wing aspect ratio. The higher values of the exponents on these two parameters (relative to all others) indicates that they have the strongest influence on weight. Thus, wing weight increases with increasing aspect ratio and area, which is opposite to that for L/D. Since the technological objective is to design the wing to produce the necessary lift with the least amount of weight and drag, the design engineer is caught in a dilemma. Trade-offs between the structural and aerodynamic interfaces are made to find the most suitable compromise. To illustrate the nature of these trade-offs, we will now turn attention to the third aspect of this brief, wing design.

5 R

WING DESIGN

Wing design is a complex problem. Evolving a wing for a particular airplane consumes a great deal of time in research, analysis and test. Some of the design parameters for a wing are listed here. Each one has its own unique impact and contribution and all must be decided upon.

The effect of each parameter would be a major story in itself.

MAJOR WING DESIGN PARAMETERS

➧ AREA	~	S_W
➧ ASPECT RATIO	~	AR
SWEEP	~	Λ
THICKNESS	~	t/c
TAPER	~	λ
TWIST	~	θ
INCIDENCE	~	i
DIHEDRAL	~	Γ
AIRFOIL		

Figure 29. Wing Parameter Nomenclature

Continuing the focus of attention on wing area and aspect ratio, the results of some design studies on transports will be shown to illustrate the effect these wing features have on airplane aerodynamic, structural and performance measures. This chart shows a matrix of wing options for a particular transport design problem.

TRANSPORT WING MATRIX

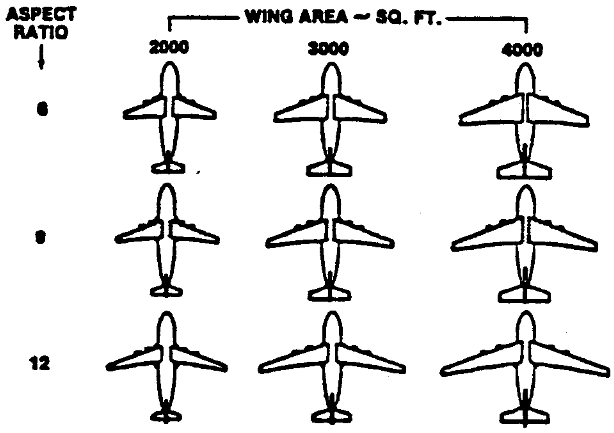


Figure 30. Transport Wing Alternatives

All of the aircraft have been designed to transport a certain payload over a specified distance. Thus, every airplane shown here will fulfill the specific job intended. All aircraft have the same engines. The wings differ only in area and aspect ratio. Areas are 2000, 3000, and 4000 sq. ft. and aspect ratios of 6, 9, and 12 are depicted.

The problem posed is, which wing should be selected?

The implications of each wing alternative are shown in the following figures. Area and aspect ratio effects on wing weight and airplane lift to drag ratio are shown first.

WING WEIGHT

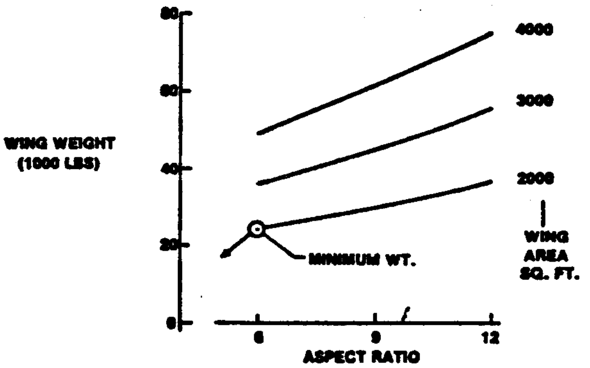


Figure 31. Effect of Aspect Ratio & Area On Wing Weight

AIRPLANE LIFT TO DRAG RATIO

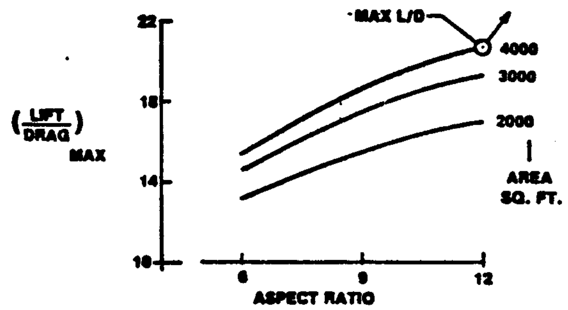


Figure 32. Effect of Aspect Ratio & Area On Max. L/D

Note the contrariness, minimum wing weight occurs at the smallest area and lowest aspect ratio, but maximum L/D is favored by the largest wing and highest aspect ratio.

The airplane gross weight to do the job intended minimizes at an aspect ratio of 9.5, with the smallest area.

AIRPLANE GROSS WEIGHT

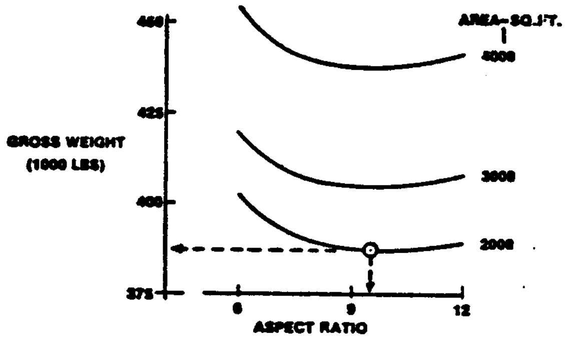


Figure 33. Effect of Aspect Ratio & Area On Gross Weight

Looking at empty weight, it minimizes at the lowest aspect ratio and wing area and is a key consideration, since lowest acquisition cost is likely to occur at the lowest empty weight.

AIRPLANE EMPTY WEIGHT

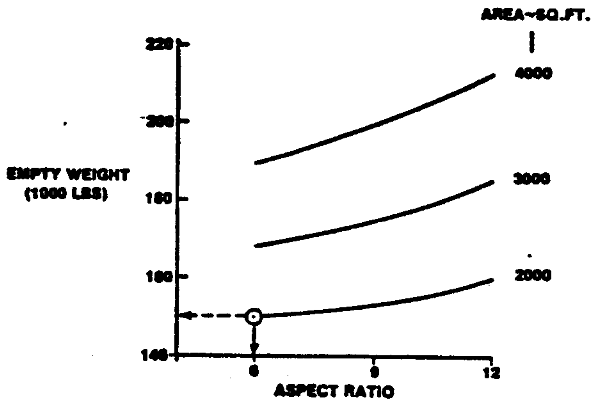


Figure 34. Effect of Aspect Ratio & Area On Empty Weight

Given the strong considerations of fuel conservation, lowest fuel consumption occurs at an intermediate wing area with the highest aspect ratio.

FUEL REQUIRED

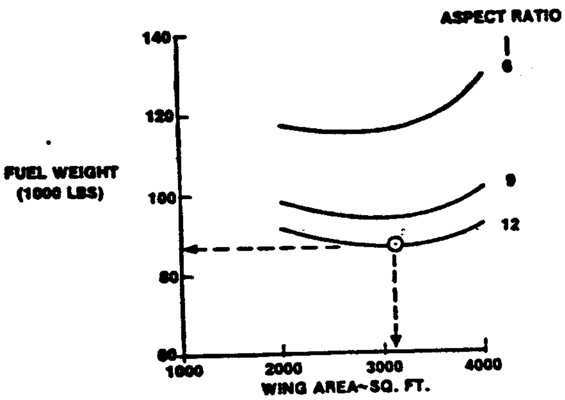


Figure 35. Effect of Aspect Ratio & Area On Fuel Required

The payload-range capabilities, as influenced by the extremes of various wing alternatives are shown. Note the differences in off-design capability, even though they all satisfy the same single design point.

OFF-DESIGN CAPABILITIES

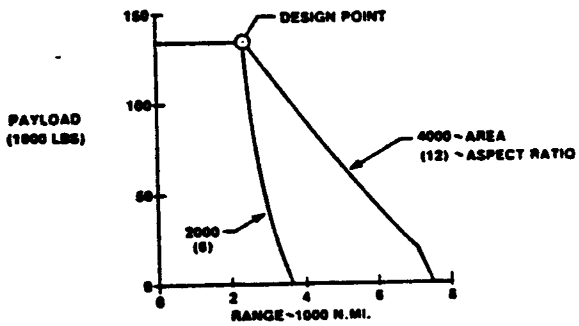


Figure 36. Effect of Wing Alternatives on Payload-Range

The ability to tailor the wing (in area and aspect ratio) to meet overall needs is evident. Additional airplane performance measures affected by the wing, include take-off and landing distance, rate of climb, and cruise altitude.

These figures show the strong influence of wing area on take-off and landing distance.

On these performance measures, the effect of aspect ratio is small.

TAKE-OFF DISTANCE

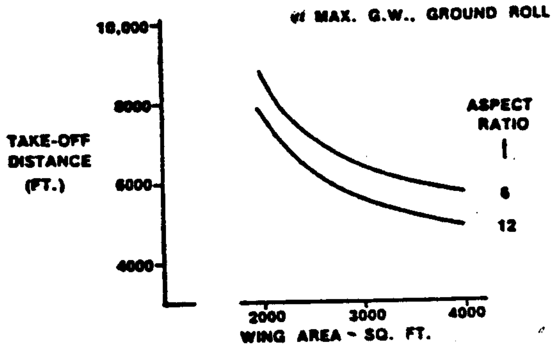


Figure 37. Effect of Area on Take-Off Distance

LANDING DISTANCE

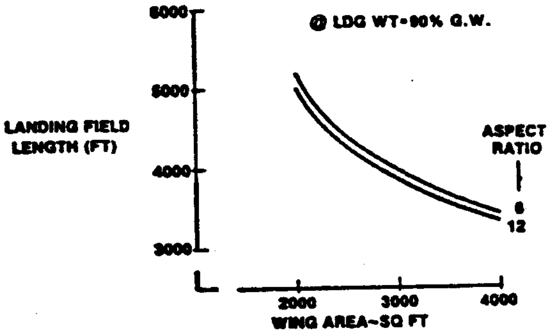


Figure 38. Effect of Area on Landing Distance

But if one examined the airplanes' rate of climb with one of its four engines in-operative, the need for a higher aspect ratio is clear.

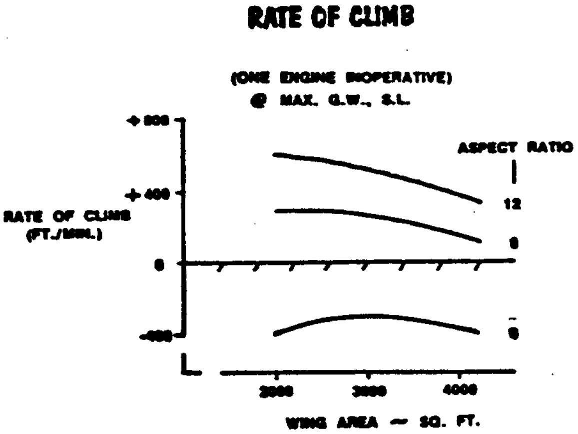


Figure 39. Effect of Area & Aspect Ratio on Rate of Climb

Many other considerations (affected by the wing) could be identified, but from that which has already been shown, it can be seen that the wing has both powerful and pervasive influences.

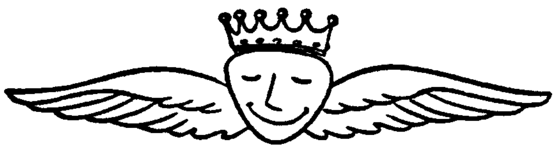
Which wing do you choose? The answer, of course, is — it all depends on what we want the airplane to do!

CLOSING REMARKS

Since wings produce the lift necessary to sustain flight, they are obviously the primary focus of attention by the designers of aircraft. The wings on all aircraft are uniquely tailored to fulfill the demands of the particular application intended. Because of the diversity of applications and the continued drive for improvements in some portion of the operational flight envelope, the wings for future aircraft will evolve to the configuration necessary to satisfy its own special needs. There are no universal "best" wings.

Broad scope trade-off studies and tests will continue to be conducted, systematically varying the key design parameters to evolve the proper values, that eventually must be fixed. For any aircraft ever built, the synthesis, analysis, iterations, and tests performed in support of wing definition; undoubtedly holds a dramatic story that tells why it is like it is. The papers to follow will provide further illumination and insight. Although there are many contributing elements that go into the make-up of an aircraft, for today:

THE WING IS KING



ACKNOWLEDGEMENTS

Contributions made by members of the Design Analysis Directorate (ASD/XRH) supporting preparation of charts for this brief are sincerely appreciated. Special recognition is made to Messrs. D. Breidenbach, E. Brown, J. Frozina, J. Snyder, G. Taylor, and Secretary Mrs. B. Tyler.

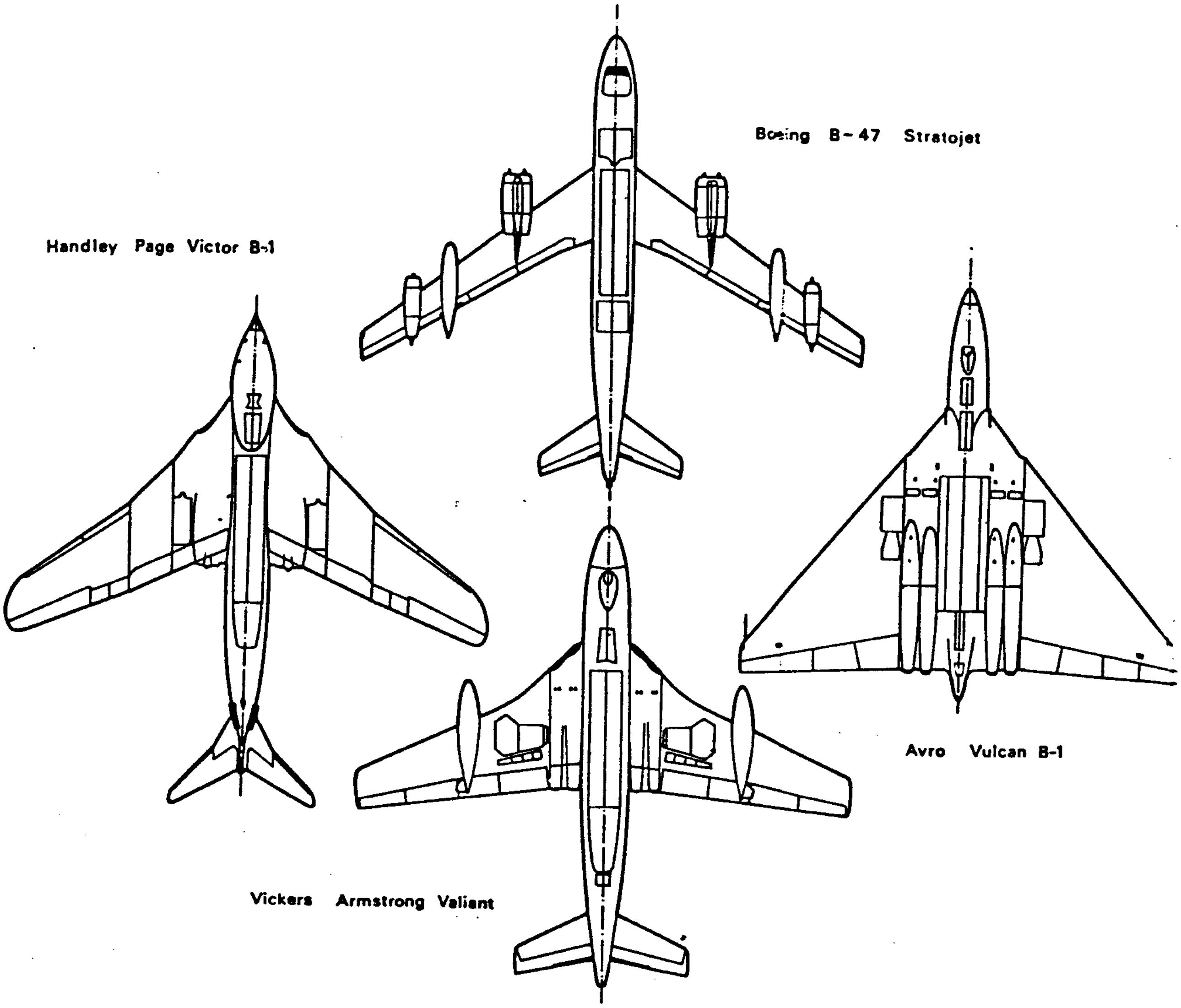


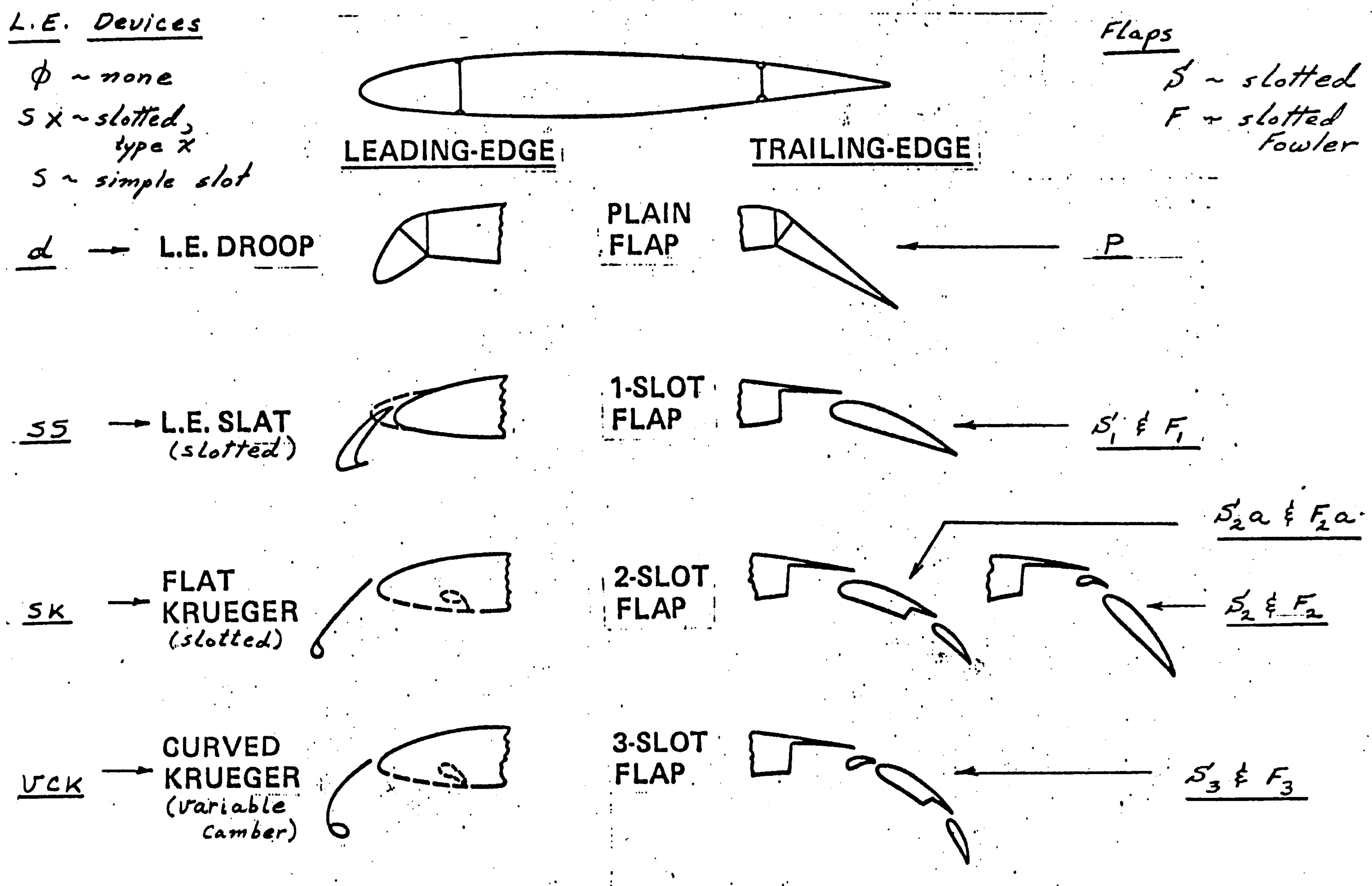
Fig. 2-12. Different configurations for aircraft designed to similar specifications

	BOEING B-47	AVRO VULCAN
GROSS WING AREA ~ ft^2 (m^2)	1430 (133)	3446 (320)
TOTAL WETTED AREA ~ ft^2 (m^2)	11300 (1050)	9500 (885)
SPAN ~ ft (m)	116 (35.4)	99 (30.2)
MAX. WING LOADING ~ lb/ft^2 (kg/m^2)	140 (690)	43.5 (212)
MAX. SPAN LOADING ~ lb/ft (kg/m)	1750 (2590)	1520 (2250)
ASPECT RATIO	9.43	2.84
C_{D_o} (ESTIMATED)	.0198	.0069
$1/\pi Ae$ (e–OSWALD FACTOR)	.0425 (.8)	[illegible] (.8)
L/D_{max} ; $C_{L_{opt}}$	17.25 ; .882	[illegible] ; .235

Fig. 2-13. Similarity in max. lift/drag ratios for two widely different configurations

NOTE A

MECHANICAL HIGH LIFT DEVICES



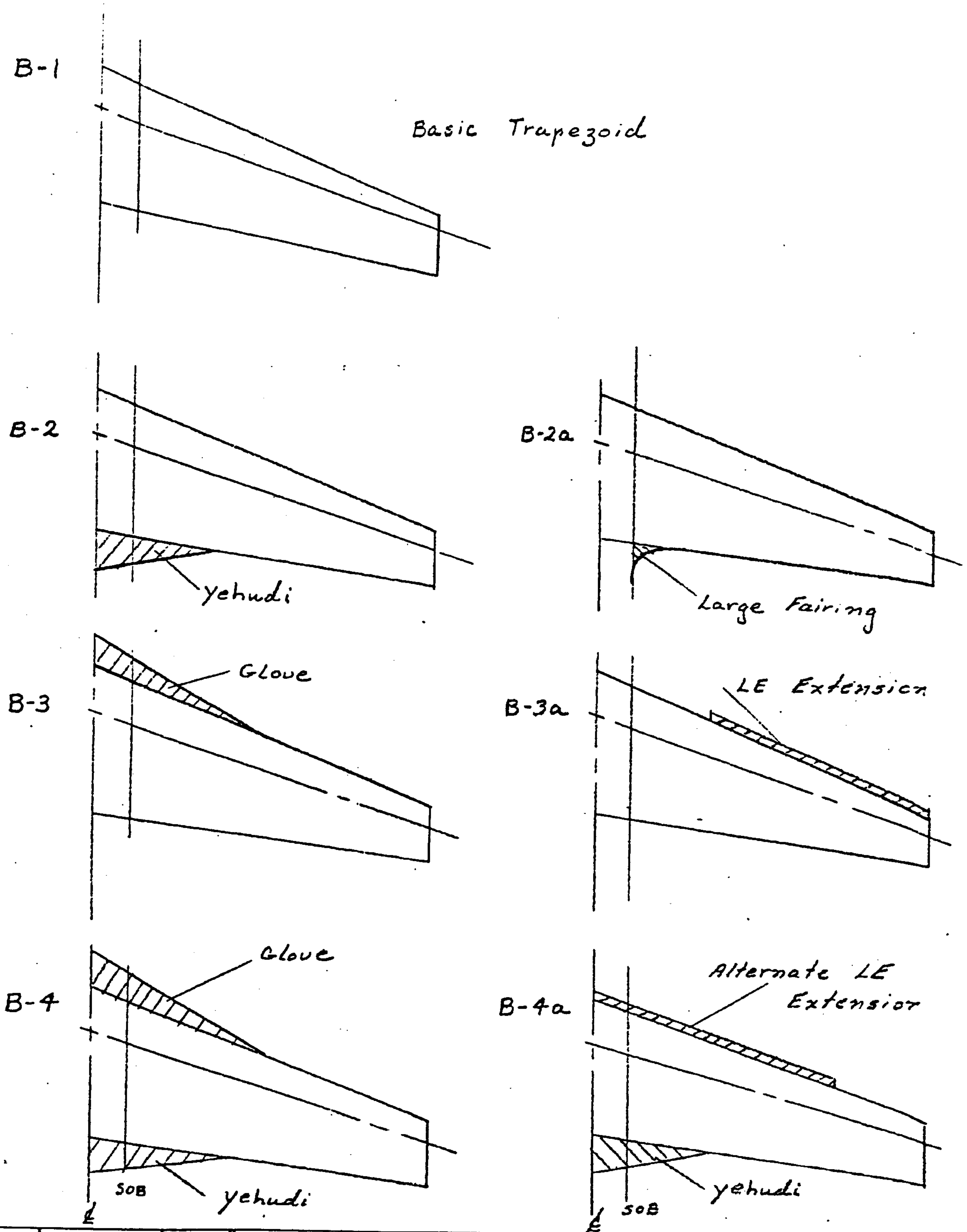
	Type	Model	First Flt.	$\Lambda_{c/4}$	S_{re}	$\Delta b_f/b$	Horiz Stab Loc.	Loadings W/S_{ref} (lb/ft²) Max TO	Max Land.	T/W_{max}
1	Caravelle	6.10	2/55	20°	15	0.57	High	64.7	66.5	0.2[illegible]9
2	BAC 111	200/400	8/63	20°	9	0.62	T	81.7	71.4	0.263
2a	BAC 111	500	/69	20°	103	0.58	T	105.3	86.7	0.240
3	VFW 614	—	6/71	15°	66	0.56	Low	63.9	63.9	0.330
4	DC-9	-10	/65	24°	93	0.56	T	97.1	87.5	0.309
5	DC-9	-30	/66	24.5°	100	0.53	T	108.9	99.8	0.268
6	DC-8	-50	12/60	30.5°	27	0.47	Low	114.4	72.8	0.222
7	DC-8	-63	4/67	30.5°	28	0.45	Low	123.2	85.0	0.214
8	HS 125	400	8/62	20°	3	0.50	High	70.8	62.3	0.300
9	Fok. F.28	2000	4/71	16°	82	0.55	T	77.0	71.9	0.303
10	VC 10	Super	5/64	32.5°	293	0.53	T	117.3	83.0	0.269
11	B 707	120B	/57	35°	243	0.51	Low	105.4	77.6	0.279
12	B 720	B	11/59	35°	24	0.51	Low	93.2	69.7	0.306
13	Falcon	20F	5/63	30°	44	0.48	High	66.0	43.7	0.300
14	Sabreliner	75A	6/71 (58)	28.5°	34	0.37	High	67.3	64.3	0.391
15	Trident	3B	12/69	35°	149	0.57	T	102.7	88.0	0.273
15a	Trident	1C	1/62	35°	133	0.62	T	88.3	77.0	0.267
16	C-5	A	6/68	25°	62	0.62	T	130.5	104.2	0.206
17	L-1011	-1	11/70	35°	34	0.66	Low	117.8	98.0	0.2[illegible]3
18	DC-10	-10	8/70	35°	35	0.64	Low	117.4	97.0	0.272
19	DC-10	-30	/71	35°	36	0.60	Low	147.0	104.9	0.260
20	B 707	-320B	1/62	35°	28	0.60	Low	106.9	68.3	0.214
21	C-141	A	12/63	25°	32	0.58	T	102.0	81.6	0.265
22	A 300	B2/4	10/72	28°	28	0.72	Low	113.2	103.4	0.322
23	Falcon	10	12/70	27.8°	2	0.59	High	75.6	71.1	0.345
24	B 727	-200	3/63	32.5°	15	0.64	T	127.1	96.8	0.243
25	B737	-100	4/67	25°	9	0.62	Low	93.4	86.4	0.266
25a	B737	-200 adv.		25°	9	0.62	Low	106.5	94.9	0.264
26	B747	-100	2/69	37.5°	550	0.59	Low	126.1	100.1	0.238
27	B747	SP		37.5°	55	0.59	Low	116.3	79.3	0.284
28	TU 154	—	10/68	35°	19	0.64	T	95.0	84.5	0.318

D] [Note E]

Calc.	JHMc	9/78	REVISED	DATE	Summa… Configu… Transpo…
Trac					
Chk					
Appr.					TH…
Appr.					

D1 4100 2860 ORIG.3/71

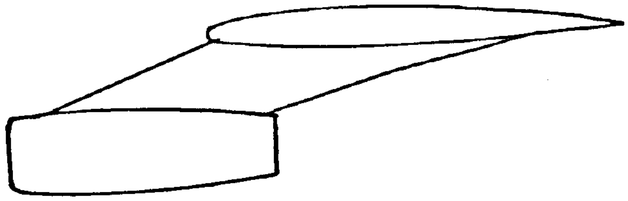
NOTE B PLANFORM TYPES



NOTE C : ENGINE LOCATION

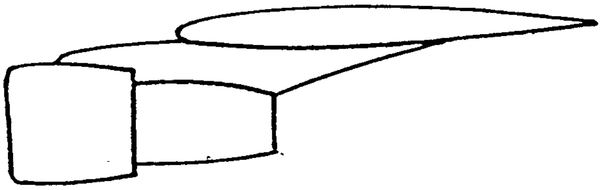
C-1 Wing Mount

C-1a



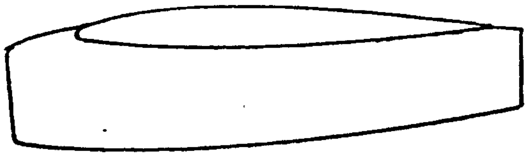
Long Strut

C-1b



Short strut

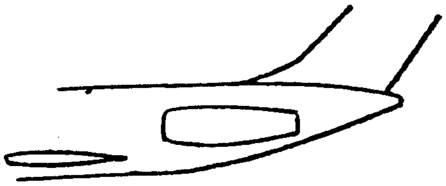
C-1c



Pod Underwing

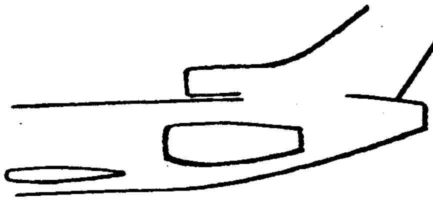
C-2 Aft Mount

C-2a



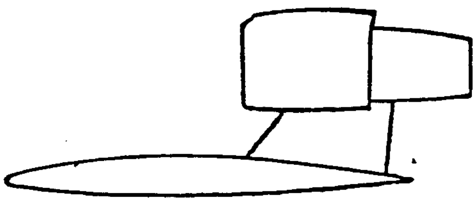
Twin Engine

C-2b



Tri-Engine

C-3



Overwing

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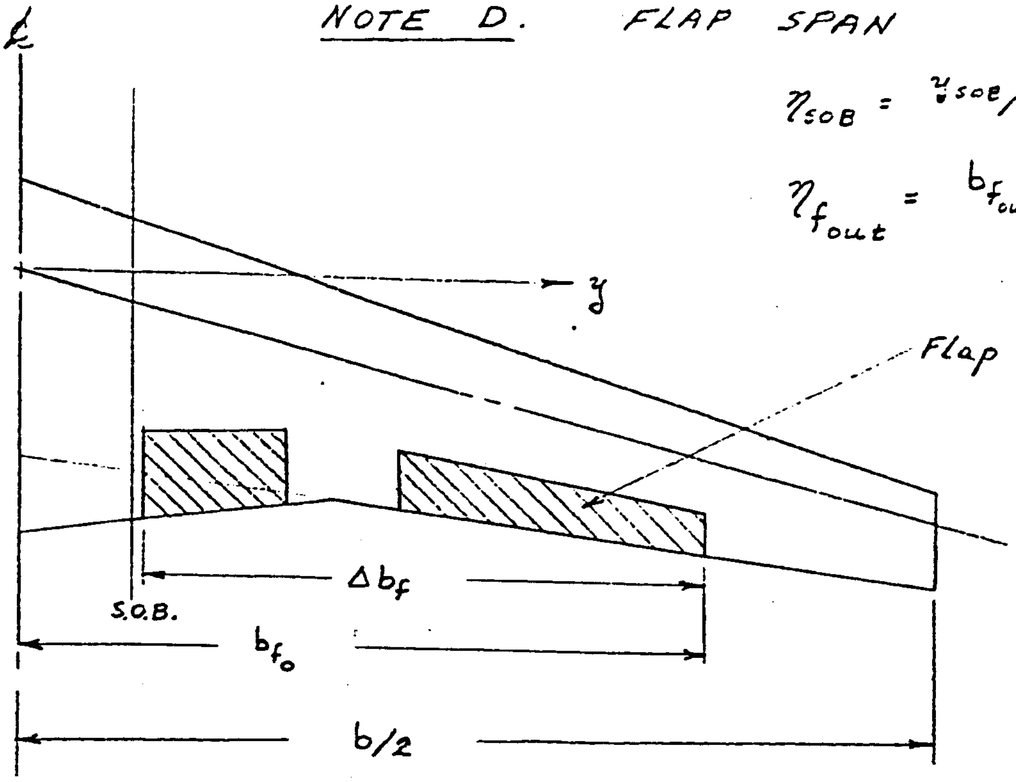
CALC						
CHECK						
APR						
APR					THE BOEING COMPANY	PAGE S.30

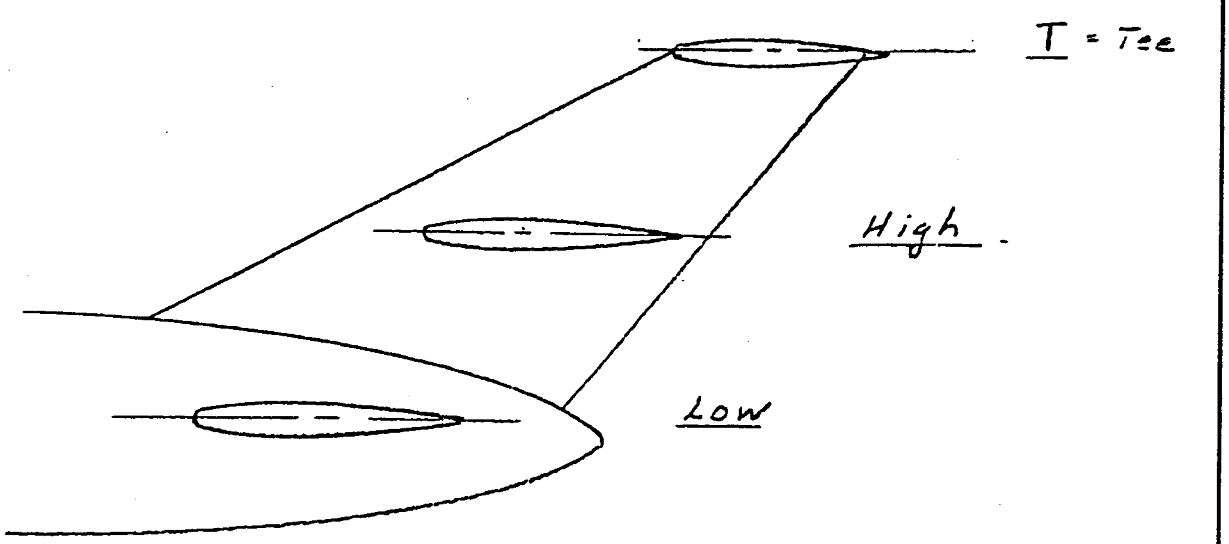
NOTE D. FLAP SPAN

$$\eta_{SOB} = y_{SOB} / \frac{b}{2}$$

$$\eta_{f_{out}} = b_{f_{out}} / \frac{b}{2}$$



NOTE E. HORIZONTAL STABILIZER POSITION



FUSELAGE DESIGN

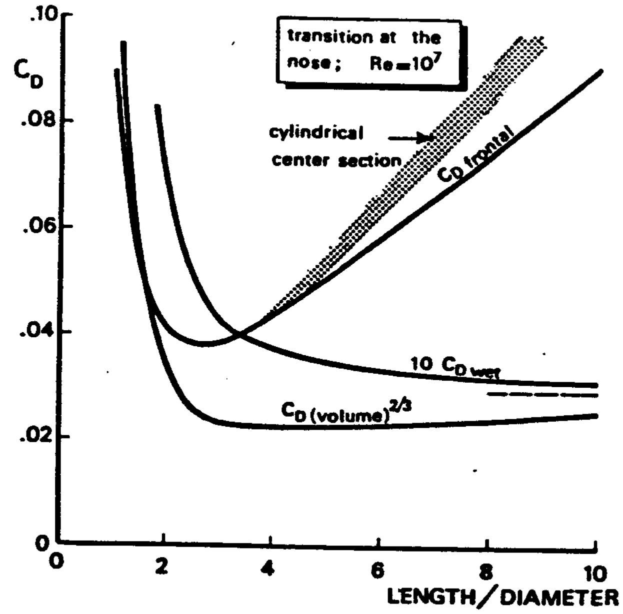


Fig. 3-2. Drag coefficient of streamline bodies of revolution at low speeds

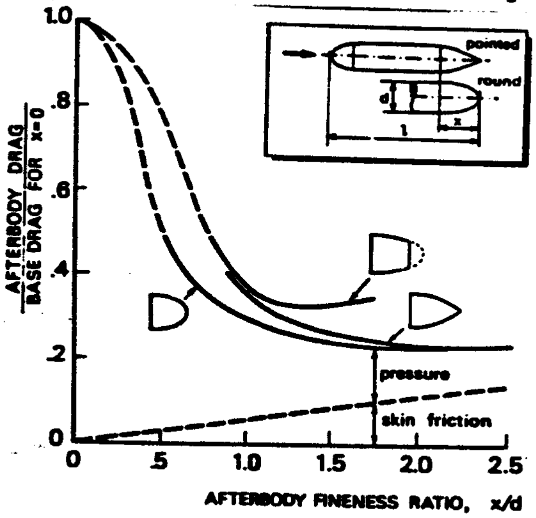


Fig. 3-4. Afterbody drag of a fuselage tail, when added to a cylindrical shape

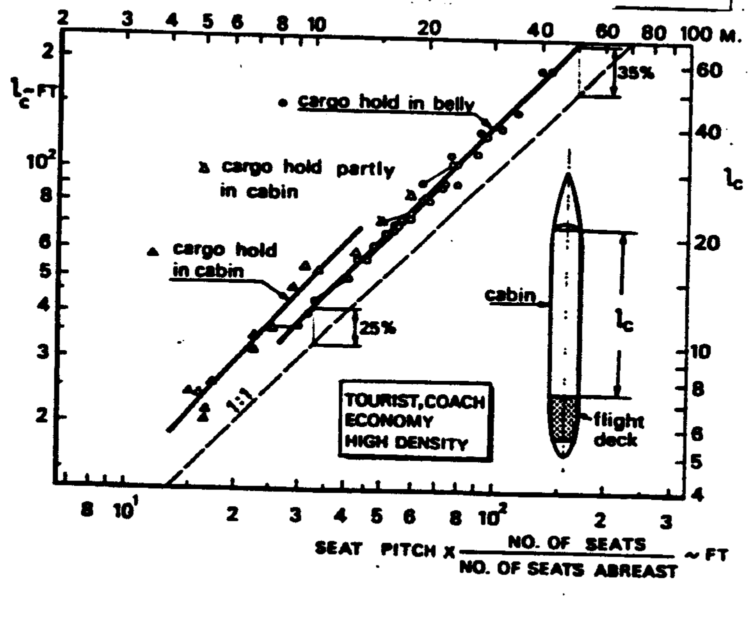
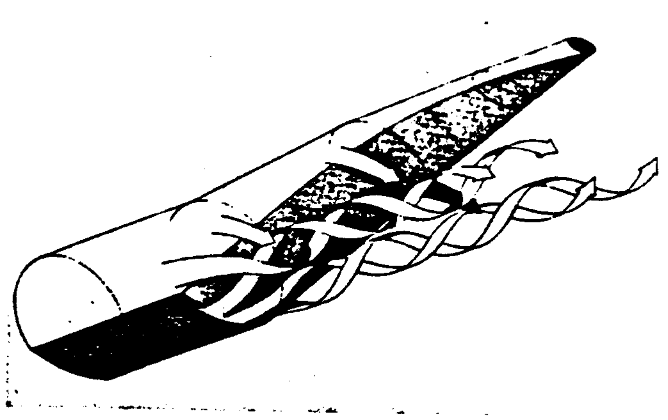
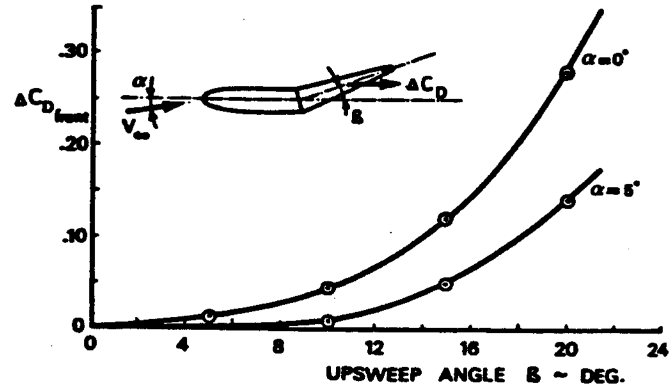


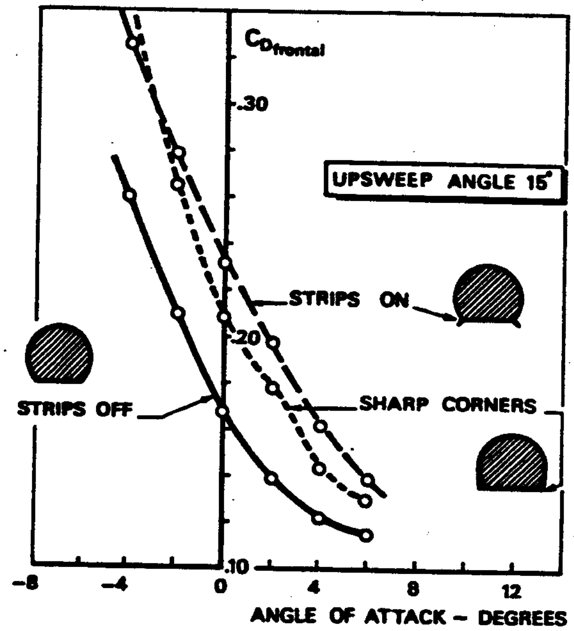
Fig. 3-13. Statistical correlation at the cabin length



a. Schematic drawing of flow separation and vortex shedding from a rear-loading fuselage (Ref.: NCR Aeron. Report LR-395)



b. Drag increment vs. upsweep angle (Ref. 3-26)



c. Effect of cross-sectional shape on drag (Ref. 3-27)

Fig. 3-27. Flow phenomena around cambered rear fuselages

5.33

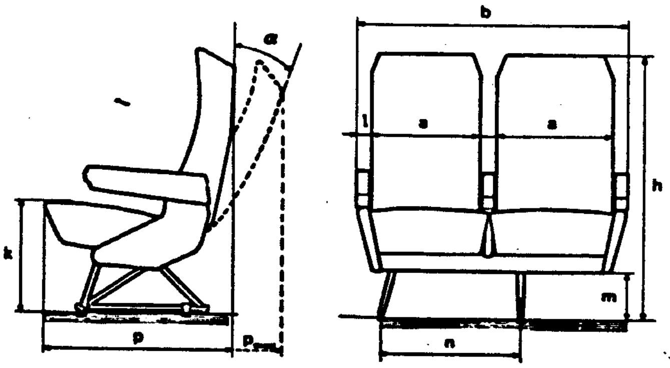


Fig. 3-15.Definitions of seat dimensions

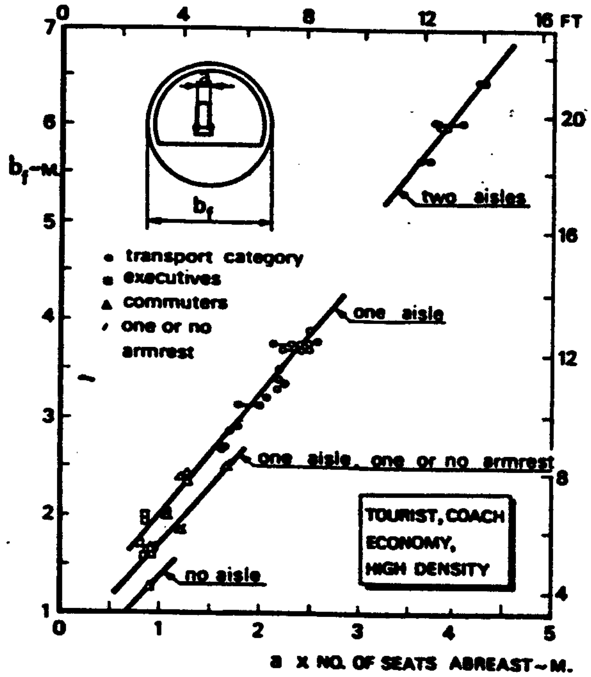


Fig. 3-12.Fuselage width vs. "total seat width"

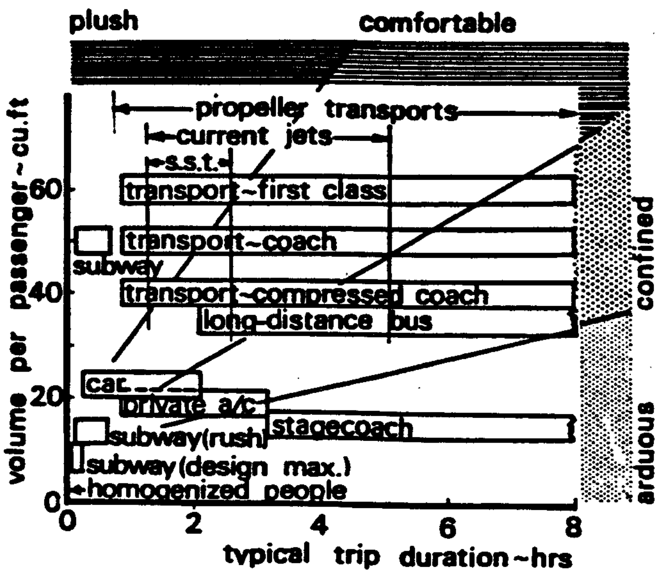


Fig. 3-6.Volume per passenger and trip duration (Ref.: The Architectural Review)

		SEAT CLASSIFICATION		
SYMBOL*	UNIT	DE LUXE	NORMAL	ECONOMY
a	inch	20(18½-21)	17(16½-17½)	16.5(16-17)
	cm	50(47-53)	43,5(42,5-45)	42(40,5-43,5)
b_2**	inch	47(46-48½)	40(39-41)	39(38-40)
	cm	120(117-123)	102(100-105)	99(47-102)
b_3**	inch	-	60(59-63)	57
	cm	-	152(150-160)	145
l	inch	2¾	2¼	2
	cm	7	5.5	5
h	inch	42(41-44)	42(41-44)	39(36-41)
	cm	107(104-112)	107(104-112)	99(92-104)
k	inch	17	17¾	17¾
	cm	43	45	45
m	inch	7¾	8¾	8¾
	cm	20	22	22
n	inch	usually 32 (24-34)		
	cm	usually 81 (61-86)		
p/p_{max}	inch	28/40	27/37½	26/35½
	cm	71/102	69/95	66/90
α/α_{max}	deg	15/45	15/38	15/38

*definitions in Fig. 3-15

**the index denotes the number of seats per block

NOTES

1. The data represent normal values and are not standard values. A statistical range is indicated in brackets.
2. In wide-body aircraft, seats are used in the tourist/coach class with a=19" (48 cm), b_3=66" (168 cm), h=43" (109 cm). In high-density arrangements the "normal" type seat is used.
3. In third-level aircraft it is customary to install seats with only one or no armrest, with typical dimensions: width 16½" (42 cm), h=35" (89 cm), p=26" (66 cm).

Table 3-1.Seat dimensions (Ref.: Seat manufacturers brochures, Flight Int., July 8, 1965)

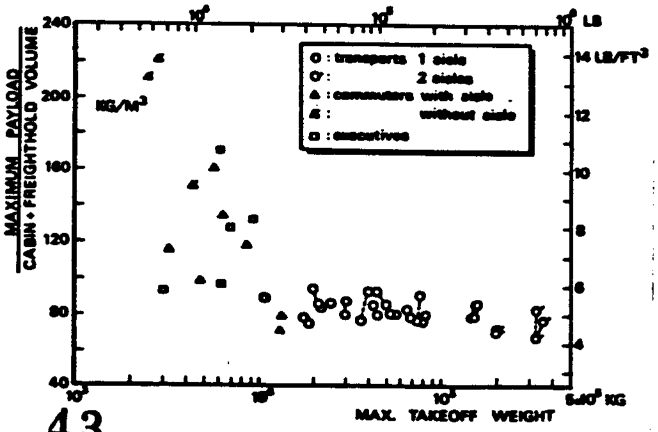
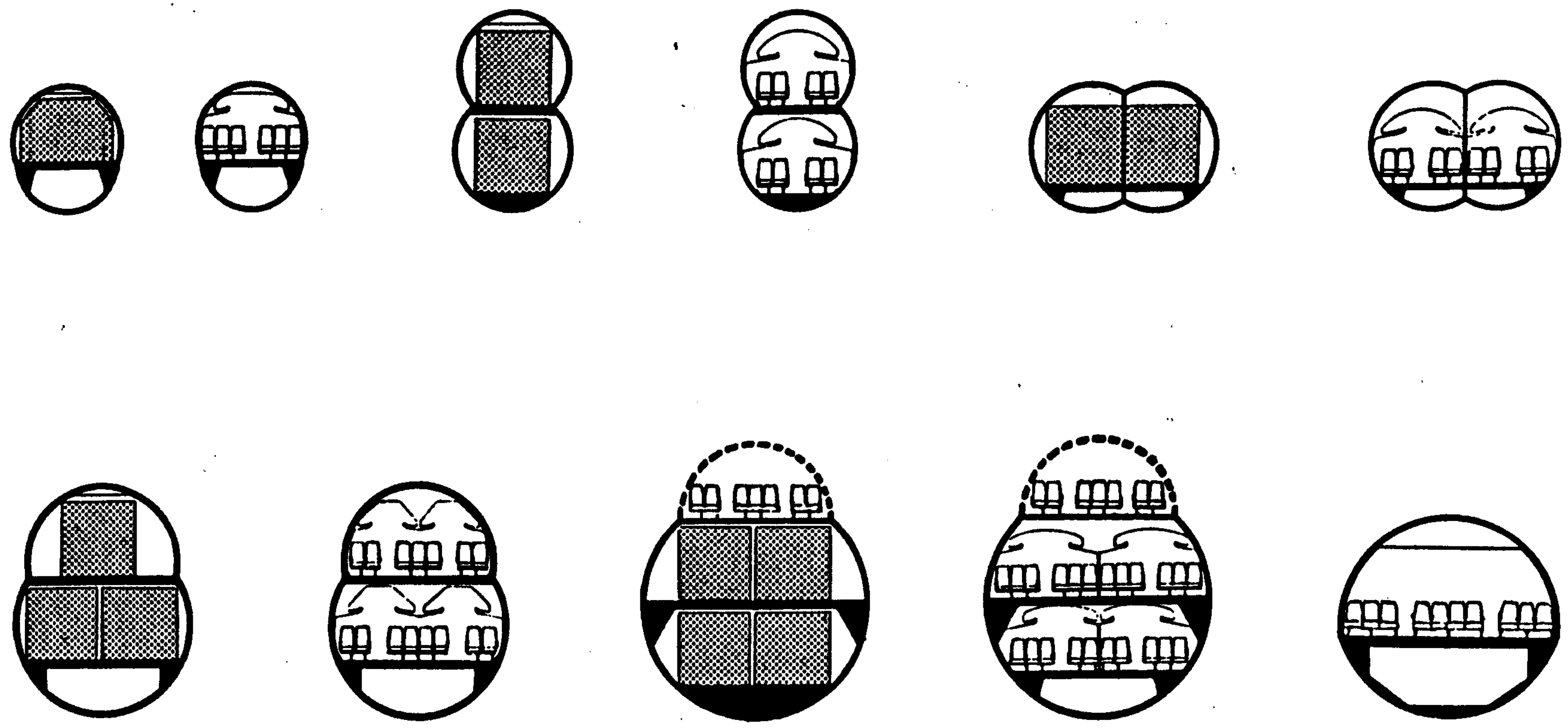


Fig. 3-7.Equivalent payload density

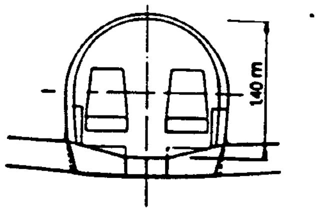
5.34



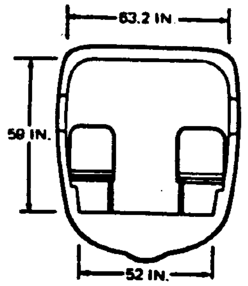
Some of the basic cross-sectional alternatives that Douglas has considered for the very large aircraft project, with a present-day cross section to provide scale. Prime requirement is for the 8 ft x 8 ft container to be carried efficiently. The bottom-row (middle) designs could alternatively be made circular, dispensing with the upper (dotted) lobe.

Fig. 3-8. Fuselage configuration studies by Douglas (Ref.: The Aeroplane, Aug. 4, 1966)

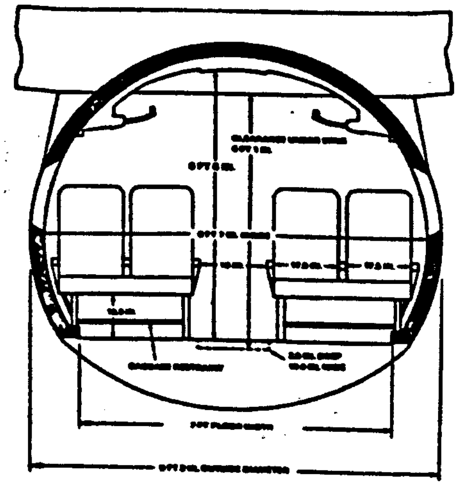
5.35



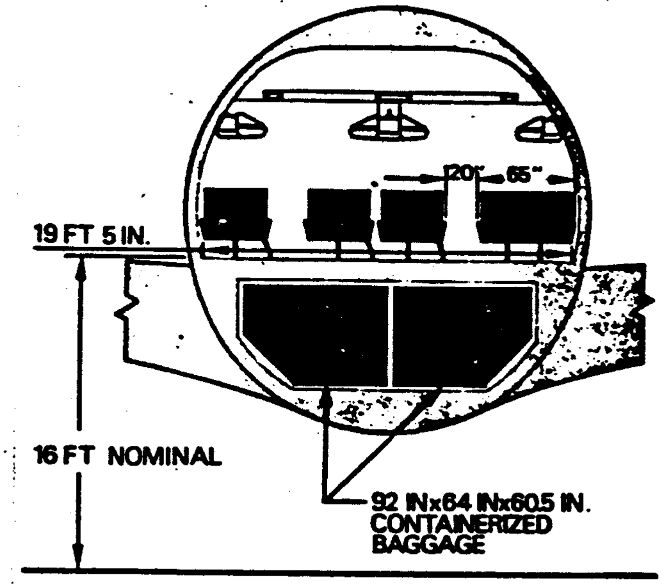
Dassault "Falcon" 10



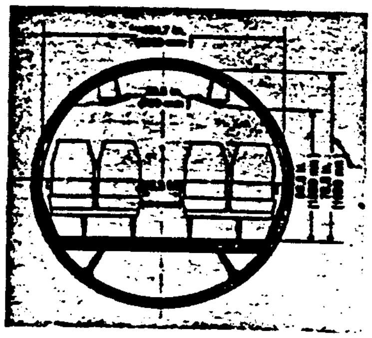
De Havilland Canada
DHC-6 "Twin Otter"



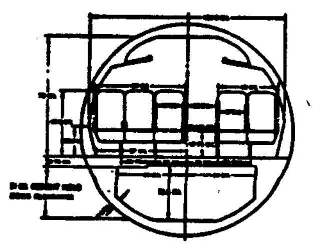
De Havilland Canada DHC-7



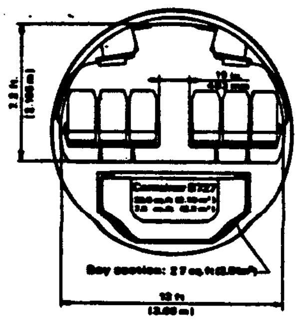
Boeing 747



VFW-Fokker 614

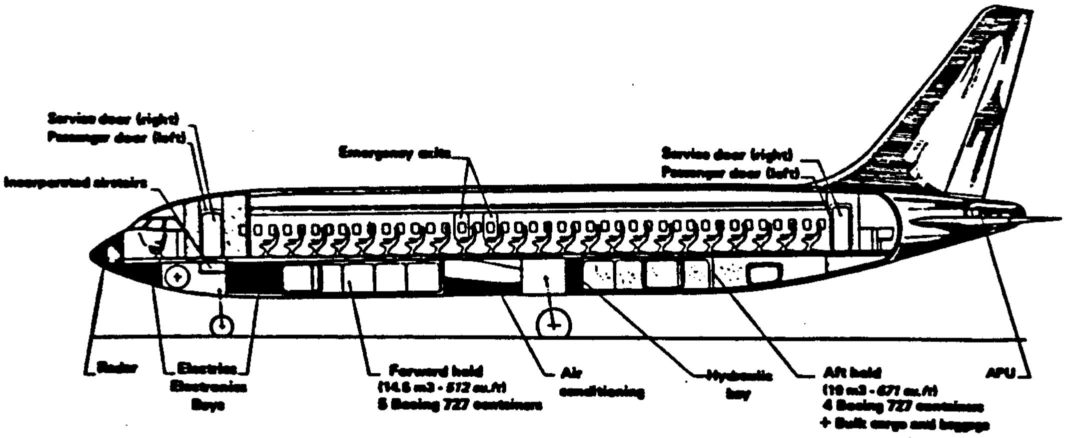


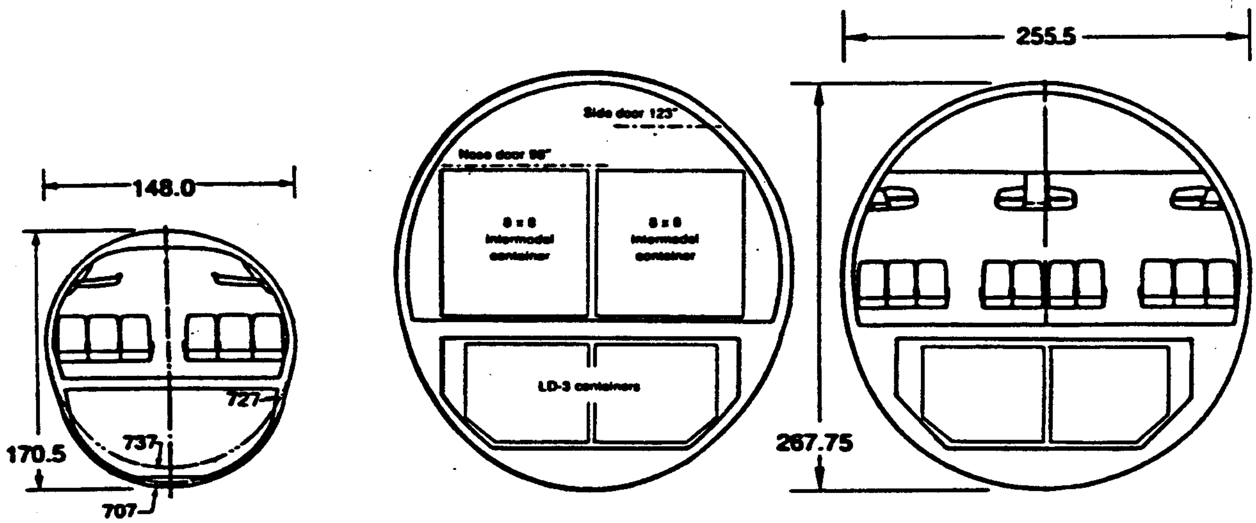
BAC-1 11



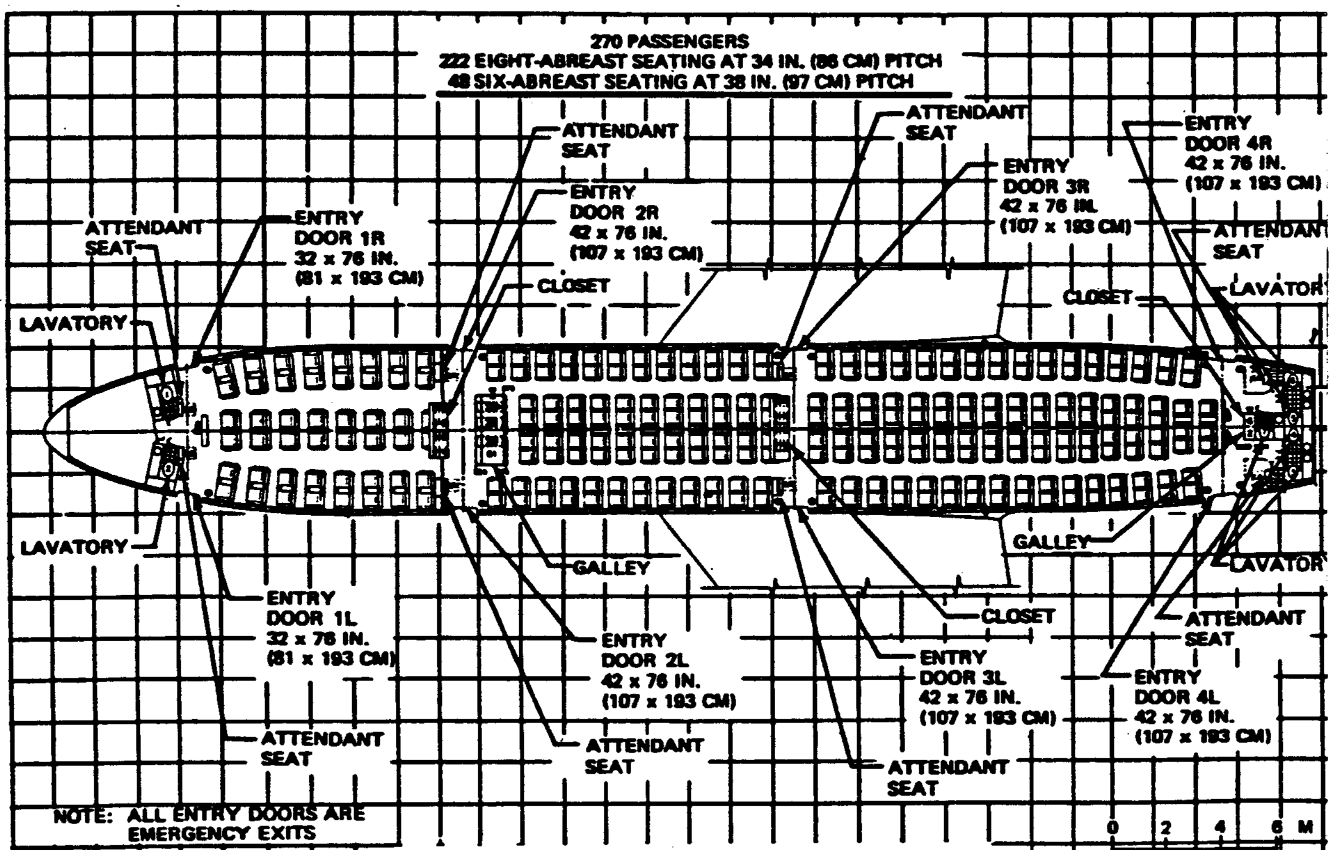
Dassault Bréguet
Mercure

Fig. 3-11. Examples of some typical fuselage cross-sections of transport aircraft





707/727/737 747F 747P

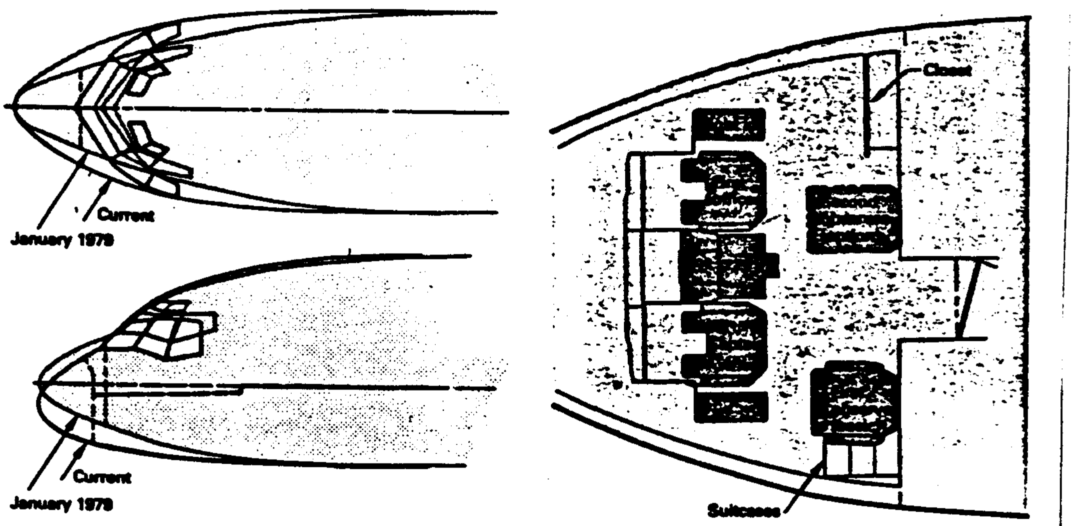


2.4 INTERIOR ARRANGEMENTS

2.4.1 PASSENGERS - MIXED CLASS

MODEL DC-10 SERIES 10, 20 AND 30

Boeing has redesigned the nose of the B.757 transport to provide improved flight deck visibility and increased crew working area, as well as more galley and toilet space. The redesign will also allow additional passenger seating. The change will result in greater commonality between the B.757 and B.767. There will, for instance, be common instrument packages for both aircraft. Boeing states that the new design will result in a quieter flight deck, as aerodynamic and air conditioning system noise will be reduced by 6 dB. The diagams below show the new design superimposed on the previous design (shaded) dating from January 1979.



INTERAVIA 11/1979

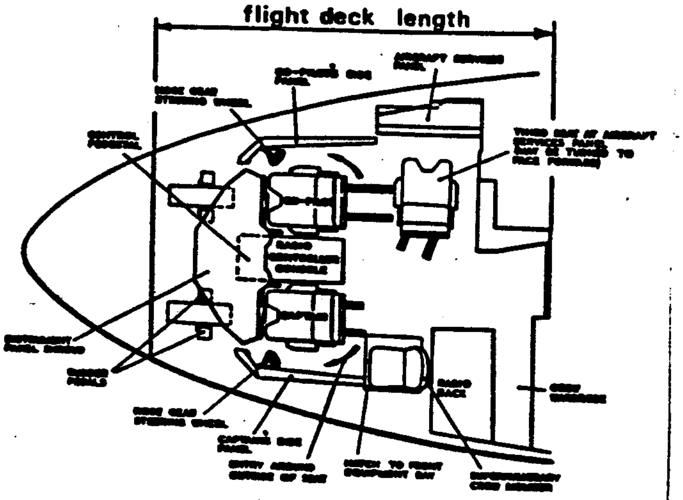


Fig. 3-26. Crew cabin layout for a medium-range transport

		Transport Aircraft			Light Aircraft
		Long Haul	Medium Haul	Short Haul	
Minimum flight crew		To be determined from the workload[1]), minimum: 2			VFR: 1 IFR: 2
Number of flight deck seats		4	3 or 4	2 or 3[3]	2
Length of flight deck[4])	Minimum	140(355)	125(317)	Low-subs. 90(228) High-subs. 105(267)	63(160)
	Average	150(380)	130(330)		70(178)
Distance between seat centerlines		42(107)	42(107)	40(102)	30(76)[5])
Number of cabin attendants	Minimum	1 per 50 passengers (PAX)			1 for 20 PAX or more
	Average	1 per 30 PAX		1 per 35 PAX	

All dimensions are typical values in inches (centimeters).

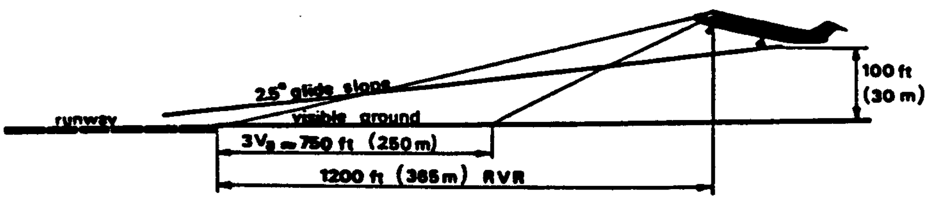
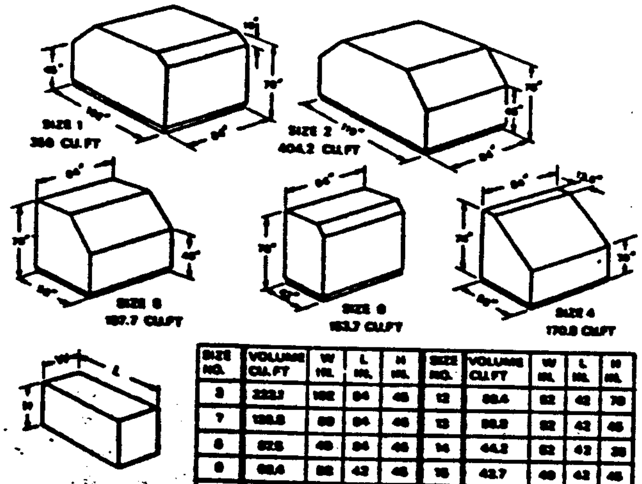
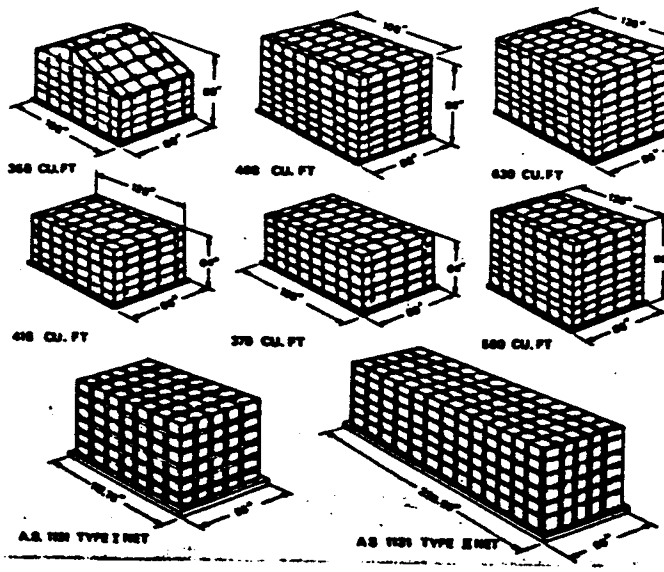
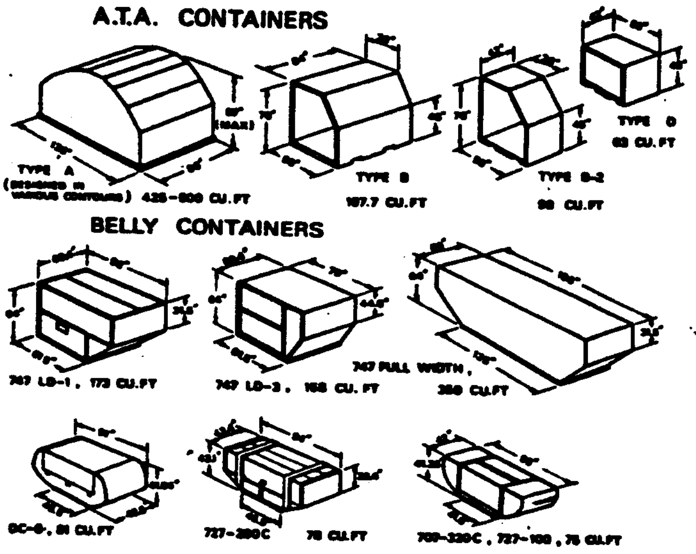


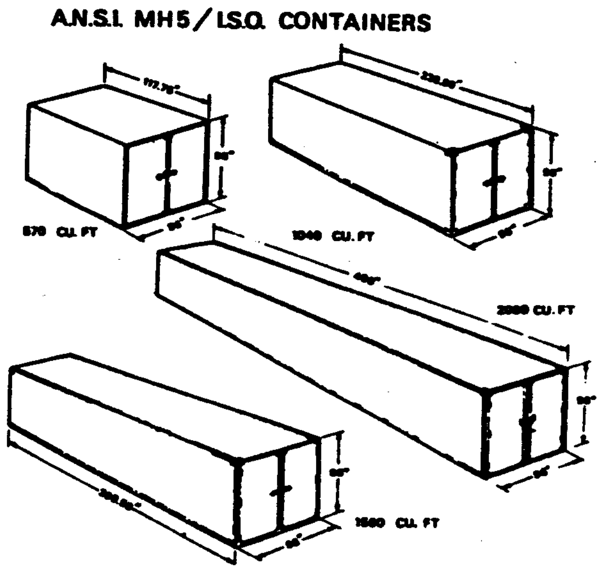
Fig. 3-25. Downward view on approach. (ref.: FAR 25.777 proposal, Jan. 1971)



SIZE NO.	VOLUME CU.FT	W IN.	L IN.	H IN.	SIZE NO.	VOLUME CU.FT	W IN.	L IN.	H IN.
3	[illegible]	[illegible]	84	46	12	88.4	82	42	70
7	[illegible]	88	84	46	13	[illegible]	82	42	46
8	[illegible]	48	84	46	14	44.2	82	42	[illegible]
9	[illegible]	88	42	46	15	42.7	40	42	46
10	[illegible]	88	42	[illegible]	16	21.7	20	42	46
11	[illegible]	82	84	70	17	[illegible]	20	42	[illegible]







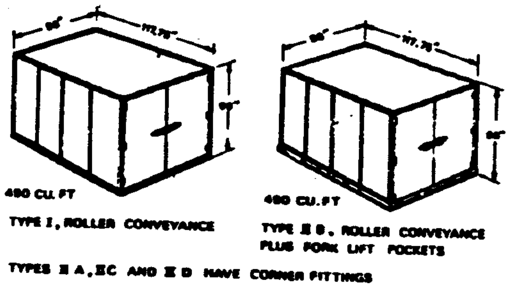


Fig. 3-20. Some standard pallets and containers

5.39

727-300

727-300B CONFIGURATION

- 240 INCH BODY STRETCH (180 INCHES FORWARD AND 60 INCHES AFT) OVER ADVANCED 727-200

- IMPROVED PERFORMANCE WING
  - MODIFIED CRUISE LEADING EDGE
  - INBOARD LEADING EDGE SLATS
  - OUTBOARD VARIABLE CAMBER LEADING EDGE FLAPS
  - MODIFIED INBOARD TRAILING EDGE
  - DOUBLE-SLOTTED INBOARD TRAILING EDGE FLAPS
  - THREE FOOT WING TIP EXTENSION PER SIDE

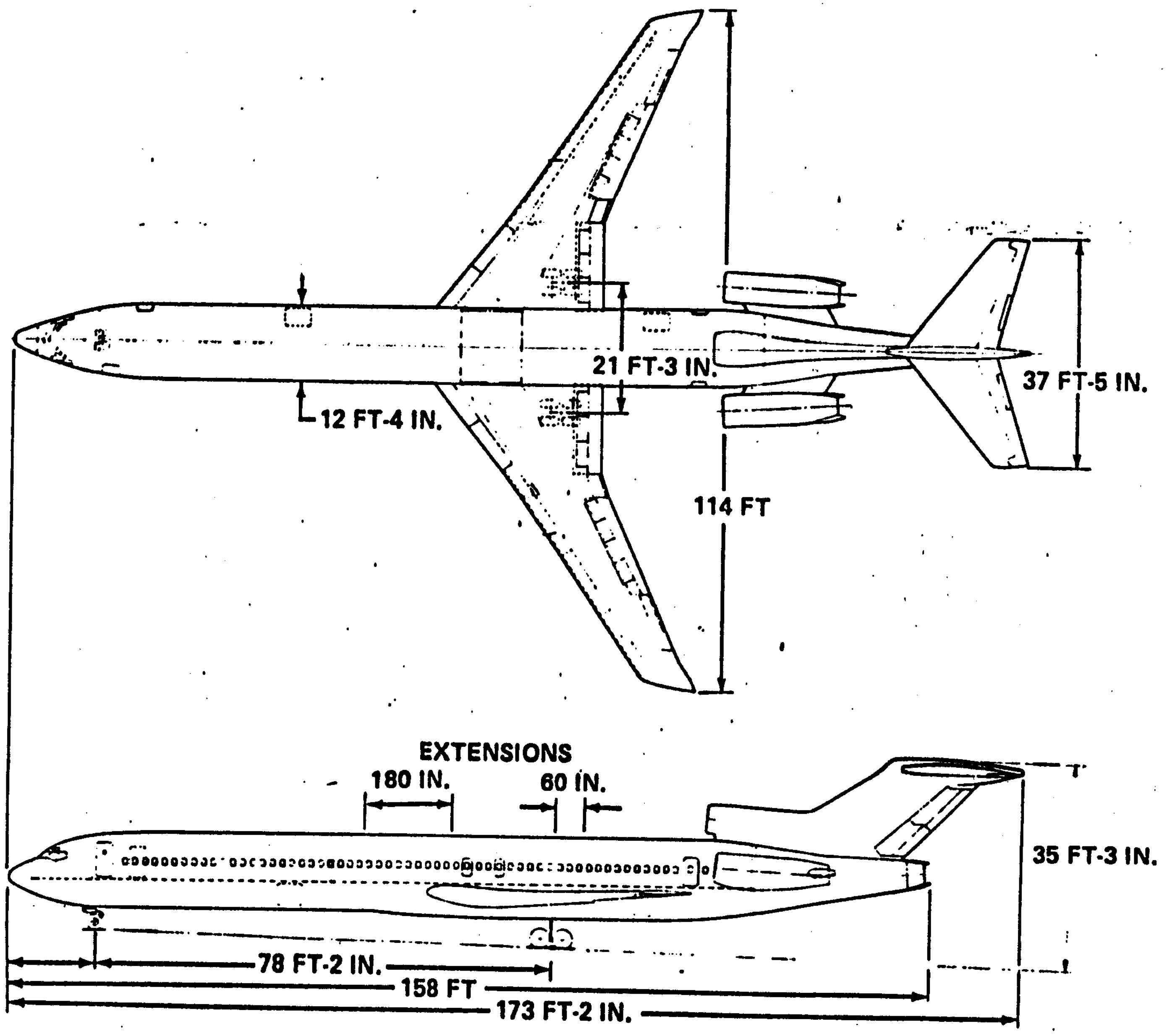
- SIMPLE FOUR-WHEEL TRUCK LANDING GEAR
- JT8D-217 WITH AUTOMATIC PERFORMANCE RESERVE
- NEW "S" DUCT AND REVISED AFT BODY
- TEN INCH HORIZONTAL STABILIZER ROOT INSERT PER SIDE
- ENGINE EXHAUST MIXER WITH TARGET THRUST REVERSER
- INCREASED TAKEOFF GROSS WEIGHT TO 222,000 LB
- VENTRAL STAIRS REMOVED

S4040R3
8-22-75

727-300

GENERAL ARRANGEMENT

727-300B



S 3540 R6
8-15-75

5.41

727-300

PERFORAMNCE SUMMARY

MIXED CLASS

	ADVANCED 727-200	727-300B		
ENGINES	JT8D-15	JT8D-209	JT8D-217	JT8D-217
WEIGHTS-LB				
MAX BRAKE RELEASE WEIGHT	190,500	210,000	210,000	222,000
MAX LANDING WEIGHT	154,500	190,000	190,000	190,000
OEW	101,216	121,060	121,200	123,230
CAPACITY				
PASSENGERS—NO. (FC/T)	134(20/114)	170(24/146)	170(24/146)	170(24/146)
FUEL-U.S. GAL (NO. AUX TANKS)	9760 (2)	8105 (0)	8105 (0)	9965 (2)
PERFORMANCE				
RANGE*-NMI				
MAX STILL AIR	2,260	1,730**	1,645**	2,060
FROM 7,000 FT RWY, S.L., 84° F	1,655	1,065	1,500	1,390
FROM 11,000 FT RWY, 5000 FT, 84° F	1,530	960	1,675**	1,585
FIELD LENGTH-FT				
TAKEOFF, MAX BRGW, S.L., 84° F	8,450	8,000	7,220	8,180
LANDING AT MISSION WT, S.L., WET	5,310	5,580	5,600	5,650
APPROACH SPEED—KEAS				
LANDING AT MISSION WT	129	139	139	140
FUEL BURN* LB/SEAT				
500 NMI TRIP	93.3	80.8	83.1	83.7
NOISE-FAR 36/ΔFAR 36-EPNdB				
TAKEOFF (WITHOUT CUTBACK)	109.8	102.0	101.6	102.7
(WITH CUTBACK)	100.0/+.3	98.5/-1.9	95.9/-4.9	97.8/-3.0
SIDELINE	102.2/-2.5	95.6/-9.4	97.6/-7.4	97.4/-7.7
APPROACH (30° FLAPS)	100.4/-4.3	103.1/-1.9	103.1/-1.9	103.1/-2.0

S5589
10-1-75

* LRC, STD DAY, ATA DOMESTIC RESERVES (200 NMI ALTERNATE), FULL PASSENGER PAYLOAD

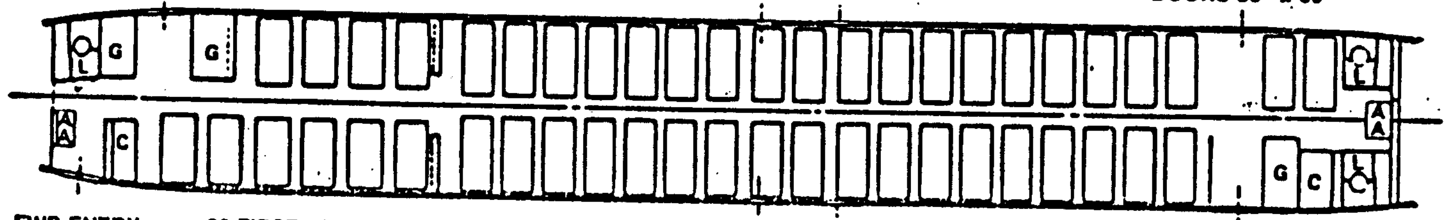
** FUEL VOLUME LIMITED

5.42

INTERIOR COMPARISON

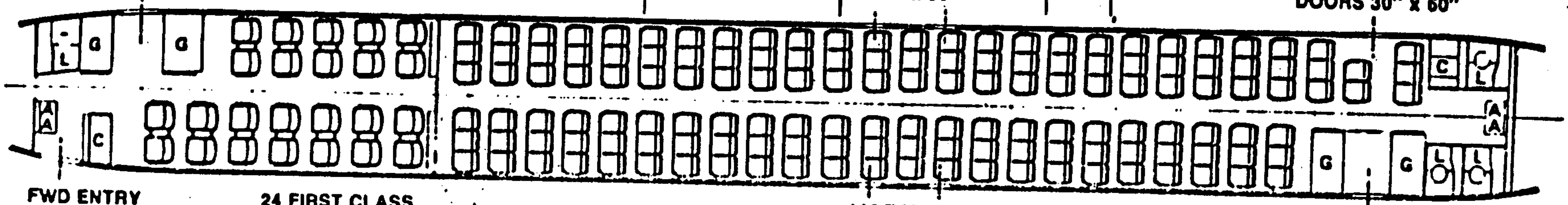
MIXED CLASS

ADVANCED 727-200



134 PASSENGERS

727-300B

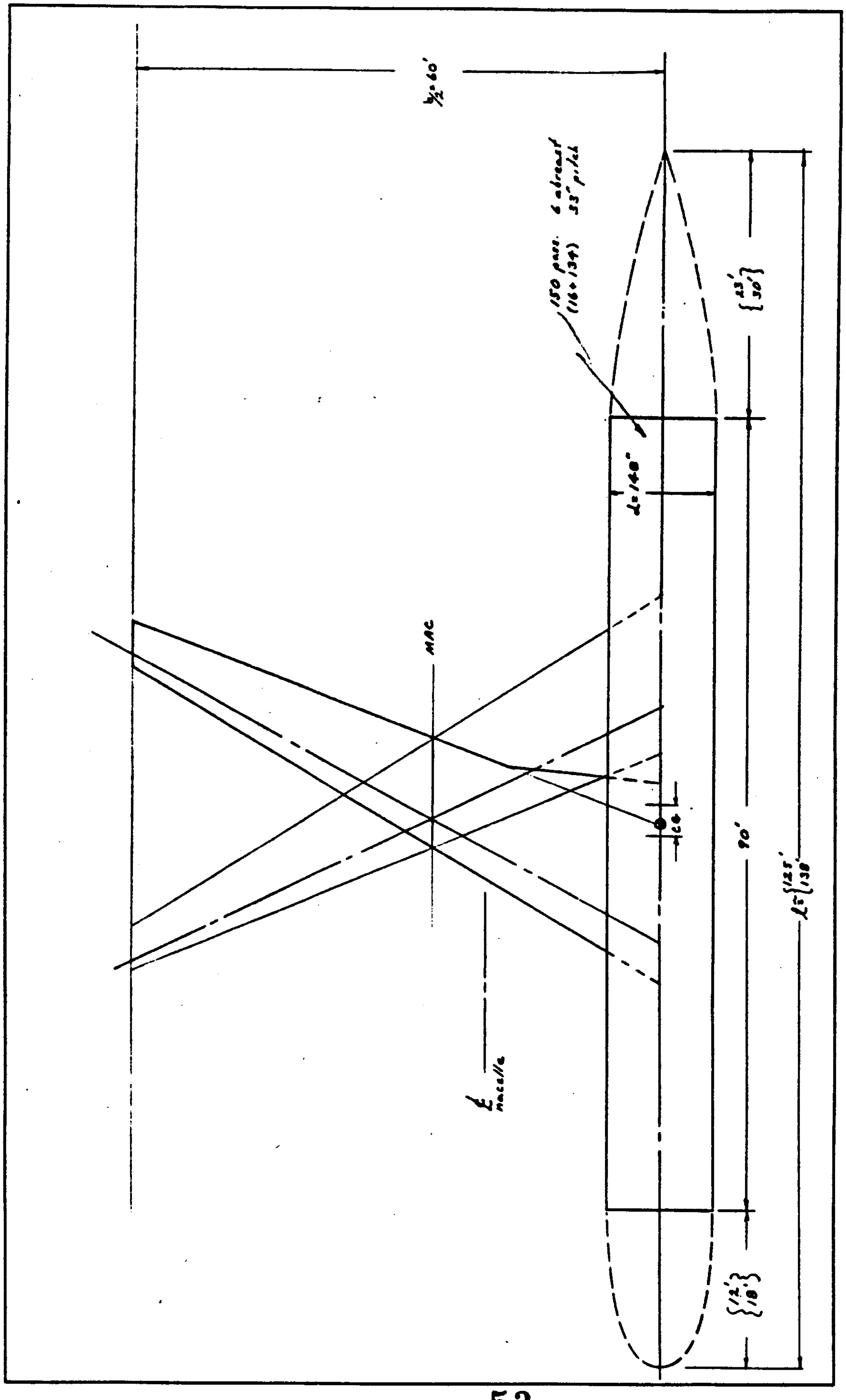


170 PASSENGERS

S5363
9-2-75

5.43

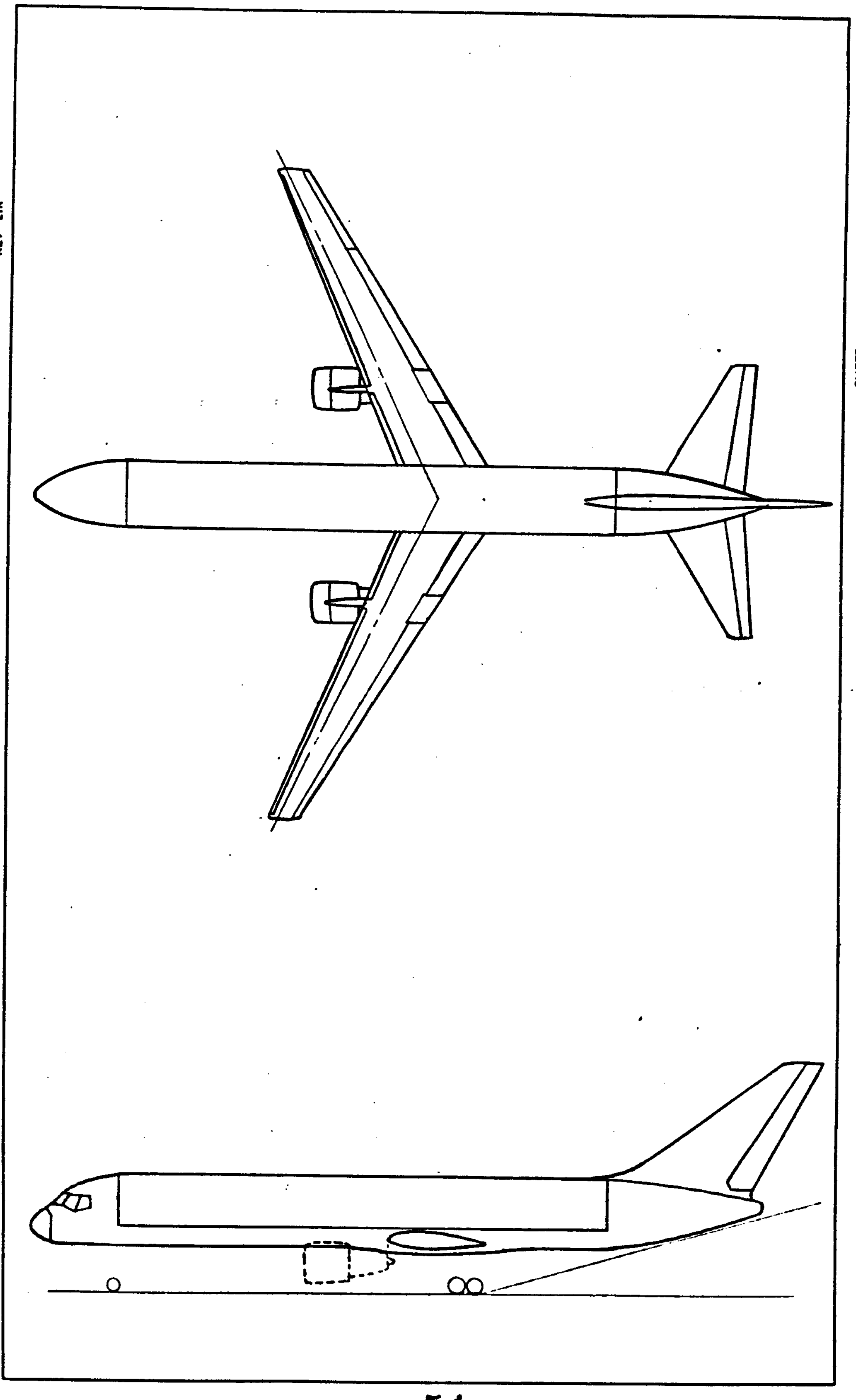
NUMBER
REV LTR



SHEET

D1 4002 8100 REV. 8/71

5.44



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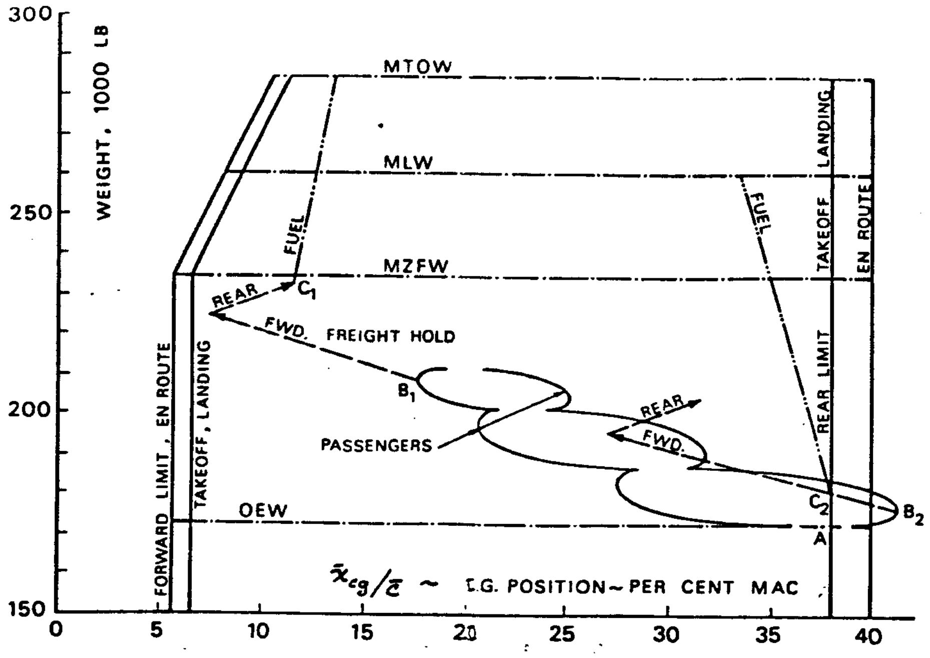


MTOW - Max. Take-off Weight (max. wt. at instant of brake release)

MLW - Max. Landing Weight (max. wt. at instant of touch down)

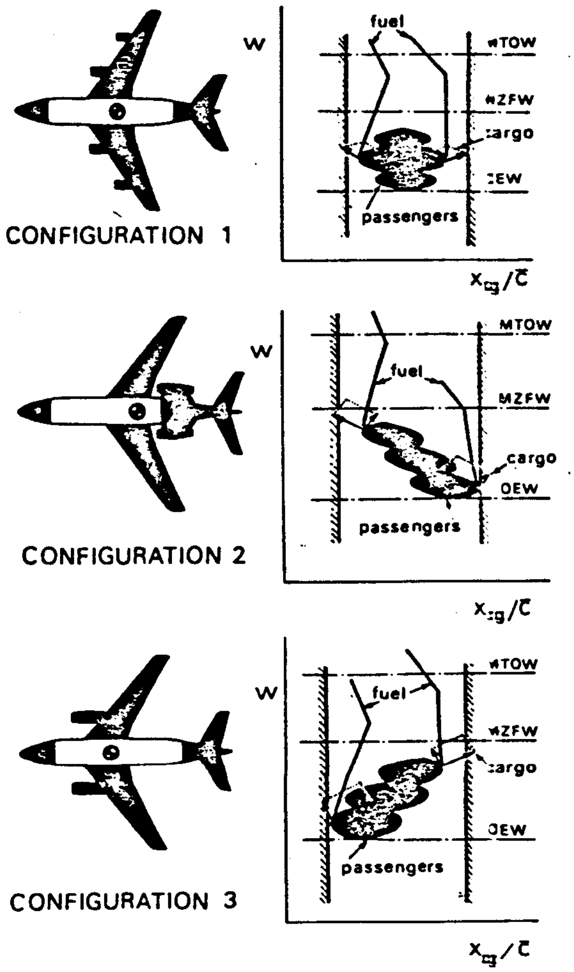
OEW - Operating Empty Weight (no payload or fuel)

ZFW - Zero Fuel Weight (OEW plus payload)

$\bar{c}$ - *mean aerodynamic chord*

Load and balance diagram for a short-haul wide-body airliner

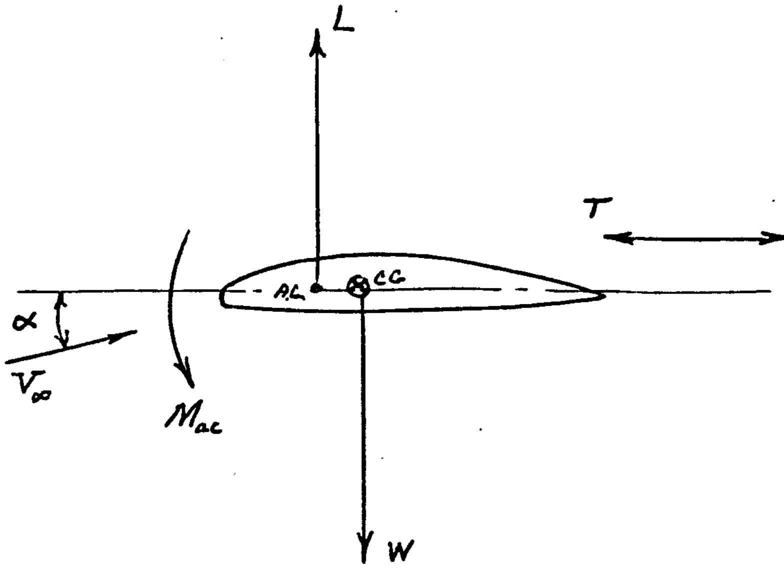
Weight & Balance



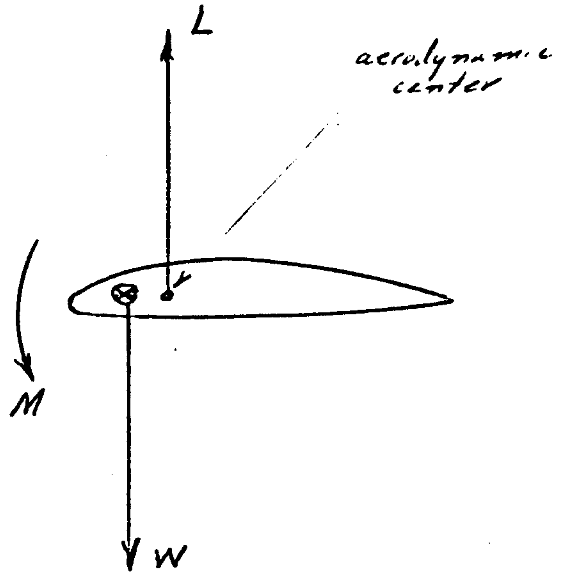
Loading Case	Underwing Podded Layout	Rear-Engined Layout
Empty aircraft	No problem. Approx. center of range	Rear limit based on this case
Very high-density full aircraft	No problem. Approx. center of range	Forward limit based on this case
Full one-class aircraft	No problem.	No problem
Full tourist. Very light first class	Rear c.g. critical Requires special baggage disposition	No problem
Full first class Very light tourist	Forward c.g. critical Requires special baggage disposition	No problem
Partially full one-class aircraft; passengers seating from rear. Window seating only	Forward limit based on this case	No problem
Partially full one-class aircraft; passengers seating from front. Window seating only	Forward limit based on this case	No problem

Effect of the general arrangement on load and balance

Summary of critical loading conditions for two airplane configurations (Reference: Aerospace Engineering, October 1960, page 74)

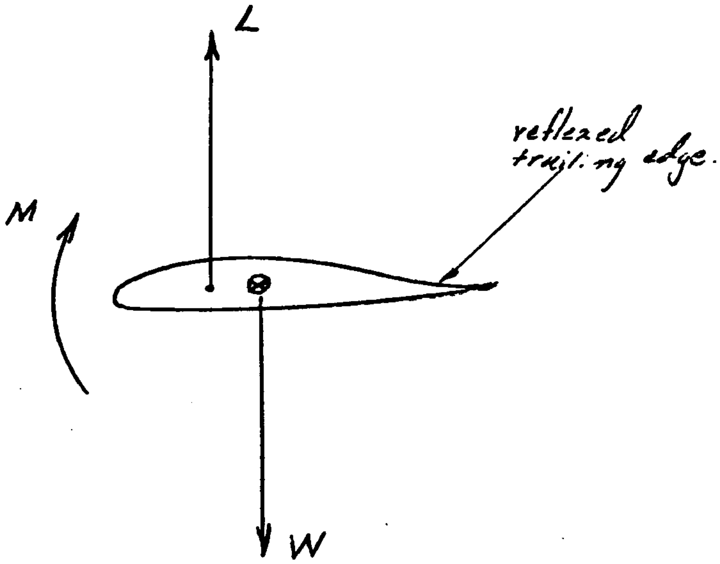


Trimmed, unstable

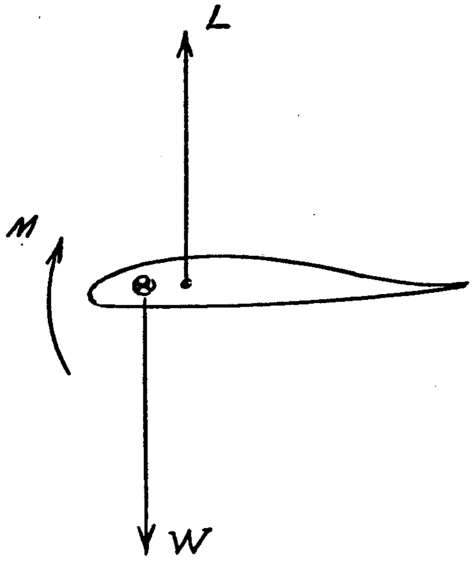


Untrimmed, unstable

Wing Alone Static Stability

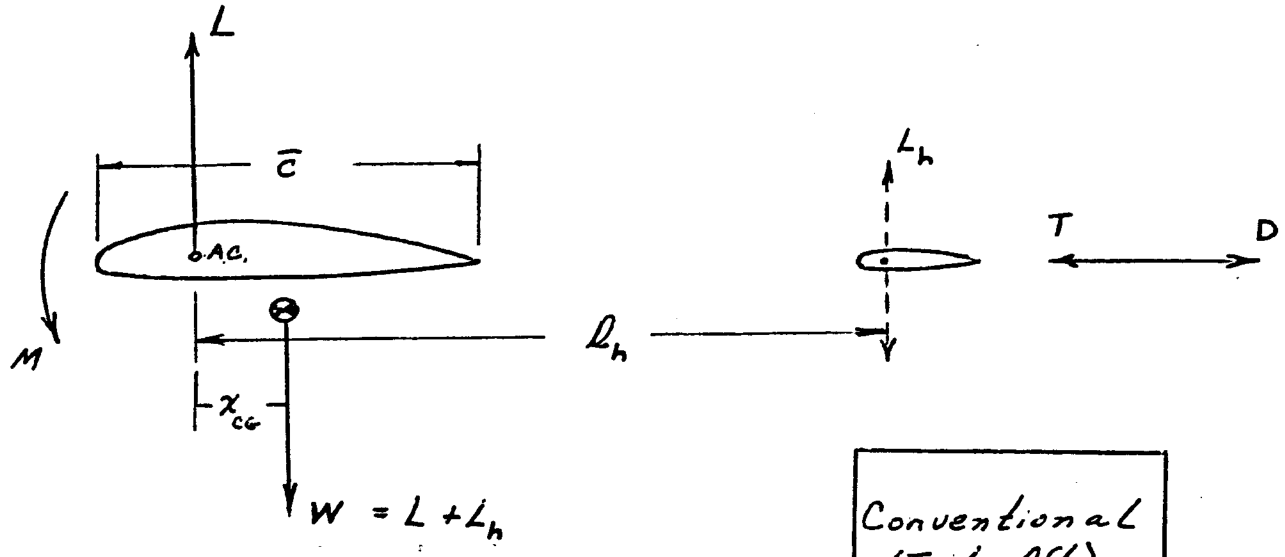


Untrimmed, unstable

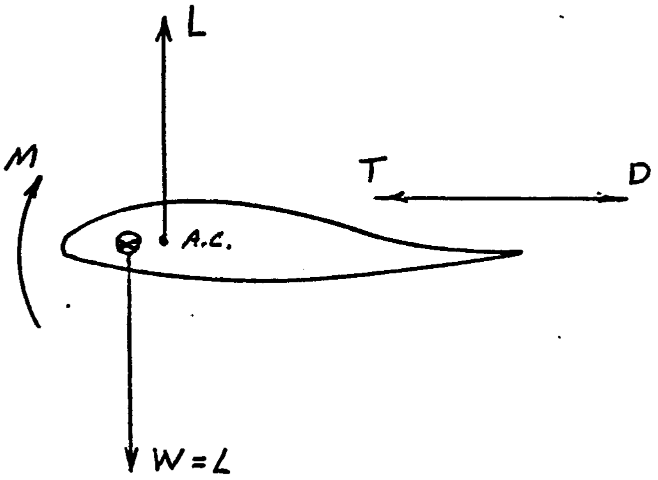


<u>Trimmed</u>, <u>stable</u>

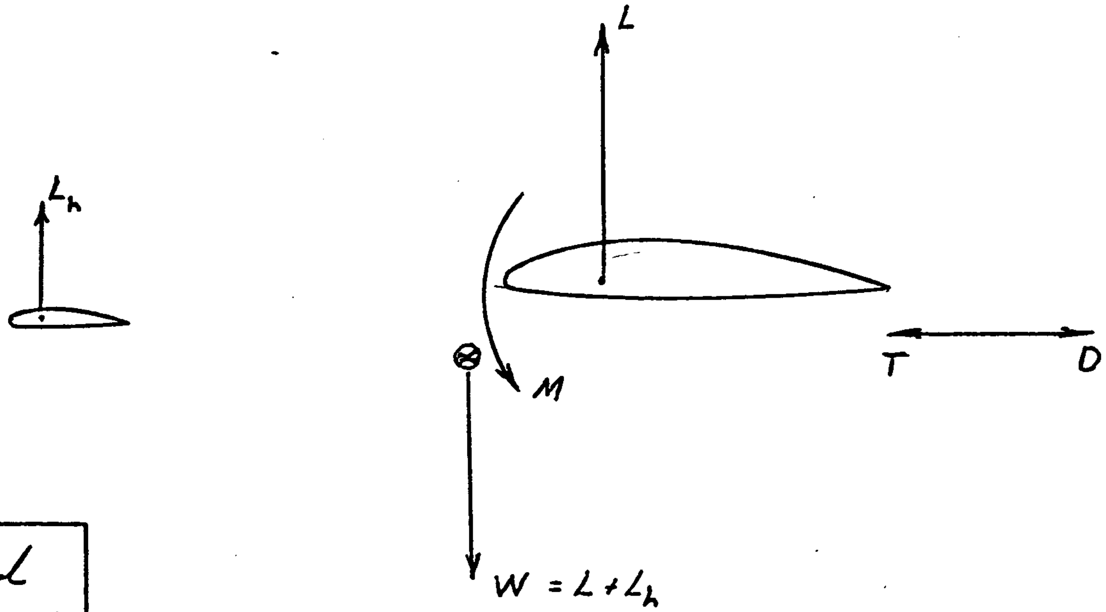
$$\left.\begin{array}{l} L \text{ varies with angle of attack } (\alpha) \\ M_{a.c.} \text{ approximately constant for variable } \alpha \end{array}\right\} \text{at } V_\infty \text{ const.}$$



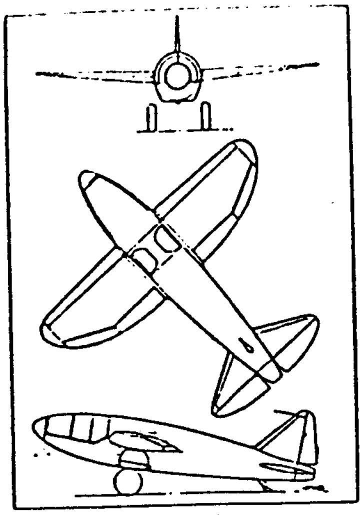
Conventional
(Tail Aft)



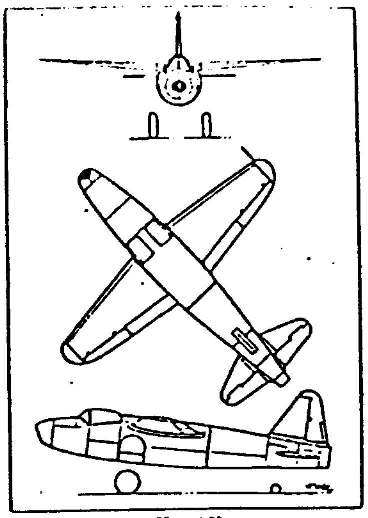
Flying Wing
(Tailless)



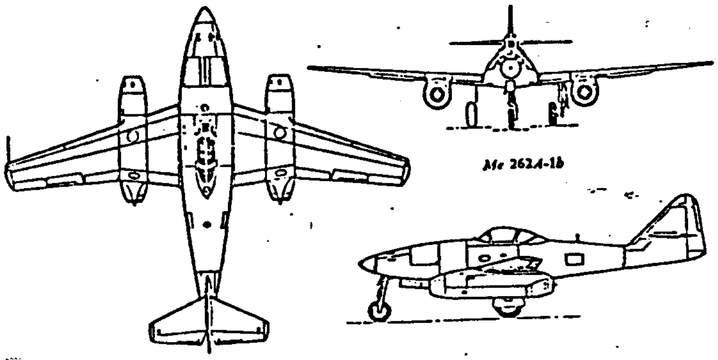
Canard
(Tail First)



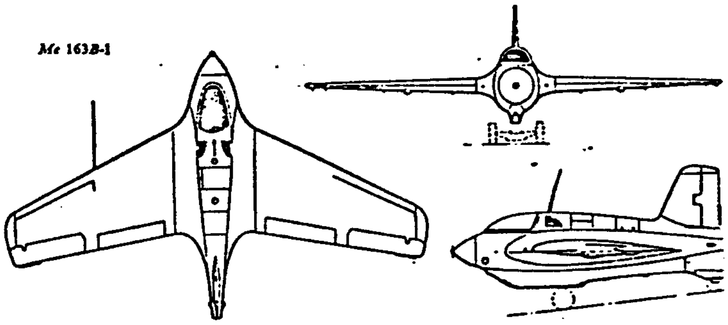
(He 176)



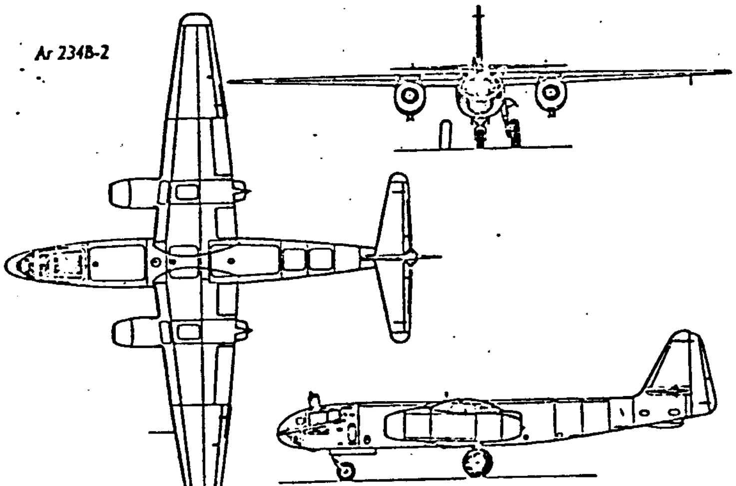
(He 178)



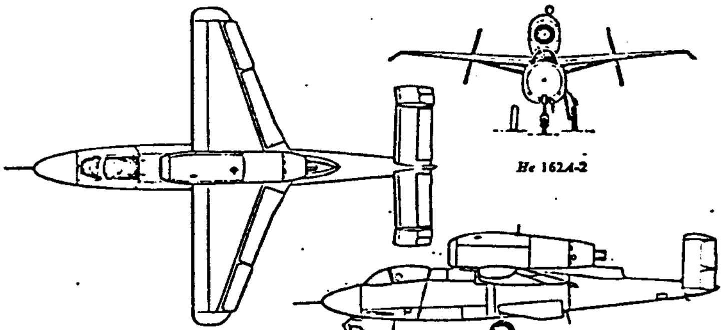
Me 262A-1b



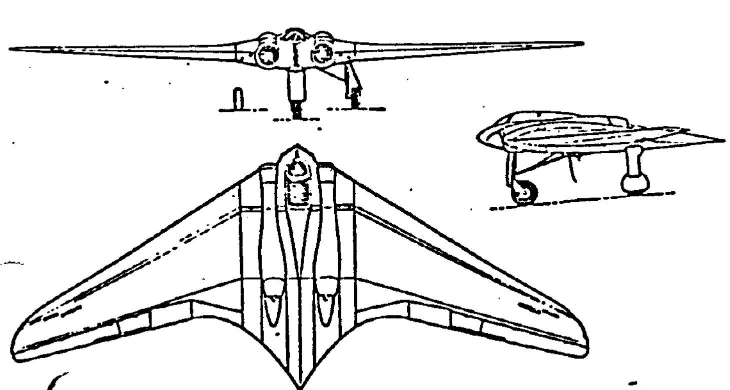
Me 163B-1

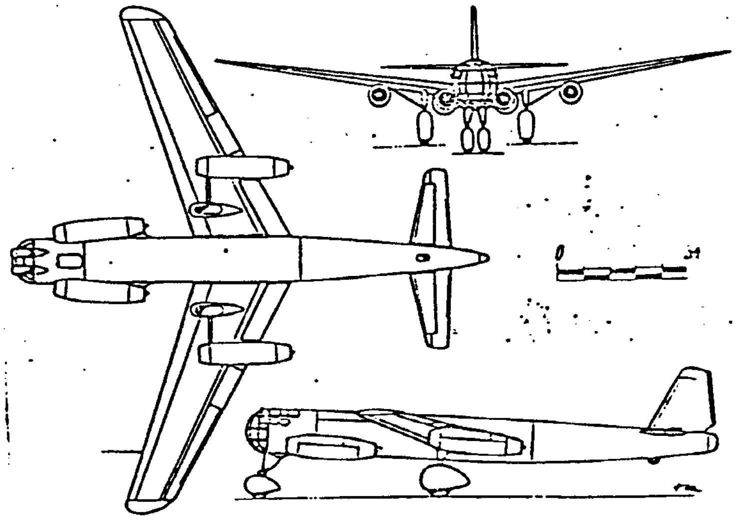


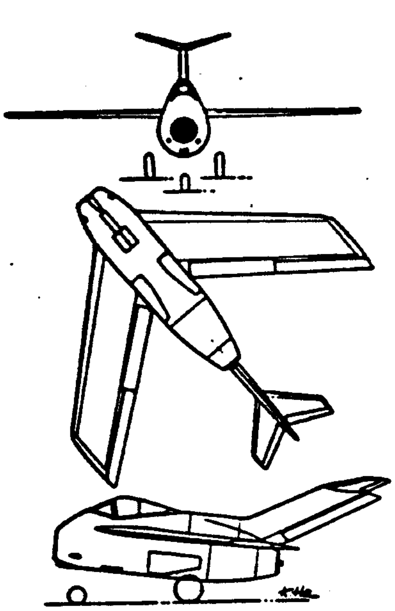
Ar 234B-2

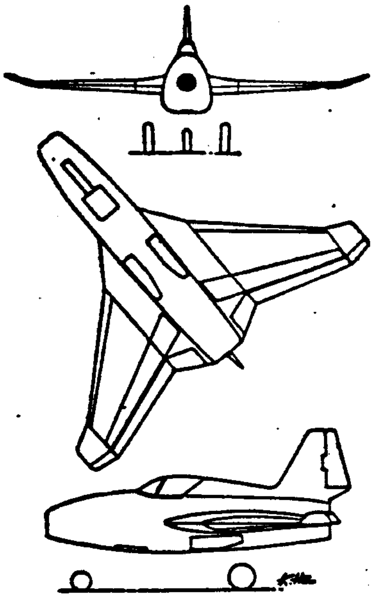


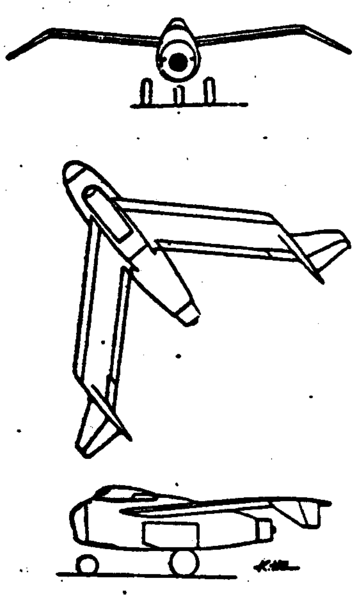
He 162A-2

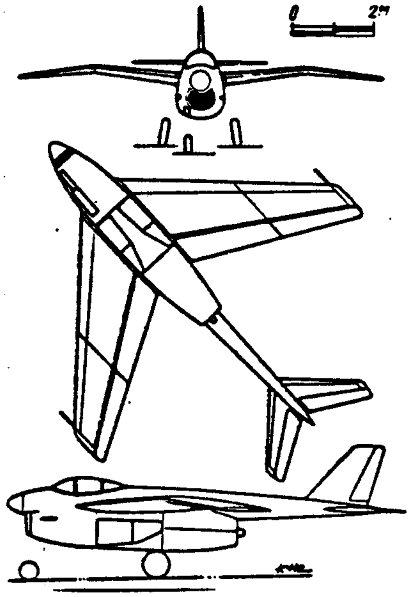


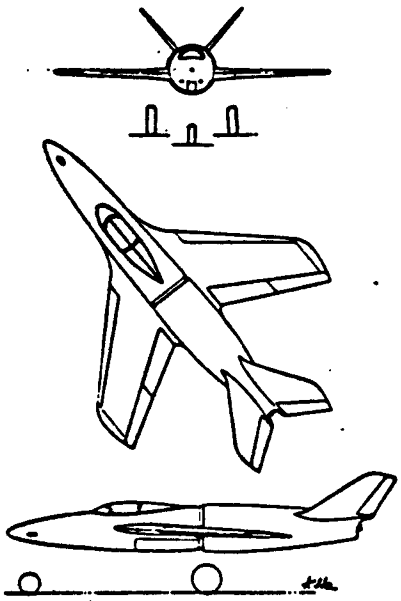


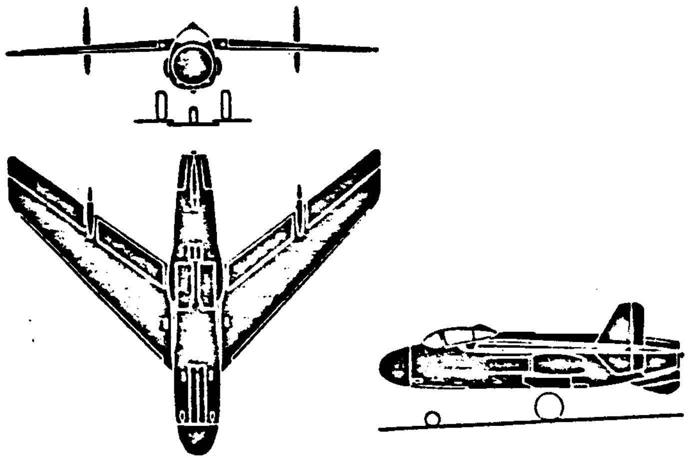


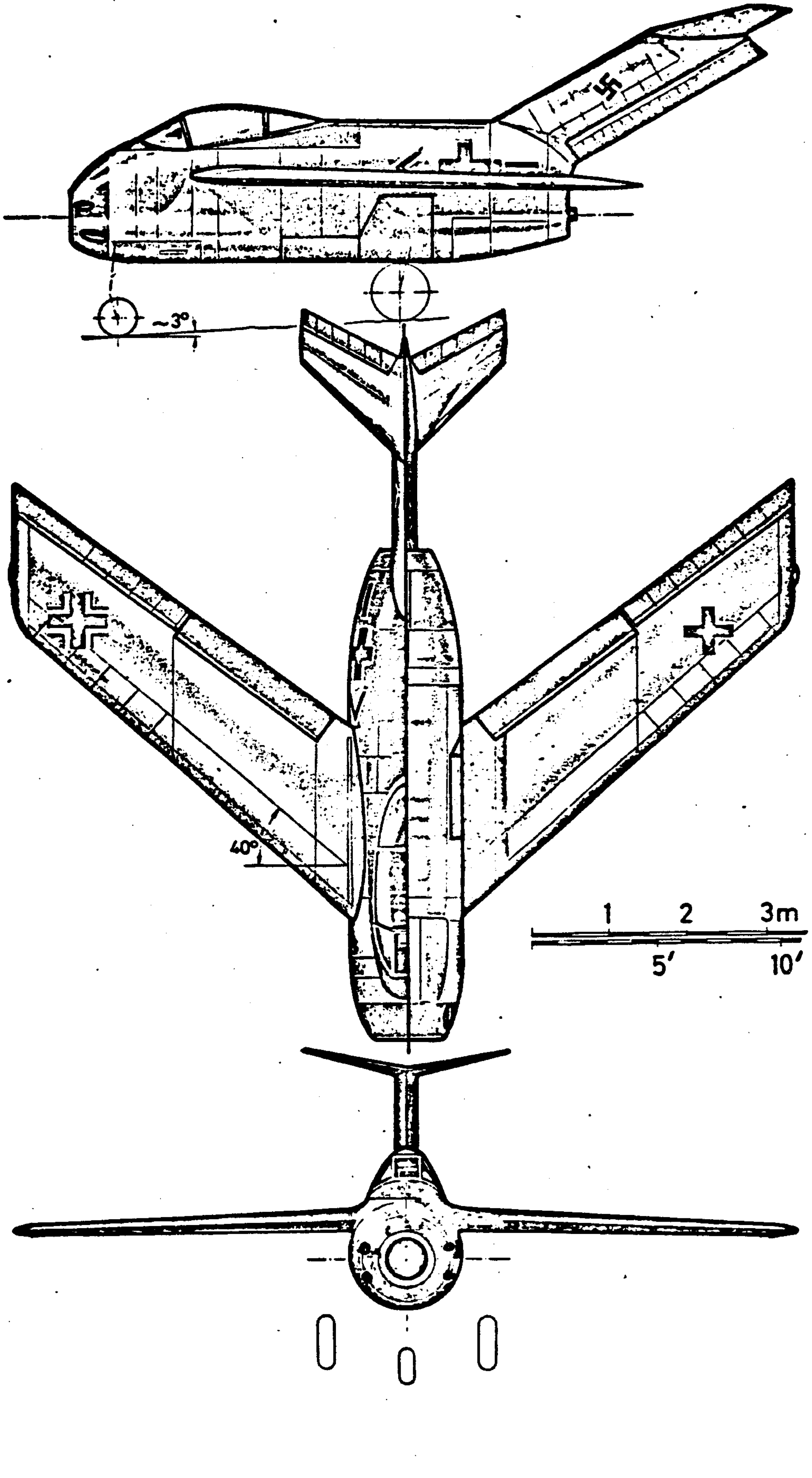





0
2m







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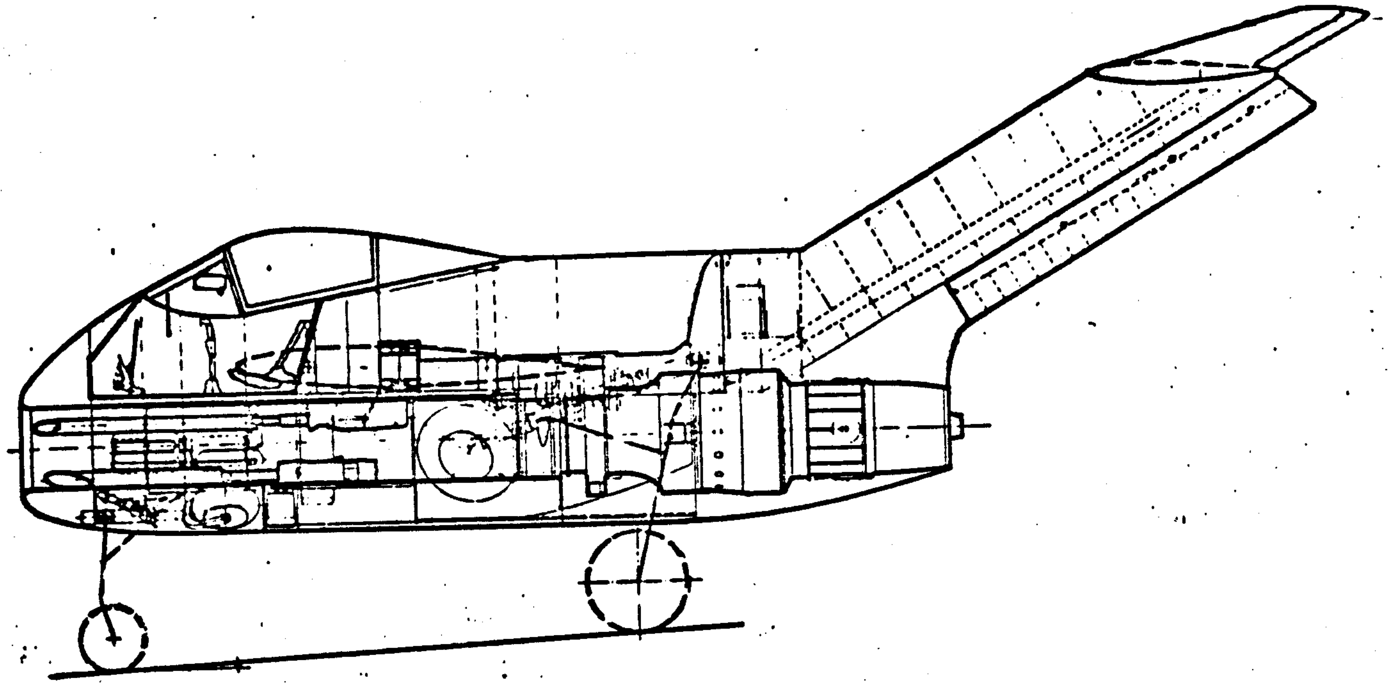
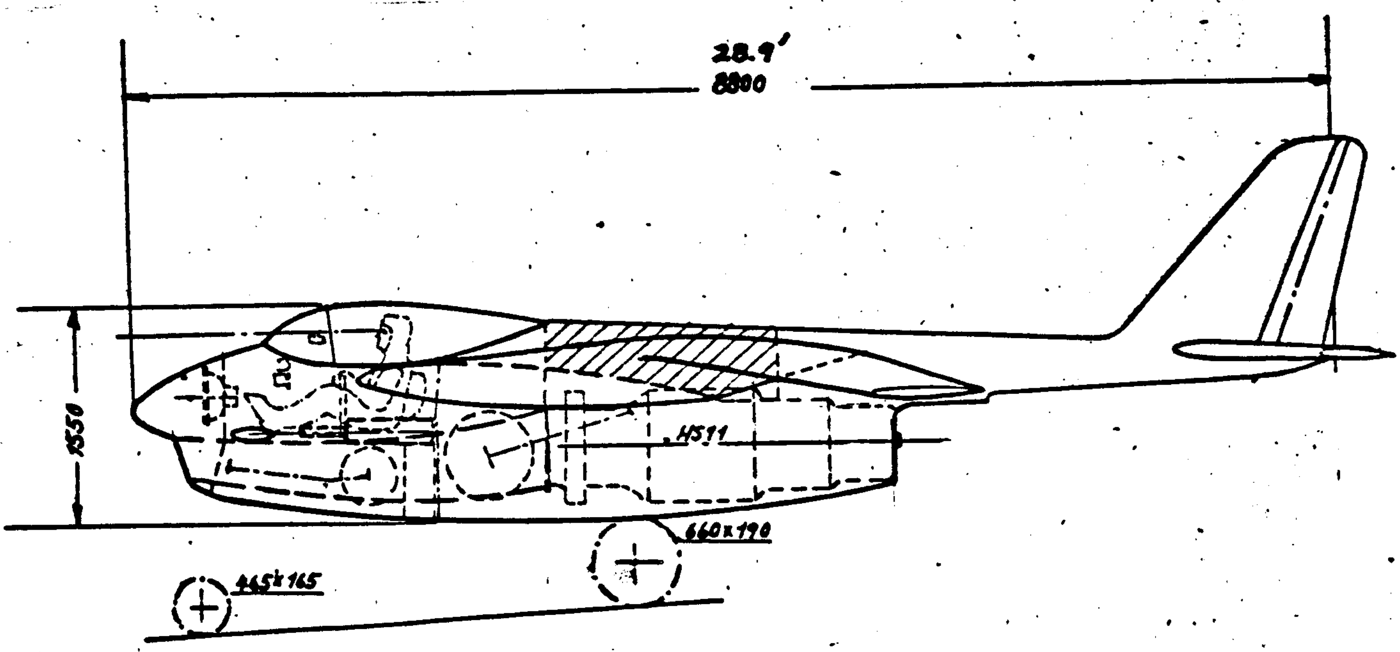
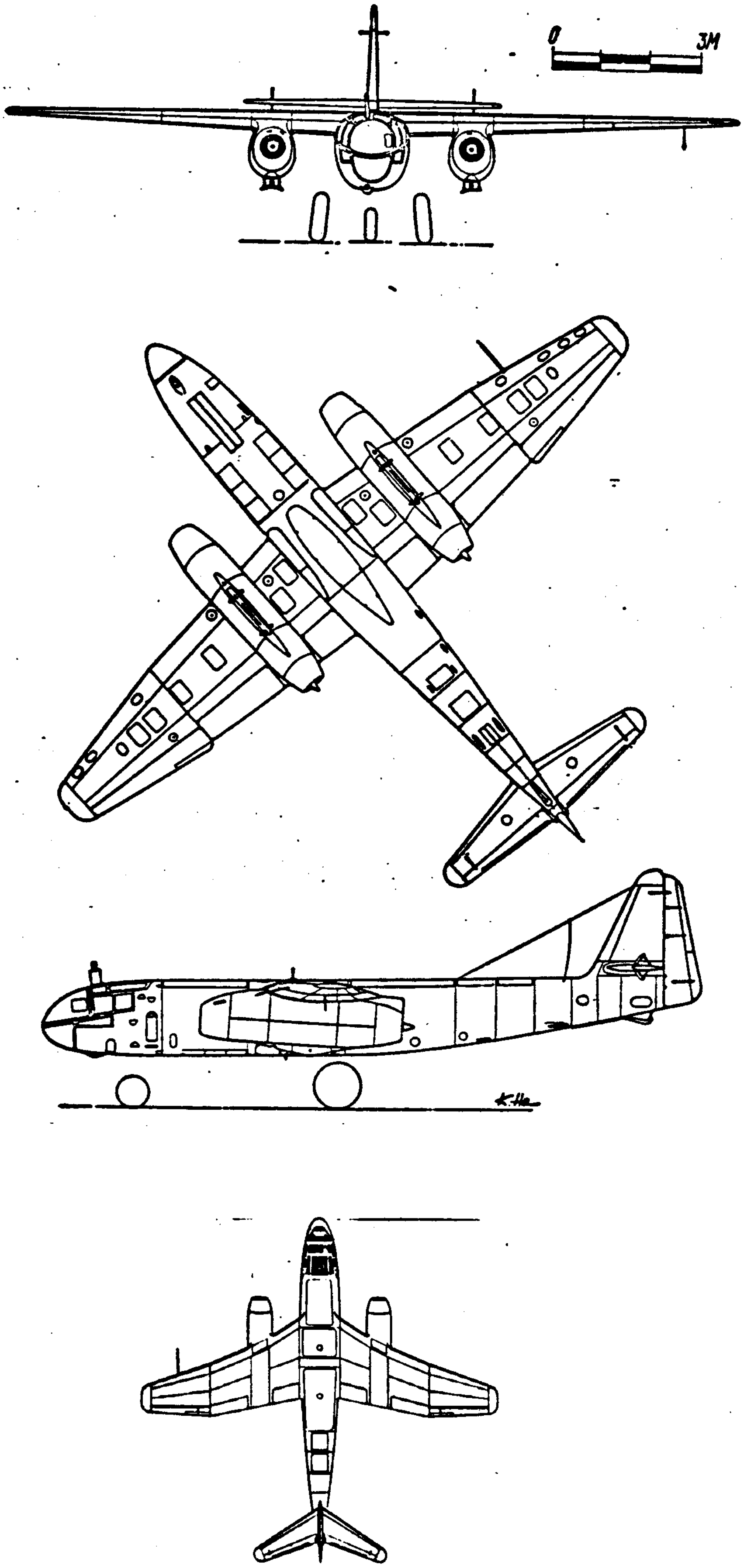
5.50





5.52
~~5.67~~

Arado Ar 234 B Blitz



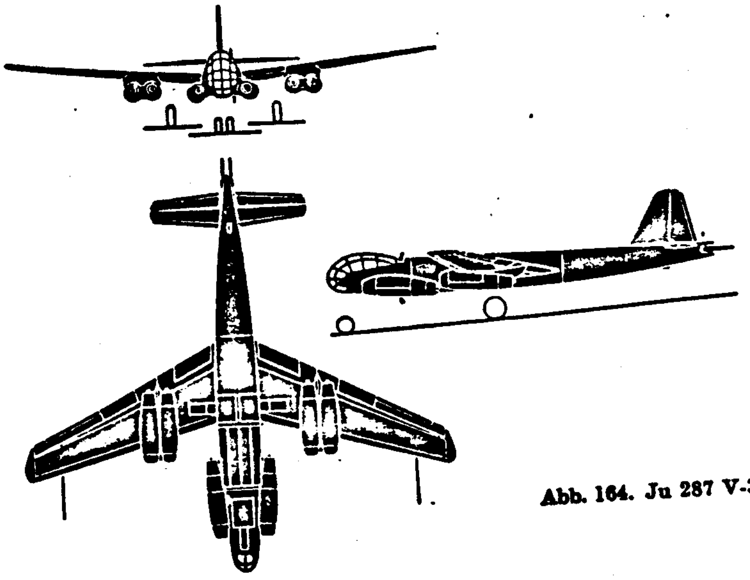


Abb. 164. Ju 287 V-3

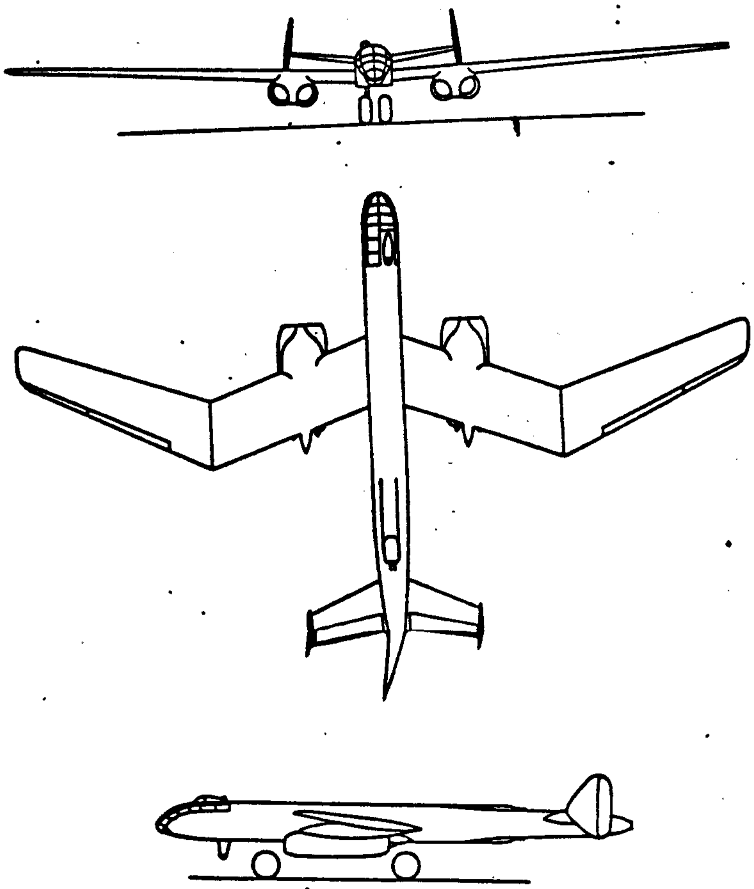


Abb. 701. Blohm & Voß P. 188.04-01

VI. Conceptual Design II - Sizing (3 hours)

A. Sizing Methods

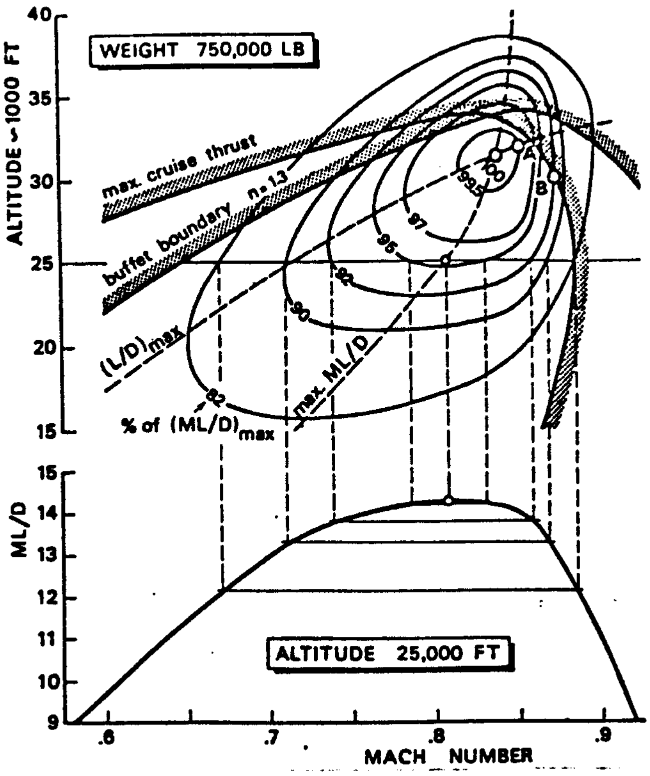
1. Thumbprint - Analysis

2. Geometric Programming - Synthesis

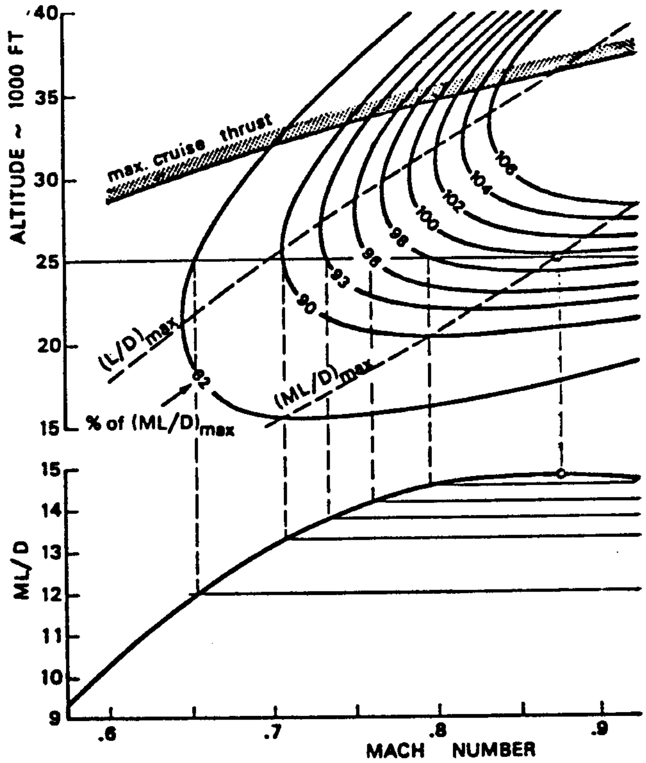
B. Examples

1. Jet Airplane optimized for cruise

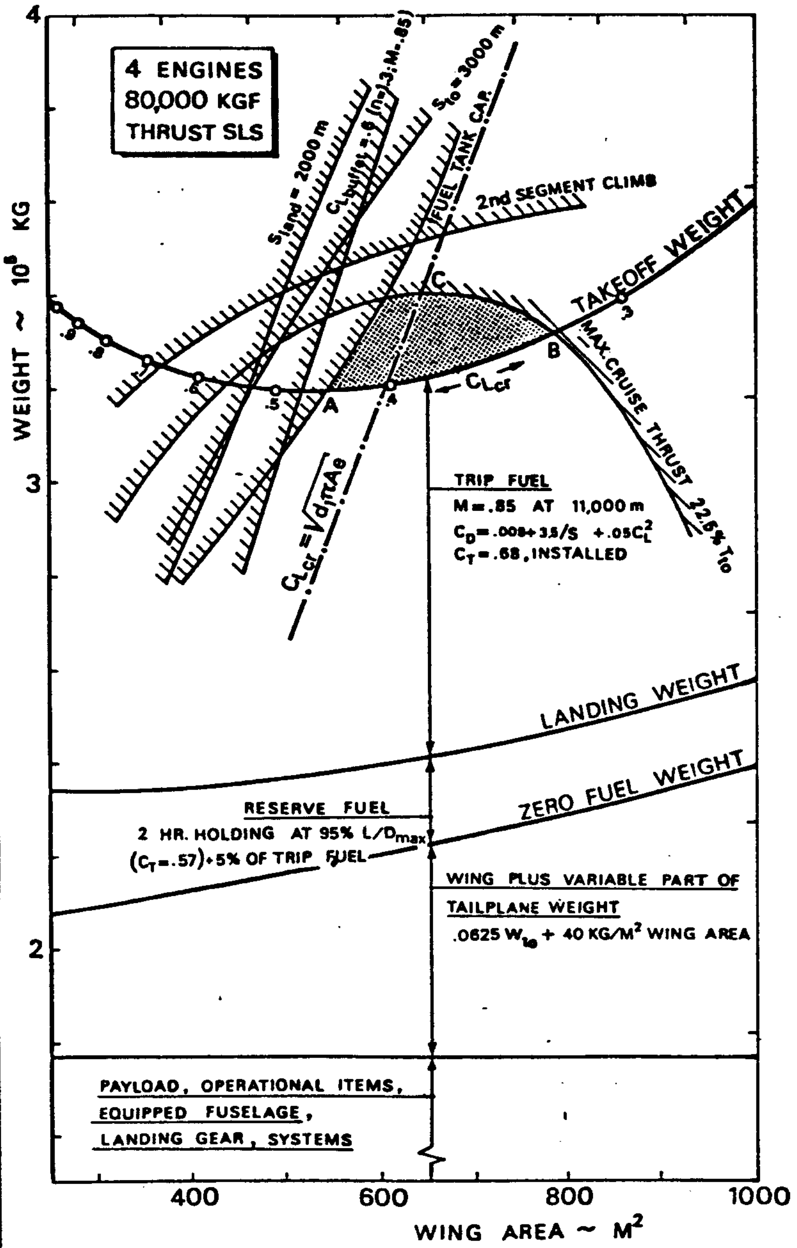
2. Laminar sailplane



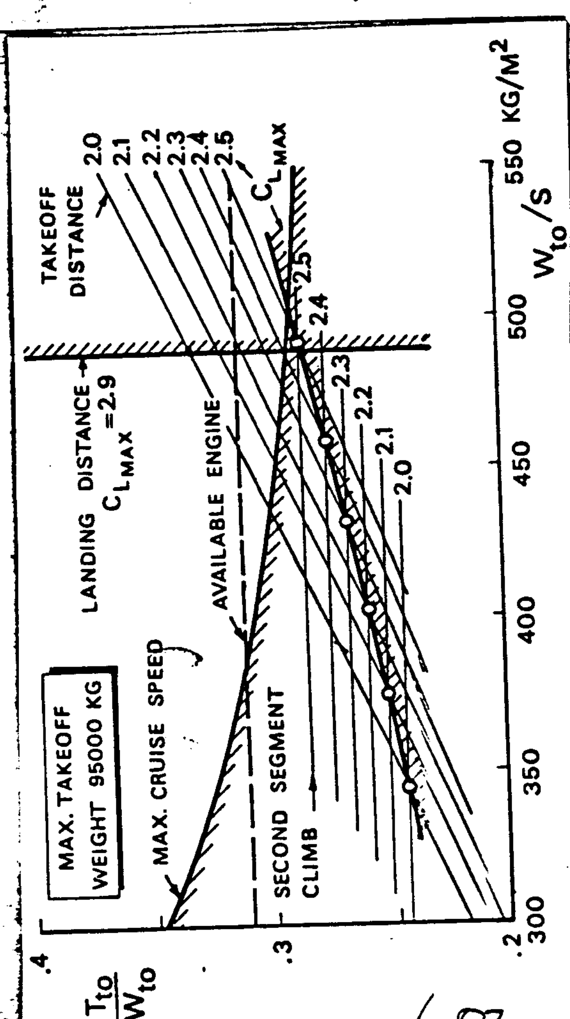
WEIGHT 750,000 LB
ALTITUDE ~ 1000 FT
max. cruise thrust
buffet boundary n=1.3
$(L/D)_{max}$
max. ML/D
% of $(ML/D)_{max}$
ML/D
ALTITUDE 25,000 FT
MACH NUMBER



ALTITUDE ~ 1000 FT
max. cruise thrust
$(L/D)_{max}$
$(ML/D)_{max}$
% of $(ML/D)_{max}$
ML/D
MACH NUMBER



4 ENGINES
80,000 KGF
THRUST SLS
WEIGHT ~ 10^5 KG
S_{land} = 2000 m
S_{to} = 3000 m
FUEL TANK CAP.
2nd SEGMENT CLIMB
TAKEOFF WEIGHT
MAX. CRUISE THRUST
22.5% T_{to}
$C_{L_{cr}} = \sqrt{C_{D_0} \pi A e}$
$C_{L_{cr}}$
TRIP FUEL
M = .85 AT 11,000 m
$C_D = .008 + 3.5/S + .05 C_L^2$
C_T = .68, INSTALLED
LANDING WEIGHT
ZERO FUEL WEIGHT
RESERVE FUEL
2 HR. HOLDING AT 95% L/D_{max}
(C_T = .57) + 5% OF TRIP FUEL
WING PLUS VARIABLE PART OF
TAILPLANE WEIGHT
.0625 W_{to} + 40 KG/M² WING AREA
PAYLOAD, OPERATIONAL ITEMS,
EQUIPPED FUSELAGE,
LANDING GEAR, SYSTEMS
WING AREA ~ M²



TAKEOFF DISTANCE
$C_{L_{MAX}}$
W_{to}/S KG/M²
LANDING DISTANCE $C_{L_{MAX}}$ = 2.9
AVAILABLE ENGINE
MAX. CRUISE SPEED
SECOND SEGMENT CLIMB
MAX. TAKEOFF WEIGHT 95000 KG
T_{to}/W_{to}

6.07

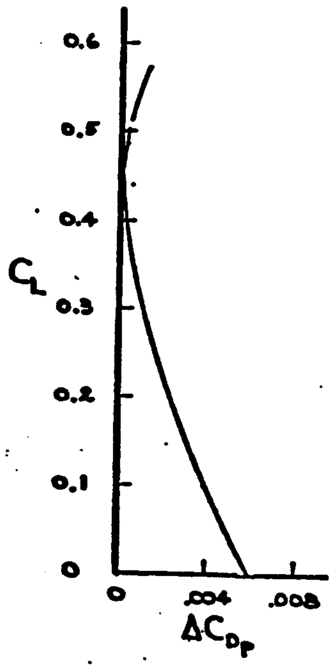


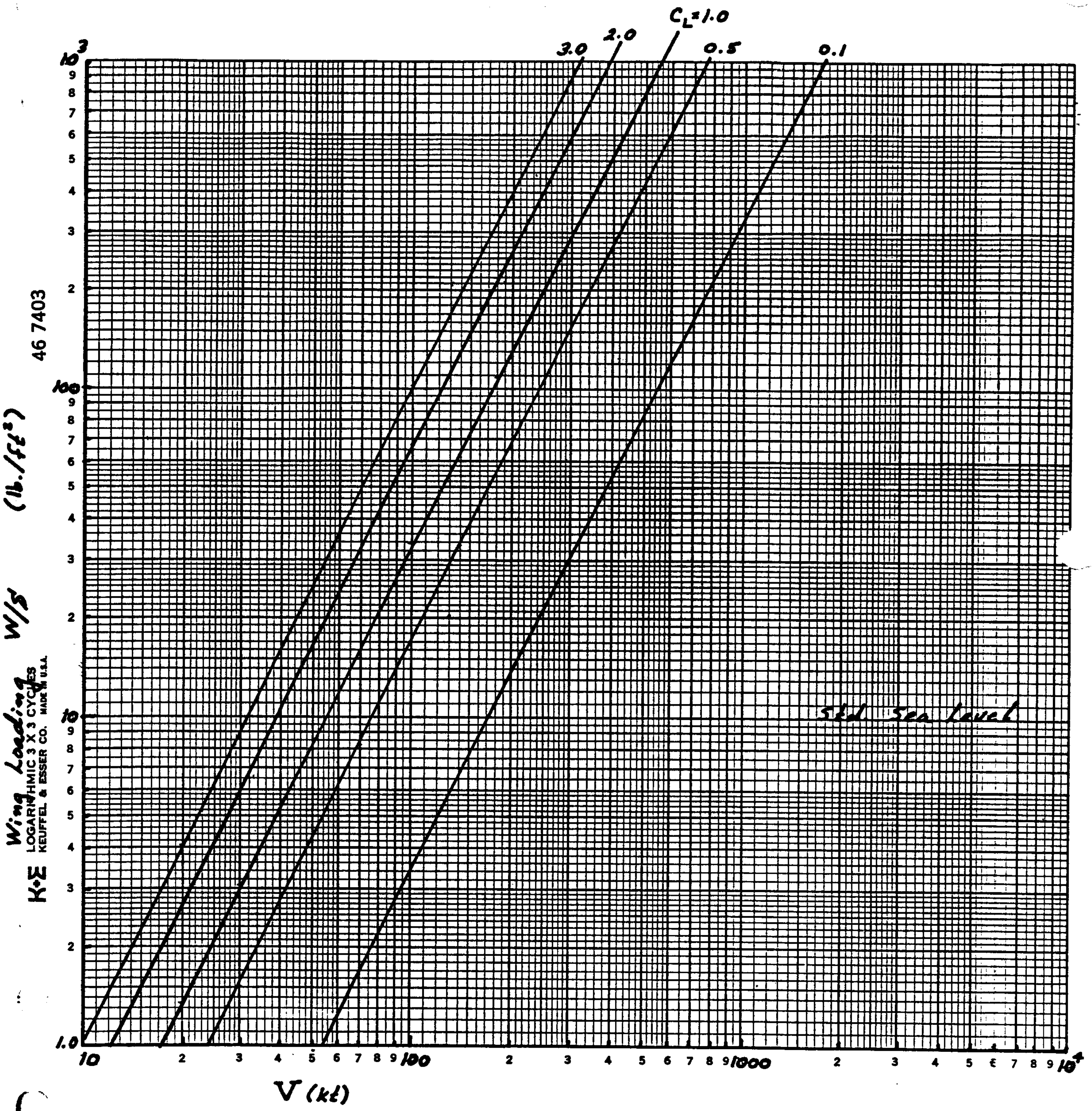
Figure 5. Learjet Model 25 Profile Drag Variation with Lift

$M = 0.75$ $C_L = 0.336$ $C_D = 0.0338$

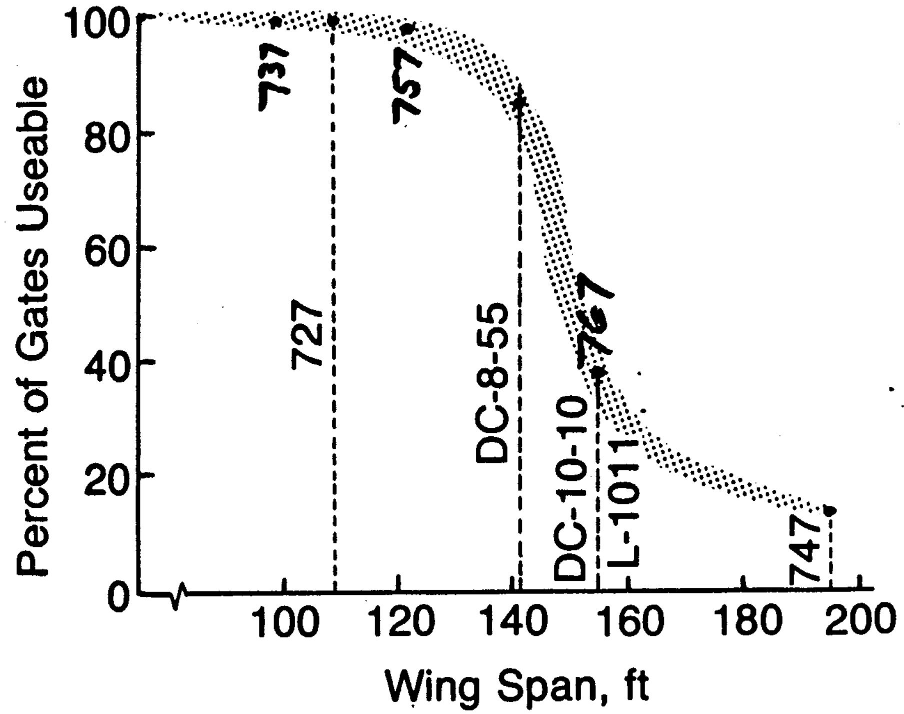
Source	ΔC_D	% of Total
Profile drag (skin friction)	.0180	53.25
Profile drag variation with lift	.0007	2.07
Interference drag	.0031	9.17
Roughness and gap drag	.0015	4.44
Induced drag	.0072	21.30
Compressibility drag	.0028	8.28
Trim drag	.0005	1.48
TOTAL	0.0338	100.00

Figure 6. Cruise Drag Breakdown

$$W/S = \frac{\rho}{2} C_L V^2 \cong 0.00338\, C_L\, V_{kt}^2$$
at std. sea level



Major Hub Gate Compatibility



6.06

HIGH LIFT SYSTEM DEVELOPMENT FOR TRANSPORT AIRCRAFT

JERRY L. LUNDRY

THE BOEING COMPANY
BCAC AERODYNAMICS RESEARCH GROUP

Presented At A Short Course In

HIGH LIFT TECHNOLOGY

At The University of Tennessee Space Institute
Tullahoma, Tennessee

March 6-10, 1978

ABSTRACT

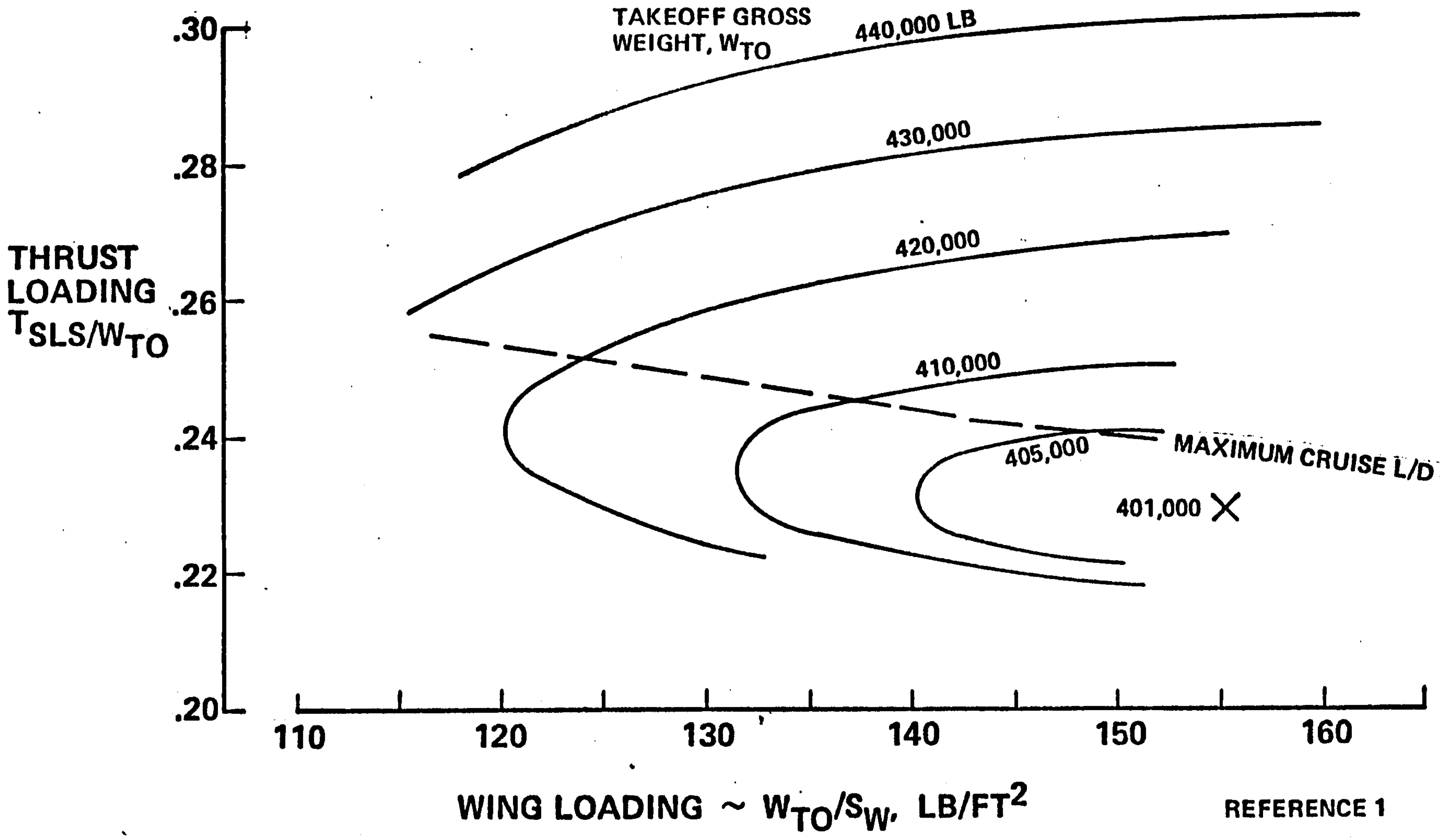
A brief overview of high lift system design requirements and constraints for transport type aircraft is presented. Typical high lift system characteristics including mechanical, boundary layer control, and powered lift types are reviewed.

A more detailed discussion is then given concerning the development of Upper Surface Blowing powered lift including theoretical and experiment results. Specific examples are taken from design data for the USAF/Boeing YC-14 prototype. The presentation closes with a movie showing flight characteristics of the YC-14.

6.07

CRUISE PERFORMANCE

PAYLOAD = 42,000 LB
RANGE = 5700 NM
MACH NO. = 0.815 (LRC)

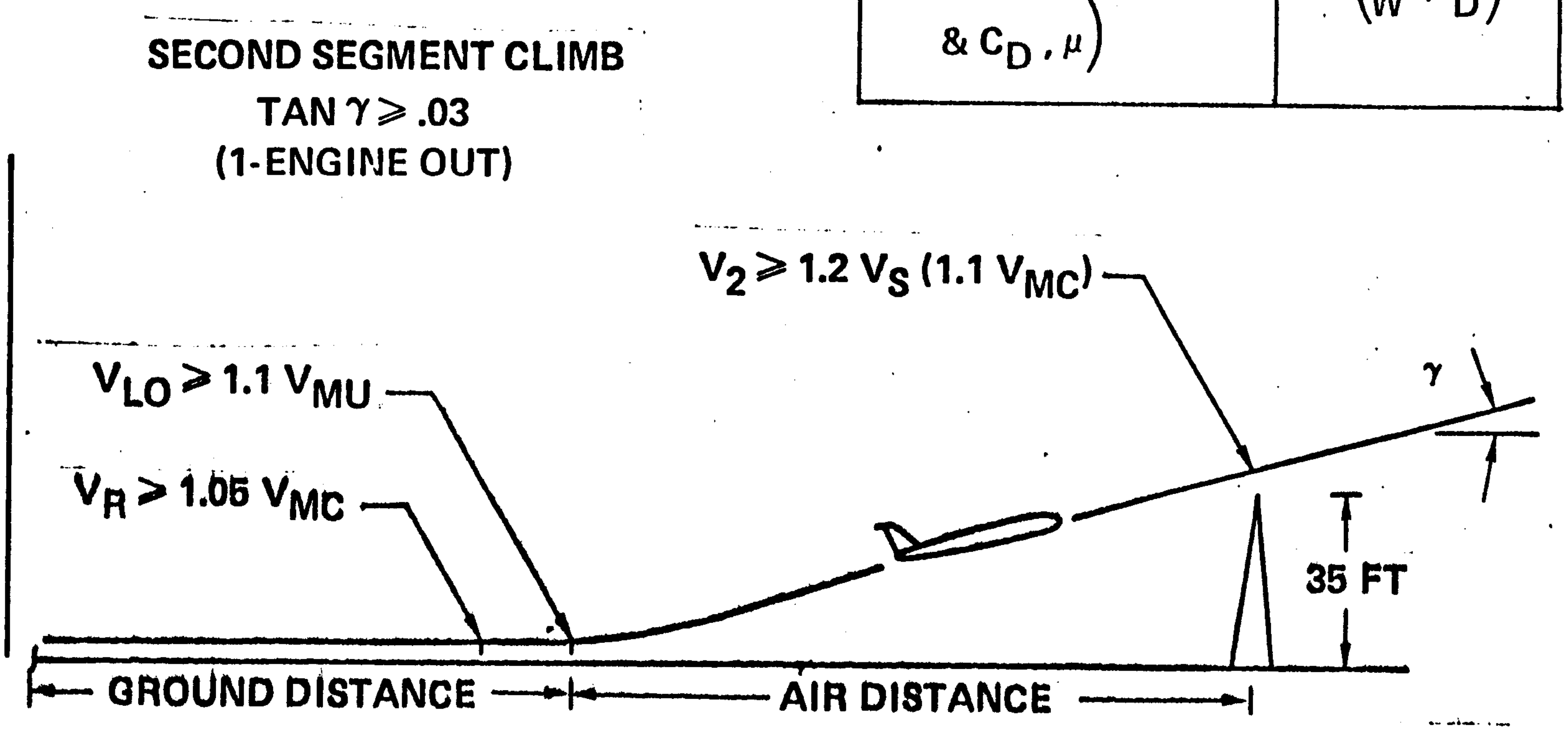


REFERENCE 1

6.08

TAKEOFF PROFILE

GROUND RUN	CLIMBOUT
$f\left(\frac{T}{W}, \frac{W}{S}, C_{L_{MAX}} \ \& \ C_D, \mu\right)$	$f\left(\frac{T}{W}, \frac{L}{D}\right)$



REFERENCE 1

6.09

LANDING PROFILE

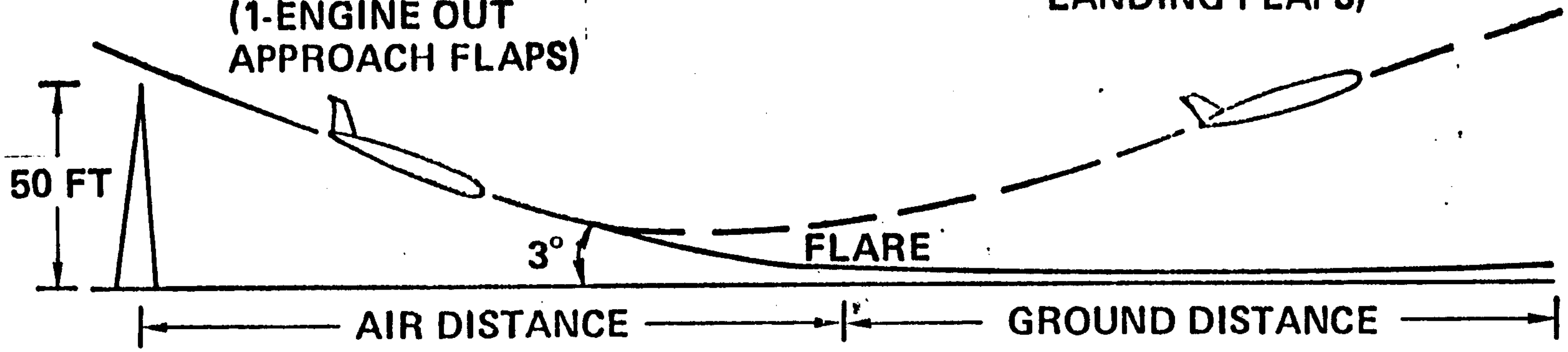
APPROACH	GROUND RUN	GO-AROUND
$f(C_{L_{MAX}}, \frac{W}{S}$ & $L/D)$	$f(C_{L_{MAX}}, W/S$ & $C_D, \mu, T_{REV})$	$f\left(\frac{T}{W}, \frac{L}{D}\right)$

APPROACH

$V_{APP} = 1.3\, V_S$
TAN $\gamma_1 \geq .027$
(1-ENGINE OUT APPROACH FLAPS)

GO-AROUND

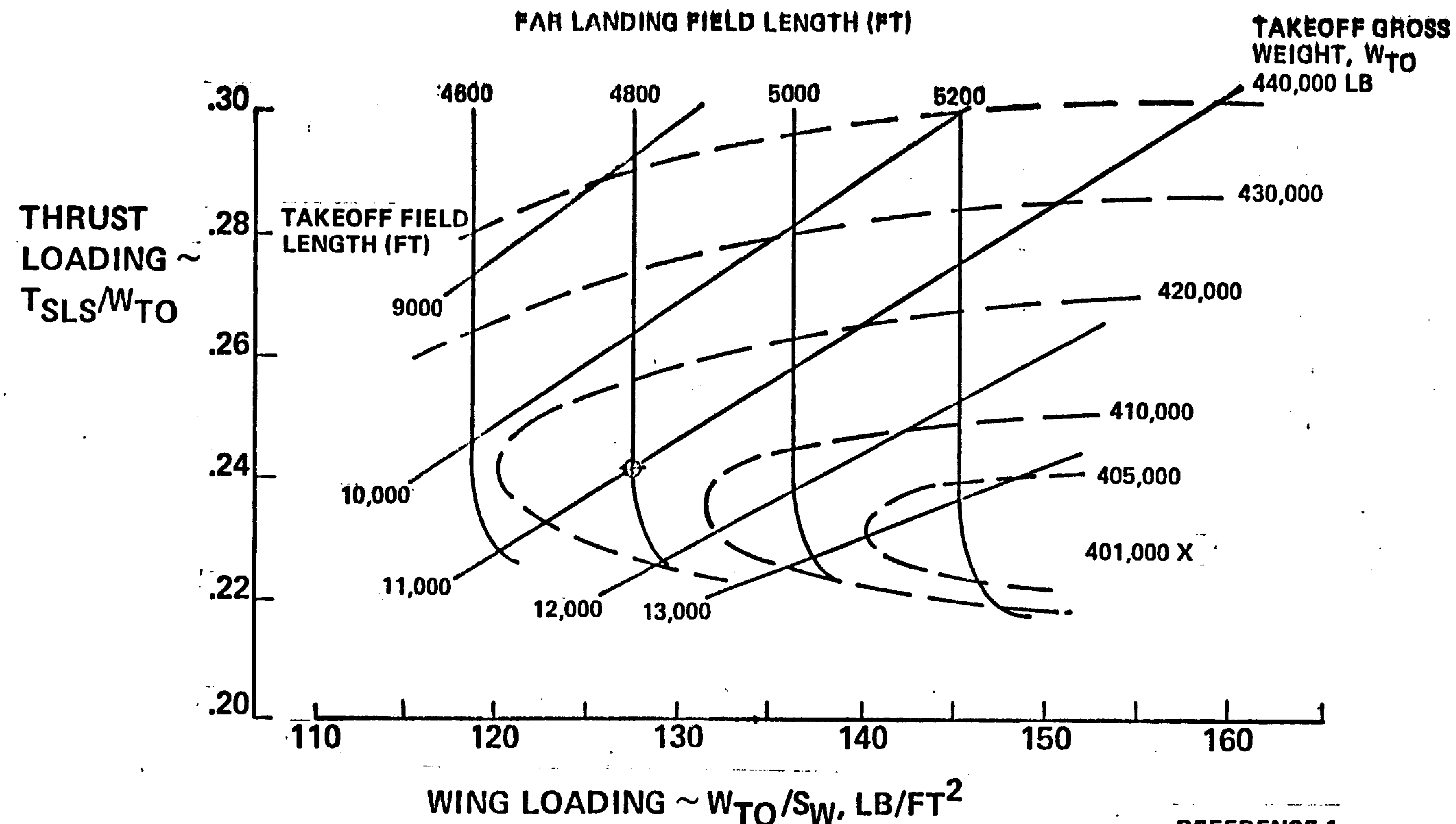
TAN $\gamma_2 \geq .032$
(ALL ENGINE LANDING FLAPS)



REFERENCE 1

LOW-SPEED PERFORMANCE

PAYLOAD = 42,000 LB
RANGE = 5700 NM
MACH NO. = 0.815 (LRC)



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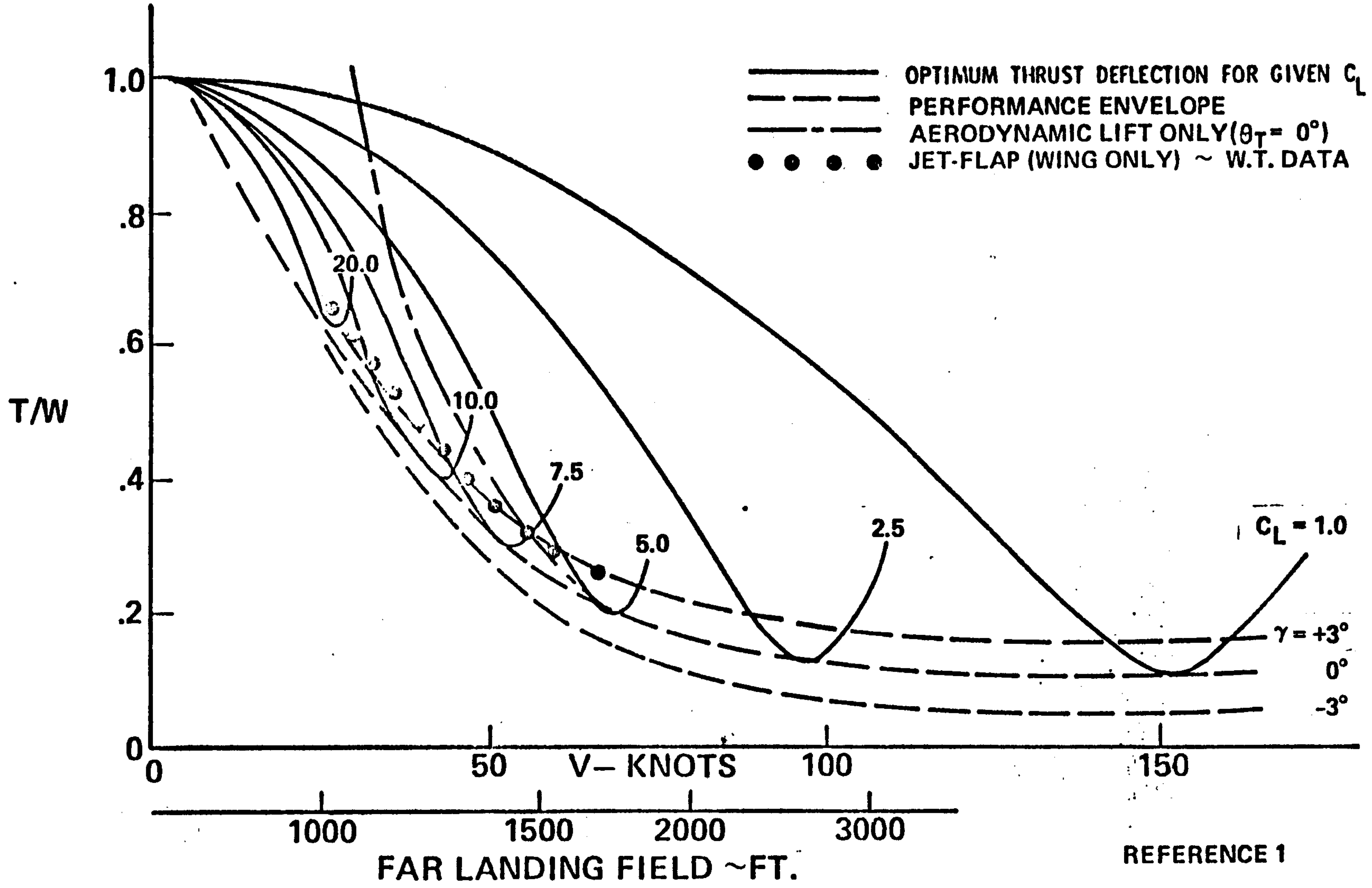
6.11

STOL LANDING PERFORMANCE. WING LOADING = 80 PSF



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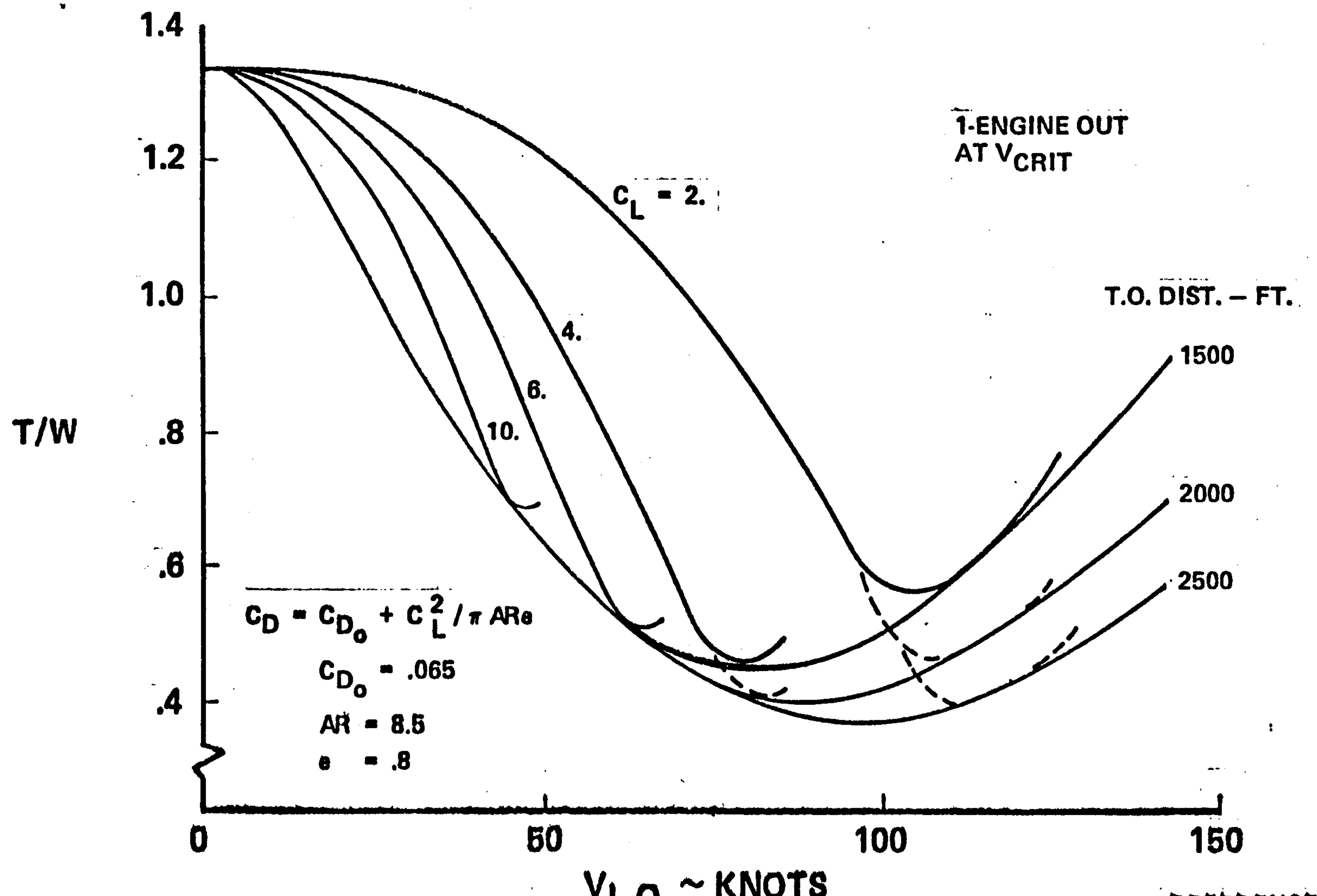
STOL TAKEOFF PERFORMANCE. WING LOADING = 90 PSF



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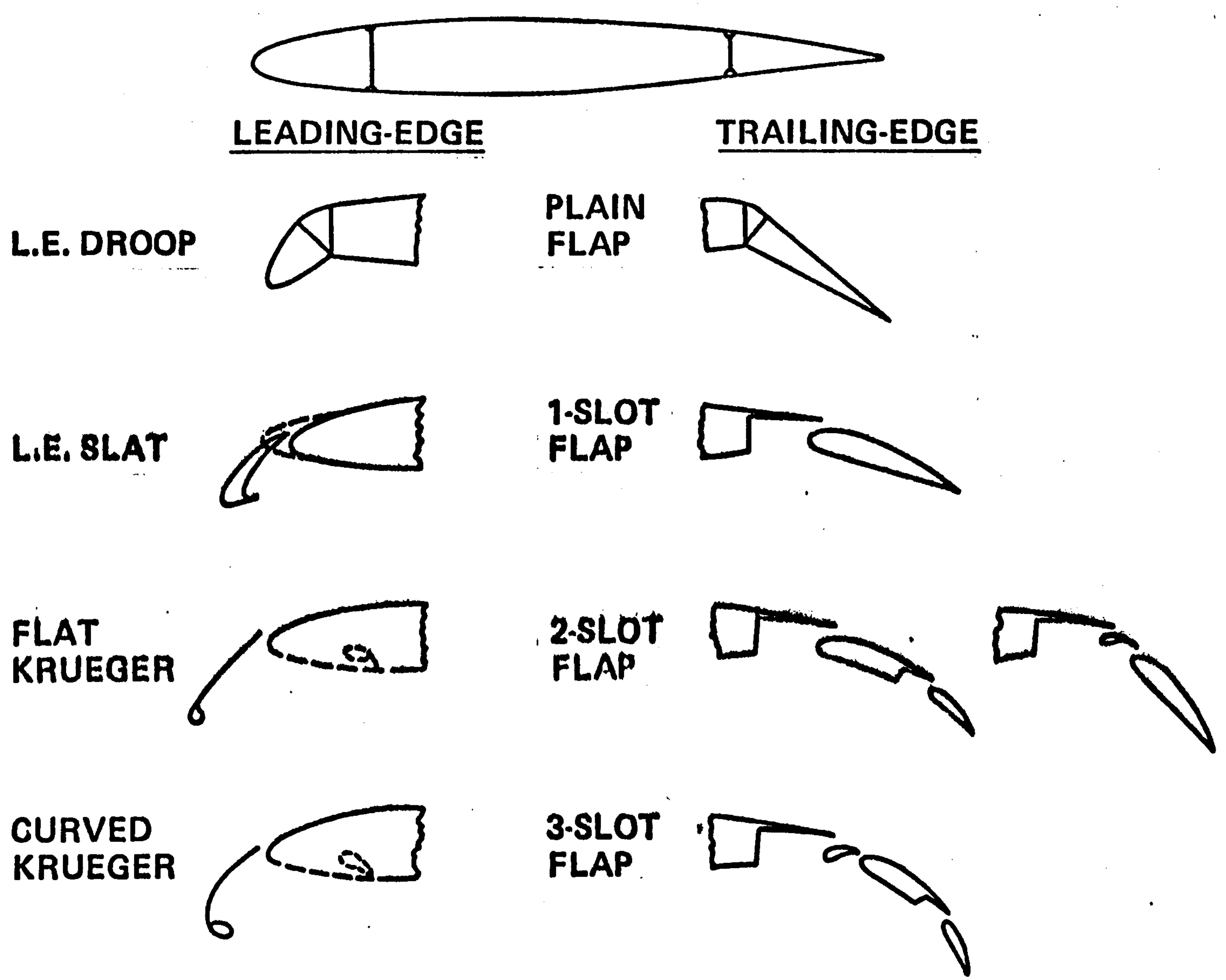
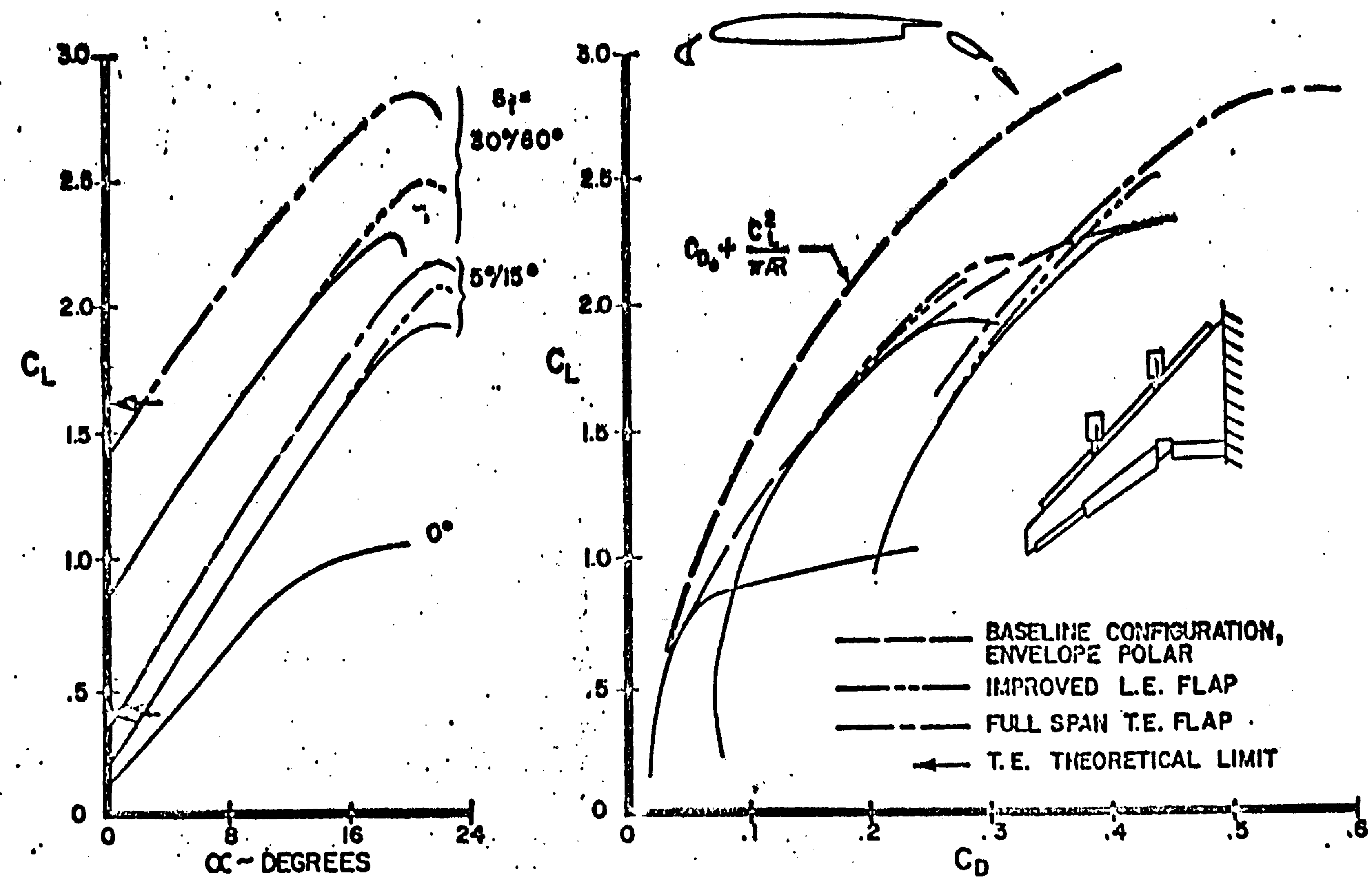
6.13

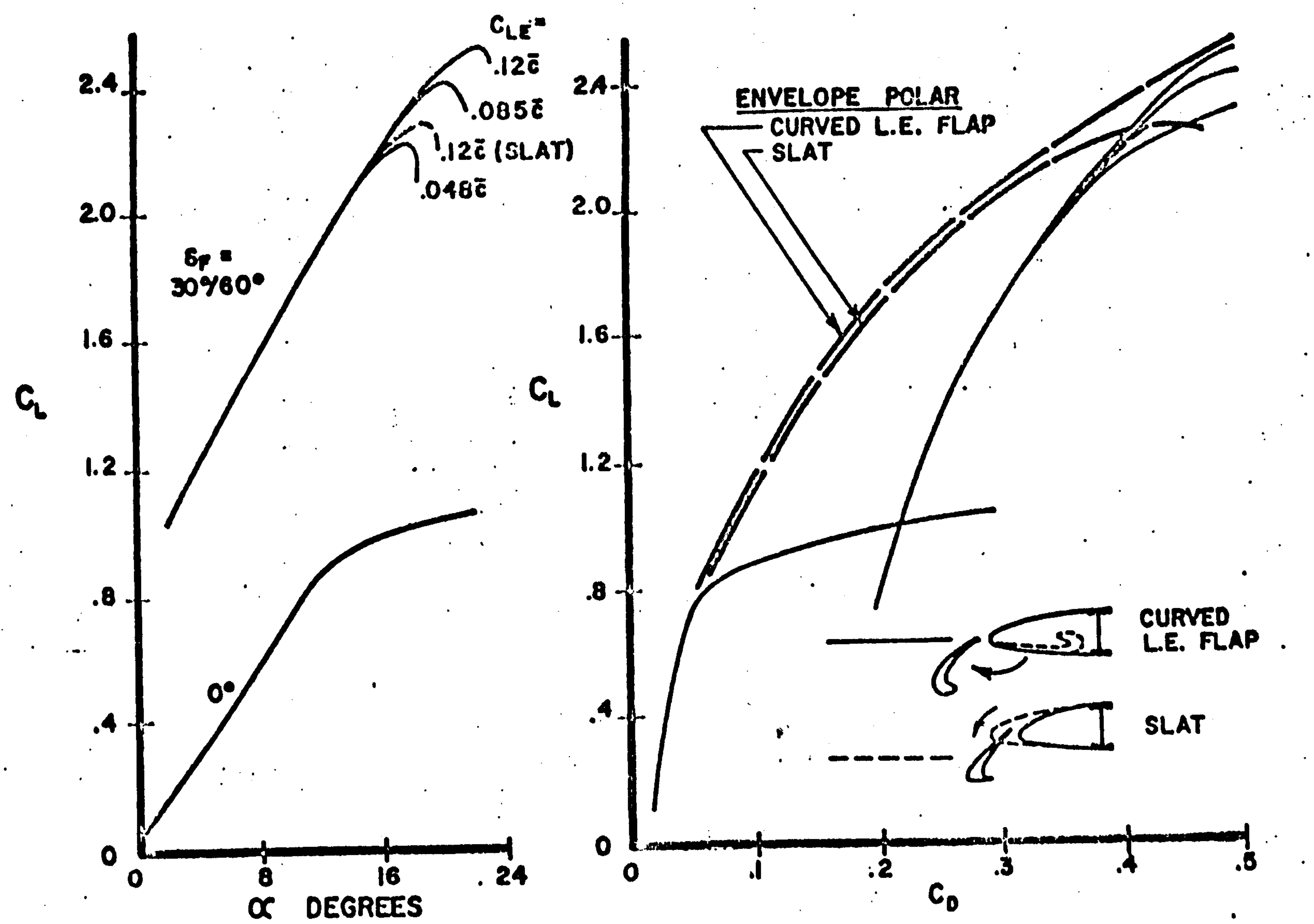
MECHANICAL HIGH LIFT DEVICES



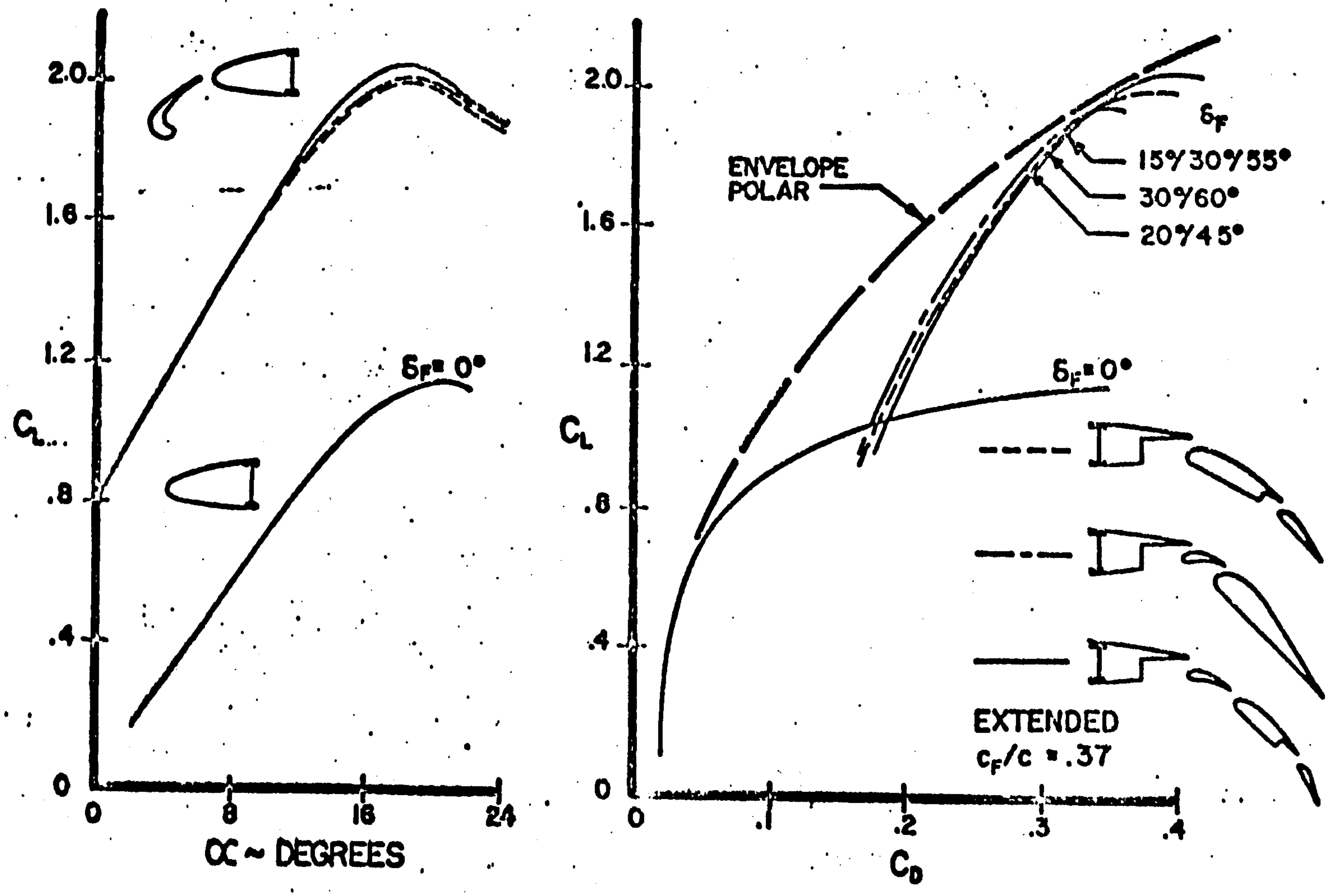
LOW SPEED DRAG POLAR



LEADING-EDGE EFFECTS

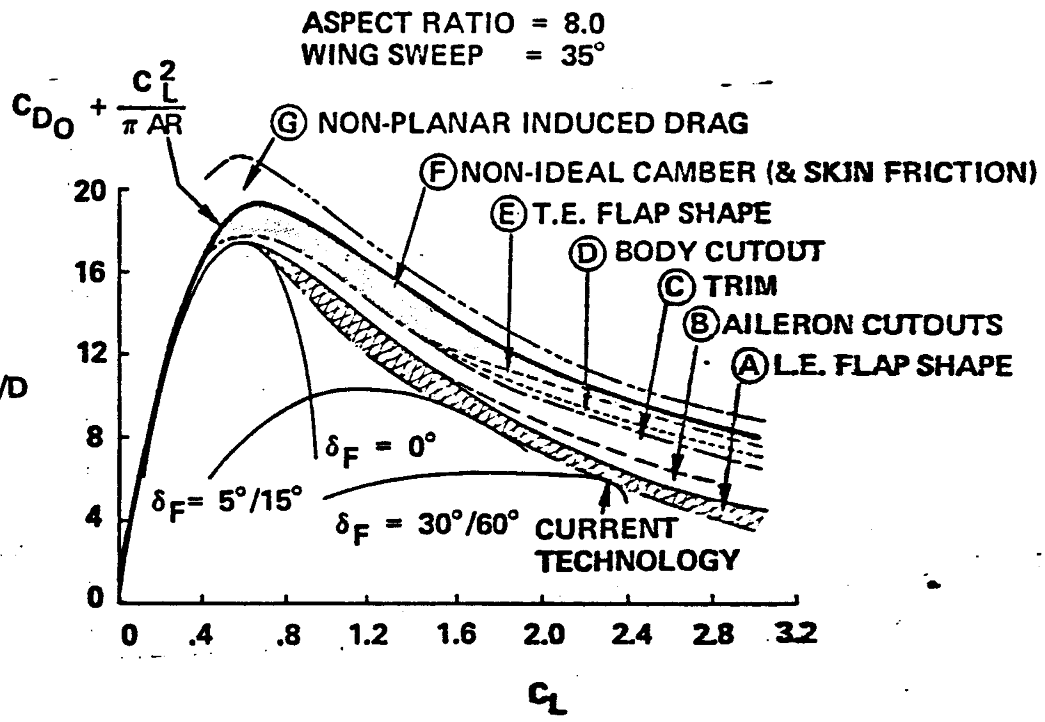


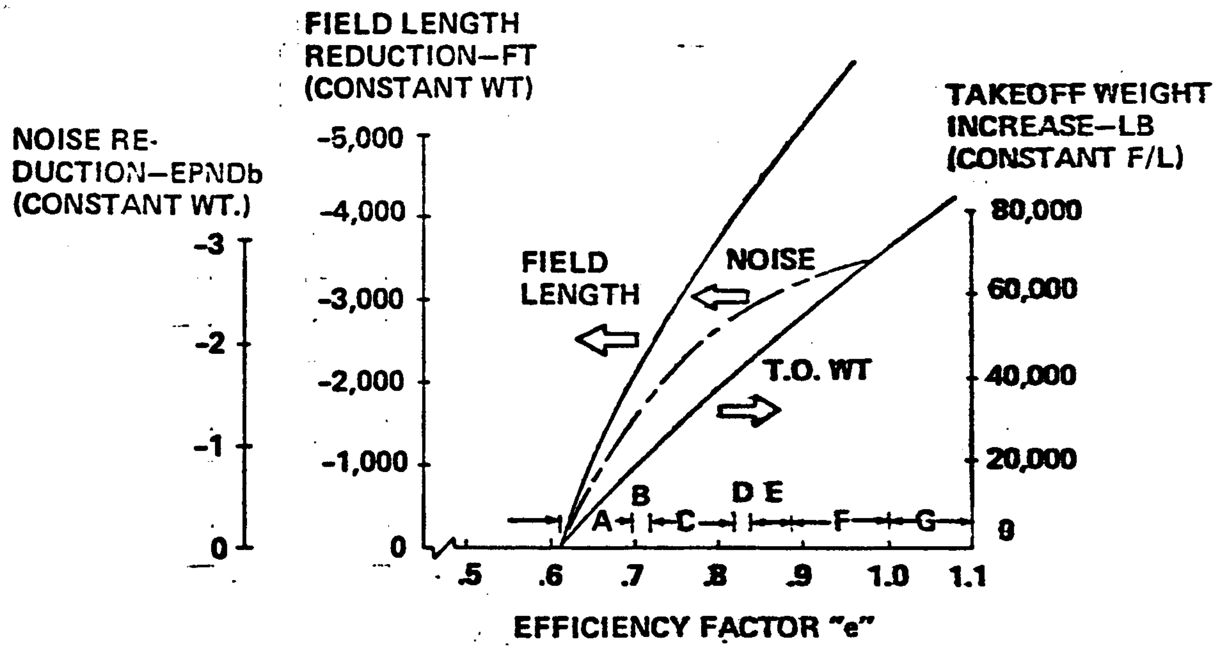
TRAILING-EDGE EFFECTS



6.17

MECHANICAL HI-LIFT POTENTIAL PERFORMANCE IMPROVEMENTS



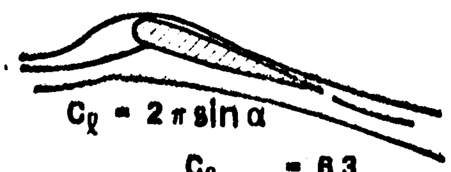


REFERENCE 1

6.18

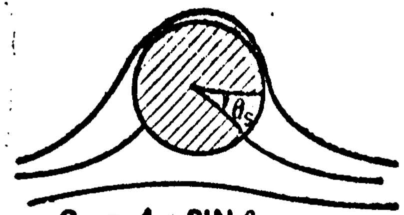
$C_{L_{MAX}}$ "LIMITS"

2-D THEORETICAL



$C_\ell = 2\pi \sin\alpha$

$C_{\ell_{max}} = 6.3$



$C_\ell = 4\pi \sin\theta_s$

$C_{\ell_{max}} = 12.6$ BASED ON DIAMETER

$C_{\ell_{max}} \simeq 4$ BASED ON CHORD

3-D THEORETICAL

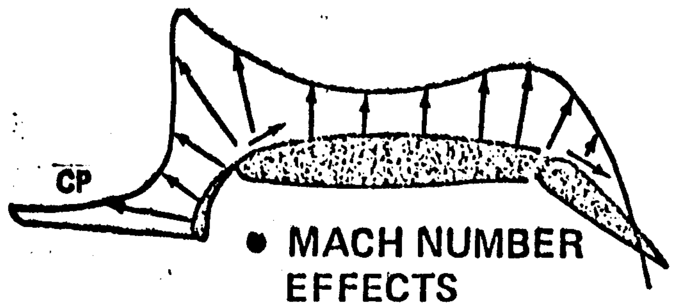
$C_{L_{max}}$ = .855 X AR (LOW AR) ⎫ = 1.21 X AR (HIGH AR) ⎭ • NON-LINEAR* WING THEORY
= 1.57 X AR • MOMENTUM THEORY*
= 1.92 X AR • HELMBOLD'S VORTEX ROLL-UP THEORY**

AR = ASPECT RATIO

*REFERENCE 2
**REFERENCE 3

PRACTICAL

- BOUNDARY LAYER SEPARATION EFFECTS

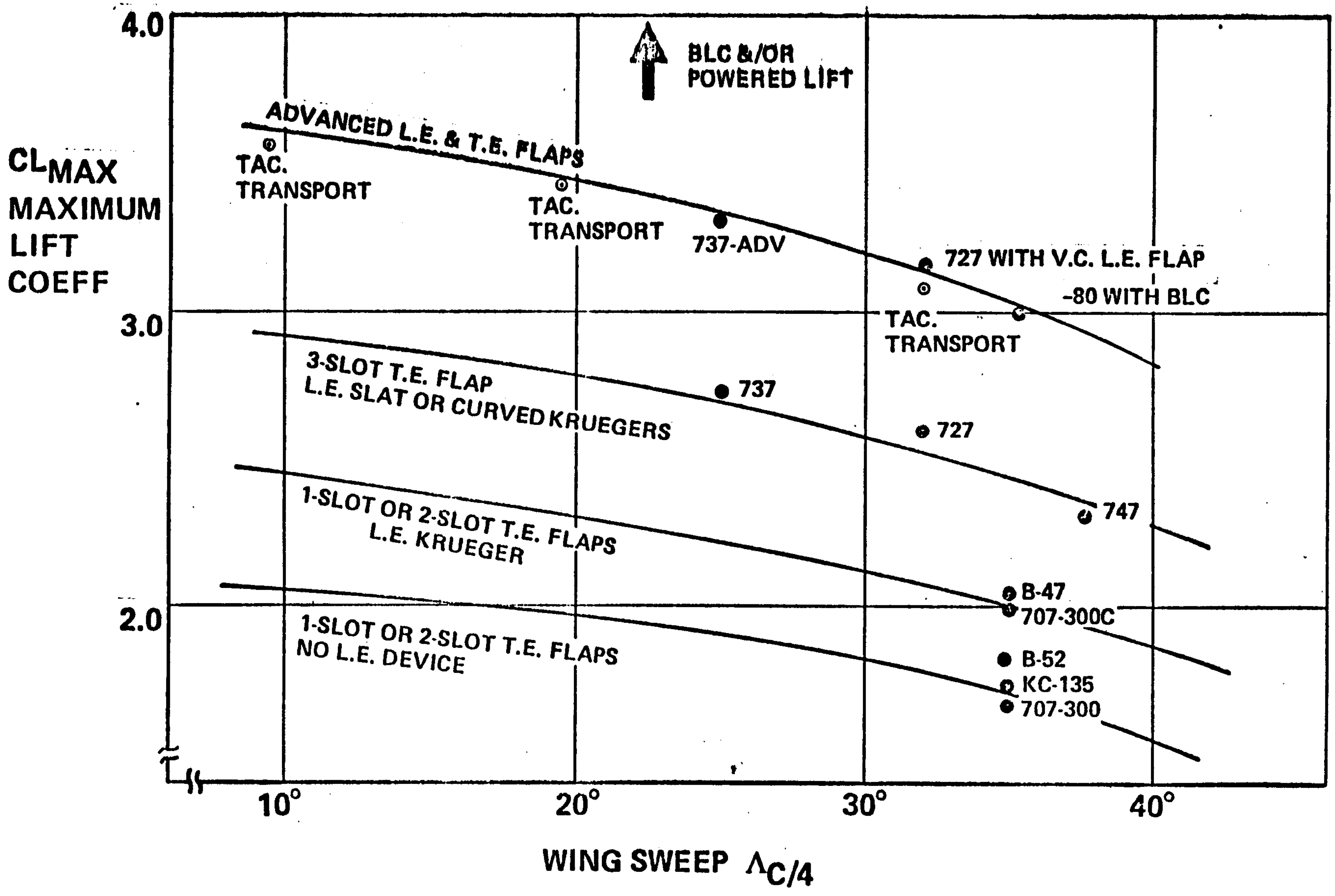


- MACH NUMBER EFFECTS

- MECHANICAL DESIGN & STRUCTURAL LIMITATIONS

6.19

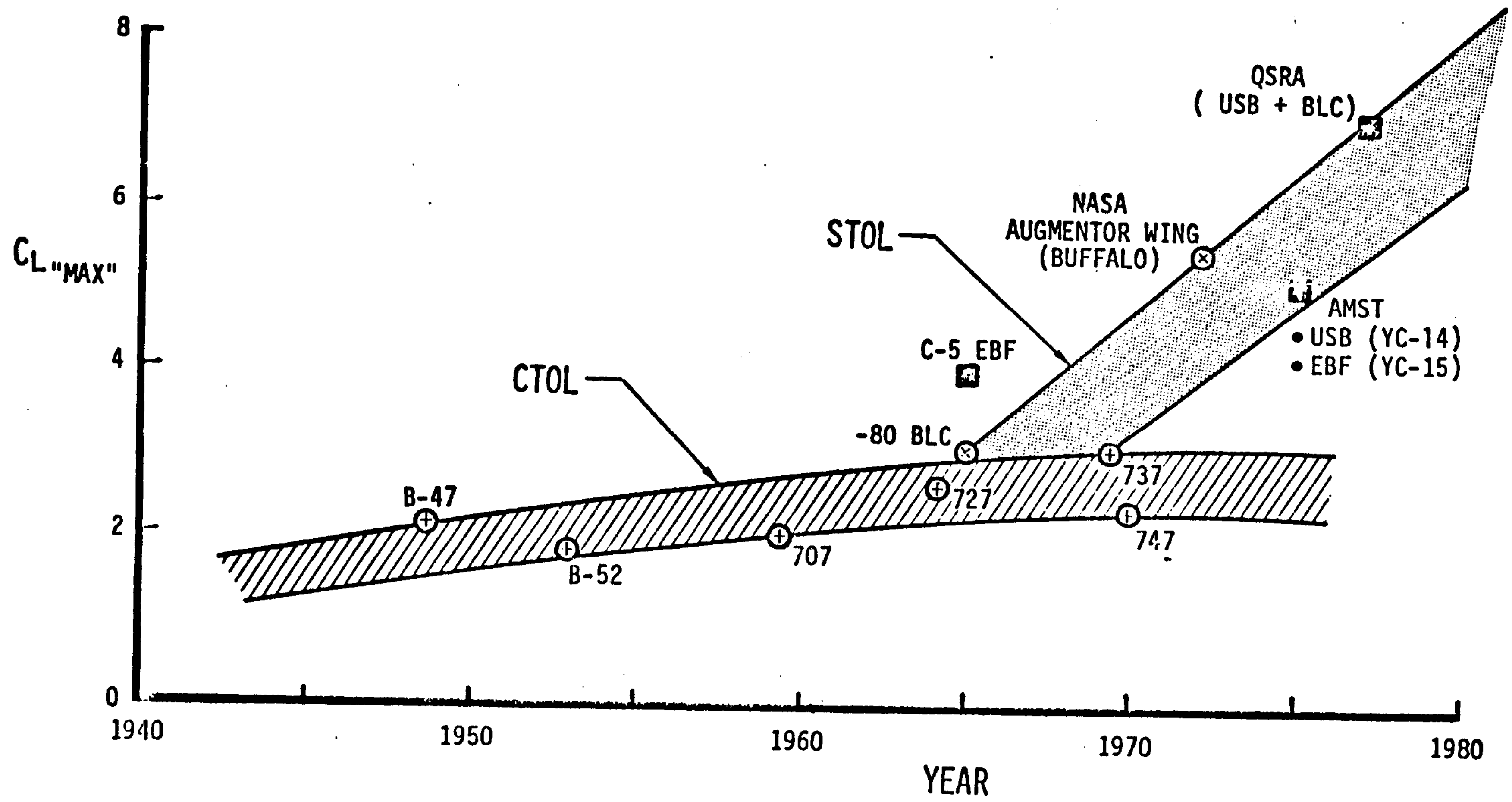
DEMONSTRATED $C_{L_{MAX}}$ LEVELS



6.20

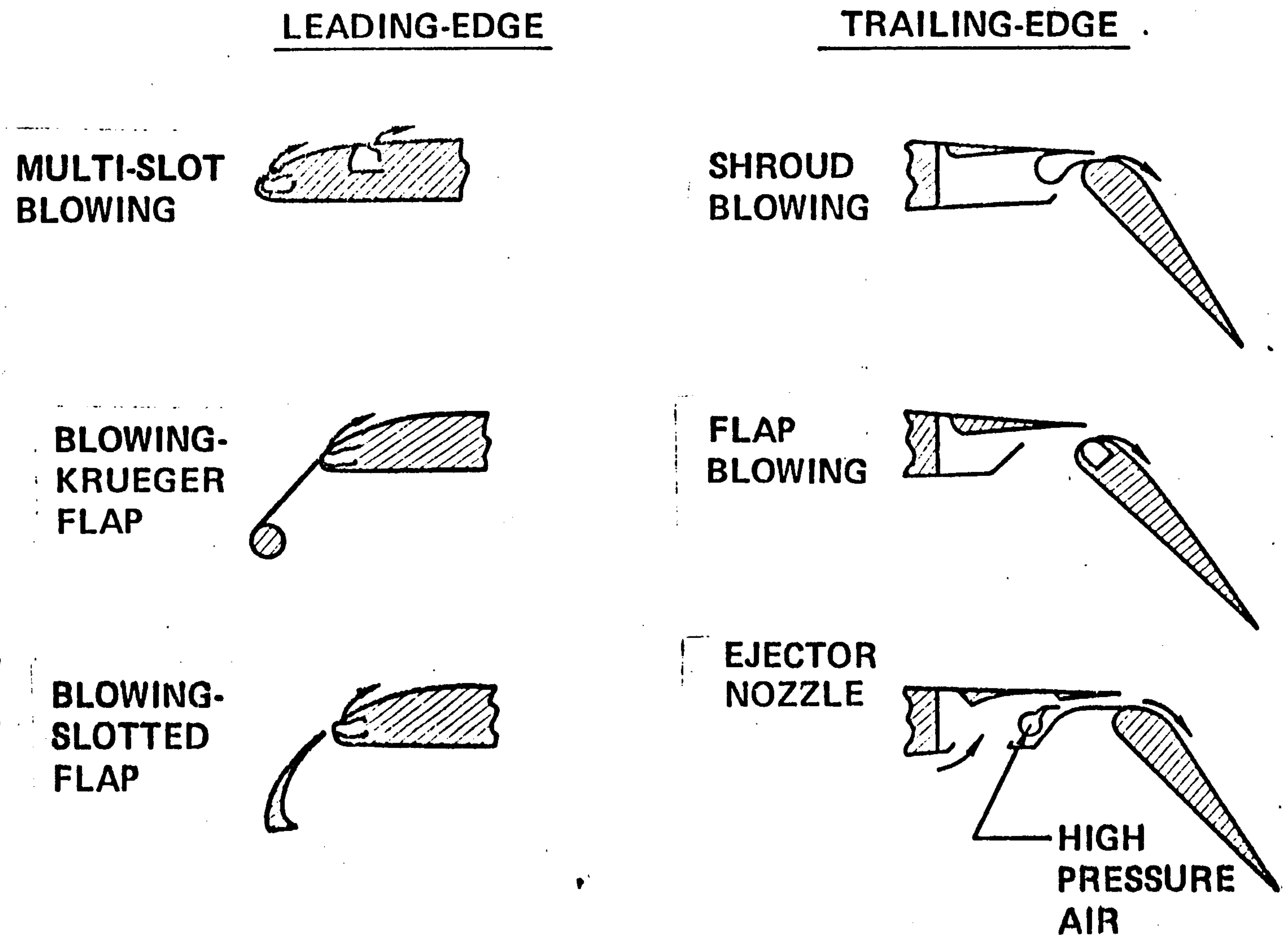
237-7417

$C_{L_{MAX}}$ TRENDS



6.20 A

BLOWING BLC CONCEPTS

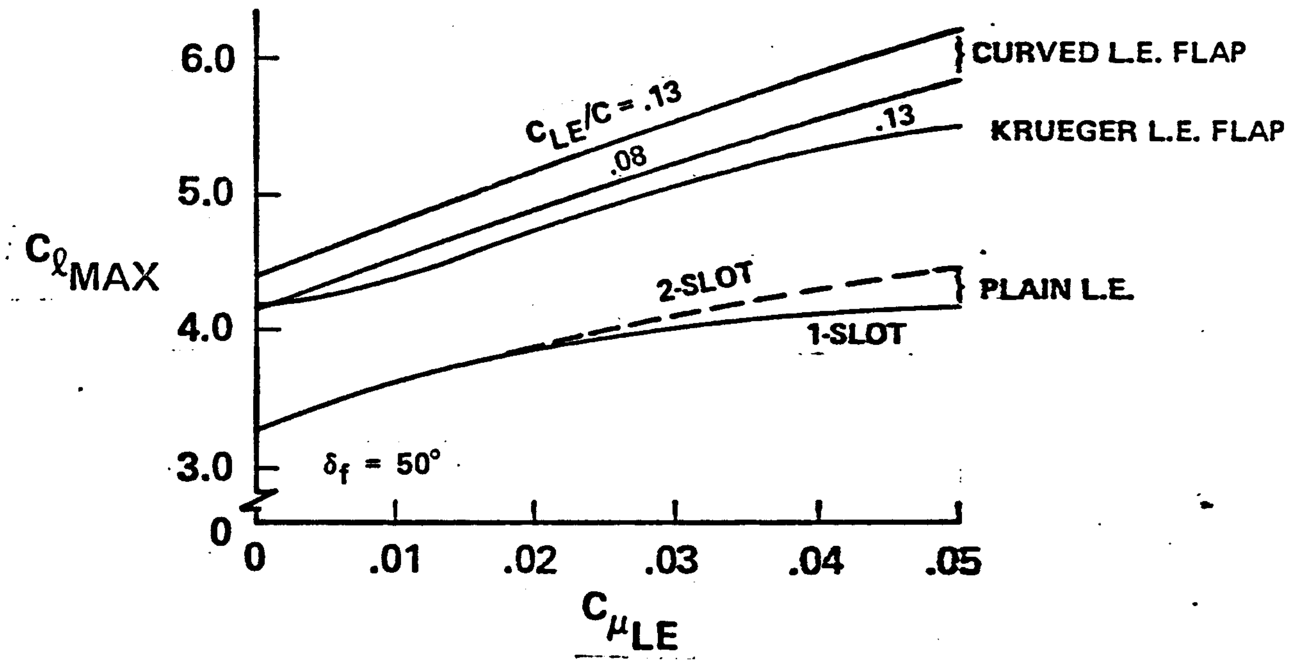


REFERENCE 1

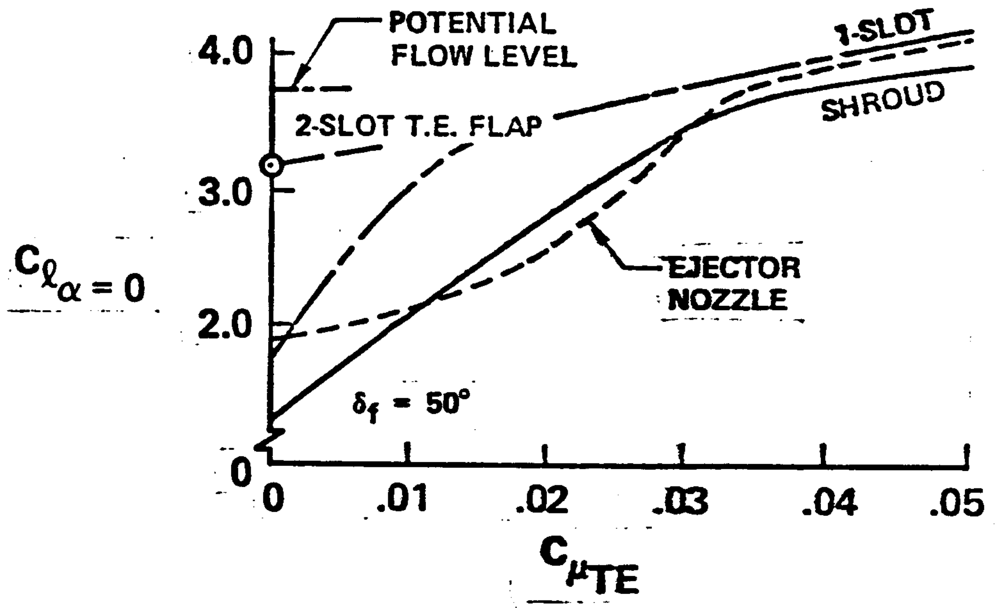
6.21

BLOWING BLC PERFORMANCE

L.E. BLC



T.E. BLC

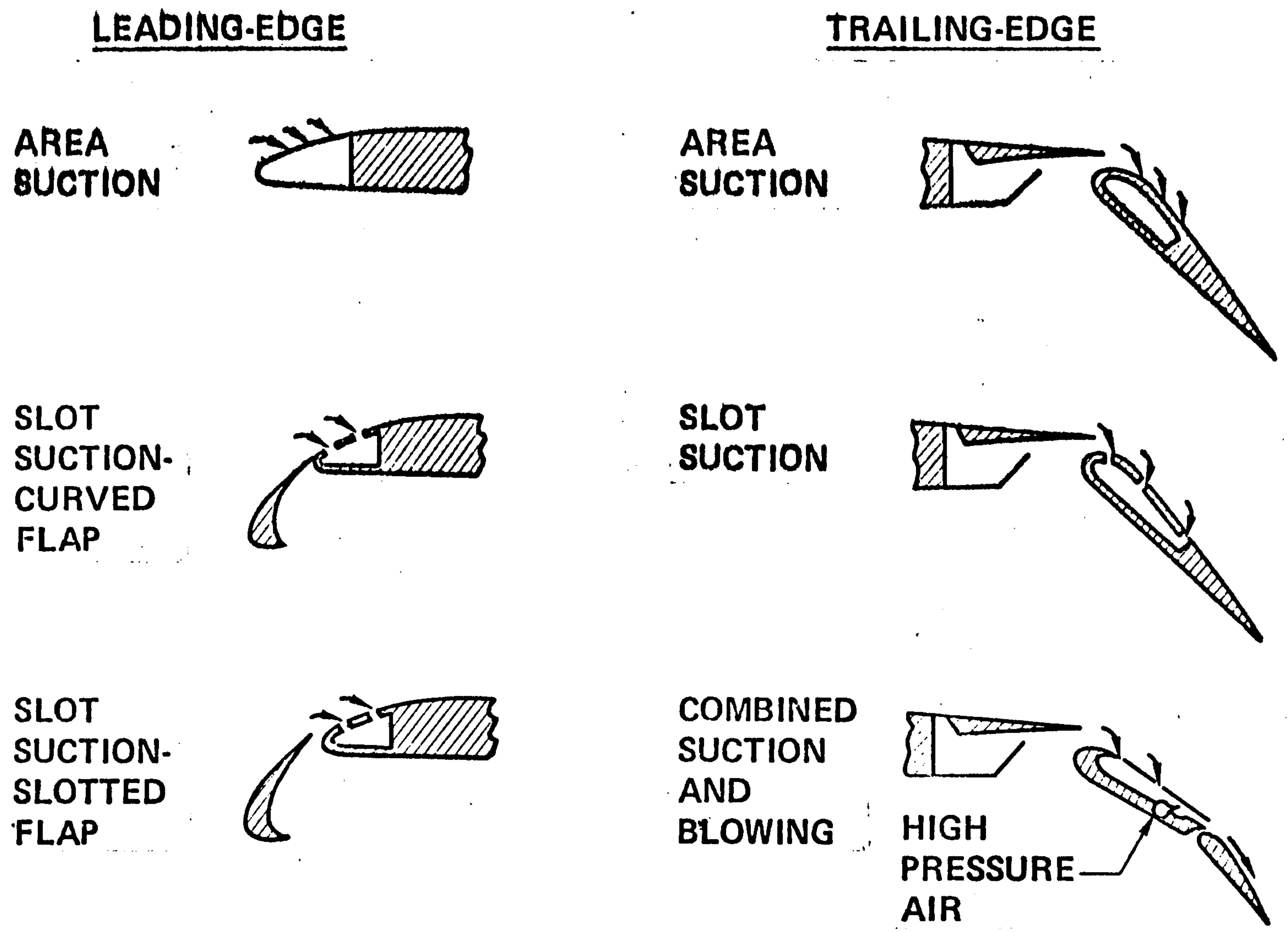


2-D W.T. DATA M = .165

REFERENCE 1

6.22

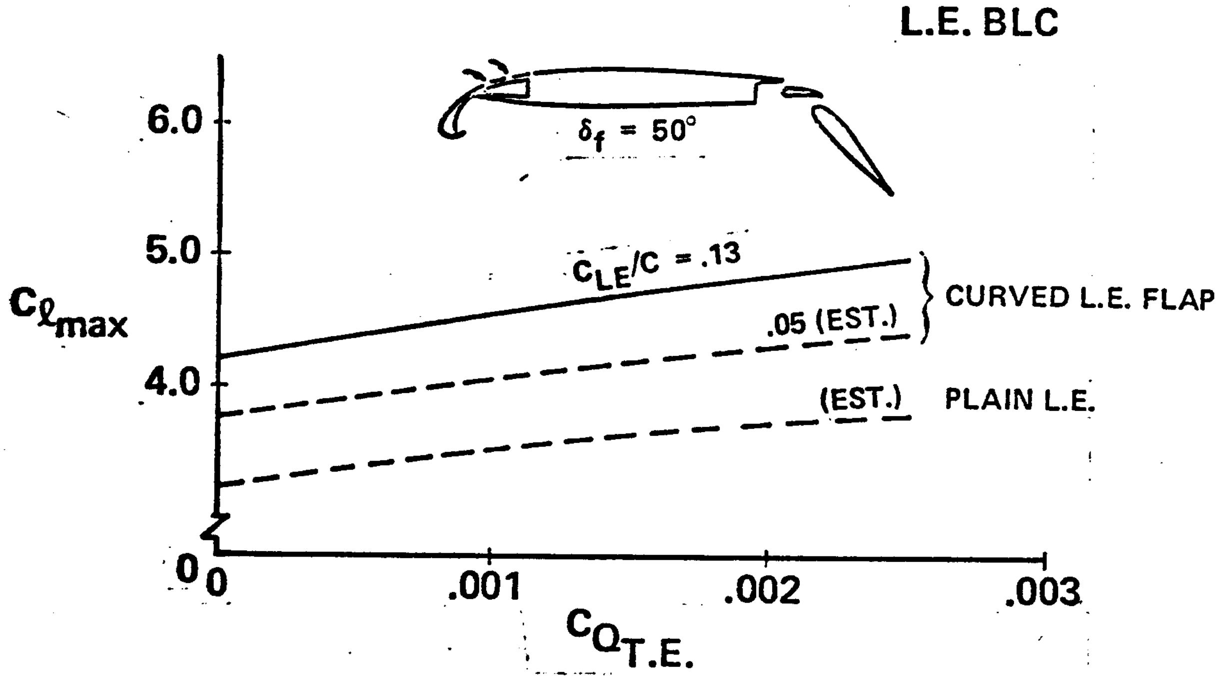
SUCTION BLC CONCEPTS

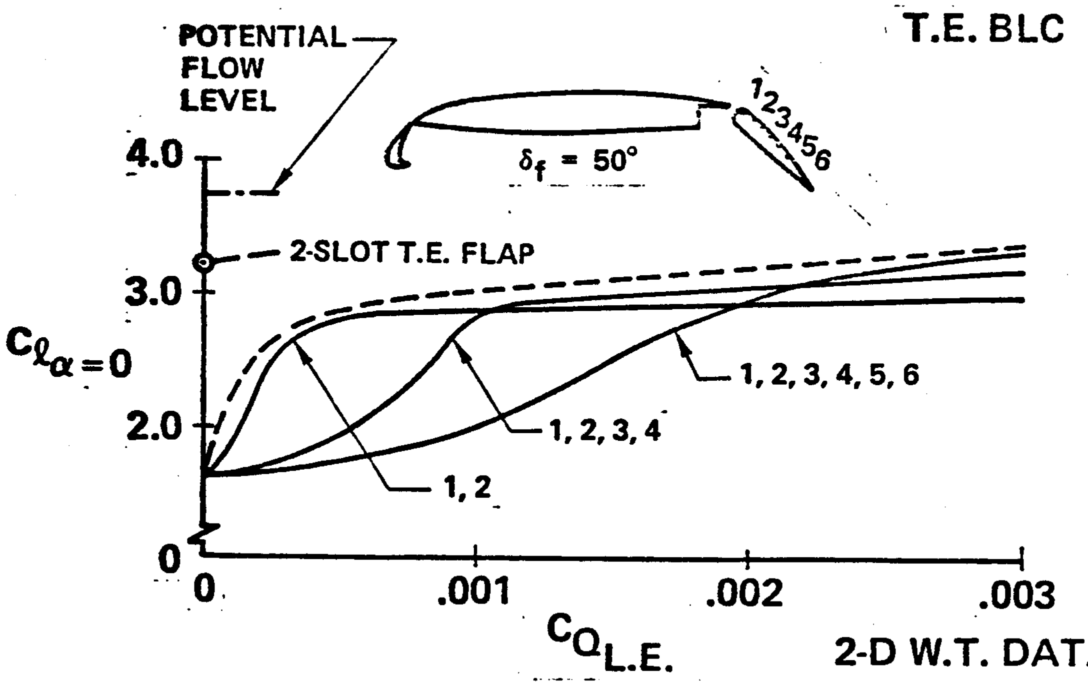


REFERENCE 1

6.23

SUCTION BLC PERFORMANCE

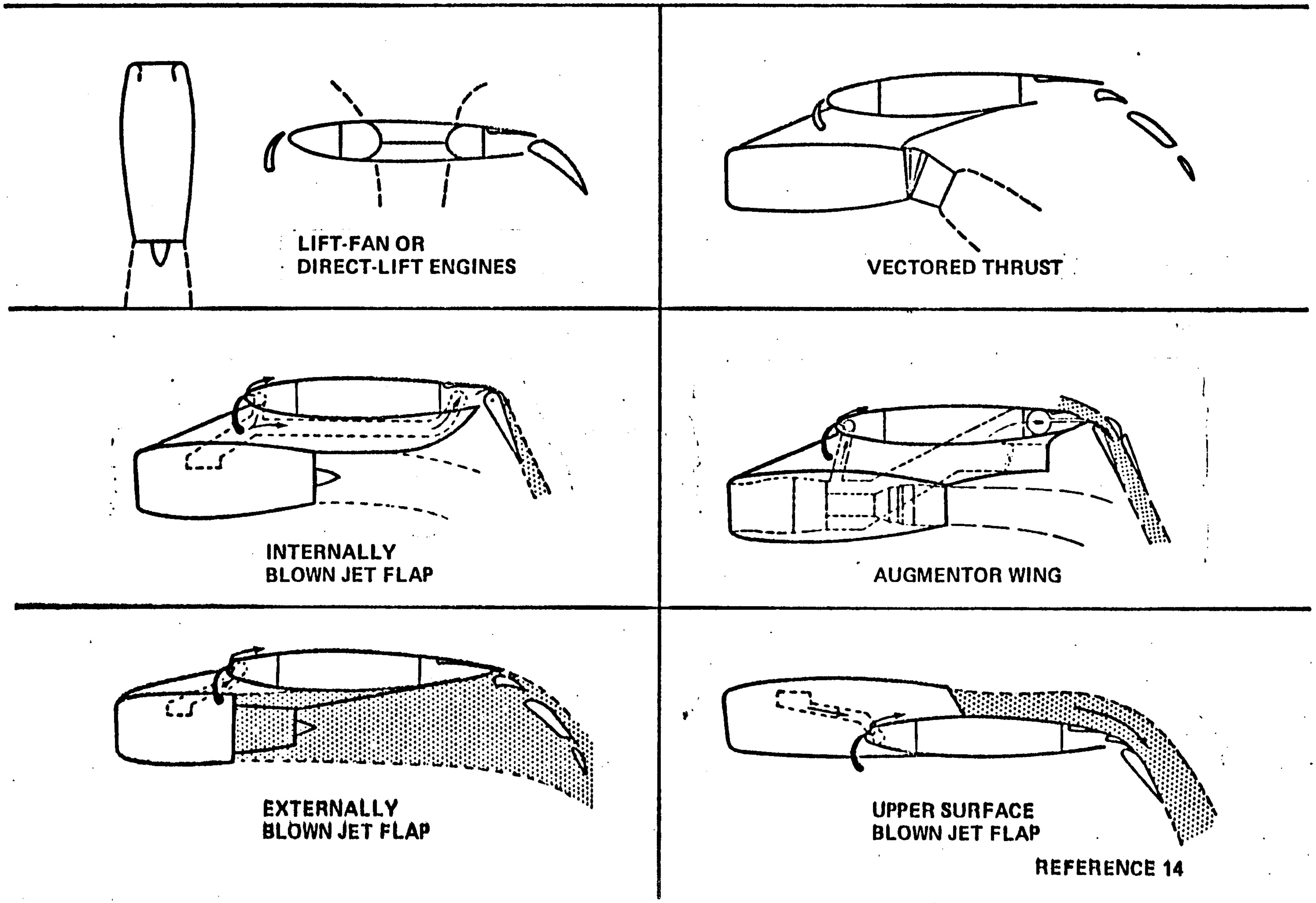




2-D W.T. DATA M = .165

REFERENCE 1

POWERED LIFT CONCEPTS



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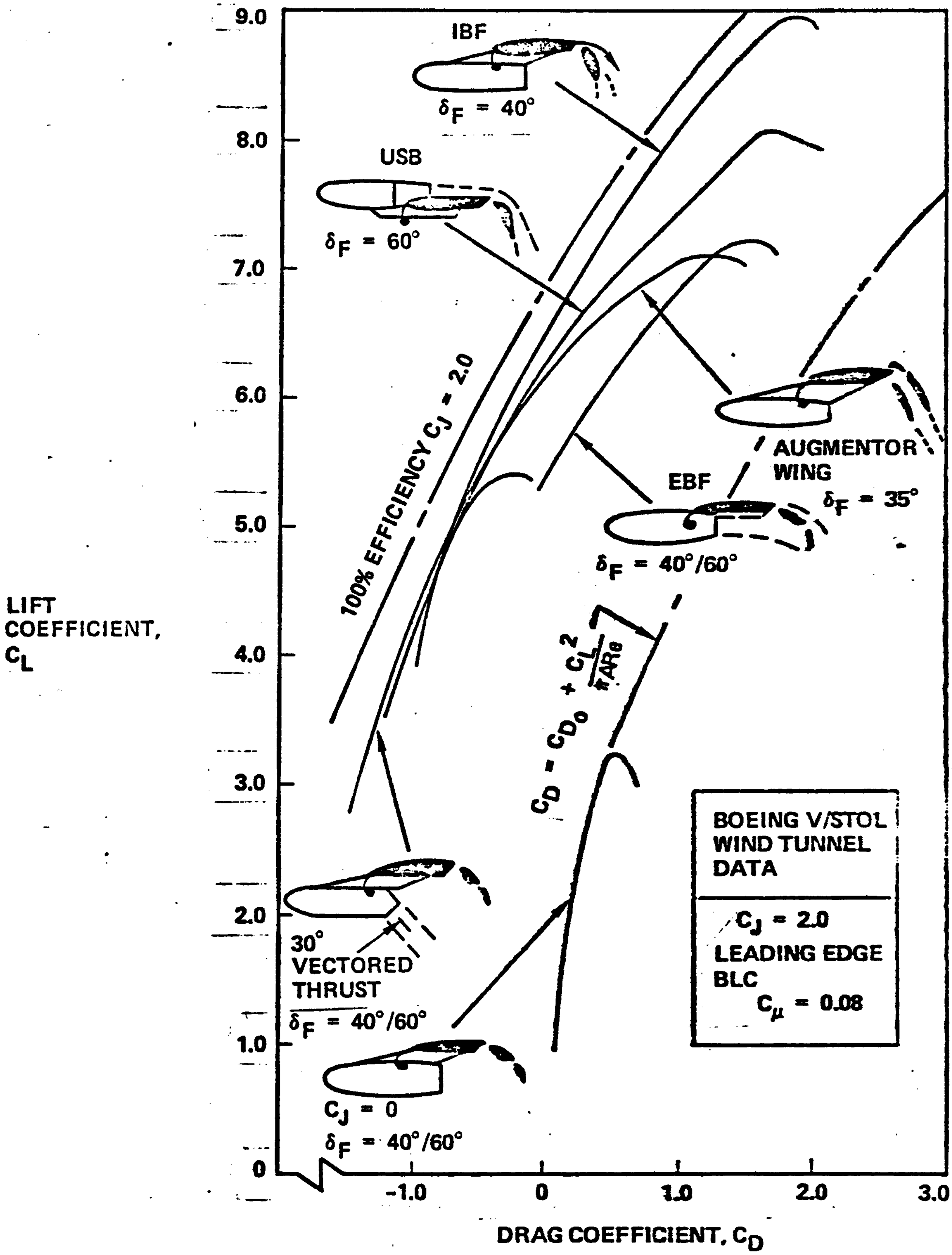
6.25

POWERED LIFT POLAR COMPARISON



REFERENCE 5 & 14

REFERENCE LIST

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AIAA'83

AIAA-83-1845
A Method for Predicting Low-Speed Aerodynamic Characteristics of Transport Aircraft

L. E. Murillo and J. H. McMasters
Boeing Commercial Airplane Co., Seattle, WA

AIAA Applied Aerodynamics Conference

July 13-15, 1983/Danvers, Massachusetts

6-28

A Method for Predicting Low-Speed Aerodynamic Characteristics of Transport Aircraft

by

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and
John H. McMasters**

Boeing Commercial Airplane Company
Seattle, Washington

Abstract

A preliminary design level methodology for predicting the global aerodynamic characteristics of transport aircraft in low-speed/high-lift configurations has been developed, based on recent advances in computational aerodynamics analysis methods. The new method involves two economical, user oriented computer programs. One, an advanced lifting surface theory for the potential flow analysis of swept-wing/body combinations with multielement high-lift devices, provides the basic theoretical structure. The second program combines the potential flow analysis results with available data from previous airplane models to predict the performance of new designs. The overall procedure is highly automated and produces generally satisfactory results for preliminary design purposes. Example results based on recent transport aircraft wind tunnel data are shown.

Nomenclature

AePP = Aerodynamic Prediction Program

AR = aspect ratio, b^2/S

b = wing span

c = wing chord

$\bar{c}$ = average wing chord S/b

C_D, C_L, C_M = configuration drag, lift and pitching moment coefficients, force/qS and moment/$qS\bar{c}$

DVM = distributed vorticity method

G, H = scaling factors (see eqn. 3)

M = Mach number

q = dynamic pressure

Re = Reynolds number

S = wing area (high-lift devices retracted)

S_f = flap area

x = longitudinal coordinate

y = lateral coordinate

α = angle of attack

Δ = differences (residuals)

δ_f = flap deflection

η = nondimensional spanwise wing station, 2y/b

Λ = sweep angle measured at wing quarter chord

Subscripts

a = baseline (input) configuration

b = configuration whose characteristics are to be predicted (output)

dvm = distributed vorticity method (potential flow lifting surface theory)

eff = "effective" viscous condition

exp = experimental (e.g., wind tunnel) value

f = flap

geo = geometric value

max = maximum

min = minimum

o = reference value (high-lift system retracted)

visc = viscous

Superscript

^ = adjusted or scaled quantity (Fig. 6)

* = critical section

Introduction

Considerable progress has been made in recent years in the development of computational methods for both the analysis and design of transport aircraft in low-speed/high-lift configurations. The development and applications of several of these computational tools to practical project level high-lift

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system design problems at Boeing have been described recently (Refs. 1, 2 and 3). The applications discussed in references 2 and 3 are typical of those that might be encountered in the detailed design phase of an overall configuration aerodynamic development process. As shown in the overview diagram (Fig. 1), at the detail design level, computational methods intended to complement extensive testing must be highly accurate. Thus costs may be high, although fully justified if an enhanced (compared with traditional cut-and-try) design process results.

In a preceding preliminary design phase of aerodynamic configuration development, computational methods are also of major importance. In this case, however, where many continually changing configuration variables must be considered and their effects on the global aerodynamic characteristics readily evaluated, the conflicting requirements of computational accuracy and ease of use, rapidity of turnaround and low cost make the development of an appropriate computational methodology challenging.

It should be observed that the flow fields associated with the complex geometry of a modern transport aircraft during takeoff and landing approach are extremely complicated and may involve numerous vortical and partially separated flows even under normal operating conditions, as shown in Figure 2. No available computational methods are capable of analyzing the full range of such flows. Similarly, past handbook build-up methods offer little hope of predicting the global aerodynamic characteristics of such configurations to the levels of accuracy and reliability required for modern preliminary design purposes.

The need for modern predictive methodology appropriate to preliminary design level aerodynamic analyses (at a point in the design process where extensive wind tunnel testing cannot be justified) remains, however. Recognizing the limitations of existing theoretical tools, better computationally based predictive methodology can be devised if one accepts certain underlying assumptions. A description of such a method is the subject of this paper.

A Preliminary Design Level Prediction Method

The method devised to fill the block for a preliminary design level predictive tool in Figure 1 is semi-empirical and relies on two computer programs. The new method is made

Design Level	Accuracy Required	Turnaround Time	Cost	Method
Conceptual	Approx. (±10-20%)	Neglig.	Neglig.	Handbook/ Calculator
Preliminary	Good (±5-10%)	Rapid	Low	Improved Semi-Empirical (Present Paper)
Detail (Project Group)	High (±2-5%)	Reasonable	High	Full Analysis & Design Viscous 2-D Inviscid 3-D Emerging (3-D Viscous)

Figure 1. – Low-Speed Aerodynamic Prediction Methods

possible and practical by the existence of a potential flow lifting surface theory computer program specifically developed for the analysis and design of multielement swept wing and fuselage combinations. This program, developed by M. I. Goldhammer (Ref. 3), will be referred to in the remainder of this paper as the DVM (distributed vorticity method) program.

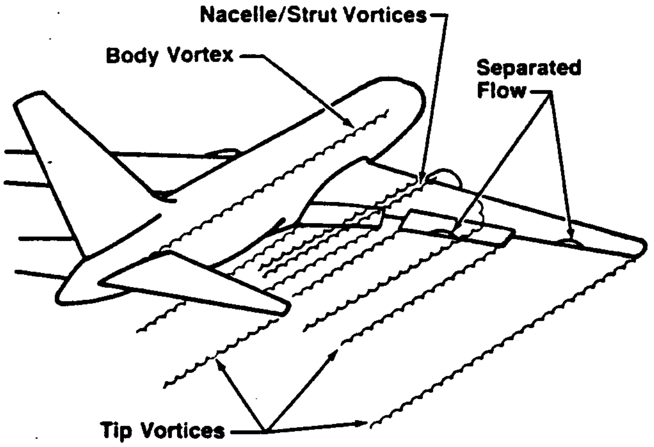

Figure 2. – Typical Flow Field Associated With a Transport Aircraft During Landing Approach

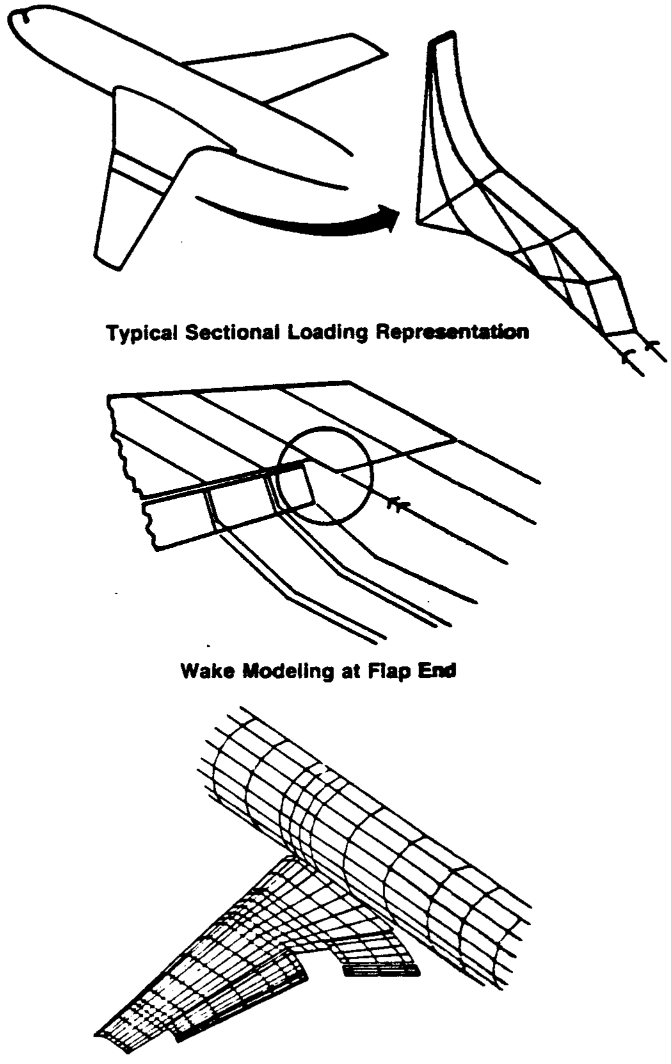

Figure 3. – Distributed Vorticity Method (DVM) Lifting Surface Theory Modeling of a High-Lift Wing

The second program in the system, developed by the authors, will be identified as AePP (Aerodynamic Prediction Program). AePP is a highly automated system of bookkeeping, interpolation/extrapolation, scaling and postprocessor routines, which produces the predictions of global aerodynamic characteristics of a configuration in a subsonic viscous flow.

The DVM potential flow lifting surface theory program uses a distributed vorticity singularity (Fig. 3) in contrast to earlier vortex lattice representations. The program is highly user oriented, and one needs only to specify gross geometric parameters for multielement wings (e.g., planform, twist, camber, flap deflections) from which the program generates its own detailed vorticity networks. The vorticity distribution automatically satisfies the Kutta condition at each trailing edge. A two-dimensional algorithm is used by the program to specify downstream wake shapes. Provision is made for multielement wings with part-span flaps, a slender body theory representation for the fuselage, and a ring-wing model of nacelles. Wing thickness is not accounted for. The DVM program also calculates and stores additional geometric parameters (e.g., flap and leading edge device areas) necessary for execution of the AePP prediction method. Thus the DVM program also provides a powerful reference geometry definition capability requiring a minimum of user manipulation and input.

The DVM program has been in routine project group use since 1979, and typical test-theory comparison results are shown in Figure 4. The remarkably good agreement in lift level

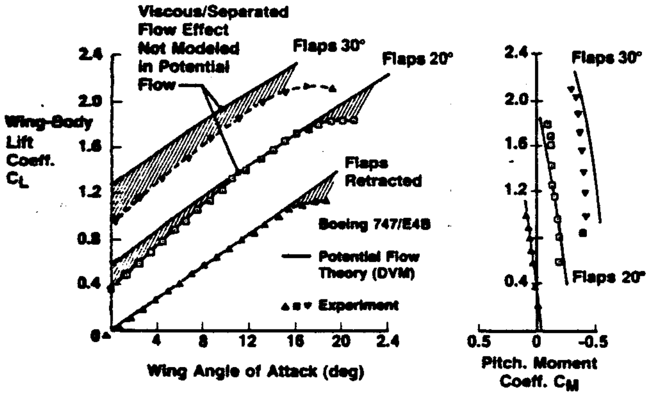

Figure 4. – Typical Distributed Vorticity Method (DVM) Results

for the cases of zero or small flap deflections demonstrated are attributable to the fact that neglect of both attached boundary layer flow and wing thickness in the potential flow analysis produces partially compensating errors. At high flap settings, wherein the flow may be partially separated and/or the boundary layers are much thicker, the potential flow results generally overpredict lift levels.

The basic structure of the overall methodology, based on a framework (in terms of potential flow lift curve, pitching moment, induced drag and span loading) provided by independent runs of the DVM program, is shown in Figure 5, and provides the engineer with two options.

Option 1: For cases where experimental data for a baseline configuration exists, the effects of changes in the baseline geometry (e.g. flap span and/or chord, number of flap elements) can be estimated with good accuracy. In this case the full procedure shown in Figure 5 is used.

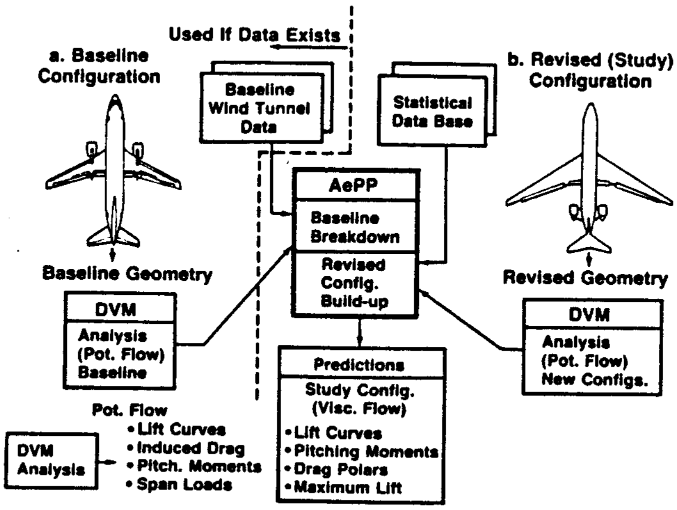

Figure 5. – Outline of the (Low-Speed) Aerodynamic Prediction Procedure

Option 2: This option pertains to cases where no explicit baseline data exists. By combining generic empirical data stored in AePP with results of DVM analyses of the study geometry, the global aerodynamic characteristics of the new configuration can be estimated with adequate accuracy.

The emphasis to date has been on development of the first option, with exploration of the empiricism required to complete the methodology at traditional wind tunnel Reynolds number levels. The problem of extending the methodology to prediction of aerodynamic characteristics of a given configuration at flight Reynolds number scale conditions from wind tunnel data remains to be fully addressed. The method has been structured to allow extension to arbitrary subsonic scale conditions, however.

Discussion of the Method

An outline of the overall method and its program elements is shown in Figure 5. How the method works, the assumption on which it is based, and some of the higher order empiricism used require further clarification and discussion, however.

The advent of methods such as the DVM lifting surface theory described above makes conceptually possible the development of better predictive methodology, if one makes a major assumption based on the following observations.:

- Past production transport aircraft have been subjected to thousands of hours of wind tunnel testing and years of development. While each model may differ dramatically from previous designs in detail, all tend to be of generally similar configuration (e.g., swept wings, empennage aft). Further, within a given company (e.g., Boeing), certain philosophies regarding high-lift system design in terms of performance and associated airplane handling characteristics goals produce a level of underlying commonality.

- The "ideal" performance (lift, induced drag and moments) of a configuration such as is shown in Figure 2 can be calculated reasonably well with inviscid methods. Extensive experience with the DVM program has demonstrated this (cf. Fig. 4 and Ref. 2).

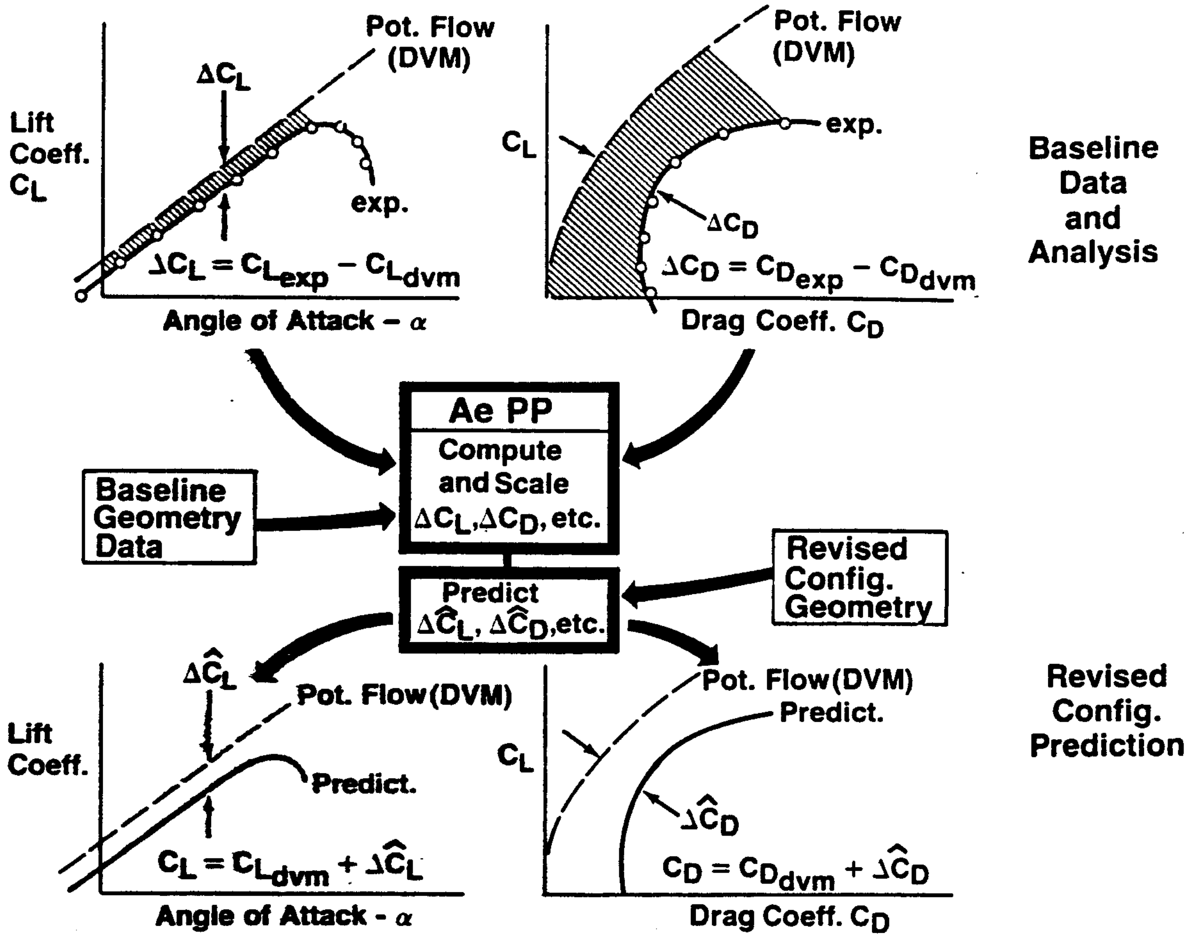

Figure 6. – (Low-Speed) Aerodynamic Prediction Procedure Outline

- Deviations from this ideal performance are attributable to thickness, viscous effects and flow separations – most of which cannot be explicitly calculated with existing methods.

Thus the central assumption of the present method is: If the configuration whose characteristics are to be predicted does not deviate "too much" from past configurations that have been well designed, then deviations from ideal performance in a new configuration will be of comparable order, and arise from similar sources. Thus, estimates of acceptable accuracy of new configuration performance at comparable viscous and compressibility scale conditions can be made *if* careful attention is paid to the geometric details of the various baseline and new configurations.

This last point implies that any computational tools employed must have the ability to adequately account for relatively small but significant geometry changes. In general, handbook methods cannot. The DVM program can.

If one accepts the validity of the basic assumption above, it becomes possible to construct a semi-empirical predictive methodology involving AePP built on the theoretical base provided by the DVM program (Figs. 5 and 6). The outline shown in Figure 6 is based on the case where experimental (e.g., wind tunnel) data for a baseline configuration exists. The initial step in the procedure is to analyze the baseline configuration(s) in potential flow, including use of some empiricism to be described presently, to obtain lift curves, induced drag polars and pitching moment curves. These data, together with detailed geometry data (e.g., individual flap element areas) obtained from the DVM program, are then stored in a data base for access by AePP.

At this point the decomposition mode of AePP compares the DVM analysis data (labeled by the subscript dvm) with the experimental data for that same configuration at equal angles of attack and extracts a set of residual terms for lift, drag and pitching moment coefficients as per the following formulas:

$$\begin{aligned}\Delta C_L\,(\alpha, \delta_f, Re, M) &= C_{L_{exp}} - C_{L_{dvm}}\\ \Delta C_D\,(\alpha, \delta_f, Re, M) &= C_{D_{exp}} - C_{D_{dvm}} \qquad (1)\\ \Delta C_M\,(\alpha, \delta_f, Re, M) &= C_{M_{exp}} - C_{M_{dvm}}\end{aligned}$$

A set of these residuals are then stored as baseline viscous data as functions of angle of attack and flap deflections. Identifying the basic residuals obtained from eqn. (1) for the cruise configuration ($\delta_f = 0°$) by the subscript o, a second set of baseline residuals that reflect coefficient increments due to high-lift system deflection is obtained from:

$$\begin{aligned}\Delta C_{L_f}\,(\alpha, \delta_f, Re, M) &= \Delta C_L - C_{L_o}\\ \Delta C_{D_f}\,(\alpha, \delta_f, Re, M) &= \Delta C_D - C_{D_o} \qquad (2)\\ \Delta C_{M_f}\,(\alpha, \delta_f, Re, M) &= \Delta C_M - C_{M_o}\end{aligned}$$

6.32

The residuals in eqn. (2) can be interpolated to obtain other values for flap deflection increments as required. It should also be noted that in addition to the parameters listed these flap increment residuals are functions of high-lift system complexity. Thus additional adjustments must be made if, say, the baseline has a single-slotted flap system, and the study configuration has double-slotted flaps.

To complete the procedure and produce the desired predictions for the new configuration (b), the new geometry is analyzed in the DVM program. The resulting lift curves, induced drag polars, pitching moment curves and required geometry data are then combined with the appropriate baseline (a) data according to the typical relation:

$$C_{L_b} = C_{L_{dvm_b}} + \Delta C_{L_{o_a}} \cdot G + \Delta C_{L_{f_a}} \cdot H \qquad (3)$$

where

$$G = G\,(Re)$$

$$H = H\,(S_a, S_b, S_{f_a}, S_{f_b}, Re)$$

Corresponding relations can be written for drag and moment coefficients. It should also be noted that many of the scaling relations in traditional handbook methods (such as flap span ratios, etc.) are automatically accounted for in the DVM analyses. The final results of the analyses are then made available for tabular listing or graphical display through AePP postprocessing routines.

The process outlined in Figure 6 pertains to the case where experimental data for the baseline exists. If such data is not available, the process is truncated. In this case it is assumed that generic data (residuals) from aircraft of similar type exist in a previously constructed statistical data base. Here only the study configuration (configuration b) is analyzed in the DVM program and adjusted residuals obtained from the generic data are added to the basic potential flow results to yield the final predictions.

<u>Some Empiricism in the Method</u>

The accuracy of the results obtained with the above procedure is strongly dependent on the magnitude and behavior of the viscous residuals. In this connection, early experiments aimed at validating the DVM program produced an interesting result. As briefly described in Ref. 2, calculations of two-dimensional airfoil characteristics (corrected for sweep) at selected stations across the span of a flap deflected transport wing, including calculation of the separated wake shape, suggested that there is an "effective viscous" camber shape (or flap deflection) associated with a given geometric input shape. It then became possible to construct the generalized curves relating geometric and effective flap deflection for a given number of flap elements and at a given Reynolds number (Fig. 7) for a large sample of diverse Boeing transport aircraft. Input of the appropriate effective (viscous flow) camber lines into the DVM program can result in a dramatic improvement in predicted span loading (Fig. 7) and hence induced drag. Assuming the deflection curves in Figure 7 apply (approximately) to all similar airplane configurations at the scale condition at which they were derived, a very utilitarian, simple and physically sound means of reducing the magnitude of the residuals in the present method is provided. This will be demonstrated in the context of the first example to be discussed presently.

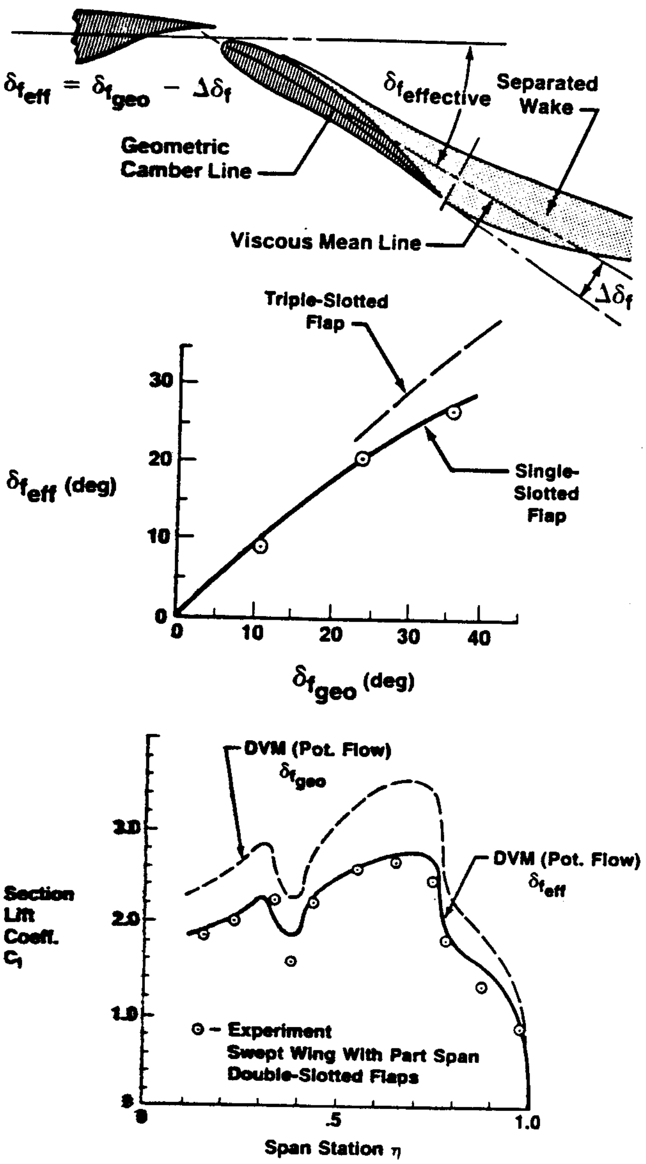

Figure 7. – Effective Flap Deflection and Its Influence on Wing Span Loading

A second empiricism has to do with the evaluation of the maximum lift capability of a given high-lift configuration. Detailed span load distributions are part of the output of the DVM program, and as demonstrated in Figure 7, are of high quality. As an alternative to simple interpolation and scaling (the default in the present method) to obtain the maximum lift coefficient and the angle at which it occurs, it is possible to compare changes in span loading between various configurations in a critical section analysis sense (Ref. 2, (Fig. 8).

Evaluation of a number of Boeing transports, using this procedure, leads to the result:

$$C_{L_{max}} = C_L^* + \text{constant} \qquad (4)$$

where C_L^* corresponds to the wing lift coefficient at which one constituent airfoil first reaches its maximum section lift coefficient. The value of the constant in eqn. (4) depends, of course, on Reynolds number.

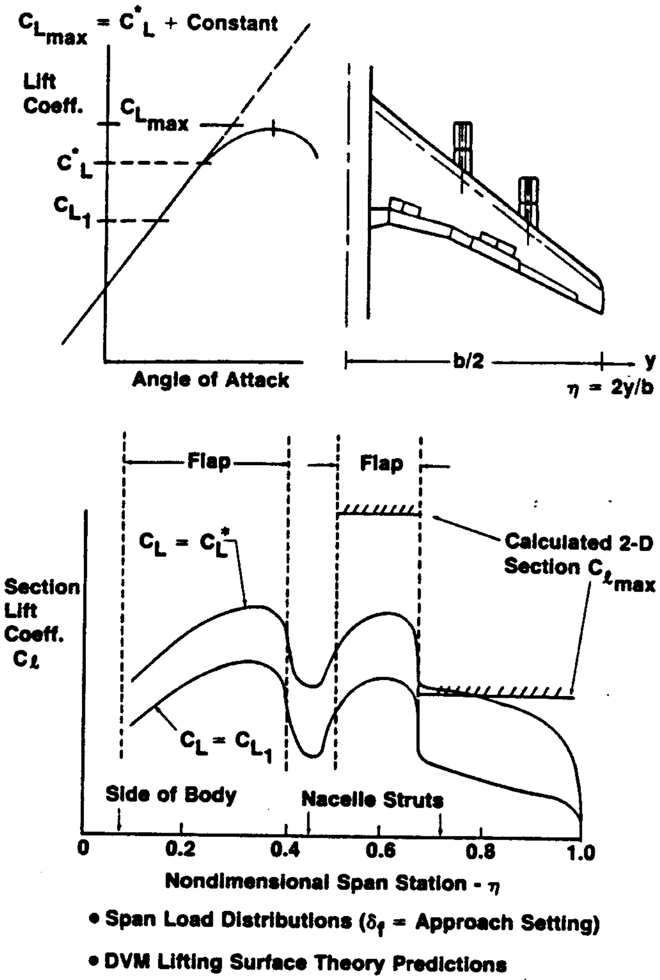

Figure 8. – Critical Section Analysis Approach to Maximum Lift Prediction

<u>Results</u>

Three example test-theory comparisons have been selected to demonstrate the capability of the prediction methodology and further clarify the way in which the procedure works. It should also be noted that the previously described empiricism was developed prior to, and independent of, the examples discussed.

<u>Example 1</u>: Reference 4 contains results of a very extensive wind tunnel test program conducted at the British Royal Aeronautical Establishment (RAE) on a 28° swept wing/fuselage model representative of a transport aircraft in a variety (92) of high-lift and flaps retracted configurations. The data presented thus forms an excellent basis for an initial validation of the prediction method. The results of one such test-theory comparison are shown in Figure 9. In this example two changes from the baseline are involved. The predictions made are for a wing with part-span flap deflected at 25° from data for configurations with full-span flaps at 0°, 10° and 40° deflection. The flaps in all cases were single-slotted but the wing did not have a leading edge high-lift device. The predictions of all characteristics, including maximum lift coefficient, are quite satisfactory.

This example also serves as a useful case to demonstrate the effect on the residuals due to the previously mentioned use of effective instead of geometric flap deflections in the analysis. Figure 10 shows a comparison of the DVM predictions of the lift curves for two of the baseline configurations calculated on the basis of both geometric and effective flap deflections, the latter value taken from Figure 7. It will be observed that the DVM program results are not a completely satisfactory match with the experimental data for any of the RAE configurations. However, the auxiliary plot of the trend in residual values versus flap deflection at a given angle of attack shows that the magnitudes of the residuals are greatly diminished for higher deflection angles when the effective flap deflections (derived from data on wings with leading edge devices) are used in place

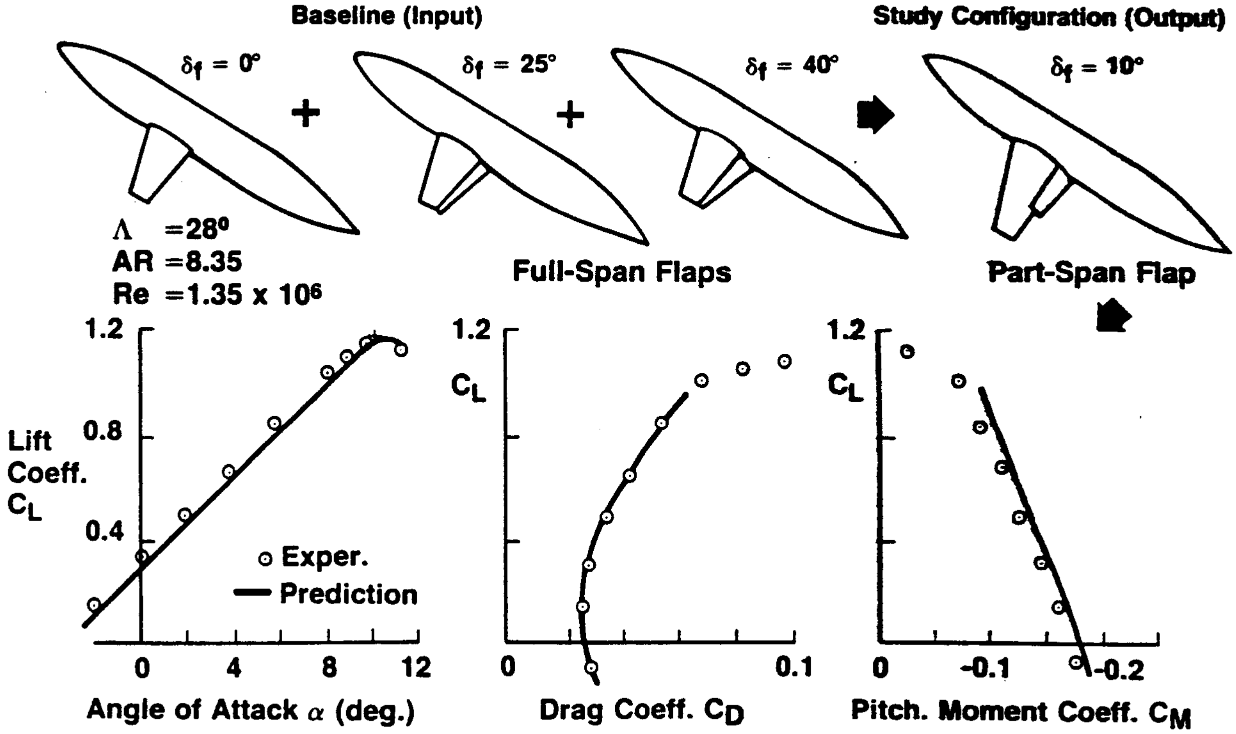

Figure 9. – Prediction of the Global Aerodynamic Characteristics of an RAE High-Lift Wing/Body Combination

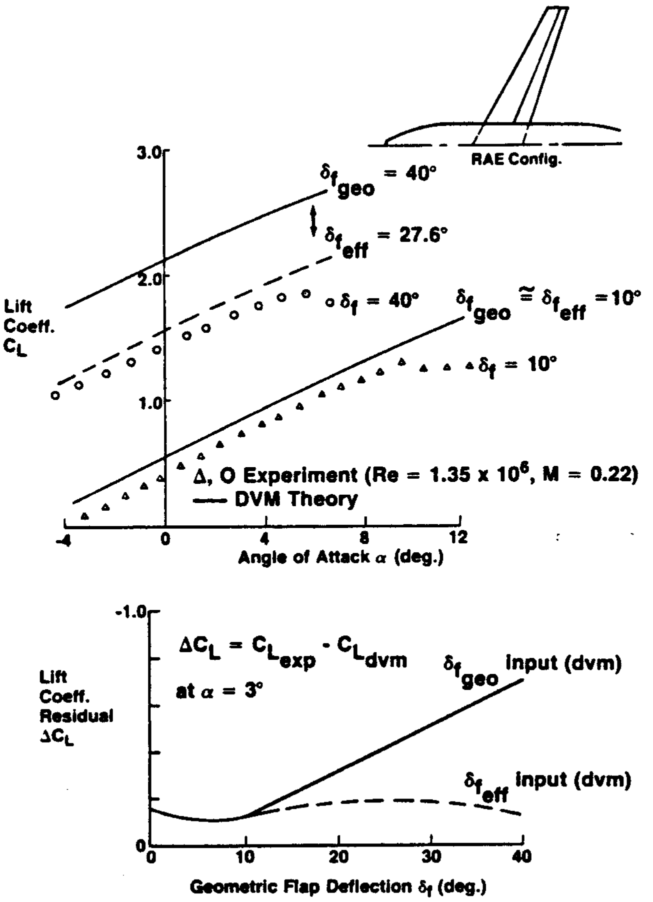

Figure 10. – Effective Flap Deflection Effects on Residual Lift Coefficient Increments

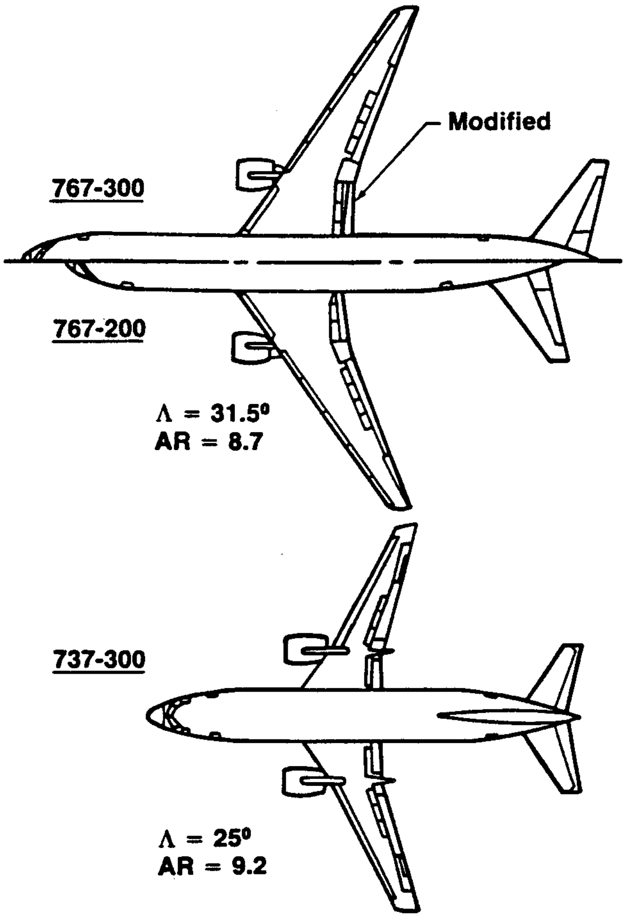

Figure 11. – Planform Comparison of Several Boeing Transport Aircraft

of the geometric values. Any subsequent adjustments of these smaller residuals, due to area ratioing etc., can be expected to lead to a better final resulting prediction.

Example 2: The final results to be presented come from applications of the methodology to recently developed Boeing transport aircraft. The geometries of the aircraft evaluated are shown in Figure 11, and the experimental data is from tail-off wind tunnel tests conducted at the University of Washington Aeronautical Laboratory (UWAL) at chord Reynolds numbers on the order of 1.5×10^6 and a Mach number of about 0.2.

In the first case the prediction to be made involved the characteristics of a stretched fuselage 767-300 with modified flaps deflected at 15°, from data for a baseline 767-200 with flaps deflected 0°, 5° and 20°. The results are shown in Figure 12 and agreement between test data and predictions of all aerodynamic quantities is seen to be excellent.

Example 3: As the final example, predictions of the characteristics of a completely different aircraft, a 737-300 (Fig. 11), were based on data for the 767-200 (of similar configuration but dramatically different in most details). As input in this case, the geometry of the 767-200 at flap deflections of 0°, 5°, 15°, and 20° was used to predict the characteristics of the 737-300 with flaps deflected 5°, representative of a takeoff configuration. Again the test-theory comparison results are quite satisfactory.

Conclusions

A preliminary design level method for predicting the global aerodynamic characteristics of transport type aircraft in low-speed/high-lift configurations has been described. The method is semi-empirical and the overall procedure is economical, user oriented and highly automated. Results obtained with the method so far are very encouraging, as demonstrated by the test-theory comparisons presented.

The feasibility and practicability of the overall method described strongly depends on the availability of a potential flow lifting surface theory computer program for the analysis of multielement swept wings. The DVM program used in the present method provides the basic skeletal structure of the procedure and strikes the right balance between the conflicting demands of ease of use, economy of operation, and accuracy and reliability of result. When combined with the empiricism discussed, the DVM program provides a very powerful and utilitarian engineering tool, aside from its use in conjunction with the AePP method discussed in this paper.

The main line of development of the methodology to date has been to devise a procedure applicable to cases where well defined baseline configurations and associated wind tunnel data for them exist. Predictions of the characteristics of a new configuration are then made at the same Mach and Reynolds number scale conditions. In these cases the quality of the predictions can be expected to be directly proportional to the degree of geometrical deviation of the study configuration from the baseline, as well as the quality and quantity of baseline data available.

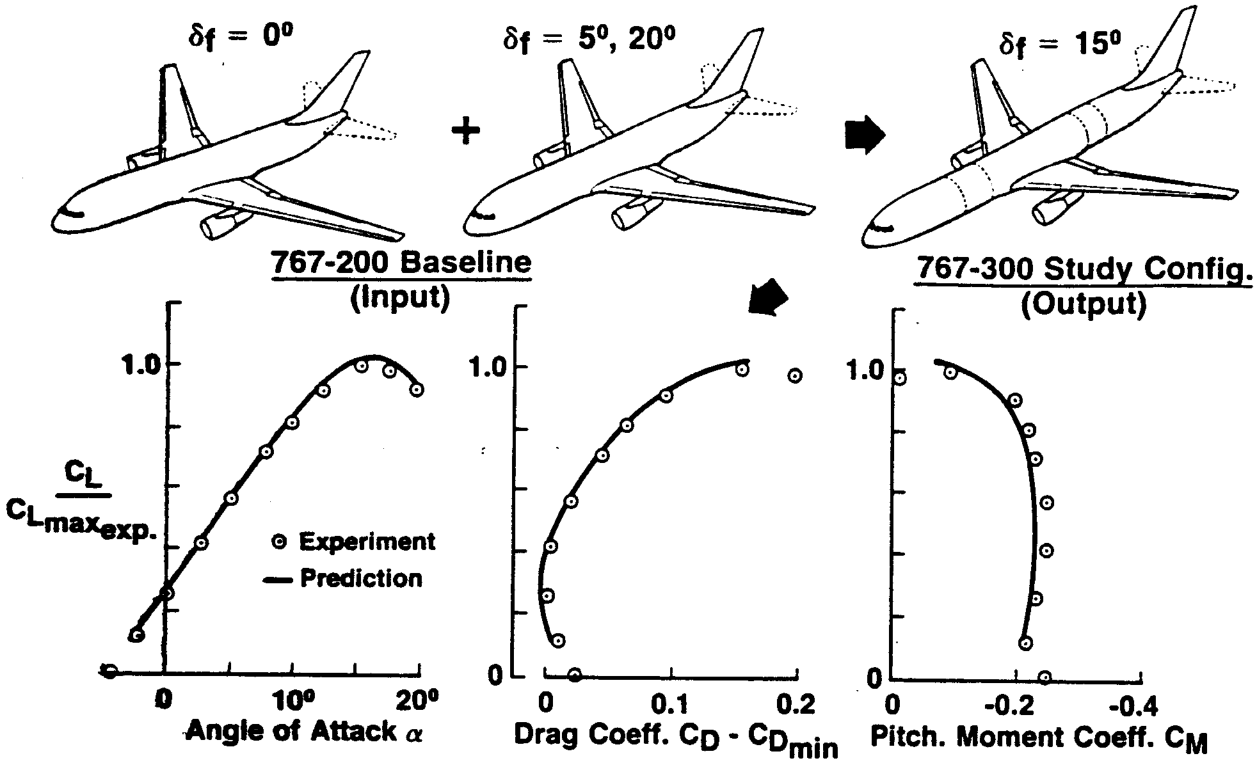

Figure 12. - Prediction of the Aerodynamic Characteristics of a Boeing 767-300 From Data for a 767-200

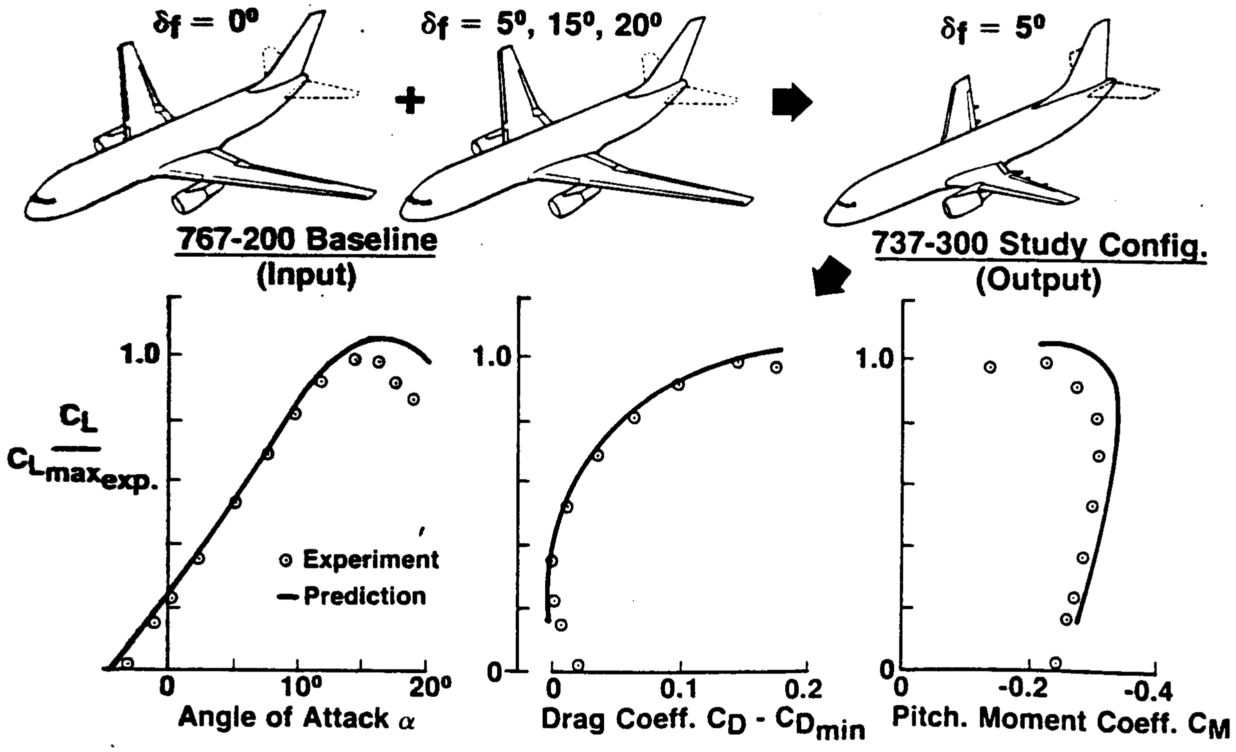

Figure 13. - Prediction of the Characteristics of a Boeing 737-300 From Data for a 767-200

Despite these limitations to the present version of the method, the procedure has been structured to provide a good deal of flexibility and potential for extension. As examples:

- Repeated applications of the present version of the method allow construction of the generic data bases necessary to provide estimates of the wind tunnel level characteristics of configurations for which no explicit baseline data exists.

- By extending the scaling laws applied to the residual coefficient values generated by AePP, estimates of the characteristics of a given configuration at flight Reynolds numbers from wind tunnel data can be made on a rational basis.

Acknowledgements

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Engine Selection Problems For Jet Powered Ultra STOL

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Engine Selection Problems For Jet Powered Ultra STOL

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ABSTRACT

This paper discusses the effect of engine size and cycle on two types of Ultra STOL aircraft. One type, a powered lift plus deflected cruise thrust vehicle, is analyzed to determine the effect of wing loading, cruise engine and lift engine thrust loading, number of lift engines, and cruise engine bypass ratio and turbine inlet temperature on airplane gross weight and cost at a 1,000-foot field length. The other type, a conventional nondeflected thrust airplane, is analyzed to determine the effect of wing loading, thrust loading, engine bypass ratio, and turbine inlet temperature on airplane gross weight and cost at field lengths from 1,000 to 2,000 feet. Results of this analysis indicate that powered lift aircraft are not cost competitive unless field lengths of less than 1,000 feet are considered.

INTRODUCTION

During the Sixties, the ability of the U. S. military and commercial transports to deliver cargo anywhere in the world increased dramatically. This capability is still growing, with the C-5 coming into military service and with the advent of the jumbo jets in commercial service. The clearing of facilities and distribution of large quantities of military supplies may be well beyond the capability of local transportation systems available in less developed parts of the world. The development of an efficient short-haul transport aircraft to operate beyond the major terminals and throughout an undeveloped region should be of considerable importance, because the benefits of strategic airlift and sealift can be realized without the long and costly development of logistic support facilities.

The proper match of powerplant and airframe is a critical interface in the development of such a transport system. The purpose of this paper is to examine two types of aircraft with Ultra STOL capabilities, and to determine the effect of engine size and cycle on each type. One type will be a conventional aircraft without thrust deflection, which achieves its field performance by a least-weight combination of low wing loading and high thrust loading. For this type of aircraft the least-weight size and cycle of the cruise engines will be determined. The other type of aircraft will use deflected cruise thrust plus vectorable wing-tip-mounted jet lift engines to help support the aircraft. For this type, the mix of lift thrust to cruise thrust that results in the lowest weight aircraft will be examined, as well as the cruise engine cycle. A cost evaluation of the two types of aircraft will also be presented.

The short-haul transport should be capable of operating into and out of small undeveloped airfields with little or no approach aids. This would require an Ultra STOL capability that, for this paper, will be defined as a requirement to achieve an all-engine distance to a 50-foot altitude from 1,000 to 2,000 feet, at a 2,500-foot field elevation, and 93°F ambient temperature. This is equivalent to about 800 to 1,500-foot distance over 50 feet at sea-level standard conditions. This level of field length performance should provide a very wide choice of operating points. Jet powerplants were chosen for the aircraft under consideration. It is felt that the modern turbofan cruise engine offers the best compromise in simplicity, reliability, economy of operation, and low maintenance costs unmatched by any other type of powerplant. A detailed examination of other types of V/STOL aircraft is presented in Reference 1.

The paper is in three parts. Part I investigates an airplane with deflected cruise thrust plus jet lift engines in wing-tip-mounted pods. Because of the complex trades involved, and because the powered lift airplane has its greatest advantage at the shortest takeoff distance, only the 1,000-foot takeoff case will be investigated for this configuration. Part II deals with a conventional-type aircraft without jet lift. This analysis will include takeoff distances from 1,000 to 2,000 feet. Part III is a comparison evaluation between the two concepts in terms of gross weight and cost trades, and will allow a determination of the cost trends with takeoff distance to be determined, and a comparison between a powered lift configuration and a conventional configuration at a 1,000-foot takeoff field length

PART I – POWERED LIFT AIRPLANE

The general configuration of the powered lift Ultra STOL airplane is shown in Figure 1. This configuration designated Model 953-403 was, in general, similar to the concept described in Reference 2, in that the cruise engine thrust is vectorable and the lift engines are mounted in pods at the wing tips. The design requirements for the aircraft are to carry 18,000 pounds of payload 750 nautical miles out of a 1,000-foot unprepared site that is at 2,500 feet altitude, with 93°F ambient temperature. The variables to be examined are:

1) Gross weight;
2) Wing loading;
3) Cruise engine/lift engine thrust mix;
4) Number of lift engines;
5) Cruise engine bypass ratio;
6) Cruise engine turbine inlet temperature.

The first four variables will be examined initially to obtain a minimum-gross-weight vehicle that has a least-weight combination of wing loading, cruise thrust loading, and lift thrust loading.

Once this airplane is defined, the cruise engine bypass ratio and turbine inlet temperature will be varied to determine the effect of cruise engine cycle on overall airplane gross weight.

For the initial configuration study, four scaled turbofans, with a bypass ratio of 6 were used, and the lift engines were scaled lift jets with a bypass ratio of zero. The configuration had an aspect ratio of 5.5 wing with a $C_{L_{max}}$ of 3.66. No noise treatment was used for these engines, because it was assumed they would not be used in a noise-critical environment. Noise effects on STOL commercial aircraft are discussed in Reference 3.

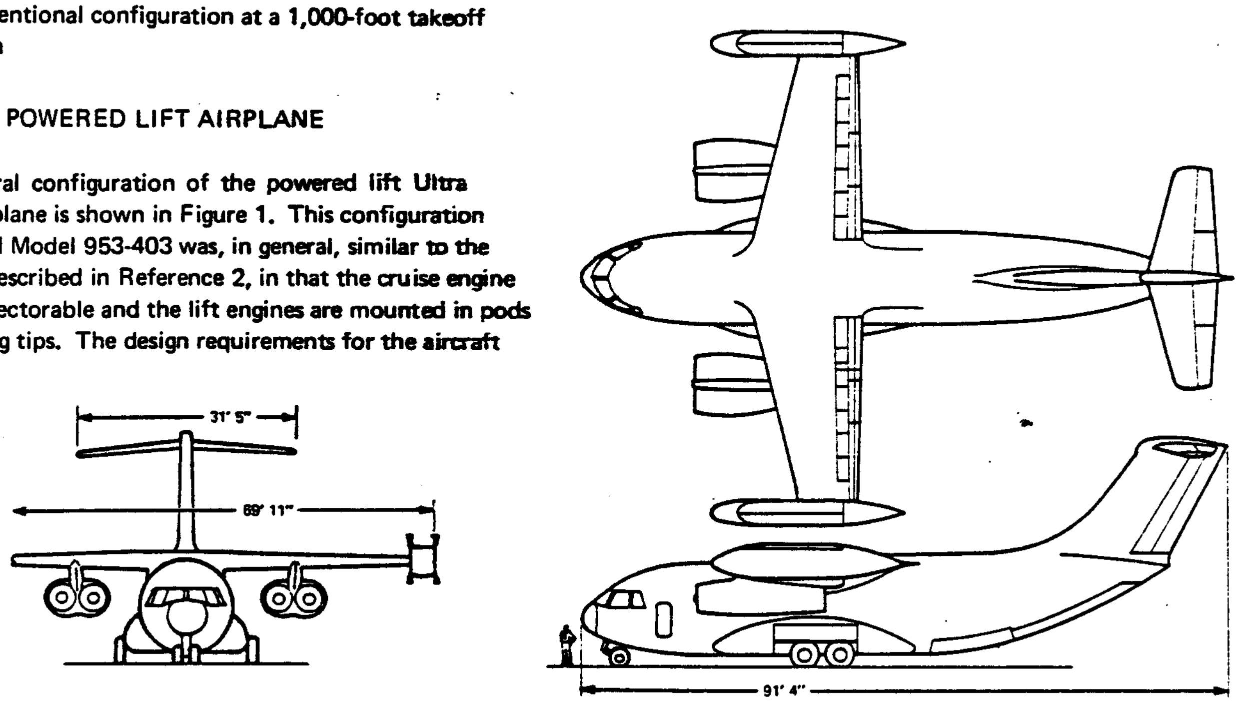

Figure 1. General arrangement–model 953-403

Takeoff speeds were selected to provide a safe takeoff with the critical lift or cruise engine failed, and the compensating opposite lift engine(s) throttled back to idle. The fuselage and box size were held constant. Full-span, triple-slotted flaps were used for the high-lift system. The cruise engines were mounted in double pods under the wing, and the lift engines were mounted in wing-tip pods. The lift engines and the cruise engine bypass thrust were vectorable. The vectorable portion of the cruise thrust was about 85 percent of the total cruise engine thrust. During the takeoff maneuver, the lift engine and the cruise bypass thrust were set at the same deflection, and maintained at this angle throughout the takeoff maneuver. The thrust deflection was that which gave the minimum distance to the 50-foot obstacle height. The deflection angle for the minimum distance was found by computing the takeoff distance versus thrust deflection for a range of wing and thrust loadings. The thrust deflection angles for minimum takeoff distances are shown in Figure 2.

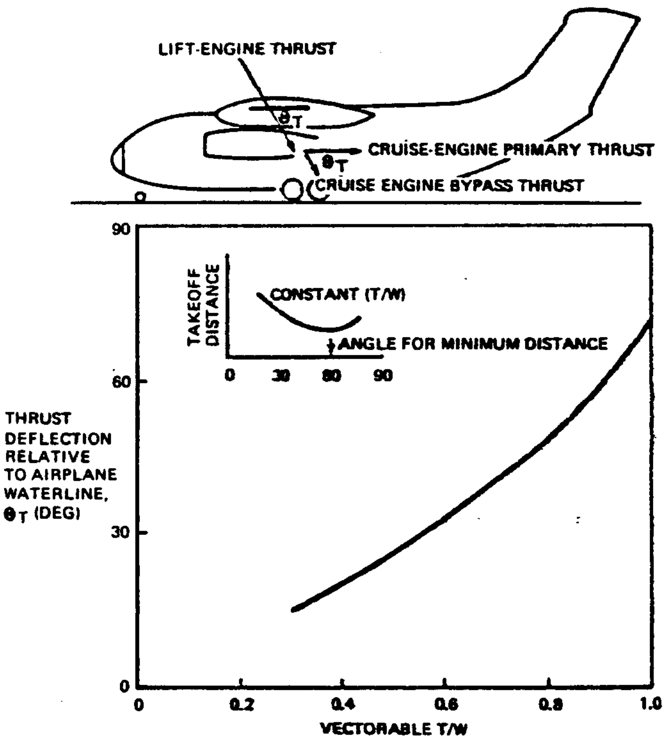

Figure 2. Thrust deflection for minimum takeoff distance

Weight trade data are shown in Figures 3, 4, and 5. The following ground rules were applied for this trade:

1) Fuselage size and geometry remained constant;
2) Wheel and gear geometry remained constant;
3) Wing geometry remained similar;
4) Horizontal and vertical tail geometry remained similar;
5) Engine and nacelle configurations remained similar;
6) Fuel system provisions remained constant.

The parametric weight data did not include the wing weight changes due to changes in the number and size of lift engines at the wing tip. More weight at the wing tips may provide weight relief for the wing, but landing dynamics may then design the wing structure. Also, a weight and lift penalty for the increased lateral control necessary to handle the increased roll inertia was not included.

The range of variables is listed below:

Gross weight	90,000 to 160,000 lb
Wing aspect ratio	5.5 (constant)
Wing loading	60 to 120 lb/ft^2
Horizontal $\bar{V}$	1.2 (constant)
Vertical $\bar{V}$	0.172 (constant)
Cruise engines	4 scaled, high-bypass turbofans, bypass ratio 6, T/W cruise 0.2 to 0.6
Lift engines	scaled, lift turbojets, T/W lift 0.3 to 0.7
Number of lift engines	6 to 14

TAKEOFF RULES

The 1,000-foot takeoff distance is defined as the all-engine distance to clear a 50-foot height. Airfield conditions are an unprepared site, 93°F day, and 2,500 feet altitude. The liftoff speed is the greater of V_{MIN} + 10 KTS or 1.15 V_{MIN} where V_{MIN} is the power on stall speed, with the critical engine and compensating opposite lift engine(s) inoperative.

The lift engines and the cruise engine bypass thrust were both vectorable, and were maintained at the deflection for minimum takeoff distance throughout the takeoff maneuver. Ten percent of the total installed thrust was used for reaction control during takeoff. (No credit was taken for this 10 percent thrust in calculating the takeoff distance.)

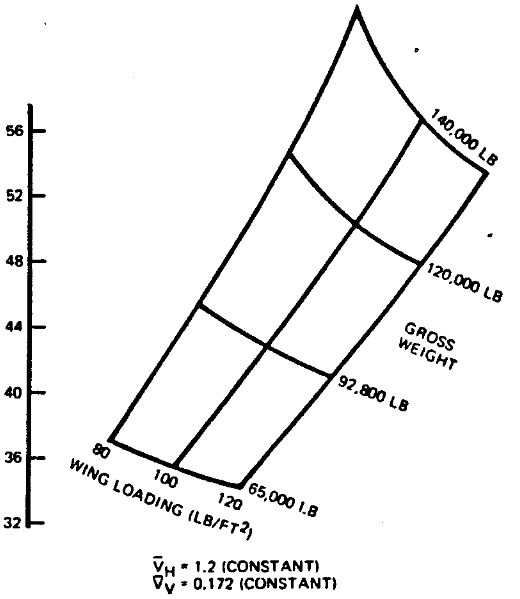

Figure 3. Variation of OEW less powerplant with wing loading and gross weight for aircraft type shown in Figure 1

MISSION AND CONFIGURATION SIZING

The 750-nautical-mile mission range was computed according to MIL-C-5011A. A takeoff fuel allowance of 5 minutes at takeoff power for the cruise engines and 1 minute at takeoff power for the lift engines was used. A brief outline

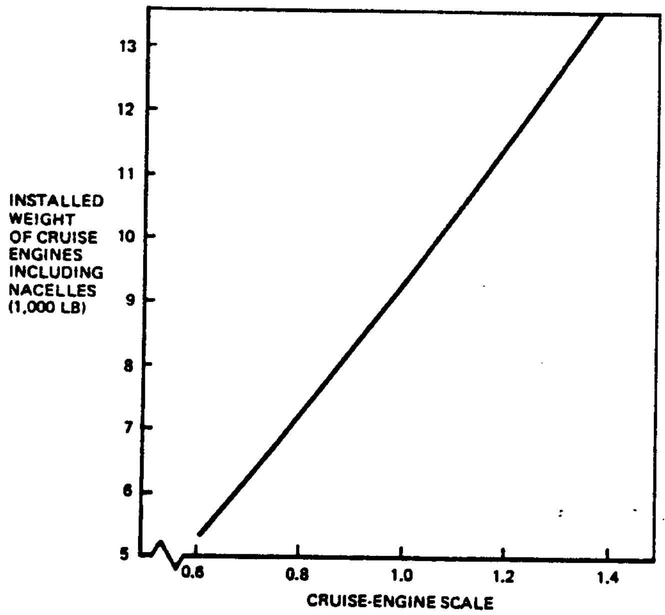

Figure 4. Installed weight of cruise engines including nacelles

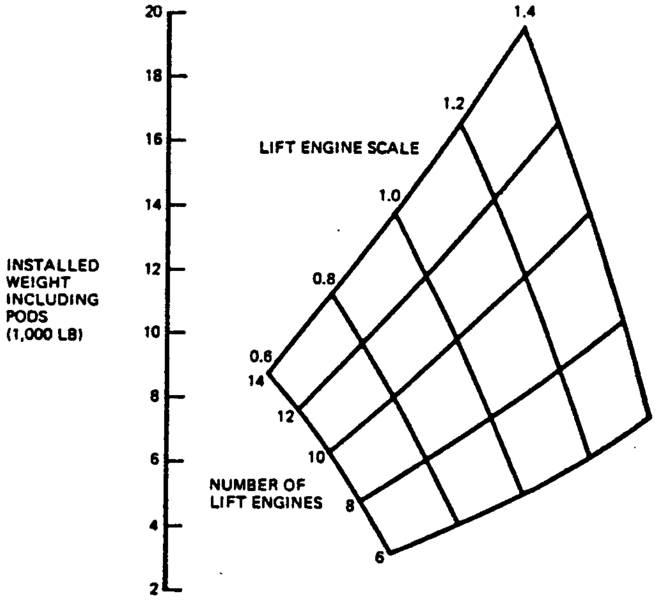

Figure 5. Installed weight of lift engines including pods

of the method for finding the gross weight that met both the takeoff and cruise requirements is given below:

1) For a given number of lift engines and different ratios of lift to cruise thrust, the wing and thrust loadings that met the takeoff requirement were found.

2) For various wing loadings, the range was computed for two weights, and the weight for the 750-nautical-mile range was found. New drags and weights were used for each configuration.

RESULTS

The gross weights required to do the design mission are shown in Figure 6. The lower dotted line on each plot is where the lift thrust per engine and the cruise thrust per engine are equal. Specific configurations, which will be used later in the cost analysis, are given individual numbers on the curves. Diminishing returns, in terms of weight per unit of installed thrust, were found at values of T_{lift}/T_{cruise} greater than 1.0 (see Figure 7). As the high values of T_{lift}/T_{cruise} per engine are approached, the reduction in cruise T/W reduces the cruise speed and altitude. This

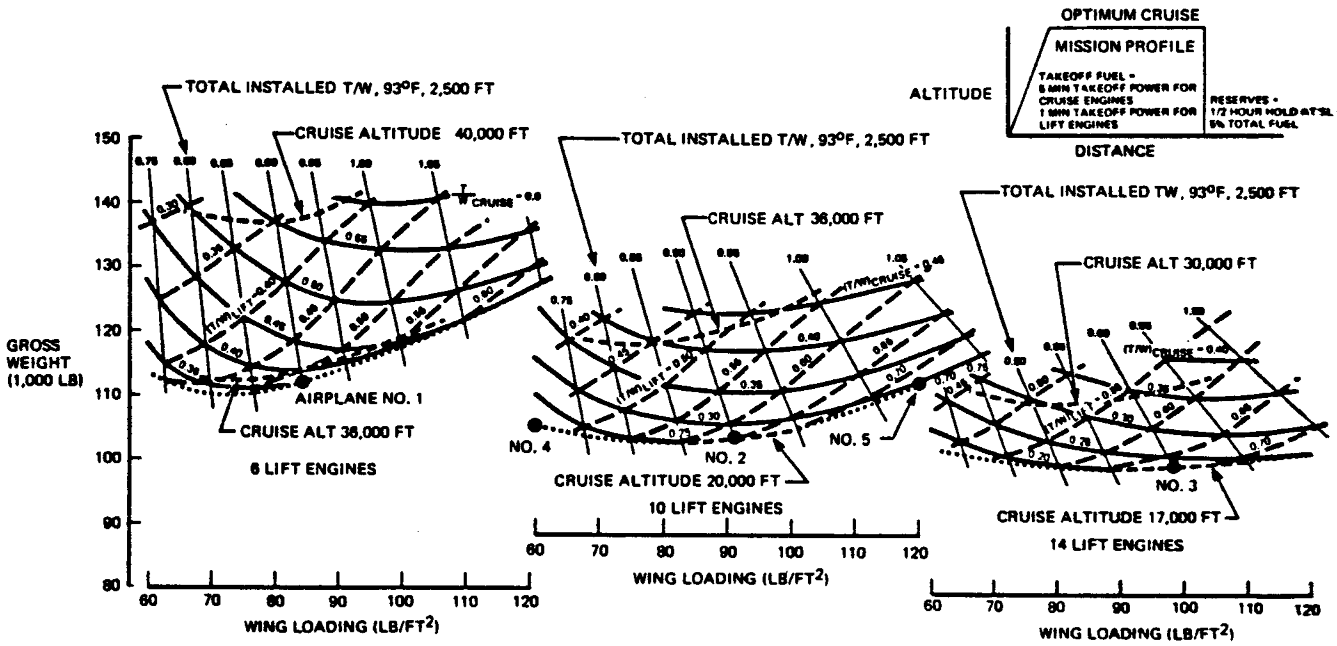

Figure 6. Configuration parametric study

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may affect the configuration choice if there are specific cruise speed and altitude requirements. Lines of typical cruise altitudes are shown in Figure 6.

The trend is toward lower gross weights with higher wing loadings and a greater number of lift engines. The reason for this is similar to that previously outlined for increasing the T_{lift}/T_{cruise} per engine ratio, plus the secondary effect of minimizing the thrust loss due to the engine-out criteria during takeoff. This allows the takeoff requirement to be met at a higher wing loading for constant T/W as the number of lift engines increases.

A preliminary cost exercise was carried out on several representative aircraft from Figure 6. This was done so that a configuration could be selected for study of the engine cycle variables. Three configurations were selected, based on constant thrust loading and varying number of lift engines. Their characteristics are as follows:

Configuration	1	2	3
Gross weight (lb)	112,500	102,500	98,000
No. of cruise engines	4	4	4
No. of lift engines	6	10	14
Cruise engine thrust (lb) (Sea-level static)	13,300	8,610	6,480
Lift engine thrust (lb) (Sea-level static)	12,300	7,950	5,980

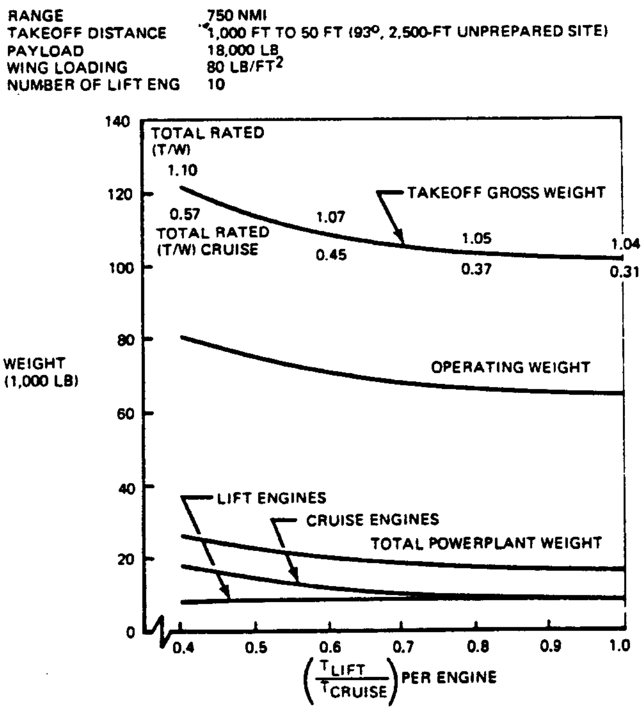

Figure 7. Change in takeoff gross weight with change in powerplant size

Resulting costs are summarized in Table I for 196 UE aircraft

Table I. Initial cost analysis (dollars x 10^6)

AIRPLANE	RDT&E	ACQUISITION	O&M	TOTAL 10 YEAR COSTS
1	842	1,900	1,959	4,701
2	641	1,680	2,056	4,377
3	542	1,581	2,174	4,297

These cost results indicate that total 10-year system costs decrease only slightly as the number of lift engines is increased past 10. Because of its unique match with an existing cruise engine, an eight-lift-engine configuration will be used in the cycle analysis to follow.

CYCLE ANALYSIS – LIFT ENGINE AIRPLANE

The four cruise engines used in the study to determine the effects of number and size of lift engines, the results of which are shown in Figure 6, were bypass-ratio-6 turbofans with a turbine inlet temperature of 2,650°R. The following analysis was made to see if reductions in gross weight could be realized if other cruise engine cycles were considered. To serve as a base for this cruise engine cycle study, the minimum-gross-weight airplane for an 8-lift-engine configuration was selected from Figure 6. This configuration represented a unique solution with existing cruise engines, which could be used as a base to determine whether a new engine cycle would reduce gross weight enough to warrant the engine development costs. The base-point airplane has the following characteristics:

Gross weight	105,000 lb
Wing loading	75 psf
Total rated T/W (SLS)	1.02
No. of lift engines	8

For a number of gross weights, the range was determined with new engine characteristics, using a total of 19 different engines, the physical and performance characteristics of which were determined during this study. Preliminary investigations with respect to overall pressure ratio indicated little benefit in using pressure ratios much higher than 20. In addition, small engines, such as are employed here, have greater difficulty in achieving compression ratios in excess of 20 to 1. Thus it was decided to reduce the number of matrix variables by defining Rp as constant and equal to 20. Engine performance characteristics were determined for engines of bypass ratios 1, 2, 4, 6, and 8 at turbine inlet temperatures of 2,450, 2,650 and 2,850°R, and bypass ratios of 2, 4, 6, and 8 at a turbine inlet temperature of 3,050°R. A typical corrected SFC versus thrust plot for a bypass-4 temperature = 2,650°R engine is shown in Figure 8.

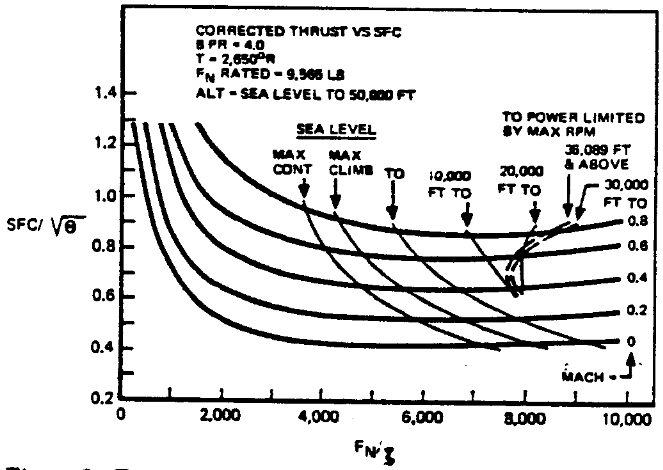

Figure 8. Typical engine characteristics

Other data assumed for these engines are tabulated in Table II.

Bare engine weights were scaled by assuming that the weight scaled equaled (scaled thrust/reference thrust)1.2. Engine installation weights were varied directly as the thrust ratio and nacelle wetted areas were scaled appropriately.

New drags and operating weights were determined for each bypass ratio and temperature. Initial cruise altitude was defined by rate of climb equal to 300 ft/min. The gross weights required for 750-nautical-mile range were then found.

Figure 9 shows the required cruise engines installed weight to accomplish the mission. Figure 10 shows the required airplane OEW. Airplane gross weight trends are shown in Figure 11. The gross weight minimizes at a bypass ratio of

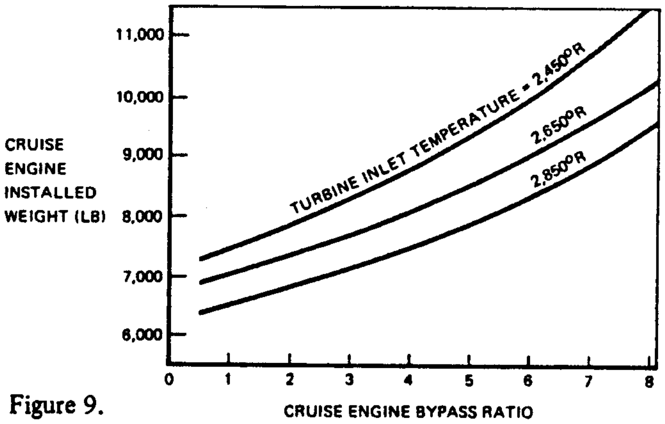

Figure 9.
Cruise engine installed weight (1,000-foot takeoff aircraft)

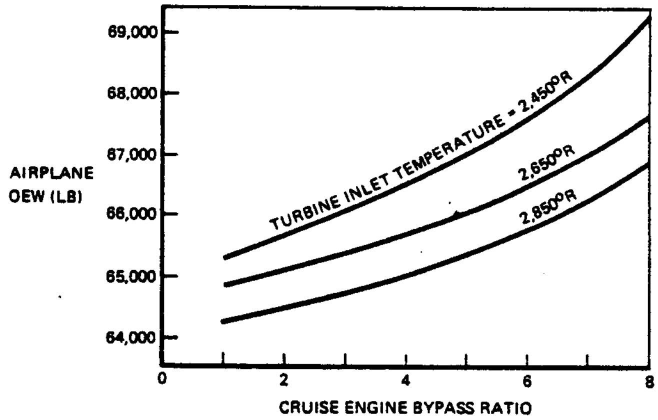

Figure 10. Airplane OEW trends

Table II. Engine characteristics

ENGINE BYPASS RATIO	TURBINE INLET TEMP °R	RATED THRUST UNINSTALLED LB	BARE ENGINE WEIGHT AT RATED THRUST LB	INSTALLED ENGINE WEIGHT AT RATED THRUST LB	NACELLE WETTED AREA AT RATED THRUST FT^2
1	2,450	8,715	1,110	1,667	95
2	2,450	8,631	1,098	1,733	106
4	2,450	8,747	1,181	1,978	128
6	2,450	8,555	1,316	2,261	143
8	2,450	9,154	1,527	2,657	165
1	2,650	9,492	1,170	1,734	95
2	2,650	9,427	1,149	1,789	106
4	2,650	9,566	1,220	2,022	128
6	2,650	9,702	1,349	2,298	143
8	2,650	10,072	1,562	2,697	165
1	2,850	10,097	1,225	1,795	95
2	2,850	10,020	1,196	1,842	106
4	2,850	10,214	1,257	2,067	128
6	2,850	10,378	1,383	2,338	143
8	2,850	10,757	1,599	2,740	165
2	3,050	10,646	1,244	1,896	106
4	3,050	10,836	1,296	2,108	128
6	3,050	11,022	1,417	2,376	143
8	3,050	11,490	1,635	2,781	165

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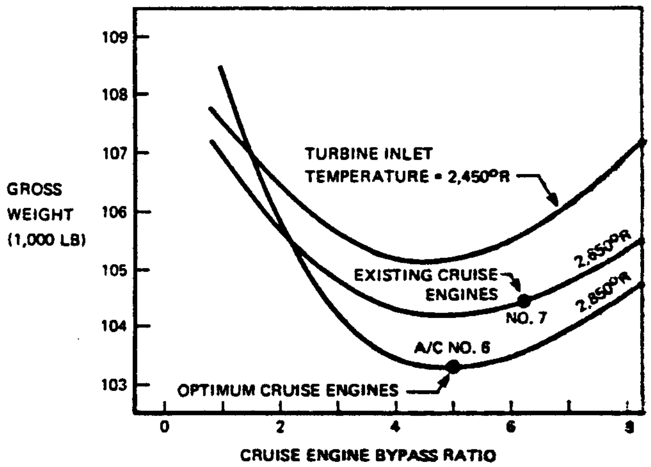

Figure 11. Aircraft gross weight trends

about 5, with a turbine inlet temperature of 2,850°R. There is little difference, however, between the gross weight required with the least-weight cycle and the existing engine cycle. In fact, the gross weight decrease is only on the order of one percent total gross weight. Final cost comparisons for these aircraft are presented in Part III.

VERTICAL CAPABILITY

It should be noted that these aircraft, when flown at standard sea-level conditions, will have some vertical capability, depending on the number of lift engines used. For a 10-lift-engine airplane, the vertical gross weight capability with two engines out at sea-level standard condition will be 91,000 pounds. This represents a full fuel load plus 3,000 pounds of payload. The conventional Ultra STOL airplanes that will be analyzed in the next section will not, of course, have this capability.

PART II – CONVENTIONAL AIRFRAME

The type of airplane to be considered as the conventional airframe base point is represented by the 953-220 configuration drawing shown in Figure 12. This airplane is a six-engine transport with high-lift mechanical flaps, producing a $C_{L\,max}$ of 4.2, high thrust-to-weight ratio, and low wing loading. It has the same cargo box size as the 953-403 configuration discussed previously. To keep the vertical tail size within reasonable limits, an engine failure during takeoff is handled by a thrust reduction on the opposite engine. The critical field length (distance to liftoff with a critical engine failure and the opposite engine throttled back to idle) is still approximately equal to the all-engine distance to 50 feet. Beneficial effects of external blowing on the full-span, triple-slotted flaps are ignored. Thrust deflection devices (other than thrust reversers) were not incorporated in this design.

The 220 configuration, shown in Figure 12, was used as a base point about which variations in engine parameters can be examined. All airplane design points represent combinations of wing and thrust loading to achieve the minimum-weight airplane with a constant box size. This is accomplished by searching the field of airplanes whose combination of wing and thrust loading match the takeoff requirement, and selecting those that represent minimum airplane size to fly the mission. Only these minimum-weight aircraft points are presented in the resulting plots. Horizontal and vertical tail volume coefficients are held constant at the same values as the -220 design. Airplane design points were selected by using the Damps II airplane PD program, which allows airplane scaling, engine scaling, mission variation, and engine cycle variables to be examined simultaneously. A flow chart of the airplane optimization procedure through the computer is shown in Figure 13.

Takeoff characteristics assumed for this type of aircraft are shown in Figure 14. This method assumes no thrust deflection and determines all engine distance to 50 feet. Airplane characteristics were determined, using the same engines as those shown in Part I, Table II.

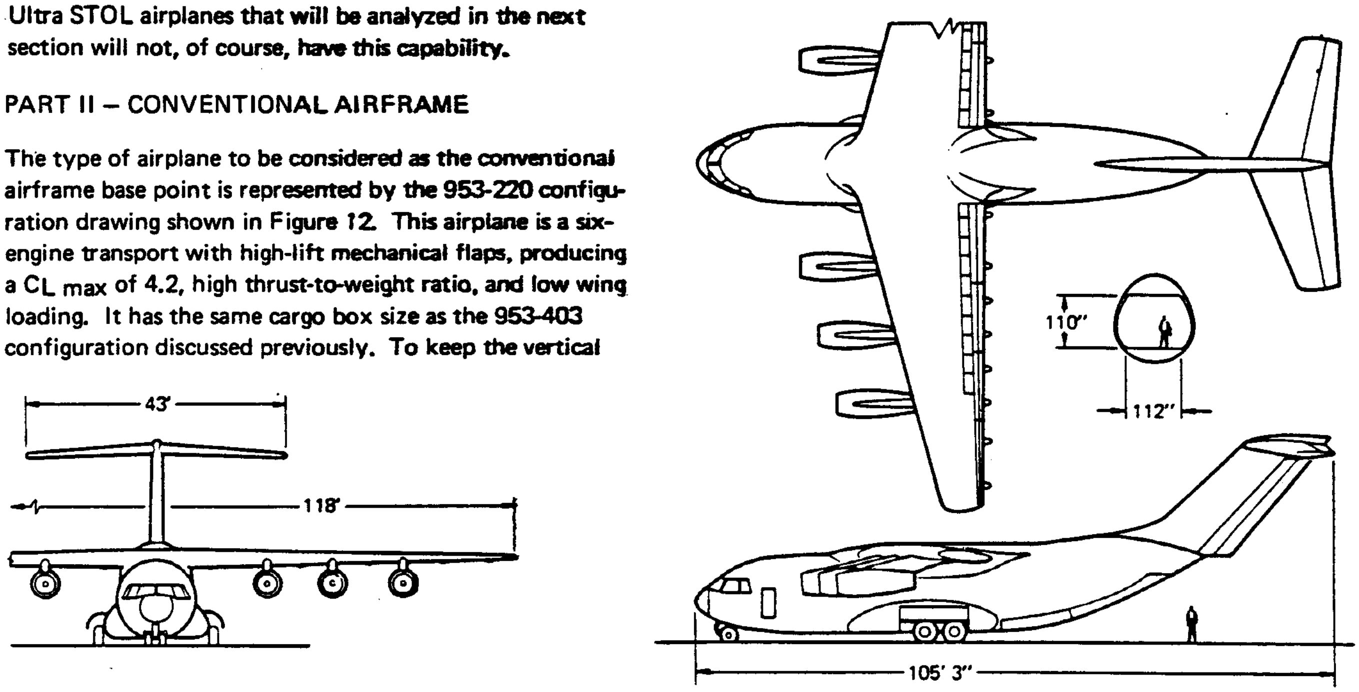

Figure 12. General Arrangement 953-220

L.45

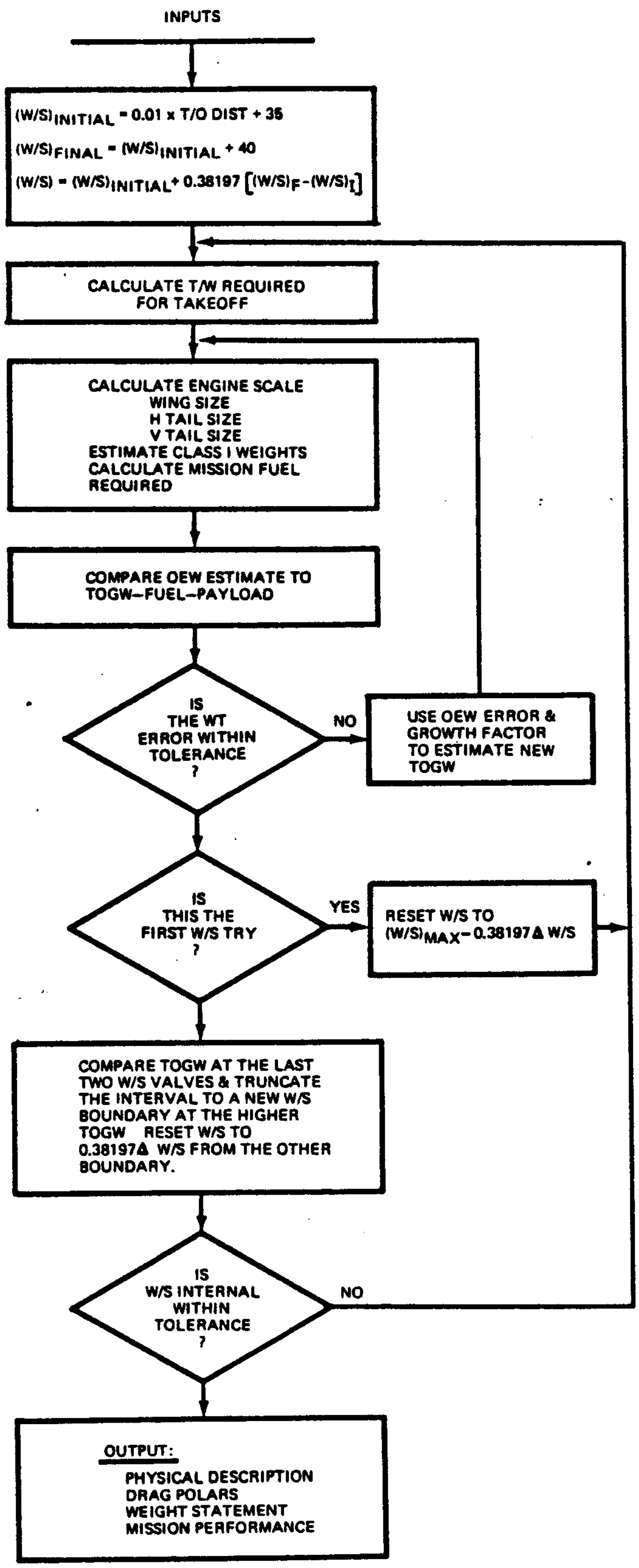

Figure 13. Computer logic flow

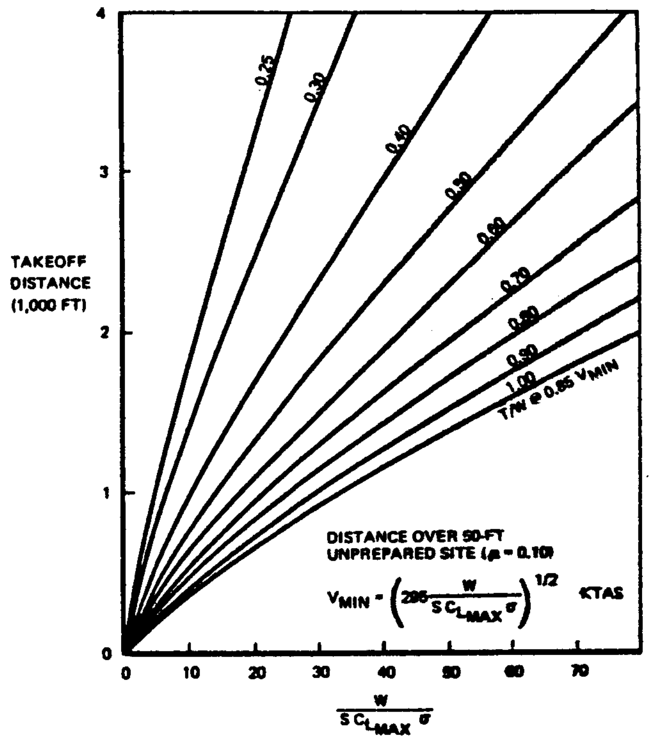

Figure 14. Generalized takeoff performance

The mission to be accomplished by the conventional airplane was the same as that for the airplane with lift engines; i.e., to carry an 18,000-pound payload out of a 1,000-foot field, at an elevation of 2,500 feet, with an ambient temperature of 93°F for 750 nautical miles, using MIL-C-5011A rules. Five minutes at normal rated power was included for takeoff fuel. The airplane size required to fly the design mission with each of the parametric engines is shown in Figure 15. Figure 15 indicates

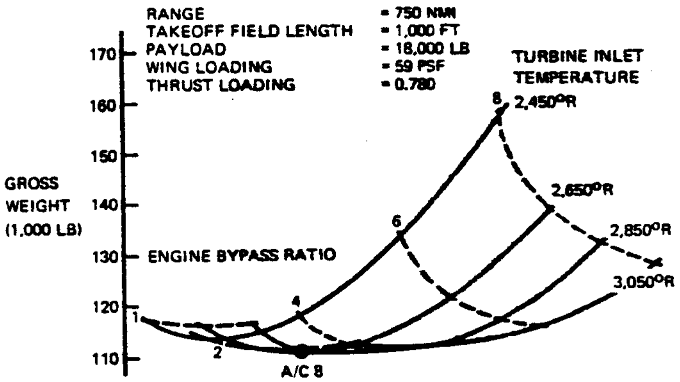

Figure 15. Effect of engine bypass ratio and turbine inlet temperature on gross weight

that the minimum-gross-weight aircraft occurs at a turbine inlet temperature of 2,850°R and a bypass ratio of 3. There is little gross weight decrease between temperatures of 2,650 and 2,850°R and some increase from 2,850 to 3,050°R.

6.46

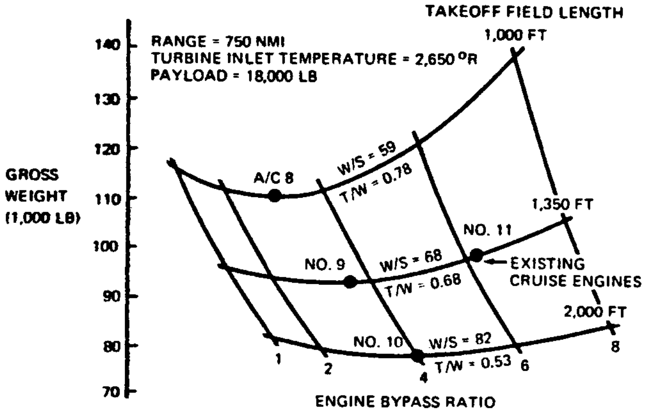

Figure 16. Effect of engine bypass ratio and takeoff distance on gross weight

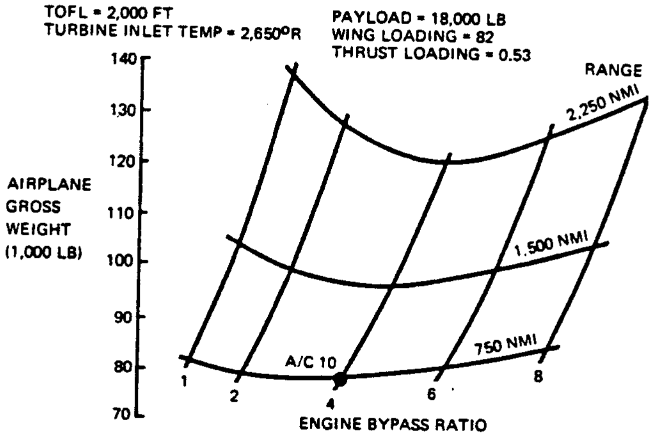

Figure 17. Effect of engine bypass ratio and range on gross weight

Figure 16 shows the effect of takeoff field length and engine bypass ratio on airplane gross weight at a constant turbine inlet temperature. From this plot, it can be seen that the bypass ratio for the least airplane gross weight is essentially constant at about 3 to 4 for a field length variation from 1,000 to 2,000 feet. It also shows that the shorter field length airplanes are more sensitive to the correct choice of engine bypass ratio than those at the longer field lengths.

Figure 17 shows the effect of mission range and bypass ratio on airplane gross weight at constant turbine inlet temperature. From this plot, it can be seen that the bypass ratio for least weight increases as the range is increased.

Cost data for the bypass-ratio-4 airplanes of Figure 16 will be developed in the next section, as well as costs for the existing engine solution at a bypass ratio of 6. These aircraft will then be compared to the lift engine configurations.

PART III – WEIGHT AND COST COMPARISON

Two types of aircraft have now been developed that are capable of accomplishing the same task. A summary of these aircraft is presented in Table III. A plot of gross weight versus field length for some of these aircraft is shown in Figure 18. The conventional airplanes with engines resulting in the lowest gross weight are the bypass-ratio-4 aircraft from Figure 16. The powered lift airplane with existing engines is an interpolated value from Figure 6 with 8 lift engines and, as such, represents a minimum gross weight aircraft with existing cruise powerplants. The powered lift airplane with engines selected for minimum gross weight is the airplane with 2,850°R turbine inlet temperature, bypass-ratio-4 engines from Figure 11. One other point of interest is at 1,350-feet takeoff and represents a conventional aircraft with existing powerplants at bypass ratio 6 from Figure 16

Table III. Airplane characteristics

AIRPLANE NUMBER	TYPE	GROSS WEIGHT	TAKEOFF FIELD LENGTH 93°F 2,500 FT	NUMBER OF CRUISE ENGINES	CRUISE ENGINE THRUST SL STATIC	NUMBER OF LIFT ENGINES	LIFT ENGINE THRUST SL STATIC	WING LOADING	TOTAL STATIC SEA LEVEL THRUST LOADING
1	LIFT ENG	112,500	1,000	4	13,300	6	12,300	86	1.12
2	LIFT ENG	102,500	1,000	4	8,610	10	7,950	91	1.12
3	LIFT ENG	98,000	1,000	4	6,450	14	5,980	98	1.12
4	LIFT ENG	104,500	1,000	4	6,650	10	6,150	60	0.844
5	LIFT ENG	111,000	1,000	4	10,650	10	9,850	120	1.275
6	LIFT ENG	103,300	1,000	4	9,200	8	8,600	75	1.020
7	LIFT ENG	104,500	1,000	4	9,280	8	8,700	75	1.020
8	CONVENTIONAL	111,300	1,000	6	14,300			59	0.76
9	CONVENTIONAL	93,035	1,350	6	10,586			69	0.68
10	CONVENTIONAL	77,993	2,000	6	6,933			83	0.53
11	CONVENTIONAL	97,800	1,350	6	9,280			57	0.57

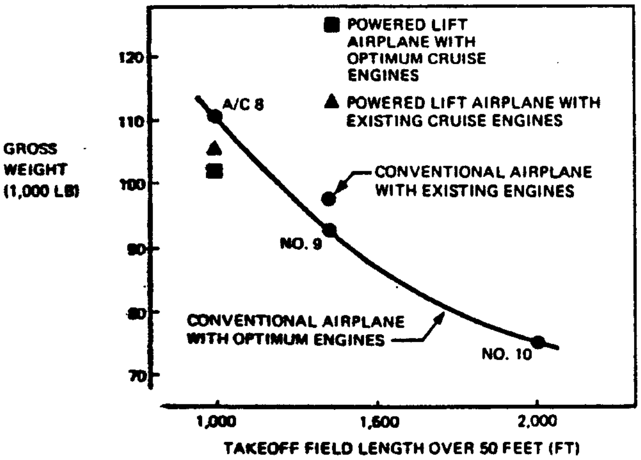

Figure 18. Gross weight comparison

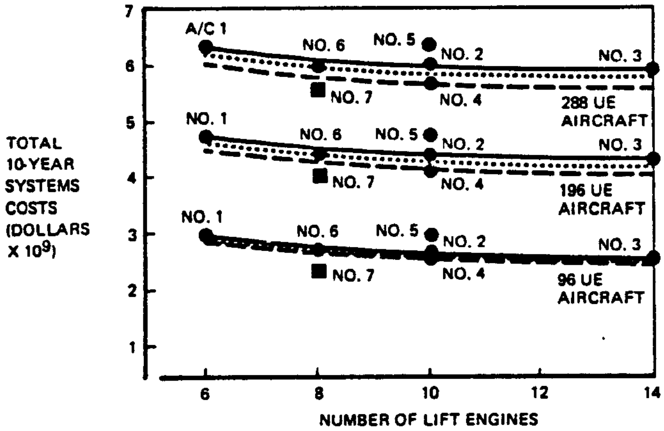

Figure 19. Total 10-year systems costs (powered lift aircraft)

As can be seen from these curves, the powered lift airplane can perform the required mission at a lighter gross weight than the conventional airplane for 1,000-foot field conditions. Whether the powered lift airplane weight advantage also represents a cost advantage remains to be determined.

Costs developed for the 11 aircraft are tabulated in Table IV. These costs were developed using the methods outlined in Reference 4, with the additional assumption that lift engines cost 0.6 times the equivalent thrust cruise engine. These costs show RDT&E, initial investment, 10-year operating, and total 10-year systems costs for 96, 196, and 288 UE aircraft.

The total 10-year systems costs for all the powered lift aircraft are plotted in Figure 19 as a function of number of lift engines. From this curve, it can be seen that costs diminish as the number of lift engines is increased up to about 14. Airplane 6 represents the minimum-weight aircraft, with 8 lift engines, from the engine cycle study shown in Figure 11. If it is assumed that the trends of cost with the increasing number of lift engines are similar for a given cruise engine with a fixed turbine inlet temperature of 2,850°R, then a dashed line can be drawn through airplane 6 parallel to the solid line through airplanes 1, 2, and 3. This dashed line can then be assumed to show cost trends with a varying number of lift engines for an airplane with bypass-ratio-5 cruise engines that have a turbine inlet temperature of 2,850°R.

Another comparison is between airplanes 2, 4, and 5. These aircraft all use 10 lift engines, but represent a trade of lift engine and cruise engine thrust with wing area. These aircraft are shown plotted against wing loading in Figure 20. This curve shows that the minimum-cost aircraft is not

Table IV. Final cost analysis

COSTS (MILLIONS OF DOLLARS)	AIRPLANE NUMBER:										
	1	2	3	4	5	6	7	8	9	10	11
RDT&E	$ 842	$ 641	$ 542	$ 550	$ 742	$ 695	$ 398	$ 685	$ 549	$ 407	$ 225
INITIAL INVESTMENT											
*114/96/2	1,162	1,023	960	935	1,148	1,052	1,024	1,090	904	726	879
225/196/4	1,900	1,680	1,581	1,527	1,888	1,726	1,677	1,747	1,449	1,163	1,405
335/288/6	2,540	2,251	2,122	2,040	2,533	2,312	2,242	2,312	1,917	1,537	1,856
10 YEARS OPTS											
*114/96/2	983	1,031	1,089	1,033	1,046	999	1,001	872	809	768	809
225/196/4	1,959	2,056	2,174	2,060	2,087	1,992	1,996	1,738	1,613	1,532	1,613
335/288/6	2,934	3,080	3,257	3,087	3,125	2,983	2,990	2,602	2,416	2,295	2,416
TOTAL 10 YR SYSTEM											
*114/96/2	2,987	2,695	2,591	2,518	2,936	2,746	2,423	2,647	2,262	1,901	1,913
225/196/4	4,701	4,377	4,297	4,137	4,717	4,413	4,071	4,170	3,611	3,102	3,243
335/288/6	6,316	5,972	5,921	5,677	6,400	5,990	5,630	5,599	4,882	4,239	4,497

* AIRCRAFT BUY/UE AIRCRAFT/BASES

6.48

the minimum-weight aircraft. From Figure 6 it can be seen that the minimum-weight aircraft has a wing loading of about 85. The minimum-cost airplane, however, has a wing loading of about 50. Airplane 4 then represents a minimum-cost aircraft with bypass-ratio-6 engines that have a turbine inlet temperature of 2,650°R. A dashed line is shown in Figure 19 through airplane 4 parallel to the line through airplanes 1, 2, and 3; this dashed line gives absolute minimum costs for the powered lift aircraft. The reason the airplane 6 cost point is not below the dashed line that passes through airplane 4 is that the cost model indicated the development of the engine for airplane 6 was enough higher than that of the type used in airplane 4 to make its total 10-year system cost higher, even though the airplane will weigh less. The absolute minimum-cost new-engine airplane will be one that has 14 lift engines and is on the

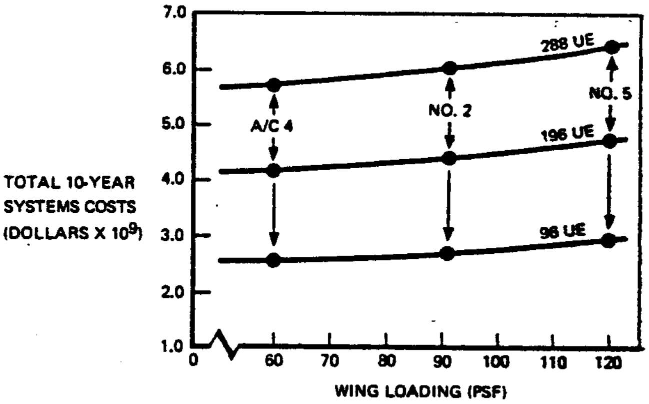

Figure 20. Total 10-year systems costs—powered lift aircraft (lift engines = 10)

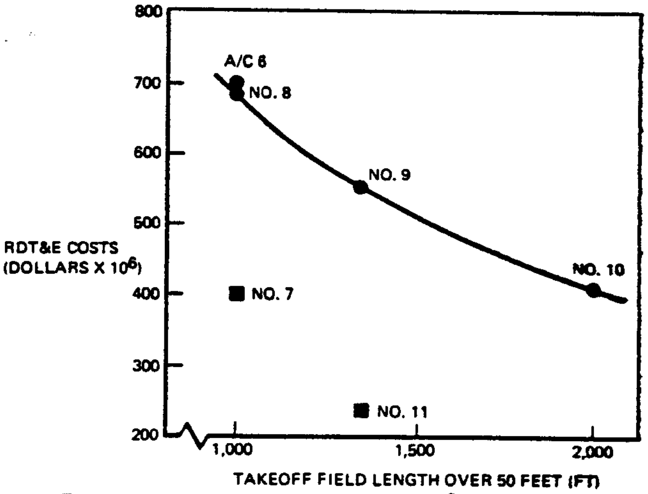

Figure 21. Aircraft RDT&E costs (powered lift and conventional aircraft)

dashed line through airplane 4. It is of interest to note that this cost point is the same in each case as that for airplane 7, which is the airplane with existing cruise engines. Thus, in subsequent graphs airplane 7 and the absolute minimum-cost powered lift airplane are represented by the same points.

Now airplane 6 through 11 will be considered. This comparison will be done as a function of field length, remembering that airplanes 6 and 7 are powered lift aircraft, and airplanes 8 through 11 are minimum-weight conventional aircraft.

Figure 21 shows the RDT&E costs for these aircraft. This figure shows the large advantage for using already developed engines. Airplanes 7 and 11 have no engine development costs, and as such enjoy much lower values of required development dollars.

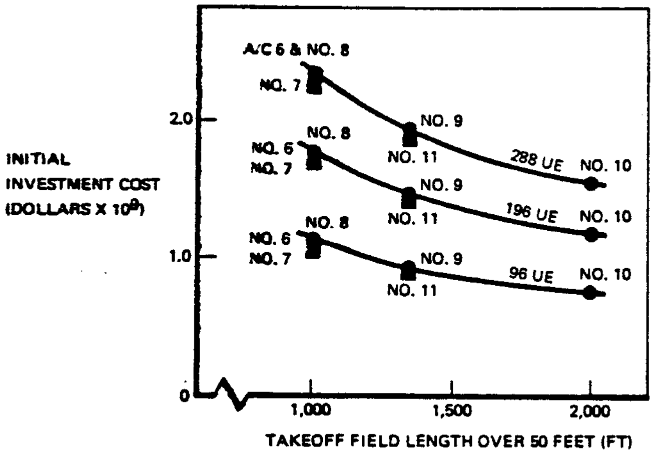

Figure 22. Aircraft initial investment costs (powered lift and conventional aircraft)

Figure 22 shows the required initial investment costs for these aircraft. These curves show that the initial investment costs are close to equal for equal field capability aircraft.

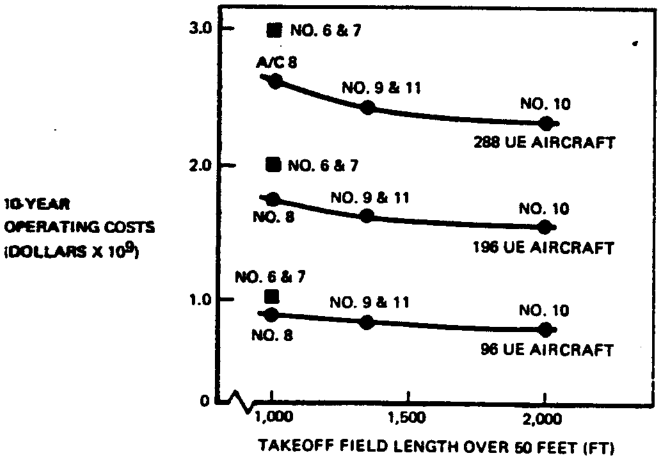

Figure 23. Aircraft 10-year operating costs (powered lift and conventional aircraft)

Figure 23 shows the 10-year operating costs for the comparison aircraft. This figure shows that the powered lift aircraft are more expensive to operate than the conventional aircraft for the 1,000-foot field case. This expense increases as the number of UE aircraft increases.

The total 10-year systems costs, which are a summation of the previous three charts, are shown in Figure 24. This chart shows that as the number of UE aircraft increases, the powered lift airplane becomes less and less competitive. Trends of costs with decreasing field lengths are shown as dashed lines through airplane 6. These trends were established in previous studies. The intersection of these dashed lines, with those of the conventional aircraft, occur at a field length of something less than 1,000 feet. From this, it can be seen that a powered lift aircraft is not cost com-

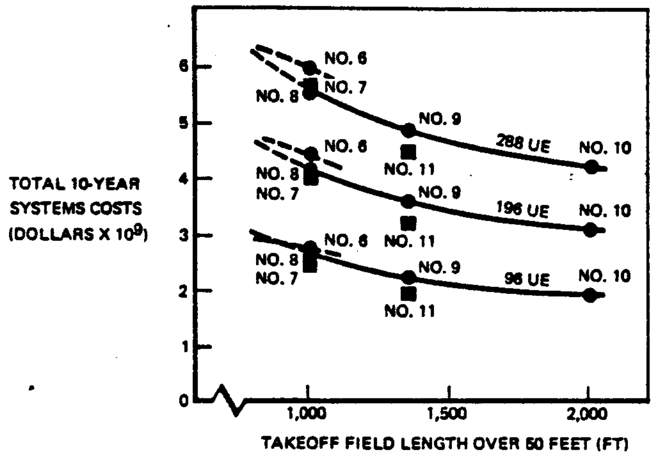

Figure 24. Total 10-year systems costs (powered lift and conventional aircraft)

petitive unless the takeoff field requirement is something less than 1,000 feet.

Finally, Figure 25 is presented to show how the minimum-cost, powered lift aircraft compare with the conventional aircraft. This curve indicates that the minimum-cost, powered lift airplane loses any cost advantage over the minimum-weight conventional airplane when more than 230 UE aircraft are considered. Note that the conventional

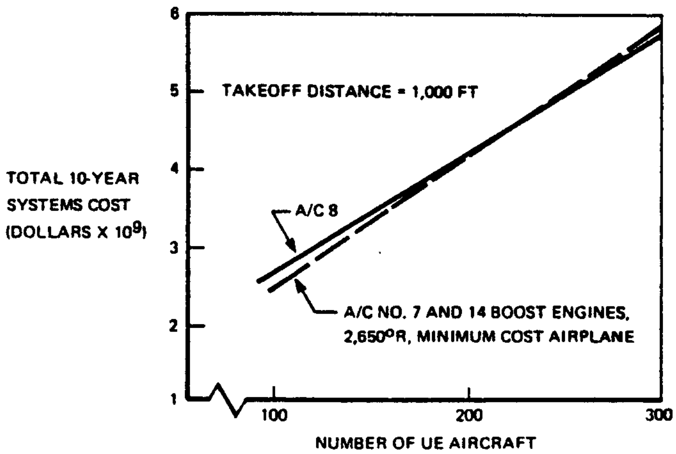

Figure 25. Cost comparison of powered lift configuration vs conventional configuration

aircraft are minimum weight and have not been given the thorough cost analysis that the powered lift aircraft have had. There are indications from airplane 11 that a slightly lower wing loading and thrust loading will result in the conventional aircraft having a cost advantage not shown in the preceding curves. Airplane 11 has a lower wing loading and thrust loading than the other aircraft that have 1,350-foot takeoff and, as can be seen in Figure 24, its initial investment is lower than airplane 9 even though it weighs more. This is because the engine costs were traded for some wing area, and wing weight is less costly than engine weight. Thus, if the conventional airplane costs were minimized like the powered lift aircraft, the total 10-year system cost comparison for the 1,000-foot field case would be very close to that shown between airplanes 6 and 8. Then the powered lift airplane would never have a cost advantage over the conventional at any field length over 1,000 feet, regardless of the number of UE aircraft considered.

As can be seen, the costs indicate:

1) Minimum-weight conventional aircraft are less costly than minimum-weight, powered lift aircraft at field lengths of 1,000 feet or greater.

2) The effect of cruise engine cycle on minimum aircraft weight is so small that using existing cruise engines results in a less costly aircraft for both conventional and powered lift concepts.

3) Achieving a takeoff field length of 1,000 feet requires an increase in program costs of 30 percent over that for a 2,000-foot conventional aircraft.

4) The point at which it is better to build a powered lift airplane than a conventional one is for a field length of something less than 1,000 feet.

5) Minimum-weight aircraft are not minimum-cost aircraft for the type under consideration in this paper.

CONCLUSIONS

PART I – POWERED LIFT AIRCRAFT

A. For powered lift aircraft, minimum gross weights result when individual cruise engine thrust is equal to individual lift engine thrust at takeoff conditions.

B. Aircraft with lift engines are relatively insensitive to cruise engine cycle.

PART II – CONVENTIONAL AIRCRAFT

A. For the type of aircraft analyzed, the bypass ratio for least weight is close to 4 for field lengths of 1,000 to 2,000 feet and ranges of 750 to 2,200 nautical miles.

B. Conventional aircraft are sensitive to proper engine cycle matching.

6.50

PART III – WEIGHT AND COST COMPARISON

A. For the ground rules imposed in this study, a new engine with its cycle selected to minimize gross weight to the mission requirement does not justify its development costs for a 1,000-foot powered lift STOL.

B. Conventional aircraft are more cost effective than aircraft with lift engines, down to field lengths of 1,000 feet.

C. The number of aircraft under consideration affects the relative cost picture between powered lift and conventional aircraft.

D. For both powered lift and conventional aircraft, minimum cost solutions are those that lie on the low wing loading side from the minimum-weight point of a wing loading versus gross weight trend line.

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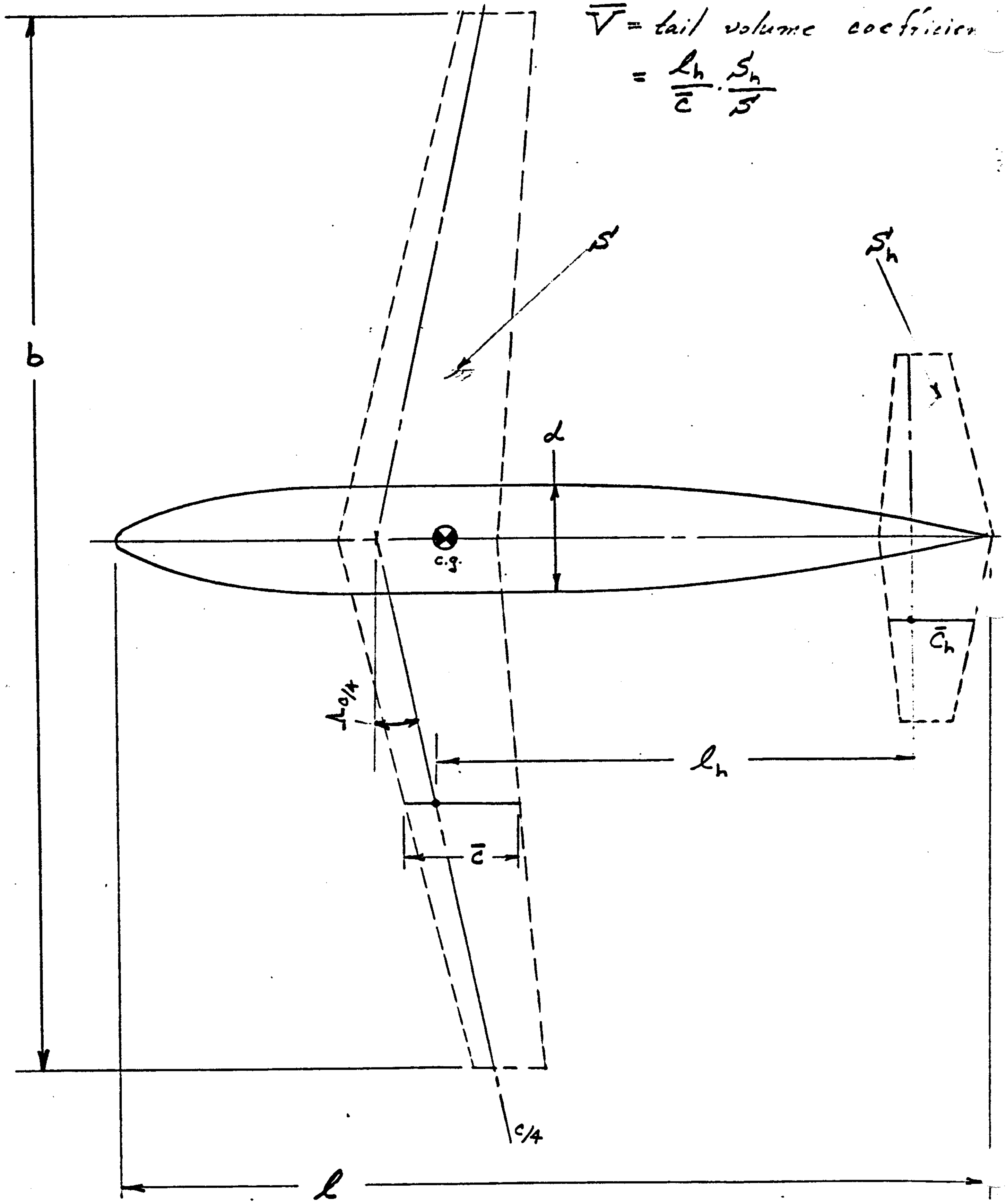
Simple Geometry Definition

$$AR = b^2/S \doteq b/\bar{c}$$

$\bar{V}$ = tail volume coefficient

$$= \frac{l_h}{\bar{c}} \cdot \frac{S_h}{S}$$



Wing Alone L/D MAX Optimization
(OR "The Optimum Bird").

Assume: $C_D \equiv \dfrac{D}{\frac{1}{2}\rho V^2 S} = C_{D_0} + \dfrac{K C_L^2}{\pi AR}$; $AR = b^2/S$

further, assume: ① $K = f$(planform, AR, airfoil section behavior with C_L & R_n) ≥ 0

② $C_{D_0} = f(R_n, S_w)$
where S_w = "wetted area"

③ Equilibrium flight, ie
$L \cong W$, $T \cong D$, $\Sigma \vec{M} = \vec{0}$

Thus: $C_{D_0} = K_0 \overline{R_n}^{\,a} (S_w/S)$

$$\overline{R_n} = \frac{V\bar{c}}{\nu} = \frac{VS}{\nu b} = \left[\frac{2}{\rho\nu^2}\right]^{1/2}\left[\frac{W}{C_L AR}\right]^{1/2}$$

$$V = \left[\frac{2W/S}{\rho C_L}\right]^{1/2}$$

for a laminar flow on a flat plate : $K_0 = 1.328$
$a = -1/2$

" " turbulent flow " " " " : $K_0 = 0.074$
$a \cong -1/5$

$$(L/D)^{-1} = \frac{C_D}{C_L} = \frac{C_{D_0}}{C_L} + \frac{K C_L}{\pi AR} = K_0\left[\frac{2}{\rho\nu^2}\right]^{a/2}\left[\frac{W}{AR}\right]^{a/2} \cdot C_L^{-\frac{a}{2}-1} + \frac{K}{\pi AR} C_L$$

(2)

Thus, the general optimization problem becomes:

$$\boxed{\begin{aligned} &\text{Minimize} \quad \phi_0 = \left(\frac{D}{L}\right) = \left(\frac{1}{L/D}\right)^{-1} = C_{01}\, C_L^{-\left(\frac{a}{2}+1\right)} + C_{02}\, C_L \\ &\qquad \text{where:} \quad C_{01} = K_0 \left(\frac{2}{\rho V^2}\right)^{a/2} \left(\frac{S_w}{S}\right) \left[\frac{W}{AR}\right]^{a/2} \\ &\qquad\qquad\qquad C_{02} = \frac{K}{\pi AR} \end{aligned}}$$

This is a trivial zero degree of difficulty geometric program. The general solution is:

$$\Phi_0^* = (D/L)_{min} = \frac{1}{(L/D)_{max}} = d(\underline{w}) = \left[\frac{C_{01}}{w_{01}}\right]^{w_{01}} \cdot \left[\frac{C_{02}}{w_{02}}\right]^{w_{02}}$$

$$\text{where:} \quad \left.\begin{aligned} w_{01} + w_{02} &= 1 \\ -\left(\frac{a}{2}+1\right) w_{01} + w_{02} &= 0 \end{aligned}\right\} \quad \begin{aligned} w_{01}^* &= \frac{2}{4+a} \\ w_{02}^* &= \frac{2+a}{4+a} \end{aligned}$$

Thus:

$$\boxed{\begin{aligned} (L/D)_{max} &\sim \left(\frac{S_w}{S}\right)^{-\left(\frac{2}{4+a}\right)} \cdot W^{-\left(\frac{a}{4+a}\right)} \cdot AR^{2\left(\frac{a+1}{a+4}\right)} \\ C_L^* &= \left[\frac{2+a}{2}\left(\frac{C_{01}}{C_{02}}\right)\right]^{2/a+4} \sim W^{a/a+4} \cdot \left(\frac{S_w}{S}\right)^{2/a+4} \cdot AR^{\frac{2-a}{4+a}} \end{aligned}}$$

Now, consider three cases:

Case I: $a = 0$ (No scale effect)

$$w_{01} = w_{02} = 1/2 \qquad ; \qquad C_{01} \begin{cases} = K_{0_0}\left(\frac{S_w}{S}\right) \\ = C_{D_0} \end{cases} \quad , \quad C_{02} = \frac{K}{\pi AR}$$

$$(L/D)_{max} = \frac{1}{2}\left[\frac{\pi AR}{K C_{D_0}}\right]^{1/2} \qquad ; \qquad C_L^* = \left[\frac{\pi AR\, C_{D_0}}{K}\right]^{1/2}$$

$$C_D^* = 2 C_{D_0} = \frac{2K}{\pi AR} C_L^2$$

Case II: Fully turbulent flow $(a = -1/5)$, $K_{0_T} = 0.074$

$$w_{01} = 10/19 = 0.526 \qquad C_{01} = K_{0_T}\left(\frac{\rho v^2}{2}\right)^{1/10} \cdot \left(\frac{AR}{W}\right)^{1/10} \cdot \left(\frac{S_w}{S}\right)$$

$$w_{02} = 9/19 = 0.474 \qquad C_{02} = K/\pi AR$$

$$(L/D)_{max} = C_{0_T}^{-1}\left[\frac{W\, AR^8}{(S_w/S)^{10}}\right]^{1/19}$$

$$C_{0_T} = \left[\frac{19}{10} K_{0_T}\left(\frac{\rho v^2}{2}\right)^{1/10}\right]^{10/19} \cdot \left[\frac{19}{9}\frac{K}{\pi}\right]^{9/19}$$

Case III: Fully laminar flow $(a = -1/2)$ $K_{0_L} = 1.328$

$$w_{01} = 4/7 = 0.571 \qquad C_{01} = K_{0_L}\left(\frac{\rho v^2}{2}\right)^{1/4} \cdot \left(\frac{AR}{W}\right)^{1/4} \cdot \left(S_w/S\right)$$

$$w_{02} = 3/7 = 0.429 \qquad C_{02} = K/\pi AR$$

$$(L/D)_{max} = C_{0_L}^{-1}\left[\frac{W\, AR^2}{(S_w/S)^4}\right]^{1/7}$$

$$C_{0_L} = \left[\frac{7}{4} K_{0_L}\left(\frac{\rho v^2}{2}\right)^{1/4}\right]^{4/7} \cdot \left[\frac{7}{3}\frac{K}{\pi}\right]^{3/7}$$

Numerical Examples

Assume: $W = 1000$ lbs

$K = 1.4$

$s_w/s = 2.5$

std. Sea Level

$\rho = 0.002378$ slugs/ft^3

$\nu = 1.564 \times 10^{-4}$ ft^2/sec.

$a=0$: $L/D_{max} = 7.964\ AR^{1/2}$

$a=-1/5$: $L/D_{max} = 9.204\ AR^{0.4211}$

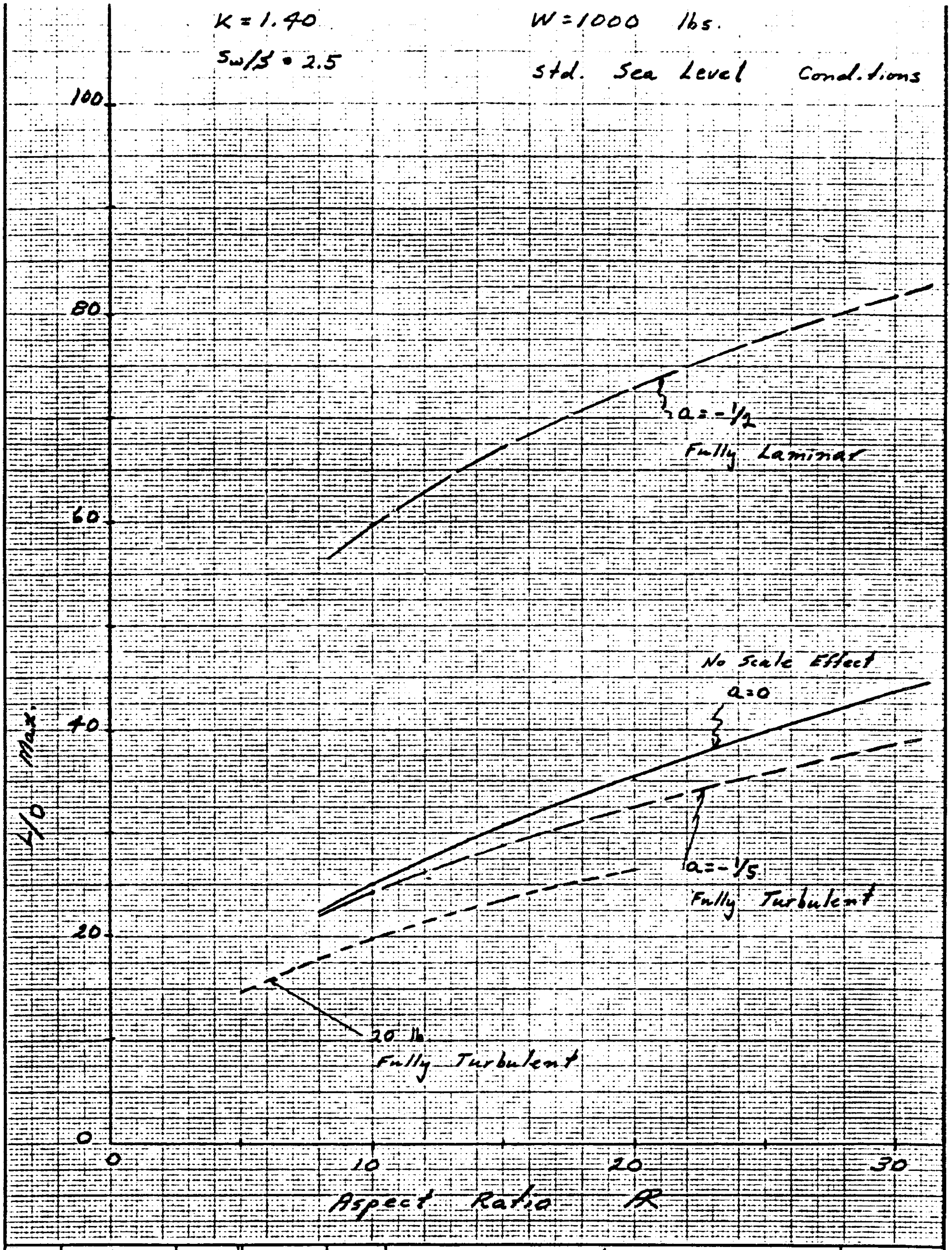
$a=-1/2$: $L/D_{max} = 30.930\ AR^{0.2857}$

when $a=0$

match: $\left.\begin{array}{l} K_{oT} = 0.074 \\ R_n = 4\times10^6 \\ a = -1/5 \\ s_w/s = 2.5 \end{array}\right\}$ $C_{D_0} = 0.00885$ = constant

		$AR=10$	$AR=15$	$AR=20$	$AR=25$	$AR=30$
No Scale Effect ($a=0$) $W=1000$ lbs	$L/D)_{max}$ =	25.18	30.84	35.61	39.82	43.62
	@ C_L^* =	0.446	0.564	0.630	0.705	0.772
	$\overline{R_n}\times10^{-6}$ =	2.78	2.02	1.65	1.40	1.22
Fully Laminar ($a=-1/2$) $W=1000$ lbs	$L/D)_{max}$ =	59.71	67.05	72.79	77.58	81.73
	@ C_L^* =	0.100	0.134	0.165	0.193	0.220
	$\overline{R_n}\times10^{-6}$ =	5.86	4.14	3.23	2.67	2.28
Fully Turbulent ($a=-1/5$) $W=1000$ lbs.	$L/D)_{max}$ =	24.27	28.79	32.50	35.70	38.55
Fully Turbulent ($W=20$ lbs)	$L/D)_{max} = 6.40\ W^{1/19} AR^{8/19}$	19.76	23.44	26.46	—	—

	(lbs)	$W=0.1$	$W=10$	$W=1000$	$W=10^4$	$W=10^5$
Fully Turbulent $AR=6$ W variable	L/D_{max} =	12.06	15.36	19.58	22.10	24.95



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K•E ALBANENE TRACING PAPER

WING DESIGN

<u>Goals</u>:

1. Adequate Cruise Performance

2. Adequate Stability & Control Characteristics and Off-Design Performance

   a. High-Speed

      - buffet

      - "Mach tuck"

   b. Low-Speed

      - Stall behavior & $C_{L_{max}}$

      - Lateral control

   c. Take-off & climb (low drag)

3. Efficient Structural Shape

   a. Strength

   b. Rigidity/Flexibility

   c. Weight

   d. Service life

   e. Accessibility

   f. Development & manufacturing cost

4. Sufficient Volume

   a. Fuel

   b. Landing gear

WING ANALYSIS & DESIGN

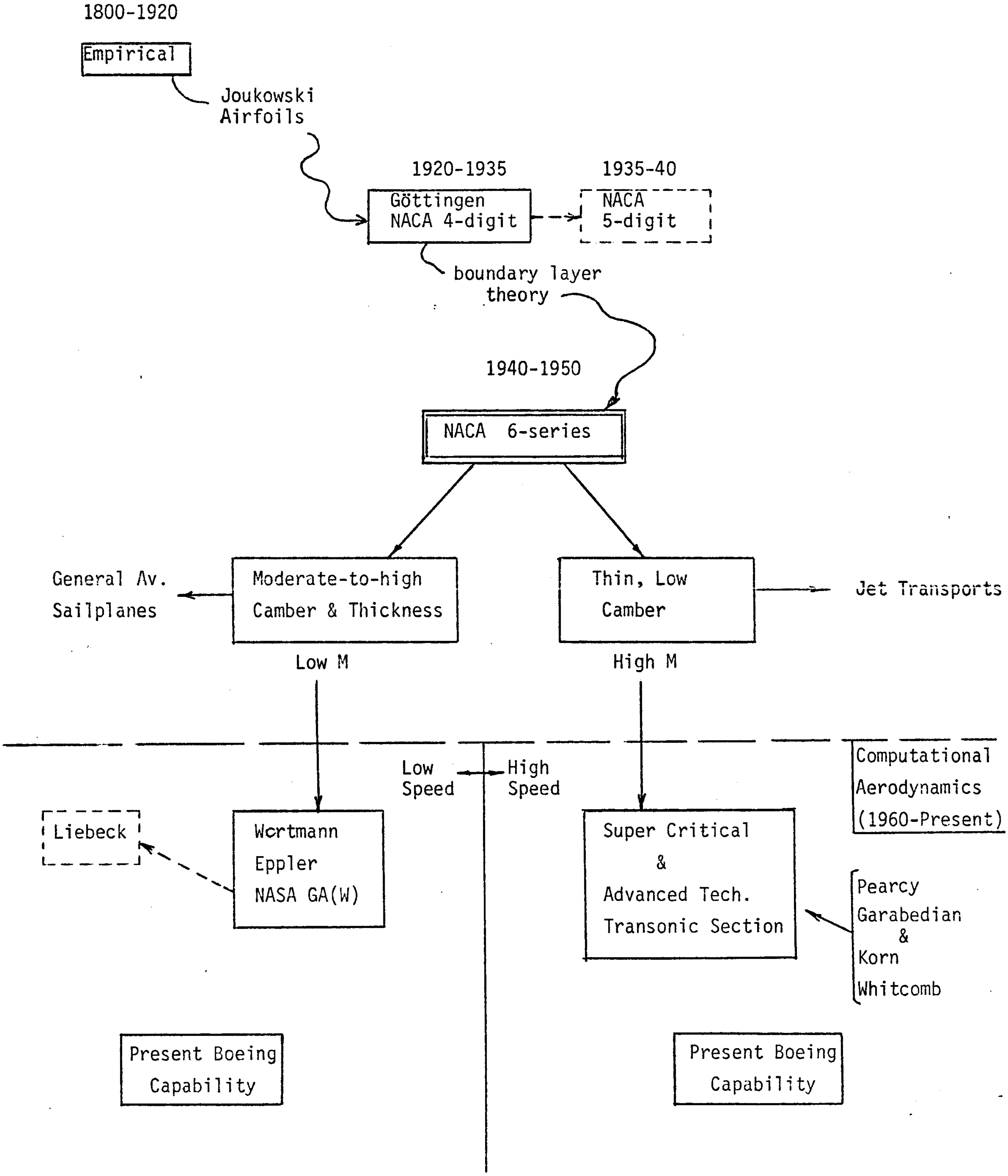
Straight Wings
(zero sweep angle)
$M \approx 0$
	- Lifting line theory
	- Planform effect ($\mathcal{R}$, taper)
	  - Lift
	  - Drag
	  - Stall ($C_{L_{max}}$)
	- Airfoil/Wing integration

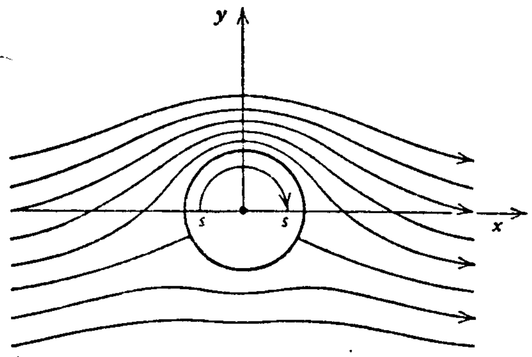
Swept Wings $M < 1$	- Simple sweep theory - Lifting surface theory - Airfoil/Wing integration - cruise (Locke's Method) - Swept wings - at high C_L
Wings With High-Lift Devices $M \approx 0$	- Straight wings - Swept wings
Supersonic Wings $M > 1$	- Conventional planforms - Peculiar planforms

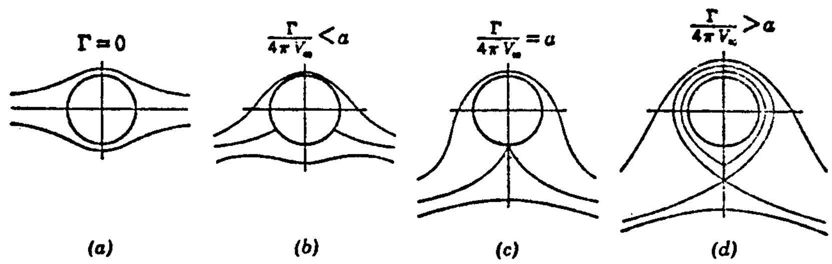
AERODYNAMIC ANALYSIS/DESIGN OF AIRFOILS & WINGS

Topic	Incompressible (Variable Rn)	Subsonic $0.2 < M < 0.7$	Transonic $0.8 < M < 1.2$	Supersonic $1.2 < M < 4\text{-}6$	Hypersonic & Rarefied Gas
2D Single Element Airfoils	1	2	3	4	--
2D Multielement Airfoils	11	--	--	N/A	N/A
3D Straight Wings (No high-lift devices-cruise)	5	7	--	14	--
3D Straight Wings (High-Lift)	6-12	--	--	--	N/A
3D Swept Wings (Cruise)	--	8	9	--	--
3D Swept Wings (High-Lift)	10-13	--	Maneuver & Buffet	Maneuver & Buffet	N/A
3D Delta Wings, Powered Lift, Etc.	--	--	--	15	16

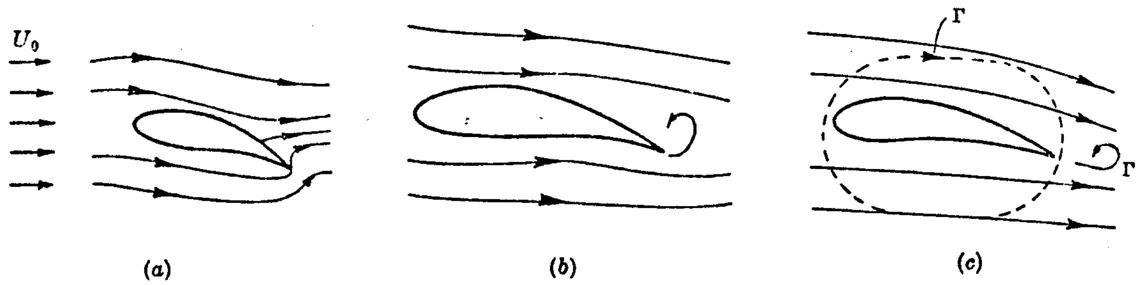
Single Element Airfoils



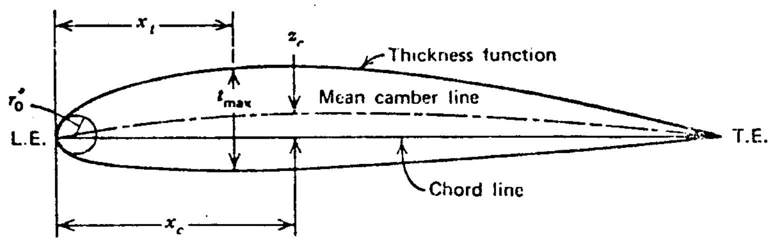


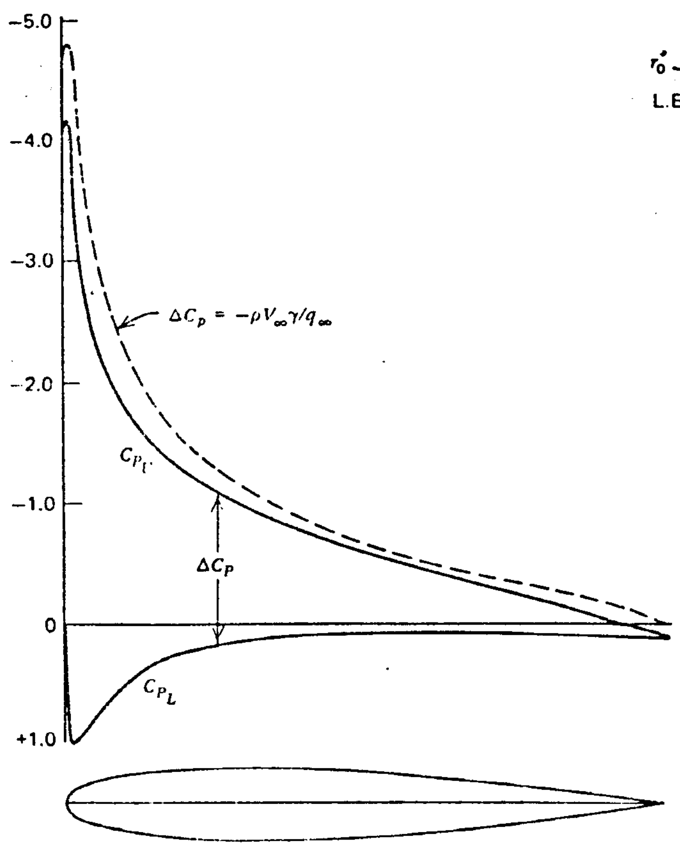


Stagnation points for several values of Γ.



Circulation and its formation about an airfoil. The stagnation point moves to the rear as the flow progresses and this transfer determines Γ. Fig. (*a*), (*b*) and (*c*) show successive stages in the startup of the airfoil.



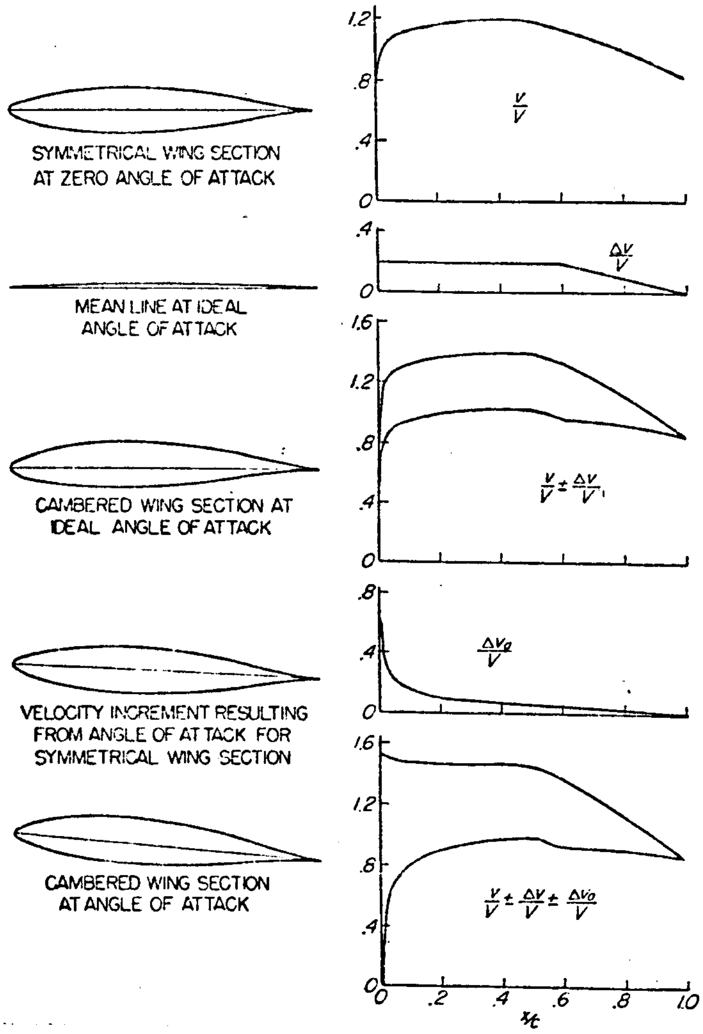


Distribution of pressure coefficient and ΔC_p on NACA 0012 airfoil at $\alpha = 9°$.

$$\Delta C_p = \frac{p_u - p_l}{q_\infty} = \frac{\Delta p}{q_\infty} \quad ; \; q_\infty = \frac{1}{2}\rho V_\infty$$

L' = lift per unit span =

$$= \rho V_\infty \Gamma = \int_0^c \Delta p \, dx = \rho V_\infty \int_0^c \gamma \, dx$$



$$\frac{p - p_\infty}{q_\infty} = 1 - \left(\frac{v}{V} \pm \frac{\Delta v}{V} \pm \frac{\Delta v_a}{V}\right)^2$$

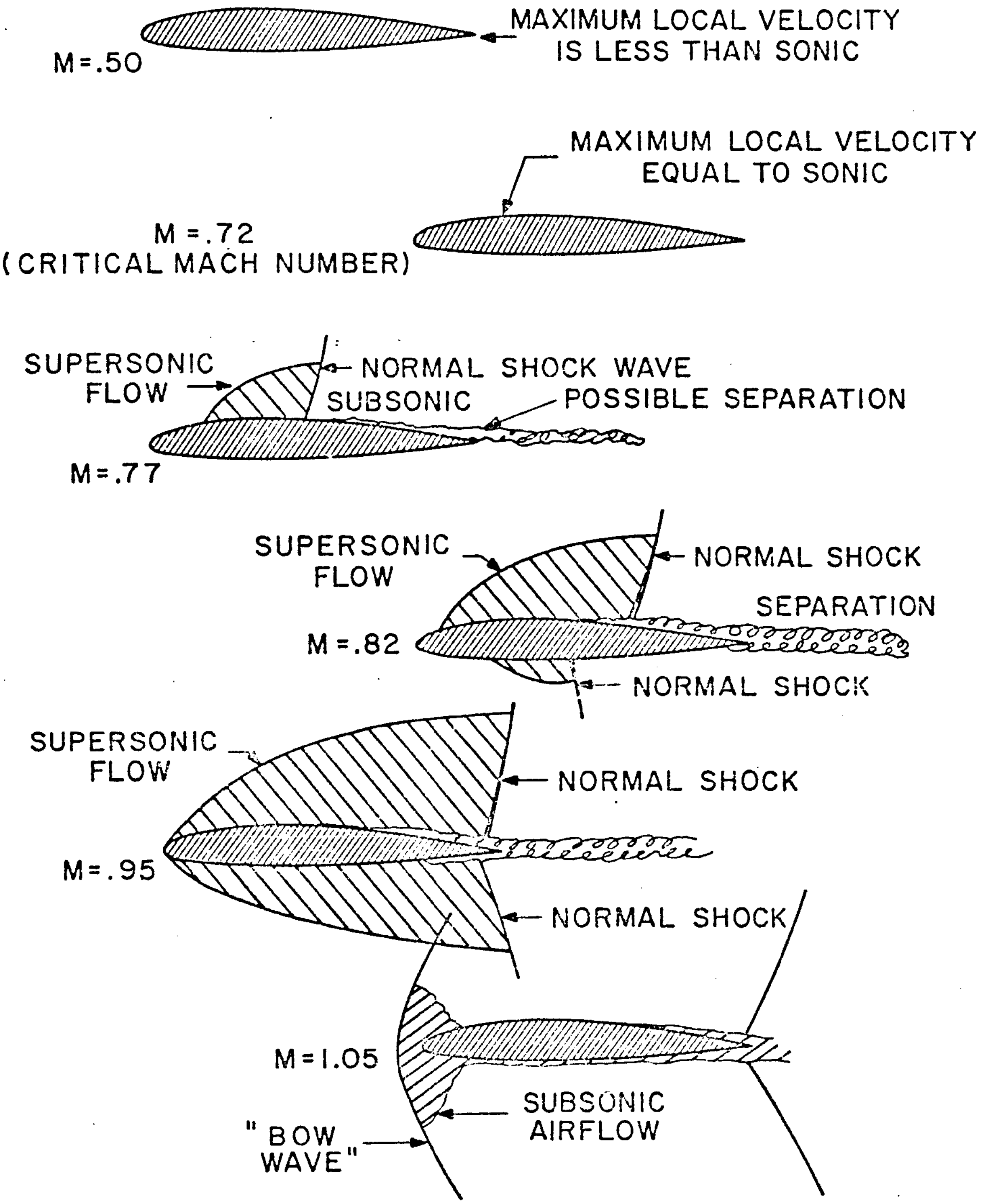


Fig. 2.9 Flow Patterns Around an Airfoil in Transonic Flow (Courtesy NASA)

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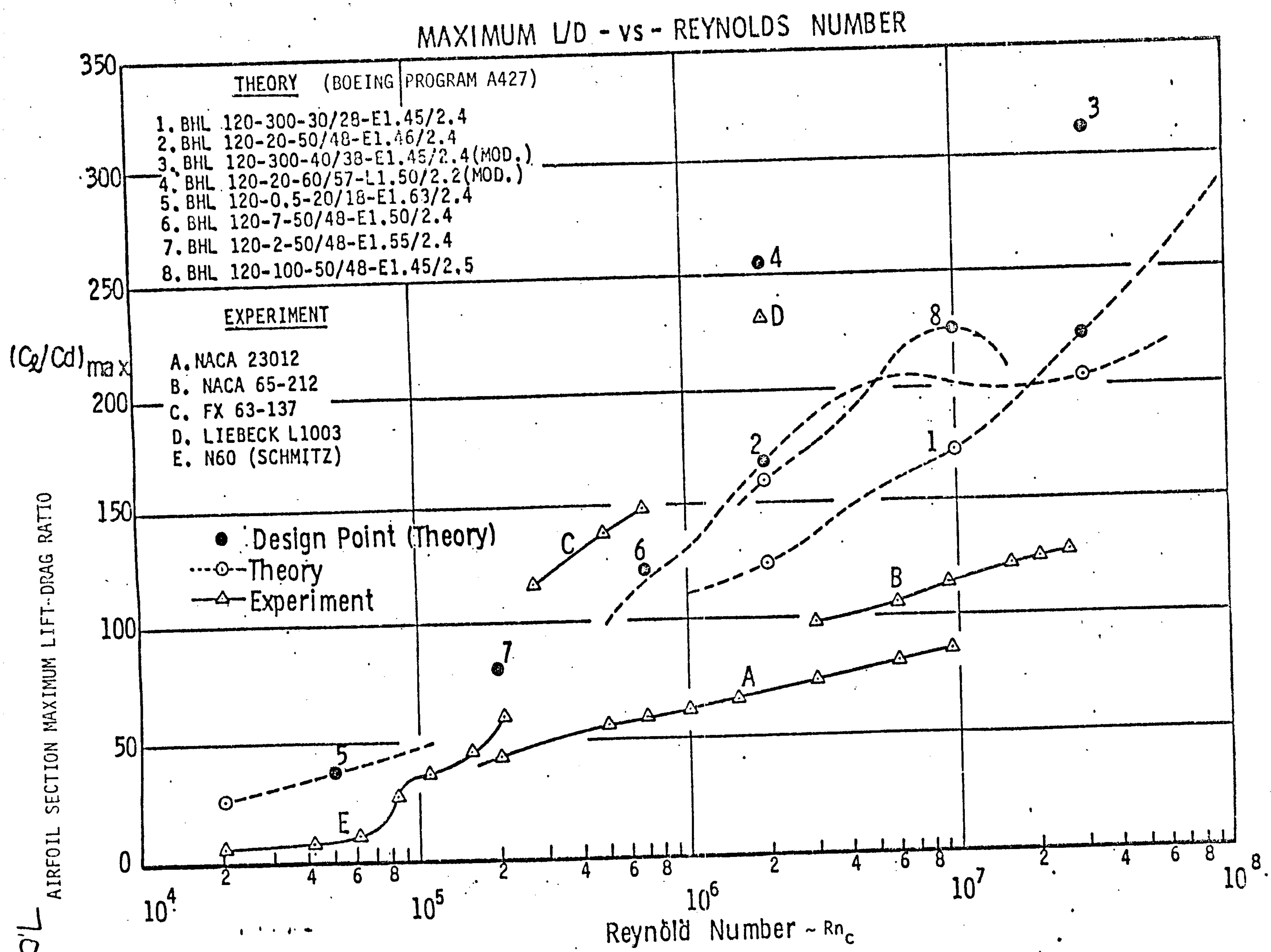


FIGURE C-4 VARIATION IN MAXIMUM AIRFOIL SECTION LIFT-DRAG RATIO WITH REYNOLDS NUMBER

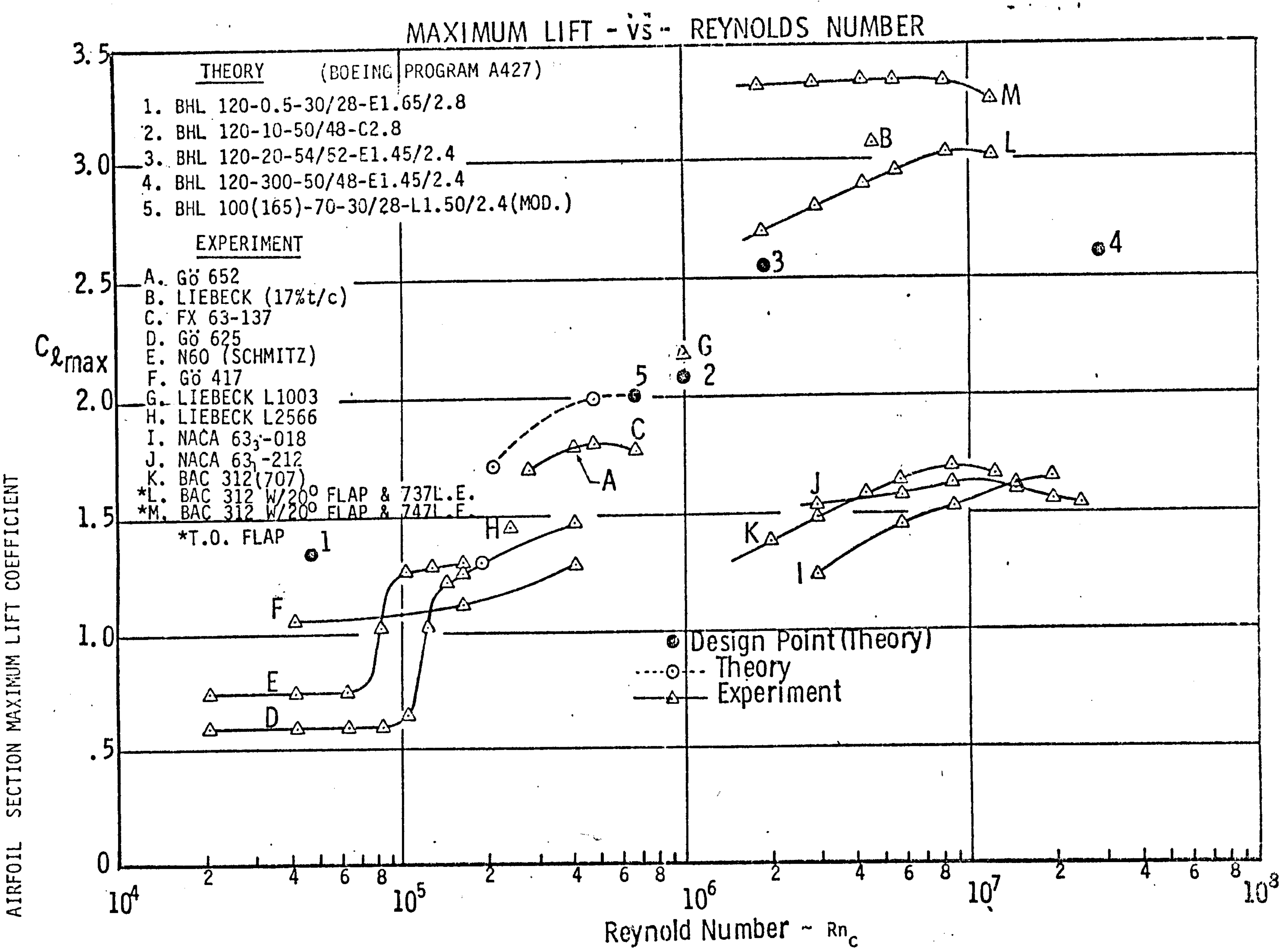


FIGURE C-5 VARIATION IN AIRFOIL MAXIMUM LIFT COEFFICIENT WITH REYNOLDS NUMBER

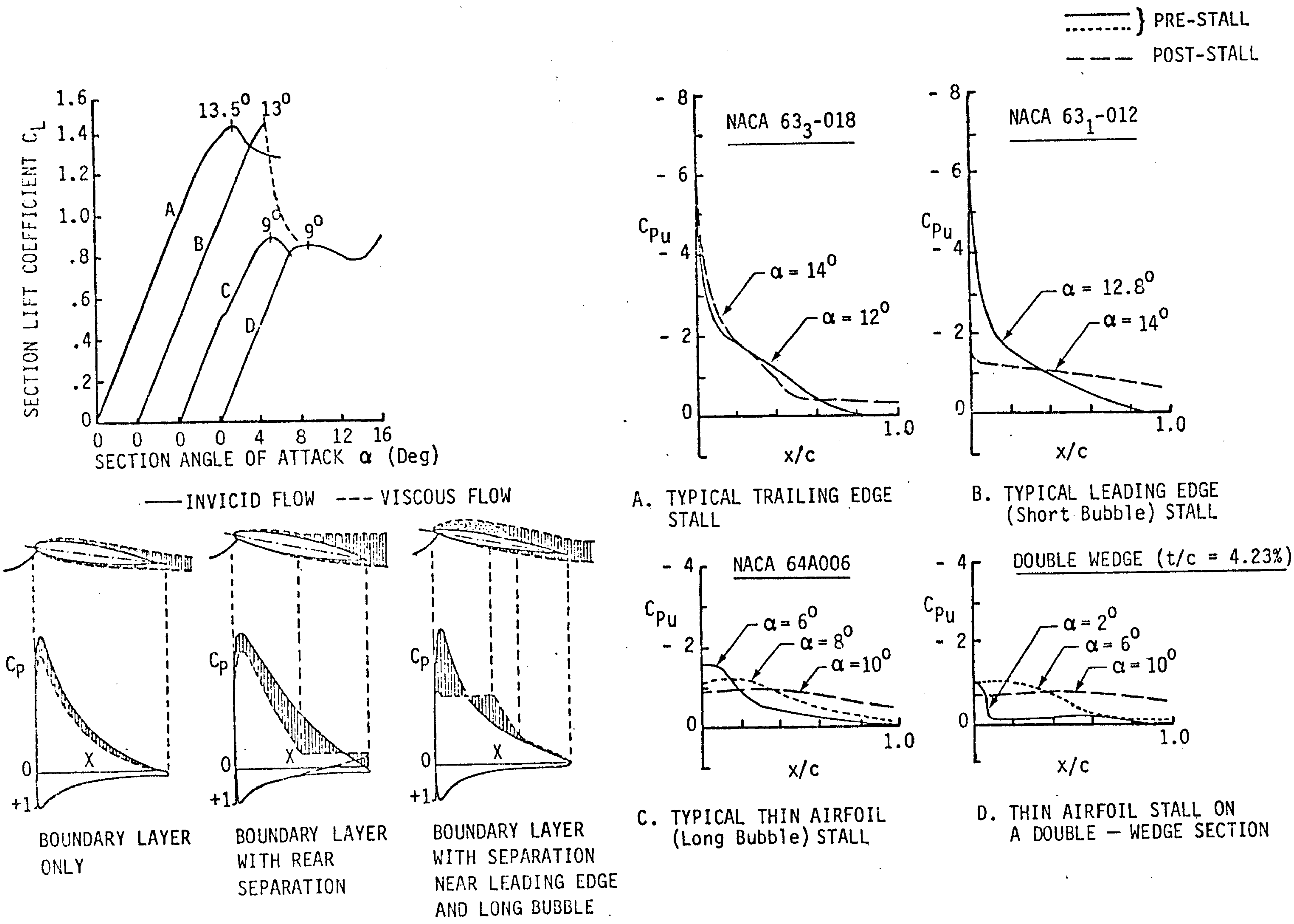


FIGURE C-10 TYPICAL SINGLE ELEMENT AIRFOIL STALL BEHAVIORS

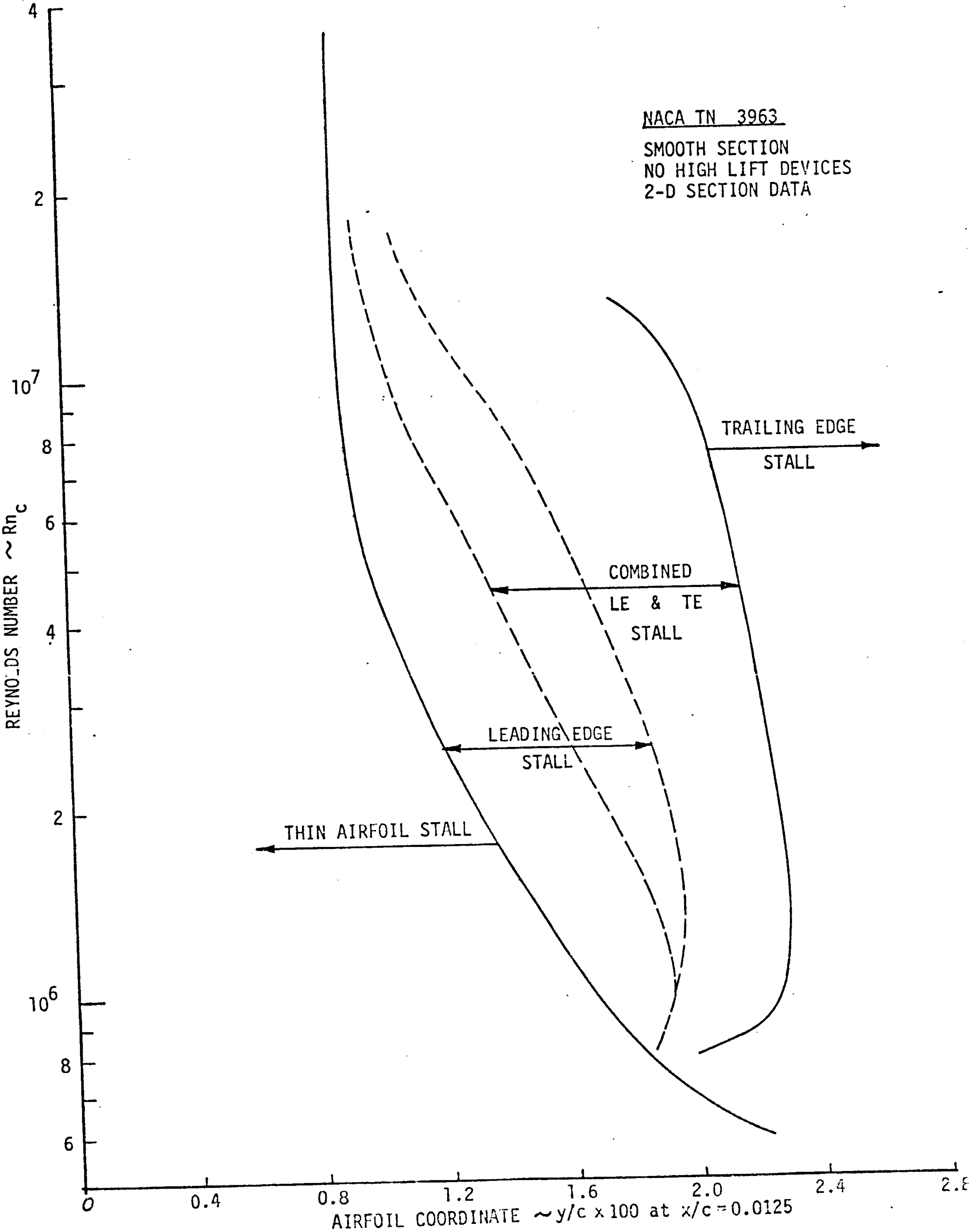
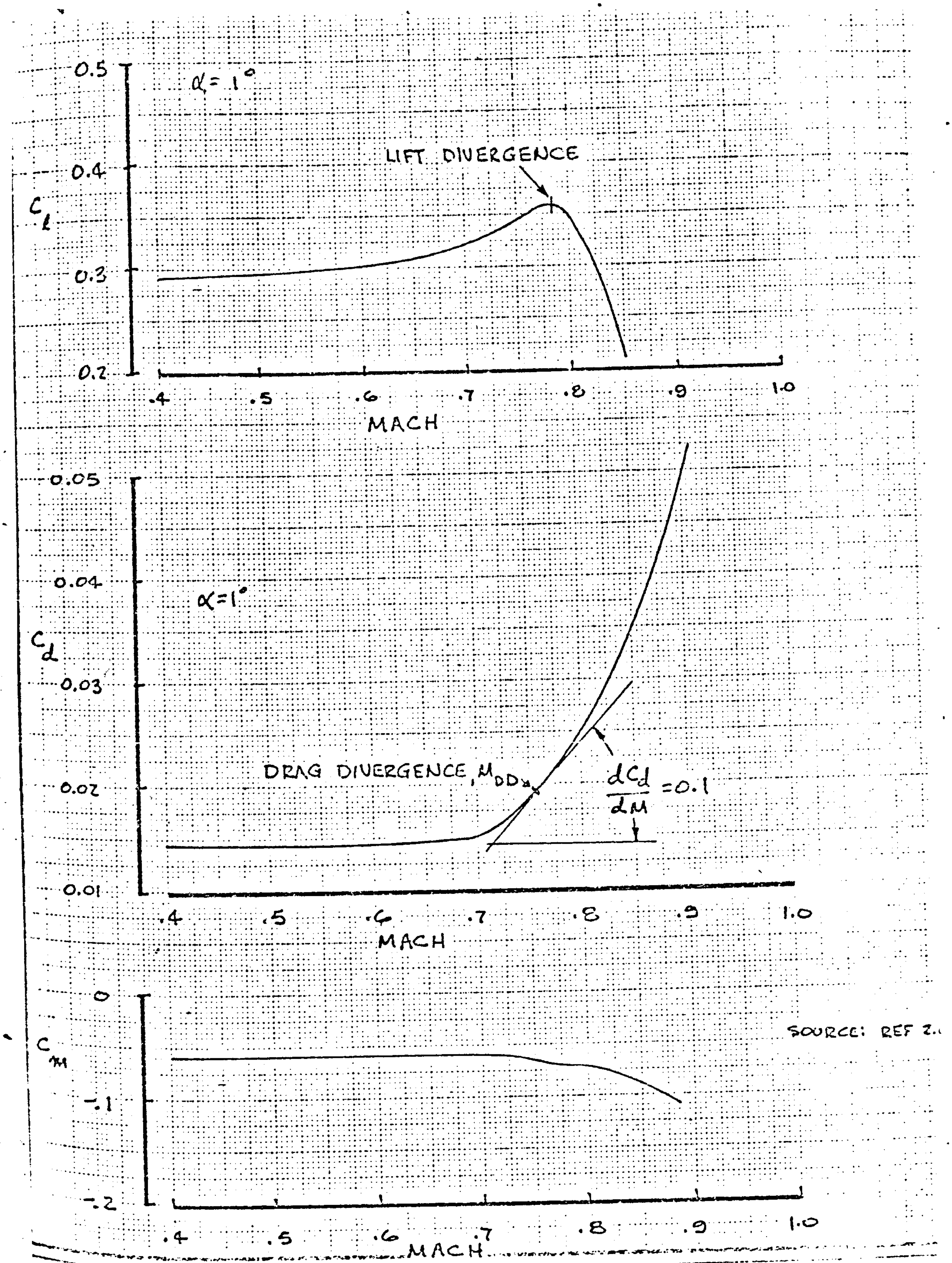


FIGURE D-3. CORRELATION BETWEEN AIRFOIL SECTION STALL BEHAVIOR AND REYNOLDS NUMBER



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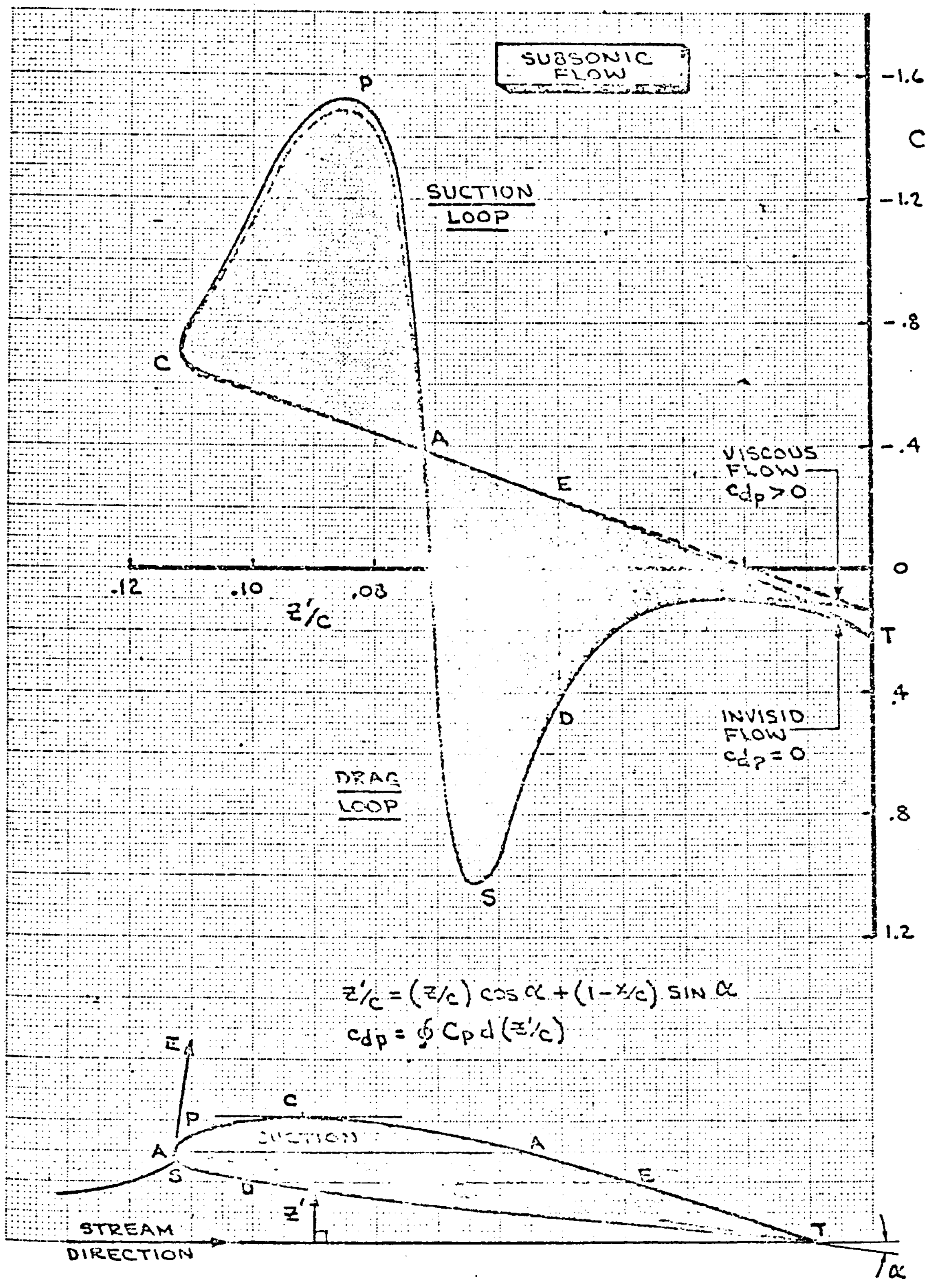
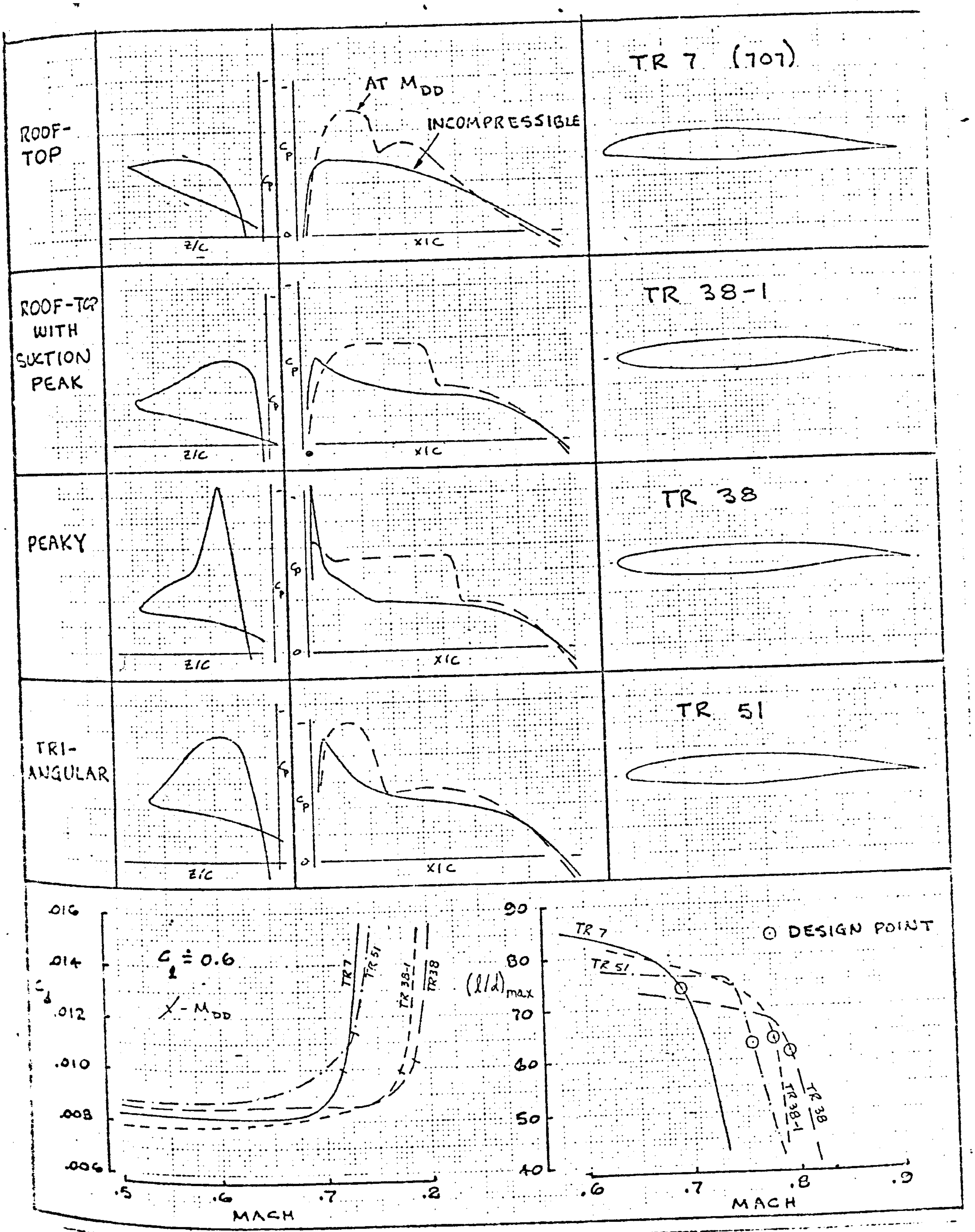
NACA 2309 TRANSONIC FORCE & MOMENT EFFECTS

FIG 2.9



C	E. GEORGE	3-75	REVISED	DATE	DRAG ANALYSIS OF SURFACE PRESSURES	FIG 2.12
CK						DC-41733 TN
D					THE [illegible] COMPANY	PAGE 2.27



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TRANSONIC AIRFOIL CLASSES FIG 2.17

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Part I

Science and Technology of Low Speed and Motorless Flight

Proceedings of a symposium held at
NASA Langley Research Center
Hampton, Virginia
March 29-30, 1979

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LOW-SPEED SINGLE-ELEMENT

AIRFOIL SYNTHESIS

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Boeing Commerical Airplane Co.

SUMMARY

Large quantities of experimental data exist on the characteristics of airfoils operating in the Reynolds number range between one and ten million, typical of conventional atmospheric wind tunnel operating conditions. Beyond either end of this range, however, good experimental data becomes scarce. Designers of model airplanes, hang gliders, ultralarge energy efficient transport aircraft, and bio-aerodynamicists attempting to evaluate the performance of natural flying devices are hard pressed to make the kinds of quality performance/design estimates taken for granted by sailplane and general aviation aerodynamicists. Even within the usual range of wind tunnel Reynolds numbers, much of the data is for "smooth" models which give little indication of how a section will perform on a wing of practical construction.

The purpose of this paper is to demonstrate the use of recently developed airfoil analysis/design computational tools to clarify, enrich and extend the existing experimental data base on low-speed, single element airfoils, and then proceed to a discussion of the problem of tailoring an airfoil for a specific application at its appropriate Reynolds number. This latter problem is approached by use of inverse (or "synthesis") techniques, wherein a desirable set of boundary layer characteristics, performance objectives, and constraints are specified, which then leads to derivation of a corresponding viscous flow pressure distribution. In this procedure, the airfoil shape required to produce the desired flow characteristics is only extracted towards the end of the design cycle. This synthesis process is contrasted with the traditional "analysis" (either experimental or computational) approach in which an initial profile shape is selected which then yields a pressure distribution and boundary layer characterisitics, and finally some performance level. The final configuration which provides the required performance is derived by cut-and-try adjustments to the shape.

Examples are presented which demonstrate the synthesis approach, following presentation of some historical information and background data which motivate the basic synthesis process.

INTRODUCTION

Since the dawn of human flight, enormous efforts have been expended on the design of efficient wings and their constituent airfoil sections. As such development became a race for ever increasing speed, the problems of low-speed flight frequently became relegated to the status of "off-design" conditions, with performance requirements met by fitting "high speed" cruise airfoils with increasingly complex and sophisticated high-lift devices. During the past forty years, relatively little attention has been given to the development of "optimized" low-speed airfoils by other than academicians and "cut-and-try" experimenters.

While frequently outside the mainstream of modern commerical interest, the range of low-speed flying devices (characterized by generally low values of the scale parameters Reynolds and Mach number) covers an enormous portion of the feasible flight spectrum. To place the subsequent discussion in a proper global context, Figure 1 has been prepared to demonstrate quantitatively the relationships between low-speed flight vehicle size and performance and the sometimes arcane parameter, Reynolds number. While "low-speed" generally implies low Reynolds and Mach numbers, it is worth noting that recent interest in ultralarge transport aircraft has now expanded the low-Mach number flight Reynolds number range from that typical of small insects ($10 < Rn < 10^4$) through devices like huge wing-in-ground effect aircraft (ref. 1) which may have chord Reynolds numbers approaching one billion at flight speeds on the order of 100 m/s ($M \sim 0.3$). Even a "small" monster like the Boeing 747 (average wing chord approximately 10 m) becomes a low-speed aircraft during approach, with typical average Reynolds numbers for the wing of 40 million at $M \sim 0.2$.

To discuss the full range of problems associated with wing/airfoil design for the range of vehicles shown in Figure 1 would require several books. The present paper is limited to a discussion of two aspects of the overall problem:

1. A brief survey of historical trends in low-speed, single-element airfoil development, culminating in a review of the present state of the art in analytic design methodology.

2. A demonstration of the value of modern computational capabilities to, first, clarify the performance characteristics of several existing low-speed airfoil sections for which experimental data exist; and then show how one may proceed to "synthesize" a suitable section for a specific application from a desired specification of boundary layer/pressure distribution characteristics.

NOTATION

AR — Aspect ratio = $b/\bar{c} = b^2/S$

b — Wing span (m)

c — chord (m)

$\bar{c}$ — Average chord = S/b (m)

C_d	Section drag coefficient
C_f	Skin friction coefficient
C_L	Wing lift coefficient = lift/qS
C_ℓ	Section lift coefficient
C_p	Pressure coefficient = $(p-p_\infty)/q_\infty$
C_m	Section pitching moment coefficient
H	Boundary layer form parameter $= \delta / \theta$
M	Mach number
p	Static pressure (N/m^2)
q	Dynamic pressure $= \frac{1}{2}\rho V^2$ (N/m)
Rn	Reynolds number = Vc/ν
S	Wing area (m^2)
t	Airfoil thickness (m)
V	Velocity (m/s)
v	Local velocity (m/s)
W	Weight (N)
x	Chordwise coordinate
z	Coordinate normal to chord

Greek symbols:

α	Angle of attack (degrees)
δ	Boundary layer displacement thickness = $\int_0^\infty \left(1 - \frac{v}{V_\infty}\right) dz$
ε	Section lift-drag ratio = C_ℓ / C_d
θ	Boundary layer momentum thickness = $\int_0^\infty \frac{v}{V_\infty}\left(1 - \frac{v}{V_\infty}\right) dz$
ν	Kinematic viscosity (1.46×10^{-5} m^2/s standard sea level)
ρ	Air mass density (1.225 kg/m^3 standard sea level)

Superscript:

( )* indicates "design condition"

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Subscript:

( )r	recovery point or region
( )tr	transition point or "trip" location
( )fp	fair point (see Fig. 9)
( )TE	trailing edge
$( )_\infty$	free-stream condition
( )u	airfoil upper surface value

HISTORICAL PERSPECTIVE

To clarify the present status of low-speed airfoil development, it is of interest to briefly review the history of how we got from there to here. A map of the route is shown in Figure 2. It is important to note that well into the present century airfoil "design" was a largely empirical process, drawing its main inspiration from natural models (i.e., birds), and only partially clarified and systematized by recourse to potential flow theory (e.g., Joukowski airfoils). Elaborate testing programs at Göttingen and by the NACA, among others, guided by intuition, experience, and inviscid theory eventually lead to the accumulation of masses of data and subsequent publication of airfoil section catalogs to aid designers.

It was not until the mid-1930's that the influence of viscous "scale effects" was appreciated, and boundary layer theory well enough developed, to allow the qualitative incorporation of viscous flow concepts into the design of "low-drag" sections. The main upshot of these new considerations was the famous NACA 6-series "laminar flow" airfoils. The accumulated results of fifty years of empiricism culminating in the matrix of 6-series sections are covered extensively in the classic catalogs by Abbott and von Doenhoff (ref. 2), Riegels (ref. 3) and reports such as those by Jacobs and Sherman (ref. 4).

The preeminence of the 6-series sections (slightly altered on occasion to the taste of the individual designer) lasted for nearly twenty years, and these sections have only been overshadowed since the late 1950's by the emergence of the revolution ushered in by the computer. While the equations of advanced potential flow methods and viscous flow theory can be concisely written, it is quite another matter to routinely solve analytically the complex flow fields around even "simple" airfoils in a real fluid. Thus, until the advent of large computers, theory could only guide what remained a largely experimental development effort.

The wind tunnel is a marvelous tool for describing <u>what</u> happens, but seldom provides much guidance on <u>why</u> a particular event (e.g., boundary layer separation) occurs. To go beyond the level of "design by testing," practical quantitative solutions to the equations of viscous flow were required to supplement empirical experience.

The remarkable success of computer based methods in improving airfoil performance beyond the NACA 6-series level is well demonstrated in the catalog of Wortmann FX-series sections (ref. 5) and the reports and papers listed in refs. 6 and 7. Despite this new progress, designers without access to a computer of sufficient size, or those lacking a sophisticated background in theoretical aerodynamics and mathematics are still forced to rely on catalog data and outmoded "simplified" theory. With very few exceptions (notably ref. 8), available good catalog data is for "ideal" surface quality wind tunnel models operating in the range $7 \times 10^5 \leq Rn < 10^7$. As a summary of the preceding historical discussion, Figure 3 shows some representative airfoils sections used, or specifically designed for, various categories of low-speed aircraft during the last eighty years. The variety of shapes even within a given category is sometimes bewildering.

LOW-SPEED AIRFOIL DESIGN

The general principles of low-speed, single-element airfoil design in light of modern theory have been discussed in detail by several authors, notably Wortmann (ref. 9-11), Miley (ref. 12) and Liebeck (ref. 13). A brief review is presented here in Appendix A.

Whether one is designing a new airfoil section or attempting to select one from a catalog, it is important that all the relevant criteria are kept clearly in mind. The author's list is as follows:

Basic Airfoil Selection/Design Criteria

1. Basic Operating Conditions (superscript * indicates design point):

   a. Lift Coefficient Range $\left(0 \leq C_{\ell_{\substack{min\\drag}}} \leq C_\ell^* \leq C_{\ell_{max}}\right)$

   b. Reynold Number Range $\left(Rn_{min} < Rn^* < Rn_{max}\right)$

   c. Mach Number Range $\left(0 \leq M^* < M_{crit}\right)$

2. Airfoil Characteristics Desired (Priorities to be established for each specific application):

   a. Low Drag (e.g., absolute minimum drag at C_ℓ^*, "low" drag over operating C_ℓ range).

   b. High Lift (e.g., absolute $C_{\ell_{max}}$, moderate $C_{\ell_{max}}$ with "gentle" stall).

c. Pitching Moment (e.g., positive moment for flying wing applications, low negative moment to minimize horizontal tail trim loads or aeroelastic effects on wing).

3. Practical Constraints:

a. Required thickness-chord ratio and/or required local structural thickness.
b. Anticipated surface quality (e.g., skin joints or slat/airfoil junctions which might force boundary layer transition).

High-Lift/Low Drag Design

From the preceeding list it can be seen that the airfoil selection/design process is complex and this partially accounts for the wide variety of section shapes shown in Figure 3, each intended to strike some particularly beneficial compromise between often conflicting requirements. It is seldom possible to state categorically that a particular section is the "best" one even for a given type of aircraft.

Within the overall low-speed performance spectrum, however, one is generally forced to bias the selection/design toward achievement of either: (a) low-drag, or (b) high-lift. No general rules can be given for how much "high-lift" one can achieve with a "low-drag" section or vice versa, although clues are beginning to emerge from modern viscous flow theory. General guidelines for good design can be formulated, and these are briefly reviewed in Appendix A.

It should be noted that the NACA 6-series airfoils are basically "low drag" sections. Their long reign is due more to the fortuitous fact that they scaled well with Mach number, rather than providing the long runs of laminar flow which was the original design objective. Only in the special case of applications to sailplane wings was the original objective met, practical construction and operational problems (bugs, paint, rivets, dimples, etc.) tending to abort the "laminar flow" behavior in other applications. None of the 6-series sections can be categorized as "high-lift" airfoils.

Empirical Data

With the preceeding list of airfoil selection/design criteria in mind, one can consult the various catalogs to see if a suitable section exists. Data from these standard sources (e.g., Refs. 2-5, 7, 8) is summarized in global terms in Figure 4.

Within the range of Reynolds number for which large quantities of data exist, a diligent searcher can find some apparently curious anomalies - specifically the "spectacular" Liebeck sections (ref. 13). That the Liebeck sections achieve the high-lift performance shown is no longer in serious question, nor are the reasons such performance is achieved. What remains unclear is the nature of the trade-offs in section characteristics which are available between the "feasible upper bound" represented by the Liebeck sections and the "top-of-the-line" conventional sections within the shaded bands shown in Figure 4.

As a prerequisite to discussion of systematic methods for evaluation of these trade-offs, some appreciation of the parameters of boundary layer theory as they relate to airfoil performance is required. Figures 5 through 8 show some examples of the boundary layer characteristics of several familiar sections and the relationships between this data; and the more traditional display of global performance data, section geometry and pressure distributions is discussed in detail in Appendix B.

AIRFOIL SYNTHESIS

To advance beyond an empirically based approach to airfoil selection, or to consider the prospect of tailoring airfoil sections to a specific application, it is necessary to understand the difference between a design approach based on "analysis" as contrasted with one based on "synthesis." The synthesis (inverse) approach to airfoil design begins with the boundary layer characteristics as they effect the pressure distribution and ultimately define and limit the performance of a section in every way. The airfoil <u>shape</u> is derived <u>last</u> in this process, and is that physically realizable contour which provides the desired flow characteristics. Synthesis is almost the direct opposite to the traditional "empirical" (analysis) approach wherein one <u>begins</u> with a <u>shape</u> which yields a pressure distribution and a set of boundary layer characteristics, and thus initial values of lift, drag and moment. Performance requirements are finally met by trial and error modification of the shape. Whether these modifications are made to a wind tunnel or computer model, the basic process is one of iterative cut-and-try until the solution "converges."

AN INVERSE AIRFOIL DESIGN TECHNIQUE

While the possibility of synthesizing an airfoil has been recognized for many years, it has only been possible to implement satisfactory inverse methods (based on modern boundary layer theory) since the advent of the computer. Synthesis approaches have been employed by Wortmann (ref. 9) and more recently by Liebeck (ref. 13). A very general technique for airfoil synthesis (applicable to both single- and multi-element section components) has recently been developed by Henderson (ref. 14), based on proven integral boundary layer techniques described largely in Schlichting (ref. 15). While the specific techniques used in the overall program may seem almost old fashioned, the program has proven to be very satisfactory in practice and is quite a powerful tool for both single and multi-element airfoil synthesis (particularly when coupled with the

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methods described in ref. 16). Details of the method are described in reference 14, and only the basic elements are listed here for reference.

Elements of an Inverse Boundary Layer Analysis and Design Technique

Component	Theory (ref. 15 except *)
Laminar Boundary Layer	Polhausen
Laminar Separation	Polhausen
Laminar Separation Bubble	Henderson (empirical)*
Transition	Granville
Turbulent Boundary Layer	Momentum integral Power law velocity profile Garner's eqn. for form parameter Ludwieg-Tillman eqn. for wall shear stress
Turbulent Separation	$H > 3.0$
Compressibility Corrections	Karman-Tsien*
Profile Drag	Squire and Young

Utilizing the methodology outlined above, it becomes possible to implement the airfoil design process shown in Figure 9. Once an "optimized" viscous flow pressure distribution and linear theory airfoil shape have been determined, the powerful methods described by Henderson in reference 16 (which also account for separated flows) are applied to arrive at the final airfoil geometry which yields that pressure distribution, and final analytic performance predictions are made.

Several points in this synthesis process need to be clarified. For example, any "airfoil" shape will produce a unique pressure distribution. The converse is not generally true. In order to assure that an initial "designed" pressure distribution will result in a closed, non-reentrant airfoil shape, an upper surface pressure distribution is designed free of geometrical constraints, and a lower surface pressure distribution is defined as that which will result in a section with an NACA 00XX thickness form. This yields a total pressure distribution which will result in a realizable airfoil of desired thickness. This initial lower surface pressure distribution and its corresponding boundary layer characteristics are usually poor. In the initial stage, however, it is the <u>upper</u> surface which is being optimized, and it is a simple matter to subsequently reconfigure the lower surface (guided by the preliminary result) to a more desirable form as indicated in Figure 9.

The program allows a rather arbitrary specification of upper surface recovery region form parameter (H) variation as a primary input. Thus one can systematically study the effect of this important parameter easily and in some detail before proceeding to more detailed design calculations. This feature will be demonstrated shortly. The significance of various form parameter variations is discussed in Appendix B.

The most difficult parameter to specify correctly at the outset is the trailing edge pressure coefficient. This parameter has a very powerful effect on the design lift level a theoretical section will achieve, and to date the determination of its final "correct" value has generally required an iterative approach. The problem is discussed at some length by Liebeck (ref. 13).

Probably the weakest part of the theoretical performance estimation procedure is calculation of profile drag. In principle, at the final stage in the design cycle one can integrate the total pressure and skin friction drag components and arrive at a total profile drag coefficient. Experience to date with viscous flow programs which accurately predict pressure distributions and hence lift and pitching moments gives generally less accurate drag estimates. This is due primarily to the fact that drag is usually two orders of magnitude lower than lift, and whereas errors in lift computations are small with a good pressure distribution predictor, errors in pressure integration (particularly in the leading edge region) tend to be on the same order as pressure drag values. Thus for simplicity, the present state of the art is to rely on the method of Squire and Young (ref. 15) for total drag prediction and, in the present case, a supplementary calculation of skin friction drag to provide a clarification of the magnitude of this component within the total drag value. This procedure has been found to be reasonably adequate, at least for purposes of <u>comparing</u> the drags of single-element sections. While <u>absolute</u> values of Squire and Young drag may sometimes be questionable, <u>anyone</u> experienced with the pecularities of two-dimensional wind tunnel testing (particularly at high-lift values) must realize the magnitude of the error band in "good" experimental drag data.

SOME RESULTS

To indicate the use of the above methodology, two examples have been chosen to demonstrate several aspects of the influence of Reynolds number on airfoil characteristics. Figure 10 demonstrates the results obtainable from a parametric study of the influence of variations of recovery point location and Reynolds number on a family of sections with simple roof-top pressure distributions (cf. Fig. 9), and a common specified exponential form factor variation in the recovery region. The principal observations to be made in this example are the significant difference in "optimum" recovery point between sections designed (for

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high lift-drag ratios) at two million and thirty million Reynolds number, and the ultimate desirability of designing to full-scale Reynolds number conditions (i.e., 30×10^6 in this case) to achieve maximum performance, despite the fact that such results may appear inferior to those obtained from a design optimized at wind tunnel conditions when both are tested at low Reynold numbers.

Figure 11 shows the effect of a systematic variation of recovery region form parameter on the shape and characteristics of three airfoils designed to the same lift coefficient level at a Reynolds number of five-hundred thousand. The performance characteristics of these sections are summarized in Figure 12, and clearly show the trades available in lift, drag, pitching moment and stall break from different specifications of recovery region characteristics.

The results shown in Figure 12 are generally nonobvious and are of some interest in view of the discussion in Appendix B and the fact that relatively little modern experimental data exists for sections designed specifically for this low value of Reynolds number. The stall behavior of the three sections can be understood on the basis of the discussion in Appendix B regarding the correlation between boundary layer form parameter (H) variation and upper surface separation progression.

A more subtle and remarkable aspect of the results shown in Figure 12 is that the net Squire-Young drag of all three sections at the design point lift coefficient is nearly the same. The rate at which the drag rises between the design point and maximum lift coefficients will be different, however, reflecting the way in which flow separation progresses on the three sections as stall is approached. The example calculations also show the relative values of upper surface recovery region (turbulent) skin friction coefficient relative to the total upper surface profile drag coefficient. Although the highly concave recovery pressure distribution of Airfoil C (which approaches a Stratford type recovery, c.f. Appendix B) shown in Figure 11 has the lowest skin friction coefficients, it also has the highest rate of growth (and final trailing edge value) of boundary layer momentum thickness. Thus while Airfoil C has the lowest skin friction drag it has the highest pressure drag and in the overall balance, all three sections exhibit similar net profile drag values. This effect is not limited to the low Reynolds number case shown. As Reynolds number increases, the pressure drag becomes the increasingly dominant drag term, and minimization of the recovery region turbulent skin friction coefficient by employing a Stratford type recovery becomes increasingly less satisfactory.

CONCLUDING COMMENTS

A review of the history and present state of the art of low-speed single-element airfoil design has been presented, leading to a description of a powerful new inverse boundary layer scheme which can be used to synthesize an airfoil section tailored to the requirements of a specific aircraft. The basic intent of this paper has been to provide background and motivation for this alternative approach to airfoil design, as contrasted with the more traditional "design by experiment/analysis" approach to the problem. Along the way (Appendix B) it has been possible to clarify the performance characteristics of sections of quite different geometry and design objectives, and indicate the influence of Reynolds number on both "low-drag" and "high-lift" sections. Several examples of parametric analyses using the "synthesis" methodology have been presented which only hint at the potential of these new techniques.

It has been shown that airfoil design (even when limited to very low Mach numbers and single-element sections) is a hugely complex problem to which no single "best" solution exists even for a single specialized category of aircraft type. On the other hand, it is clearly possible to derive a section biased and optimized to the taste of an individual aerodynamicist with a great deal more intelligence than was possible less than a decade ago. Much work still needs to be done, however, to finally free the hang glider designer from reliance on his present very slender catalog of airfoil candidates.

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APPENDIX A: BASIC AIRFOIL DESIGN

The purpose of this appendix is to provide a brief tutorial review of some of the principles of airfoil design. The discussion follows that of Wortmann (ref. 11), Miley (ref. 12) and Liebeck (ref. 13).

All practical airfoils will carry some lift loading (whether high, low, or moderate) at some desired operating condition, and this will be characterized by generation of some peak level of negative pressure coefficient on the upper surface of the section, followed by recovery to near free-stream conditions at the trailing edge. The pressure loading on the lower surface will depend on factors like required maximum section thickness, establishment of favorable pressure gradients for low-drag at the section design lift level, and the requirements of satisfactory "off-design" performance at low section lift coefficients. At some point on both surfaces of the contour, the initial run of laminar boundary layer flow will transition to turbulent flow, the particular transition points being strongly dependent on the Reynold number, the form of the pressure distribution (or the profile shape which generates it), the surface quality of the section, and the free-stream turbulence level. All other factors being equal, the natural transition point will move forward on the profile as Reynolds number increases.

At this point there is a parting of the ways as one seeks either high-lift, or low-drag performance at low-to-moderate lift coefficients. To achieve low-drag, the longest possible runs of laminar flow are desired on both surfaces of the section followed by an orderly transition to thin turbulent boundary layer flow as the pressure recovers to trailing edge conditions; and separation is to be avoided like the plague.

In the high-lift case, attention mainly focuses on the upper surface. As in the low-drag case laminar flow is sought, together with high negative pressures over the forward portion of the section. The problem in the high-lift case is not necessarily to delay the onset of turbulent flow, but rather to cause an orderly transition at some optimum point to a healthy thin turbulent boundary layer over the pressure recovery region to allow the flow to decelerate from the high peak values reached on the forward portion without significant separation. The "optimum" high-lift upper surface pressure distribution will thus be constructed to produce the highest possible loading on the forward portion of the profile, consistent with the recovery capability of the turbulent boundary layer beginning at an "optimum" transition point. At low Reynolds numbers, getting rid of laminar flow at the recovery point and avoidance of large scale laminar separation become a major consideration.

A major constraint on the high-lift section is the character of the stall break; all things being equal, a gradual stall progressing from the trailing edge is desired. It should also be noted that the bulk of "good" high-lift sections achieve their maximum lift coefficients after upper surface (trailing edge) separation has begun. Controlled laminar separation bubbles may even be tolerated if they lead to orderly transition to turbulent flow in the pressure recovery region and do not burst before trailing edge separation is well developed.

In the high-lift case, the lower surface pressure distribution will be tailored in much the same fashion as in the low-drag case, although the

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lower surface pressure distribution can be made to produce a significant portion of the net lift and/or alter the pitching moment characteristics. This factor and the influence of various forms of upper surface distribution on section pitching moment coefficients are indicated in Figures 9 through 12 and in Appendix B.

APPENDIX B: SOME RELATIONSHIPS BETWEEN AIRFOIL PERFORMANCE AND BOUNDARY LAYER CHARACTERISTICS

While most aerodynamicists have some appreciation of the section geometric parameters (e.g. thickness, camber, leading edge radius, trailing edge angle) which may influence performance, relatively few have a corresponding "feeling" for the fundamental parameters of boundary layer theory (e.g. form parameter, momentum thickness), and how these parameters are influenced by scale effects. The purpose of this appendix is to provide a brief evaluation of the boundary layer characteristics of several representative airfoils, and a description of how these parameters relate to the more familiar presentations of pressure distributions and global performance characteristics. An understanding of the connection between boundary layer behavior, pressure distribution, and section geometry as they influence performance is essential to success in the synthesis approach to design.

The performance characteristics of four familiar sections are shown in Figure 5. Two of these sections (the NACA 63_3-018 and Wortmann FX 61-184) have been designed primarily for low-drag, and the other two (the FX 74-CL6-140 and Liebeck L1003) for high-lift. These sections actually represent something of a continuum in that the NACA section is a classic "minimum drag" shape while the Liebeck is a pure "high-lift" section. The Wortmann FX 61-184 (ref. 5, 11) is a classic 1960 vintage sailplane section designed for "low-drag" over a "wide" range of lift coefficients, with a compromise struck between absolute low drag, thickness, and a very benevolent stall behavior at a moderate maximum lift coefficient.

The FX 74-CL6-140 (ref. 18) on the other hand, represents an attempt to design a section with the same level of maximum lift coefficient as the Liebeck, but with a biased compromise again being struck between thickness, maximum lift, wide "drag bucket" and satisfactory stall characteristics. All four sections are quite different in shape, and in the absence of detailed information on the types of pressure distribution and boundary layer characteristics (including an evaluation of the post-separated flow region) one is provided only superficial clues to why each of these sections exhibits such different performance characteristics.

As an aside, the influence of flow separation on the performance of a section and the importance of accurately modeling this effect in a theoretical design exercise have been graphically demonstrated by Henderson (ref. 16). Figure 6 shows an experimental lift curve for the NASA GA(W)-1 section (ref. 17) in comparison with theoretical calculations made with increasingly sophisticated analytical techniques. For this particular section, Figure 6 shows that modeling the attached boundary layer flow remains inadequate in predicting the variation in lift with angle of attack beyond 75% of the final maximum lift coefficient value. The full theory developed by Henderson (ref. 16), which models both the boundary layer and separation, provides excellent predictions however. This improved methodology (which extends to multielement sections) represents a major, and so far unique, advance in computational capability.

To better understand the differences in performance and shape between the sections shown in Figure 5, it is necessary to evaluate in detail the pressure distributions and boundary layer parameter (specifically the form parameter, H) variations for each section. Example data for the NACA 63_3-018 (ref. 19) at 2^o angle of attack (within the drag bucket of the section) are shown in Figure 7 for three widely different Reynolds numbers. The classic 6-series aft-end shape corresponds to a roughly linear rise in the recovery region pressure distribution, and consequent form parameter (H) variation shown. The influence of Reynolds number on the location of the point of natural transition is indicated, and clearly shows the difficulty of achieving long runs of laminar flow as Reynolds number increases.

As shown in Figure 11, the shape and magnitude of the form parameter (the ratio of boundary layer displacement thickness to momentum thickness) variation in the pressure recovery region of the airfoil correlate in general with the shape of the pressure distribution in this region. The specification of recovery region form parameter variation is one of the central inputs in the Henderson inverse method described previously. As discussed in Schlichting (ref. 15), laminar separation occurs when H reaches 3.5 and turbulent separation begins when H exceeds about 3.0. The influence of the H-factor variation on airfoil stall behavior will be discussed presently.

Wortmann (refs. 9-11) has argued that there are advantages to a "concave" recovery pressure distribution (with near constant value of recovery region form parameter) for drag reduction, compared to the linear or convex pressure distributions associated with earlier profiles, including many of the Göttingen/Joukowski airfoils (c.f. Figure 3). The basic principles of the design of Wortmann's sailplane and related sections (including the FX 61-184) with concave pressure rises have been thoroughly discussed in references 9 through 11, and by Miley (ref. 12). These references also discuss the importance of properly contouring both the upper and lower surfaces of low-drag profiles.

Turning attention to the high-lift airfoils cases, it is interesting to compare the pressure distributions and boundary layer characteristics of the Wortmann FX 74-CL4-140 (ref. 18) and Liebeck L1003 (ref. 13) shown in Figure 8, and contrast this data with that for the NACA 63_3-018 in Figure 7.

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The Liebeck sections are of great theoretical interest for several reasons. Members of the family apparently approach the upper limit of lift coefficient achievable with a single-element section without mechanical boundary layer control. The sections also exhibit commendably low drag coefficients in the region of the design lift coefficient and low pitching moments. In exchange for these desirable characteristics, the stall behavior is wretched and the undersurface separates at rather high (positive) lift coefficients, thus limiting "high-speed" performance. This latter factor can be partially ameliorated by use of a camber changing trailing edge flap; however, the abrupt stall behavior is a fundamental characteristic of the basic family.

The Liebeck sections have been theoretically designed by the previously described synthesis process, in this case by use of a Stratford recovery region pressure distribution (ref. 20) to establish the maximum level of negative pressure on the upper surface "roof top" region of the section. The Stratford recovery region pressure distribution is that which, for a turbulent flow, results in a boundary layer which is everywhere equally close to separation. Thus, to within the accuracy of the Stratford formulation, the recovery region boundary layer is either completely attached or completely separated - there is no (theoretical) middle ground. This factor accounts for the very abrupt stall behavior of the sections. Thus, by reliance on the Stratford distribution, Liebeck generated the single class of high lift sections which can be "optimized" and analyzed without recourse to <u>explicit</u> partially separated flow calculations. Herein lies the success Liebeck had in designing to very much higher lift coefficients and section lift-drag ratios than had once been thought possible for a single-element section. The resulting shapes and pressure distributions for Liebeck sections are entirely non-obvious and the prospects of happening on them by "cut-and-try" were remote. This example provides a strong motivation for use of inverse methods.

The experimental verification of the predicted performance of the Liebeck sections, and by extension the validation of the Stratford theory, apparently opens a whole new prospect in high-lift airfoil design. However, the inability of Liebeck's methodology to account for partially separated flows, and the resulting formal reliance on the Stratford distribution, severely circumscribe the range of sections which can be designed. The possible trade-offs in performance between the Liebeck sections and the range of conventional sections shown in Figure 4 remain obscure.

The result of a highly sophisticated attempt to design such an "intermediate" airfoil, which trades some drag and thickness for a better stall behavior, while acheiving the same high-lift level, is represented by the Wortmann FX 74-CL(X)-140 pair discussed in ref. 18. Referring to Figure 8, one sees that the Liebeck and Wortmann pressure distributions are quite different, although both have "concave" distributions in the recovery region. Where Liebeck uses a well defined "instability" region as described by Miley (ref. 12) to achieve orderly transition to turbulent flow in the recovery region, Wortmann forces the formation of a "well-behaved" thin laminar separation bubble which acts as a passive boundary layer trip.

Reviewing the performance curves for the Wortmann and Liebeck high-lift sections shown in Figure 5, one sees the consequences of the two approaches to the design problem. Looking at the resulting airfoil shapes and pressure distributions in Figure 8, one sees little in common between the two sections however. To see how "equally" high-lift coefficients are generated by two such dissimilar sections, one must refer to the details of the boundary layer characteristics for the two airfoils.

For both the Liebeck and Wortmann sections, recovery begins at about 40% of the chord aft of the leading edge. Prior to this, the "laminar H" for the Liebeck section is nearly constant through the instability region, falling abruptly to an initial "turbulent" value as the flow transitions. By contrast, on the Wortmann section the laminar H rises abruptly prior to transition until a value of H for laminar separation is reached, following which a "short bubble" is formed leading to transition and turbulent reattachment at the beginning of the recovery region.

Once into the recovery region, the turbulent form parameters on the Liebeck section rise rapidly to an initially high value and then begin a further very gradual linear rise to a point just short of the trailing edge. This recovery region form parameter variation is characteristic of a Stratford imposed pressure distribution.

On the Wortmann section, the turbulent form parameter does not jump initially, but rises instead from its starting value behind the laminar bubble at a nearly identical rate to that of the Liebeck/Stratford, until it hooks upward at the end. The result is again a generally concave pressure distribution on the recovery portion of the Wortmann section.

Comparison of these form parameter variations for two very different "looking" sections clarifies much of the difference in stall behavior between the sections. On the Liebeck section, as angle of attack is increased beyond the "design" value (design lift coefficient equal to 1.8), the recovery region form parameter level is shifted progressively upward until a value of approximately 3.0 is reached, at which point turbulent separation begins. With the Liebeck/Stratford recovery pressure distribution, the form parameter level is almost constant across the bulk of the recovery region. Thus, if nothing else (a laminar short bubble for example) interferes, the whole recovery region becomes "critical" with respect to separation at nearly the same time, and an abrupt stall subsequently occurs. By contrast, the recovery region form parameter on the Wortmann section does not reach so uniform a critical level as angle of attack is increased towards stall. This is reflected in the more gradual stall break for the Wortmann section. The existence of the short bubble ahead of the recovery point on the Wortmann section

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throughout this approach to stall clouds the issue of how the stall progresses, and the critic will note that the stall behavior is not that much better than the Liebeck. That the stall progresses non-catastrophically (at least initially) from the trailing edge is indicated (c.f. Fig. 5) by the creeping drag rise as stall is approached and entered.

The preceeding examples are intended to be illustrative of a few well known sections and demonstrate some specific trends. The results shown are not necessarily typical of wide classes of sections and the possible ranges of form parameter variation and pressure distribution are enormous. These limited examples do, however, demonstrate the level of detailed analysis which modern theory can provide, and the necessity of delving this deeply into detail in order to understand differences and similarities between airfoils with different shapes and global performance characteristics, and finally to design an optimized profile for a given application. Obviously, much more could and should be said on these topics. In addition, much needs to be said regarding the problems of "optimizing" both upper and lower surface contours, and the influence on drag of form parameter variation, boundary layer momentum thickness, transition point, etc. All of these investigations require a technique by which the important variables of the problem can be varied in an orderly and systematic fashion, particularly as a function of Reynolds number. Such a technique has been described in this paper.

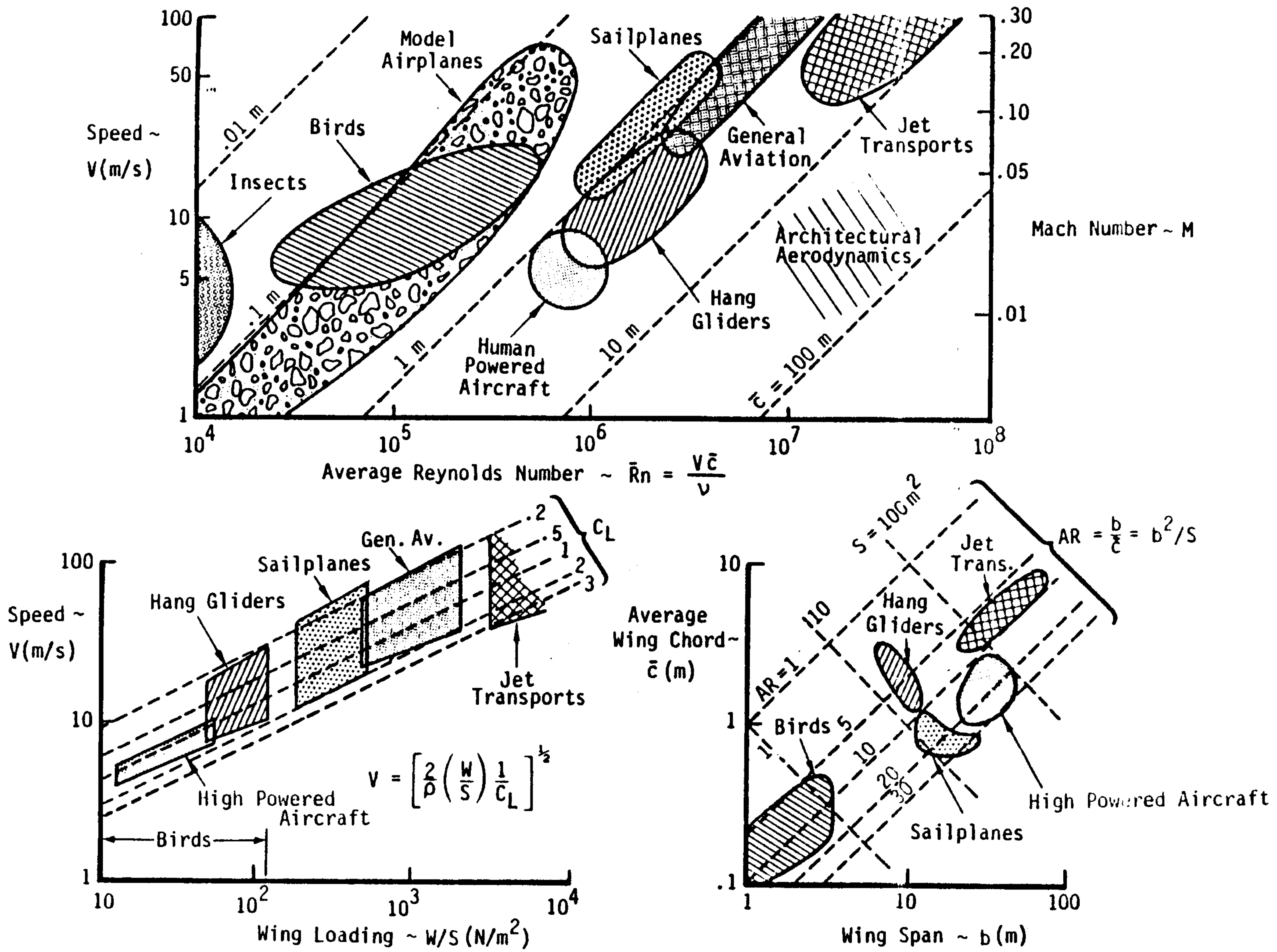

FIGURE 1 A GUIDE TO THE REGIONS OF LOW-SPEED FLIGHT AT STANDARD SEA LEVEL CONDITIONS

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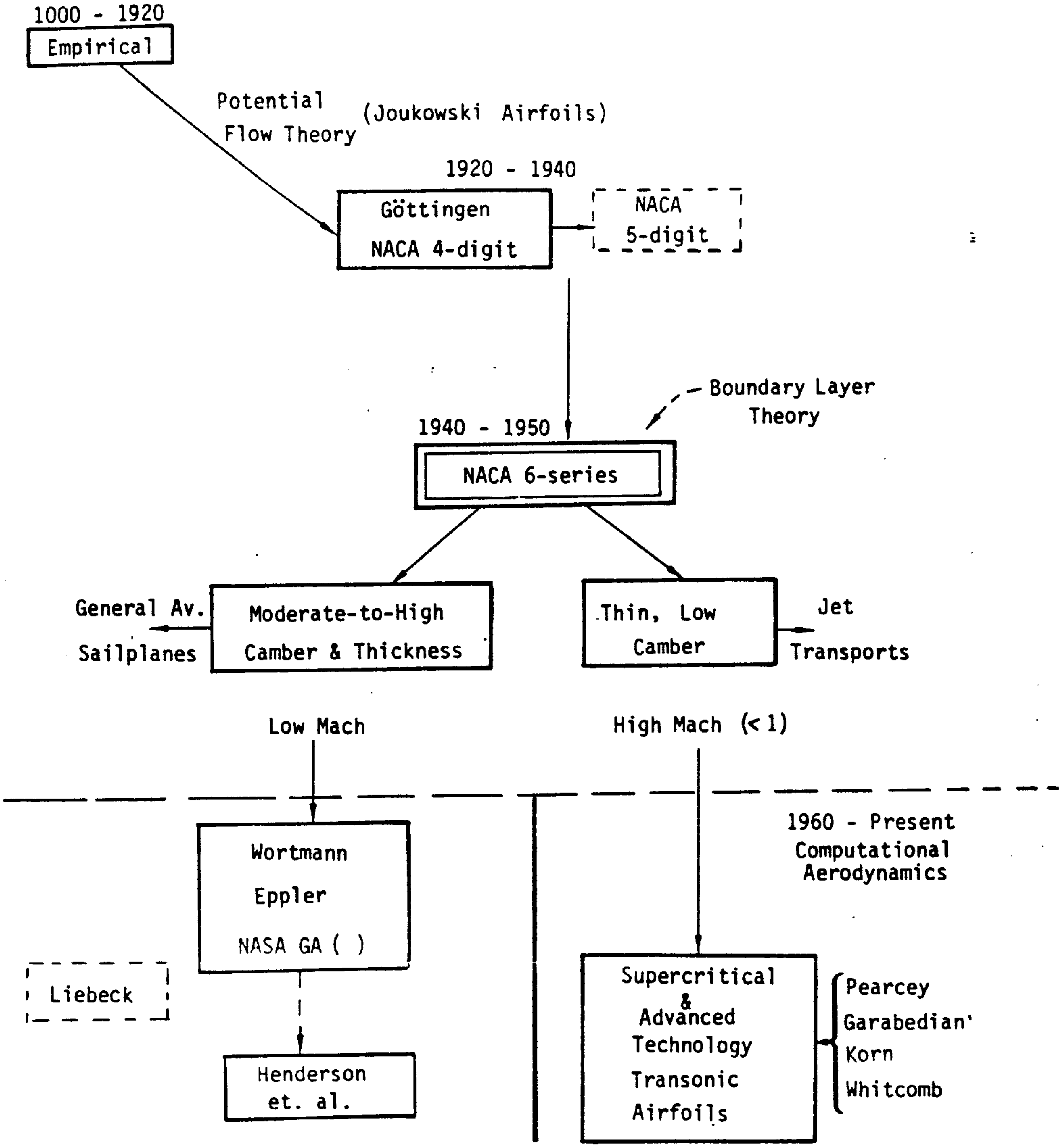

FIGURE 2 SINGLE ELEMENT AIRFOIL EVOLUTION

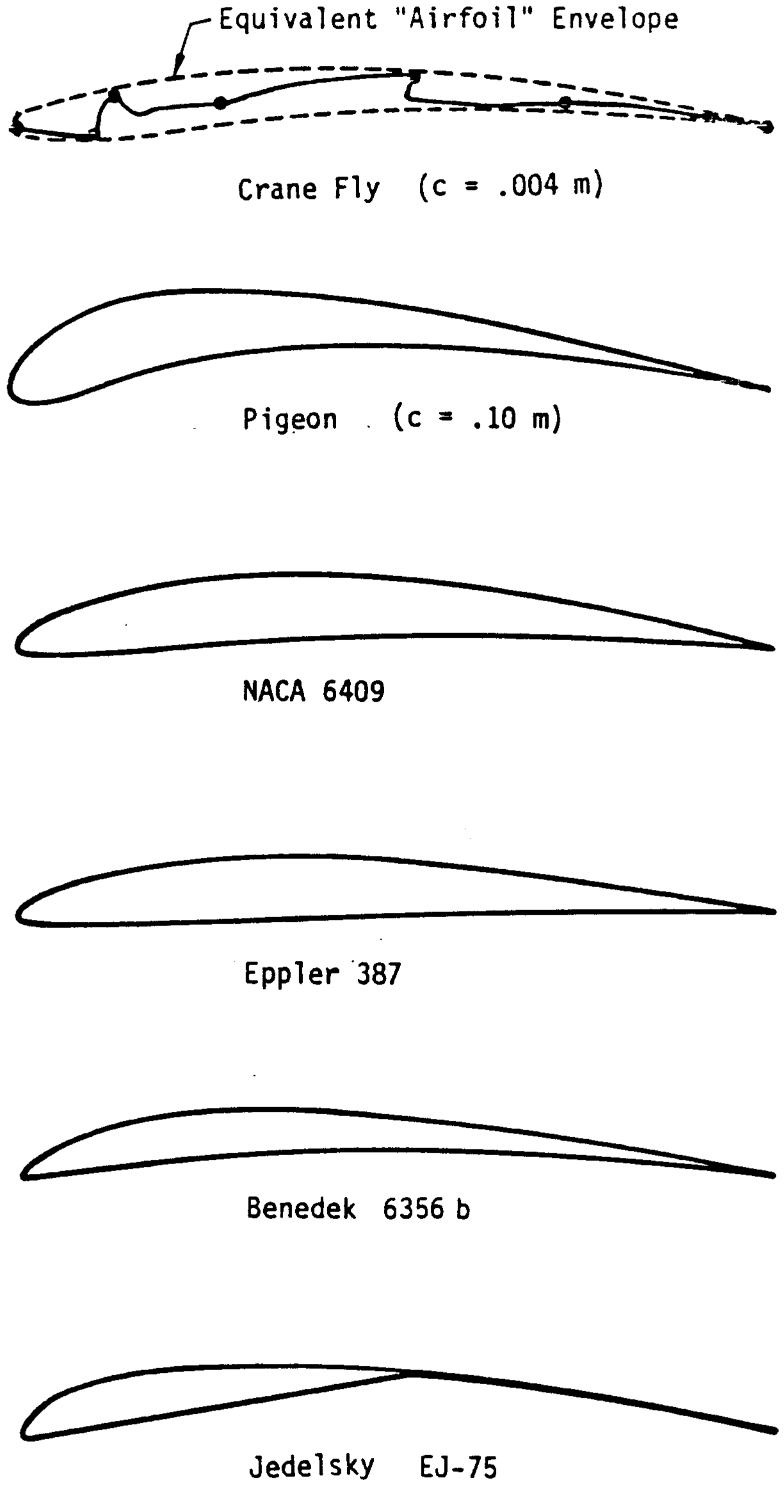

FIGURE 3a SOME VERY LOW-SPEED AIRFOILS

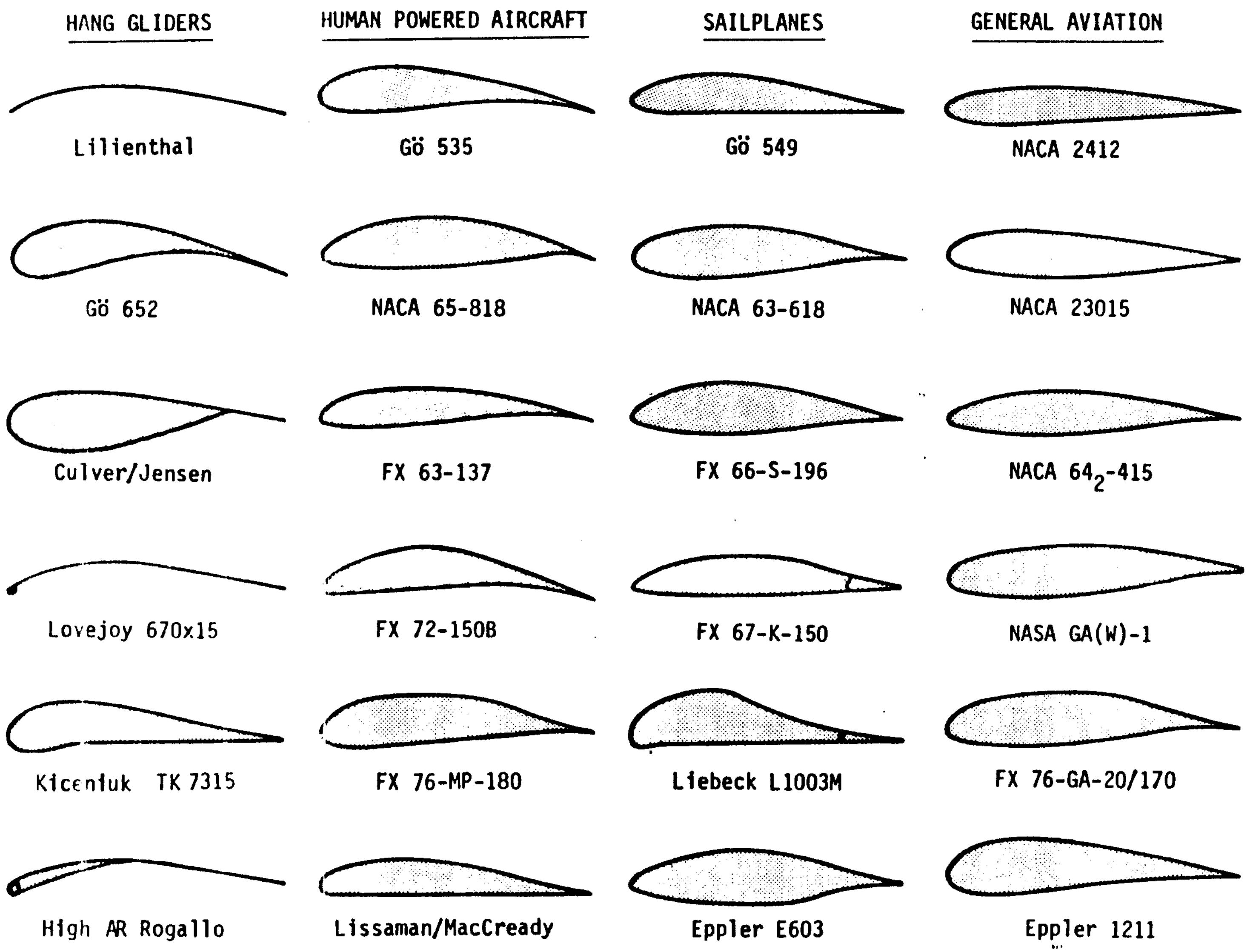

FIGURE 3b REPRESENTATIVE LOW-SPEED AIRFOIL SECTIONS

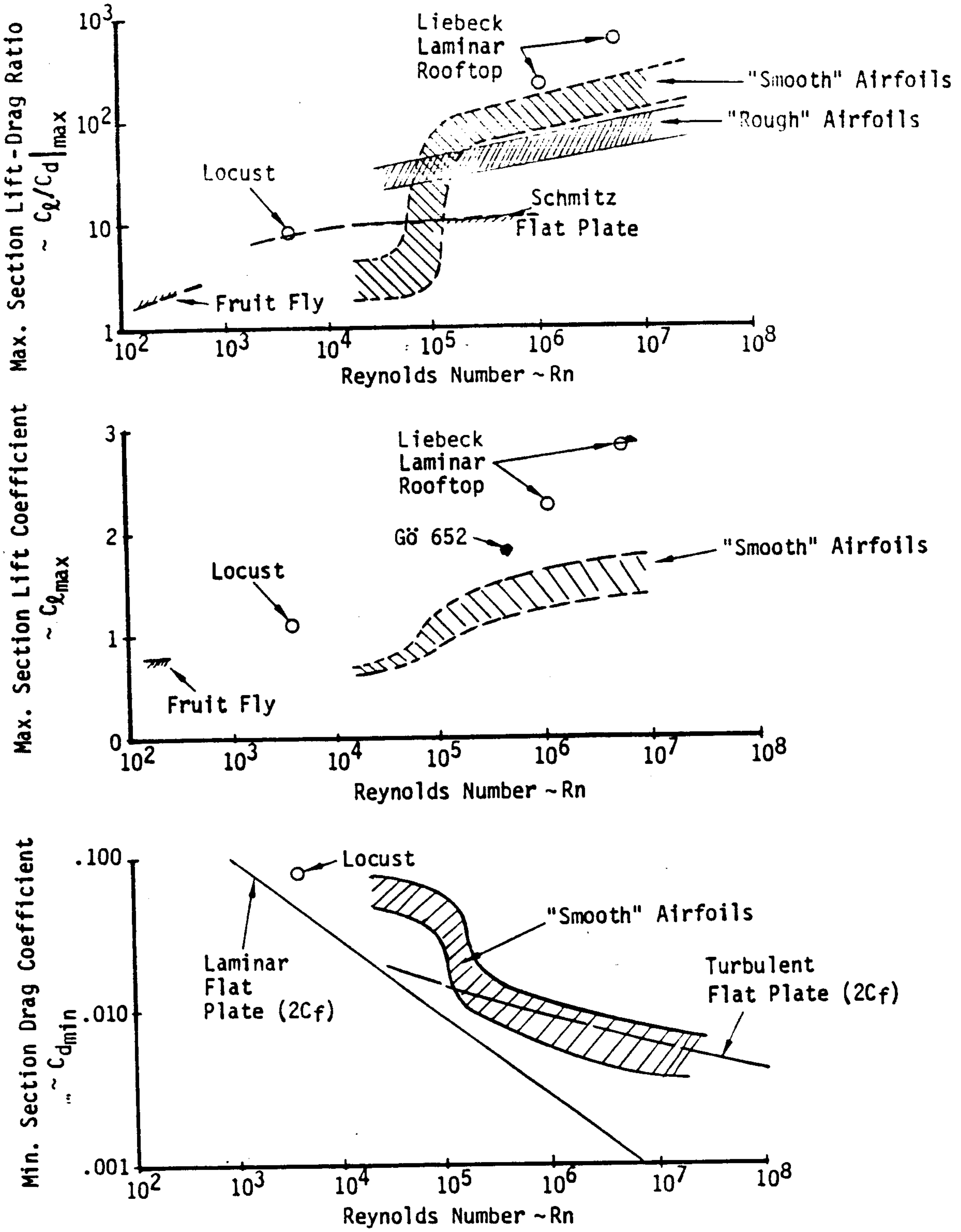

FIGURE 4 EMPIRICAL SURVEY OF LOW-SPEED SINGLE ELEMENT AIRFOIL DATA

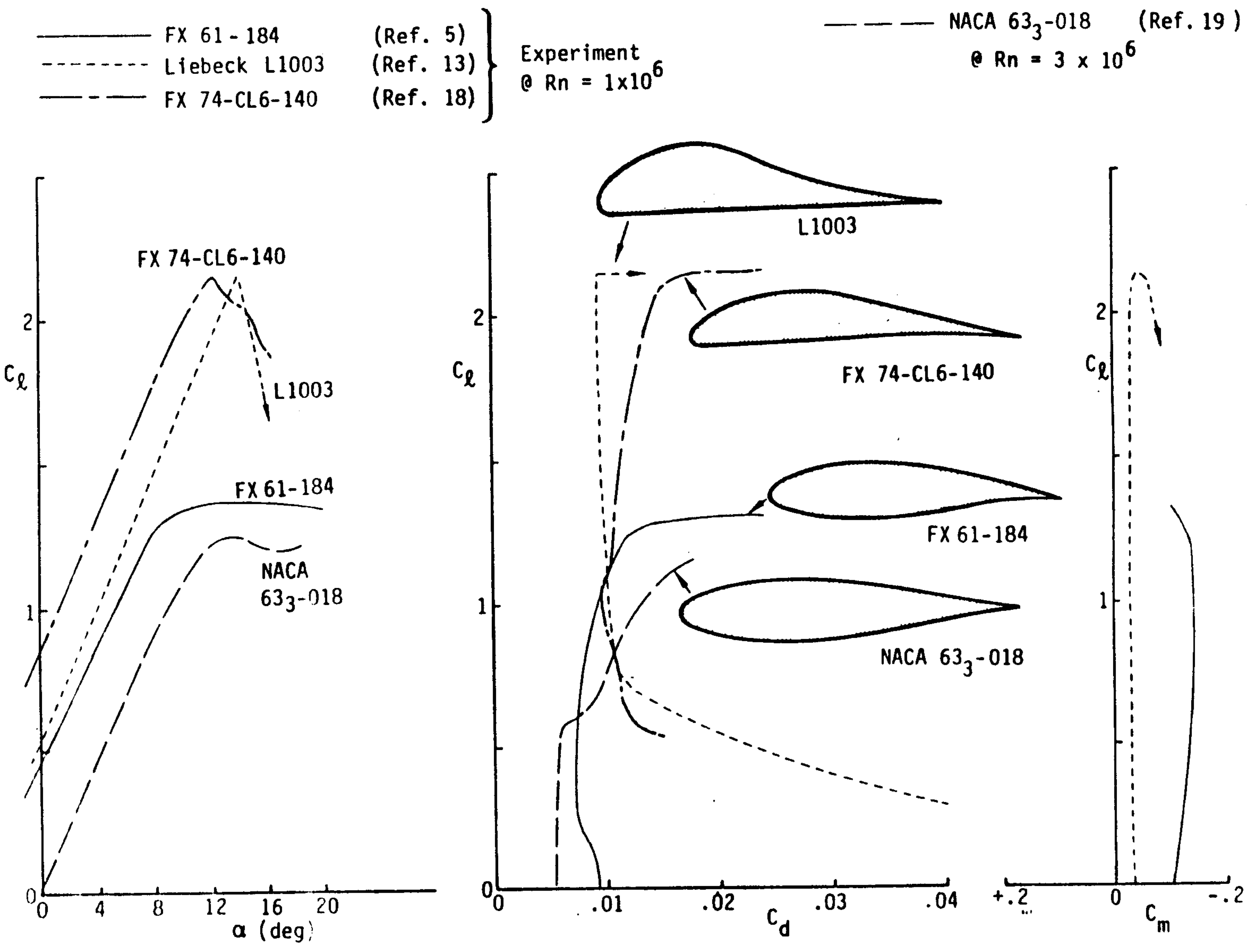

FIGURE 5 - COMPARISON OF PERFORMANCE CHARACTERISTICS OF SEVERAL AIRFOILS

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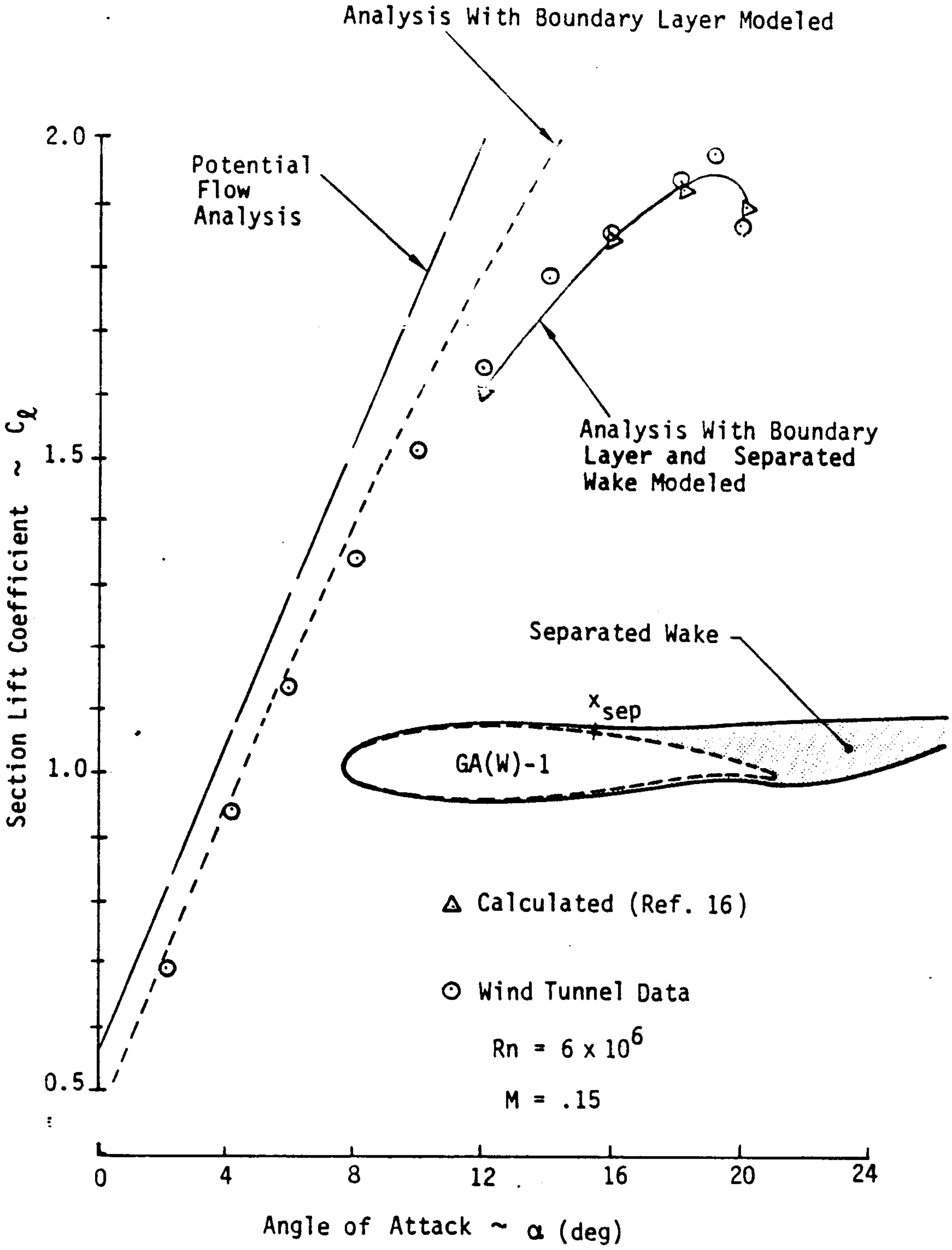


FIGURE 6. TEST THEORY COMPARISONS FOR NASA GA(W)-1 AIRFOIL

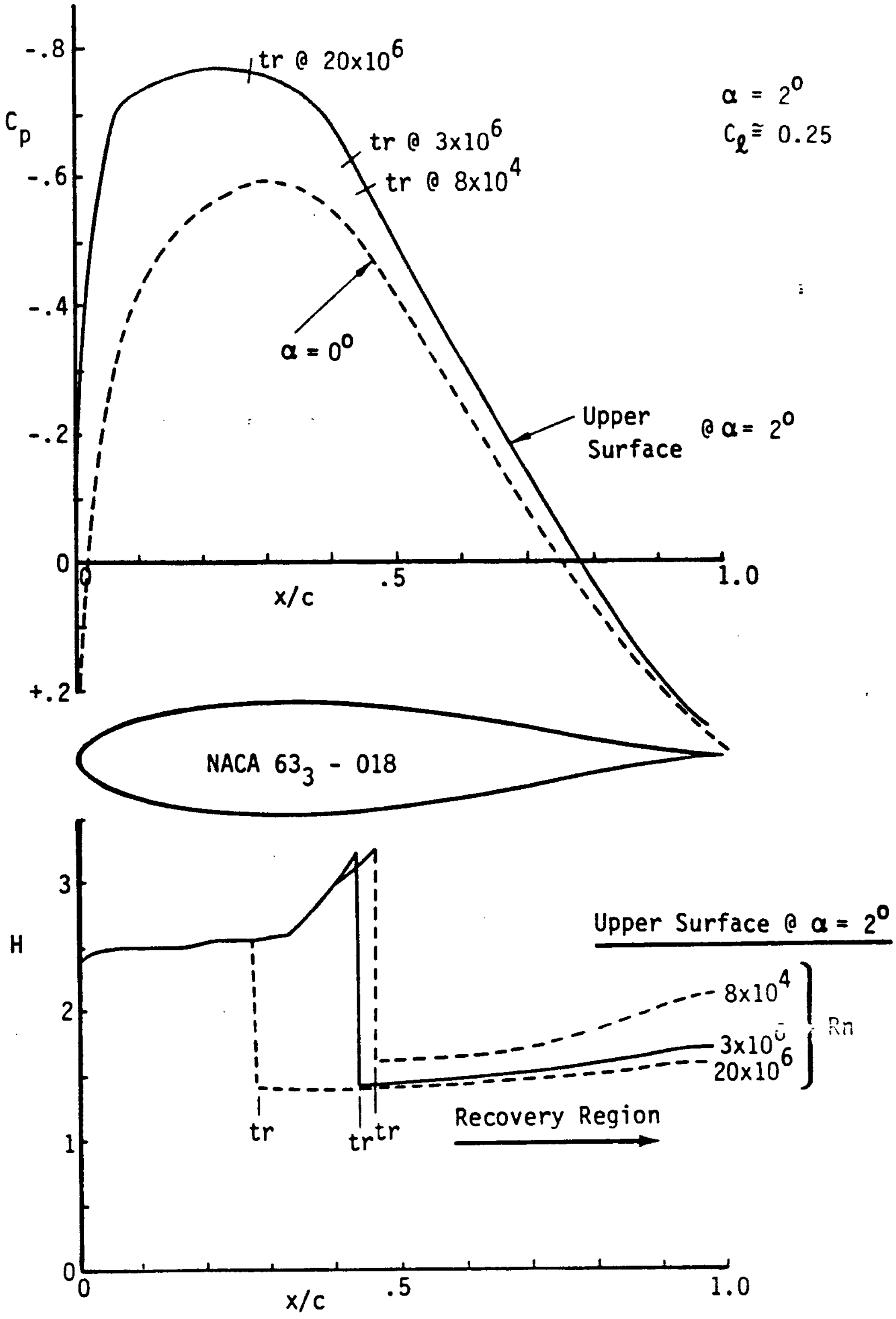


FIGURE 7 - PRESSURE DISTRIBUTION AND BOUNDARY LAYER CHARACTERISTICS OF THE NACA 63_3 - 018 AT SEVERAL REYNOLDS NUMBERS

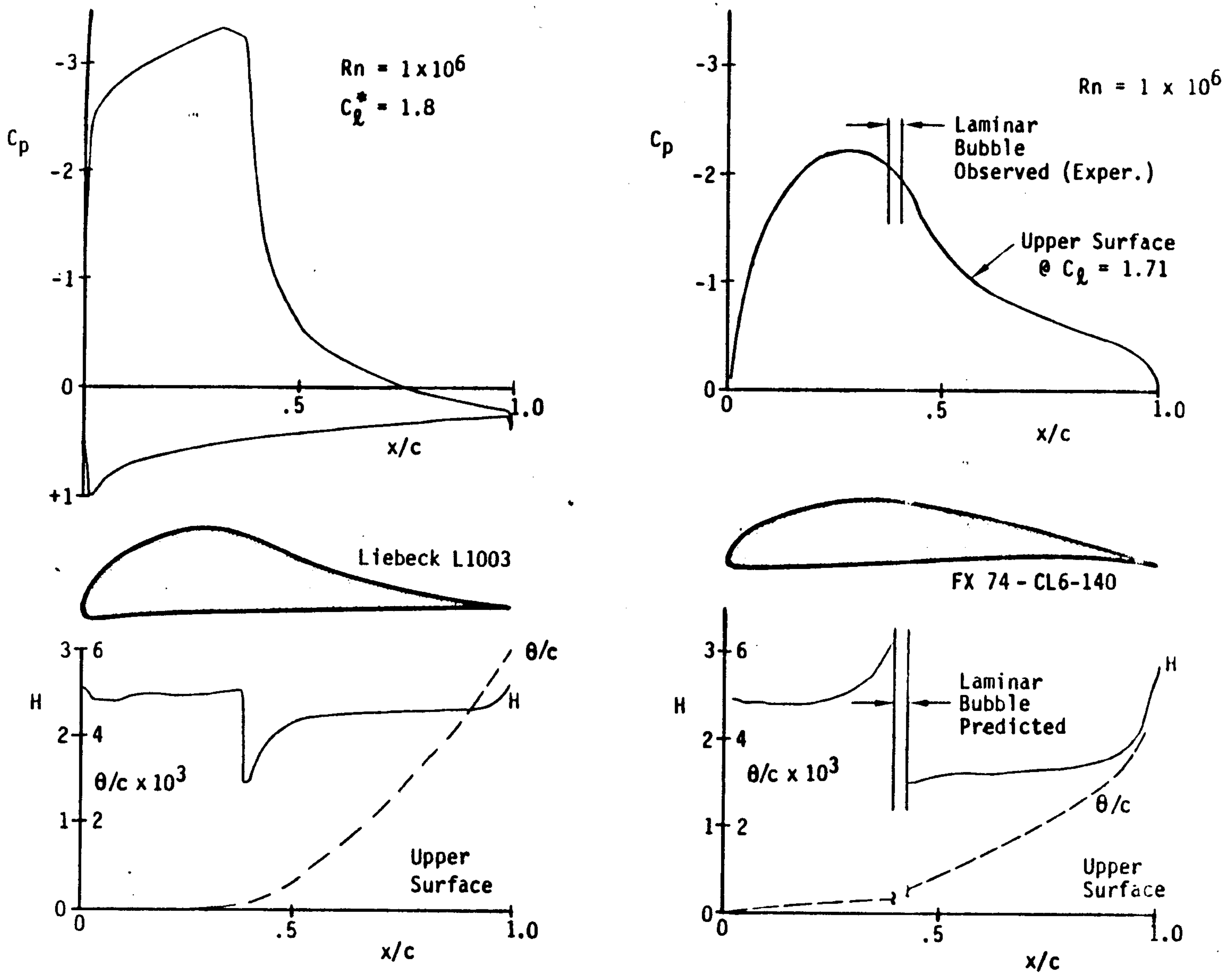


FIGURE 8 - COMPARISON OF LIEBECK AND WORTMANN HIGH-LIFT AIRFOILS

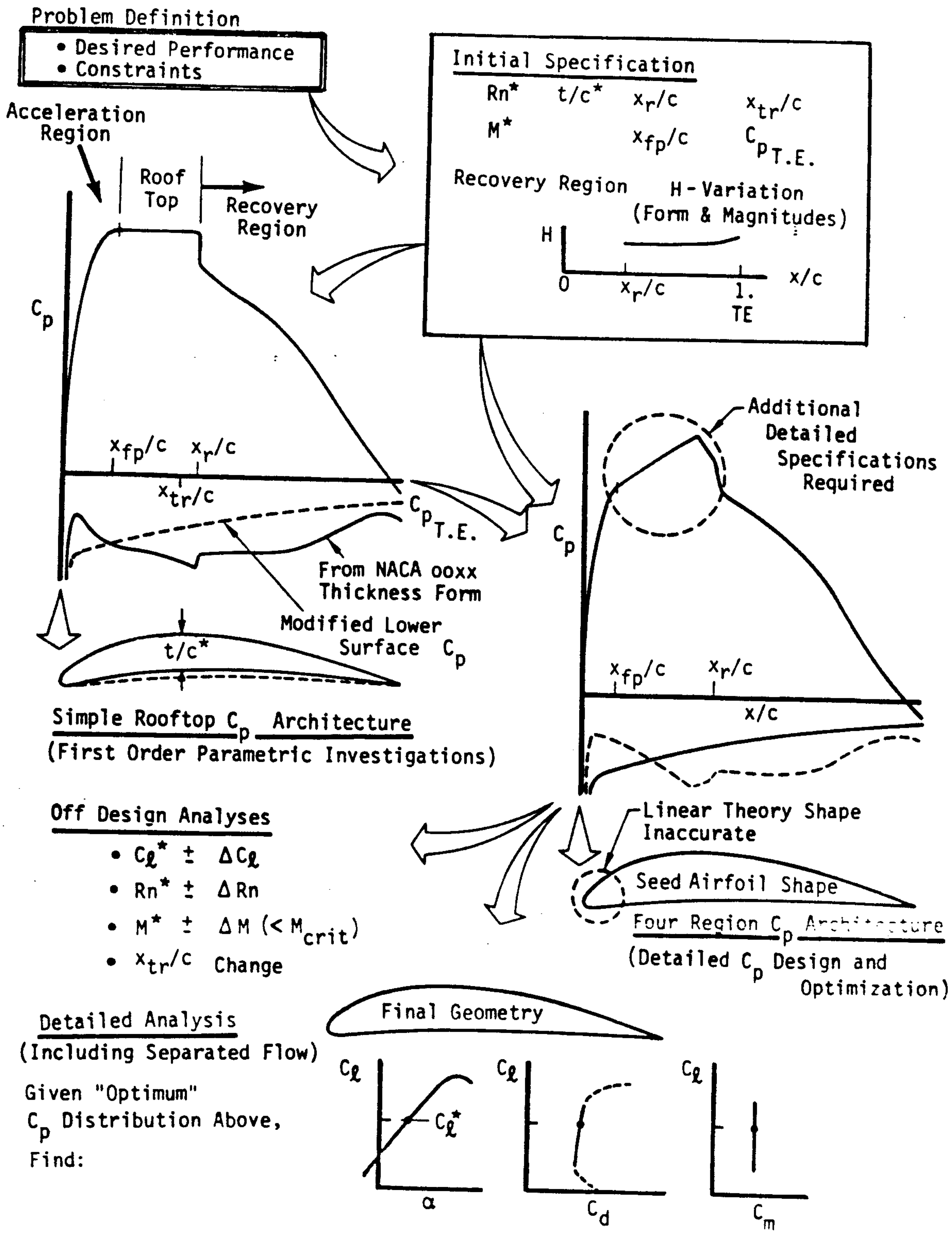


FIGURE 9 GENERAL AIRFOIL DESIGN PROCEDURE

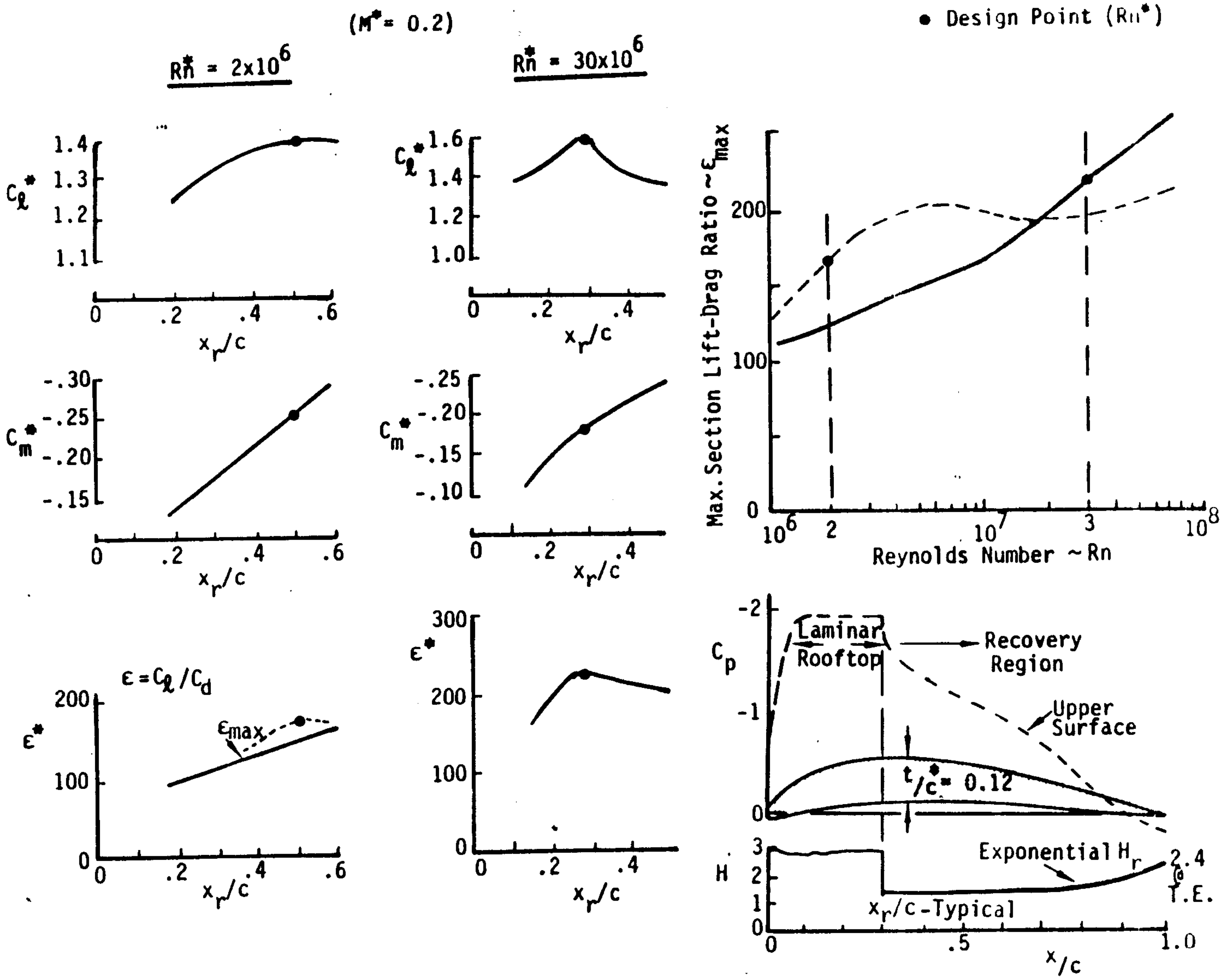


FIGURE 10 INFLUENCE OF VARIATIONS IN RECOVERY POINT LOCATION AND REYNOLDS NUMBER ON A FAMILY OF ROOF-TOP AIRFOILS

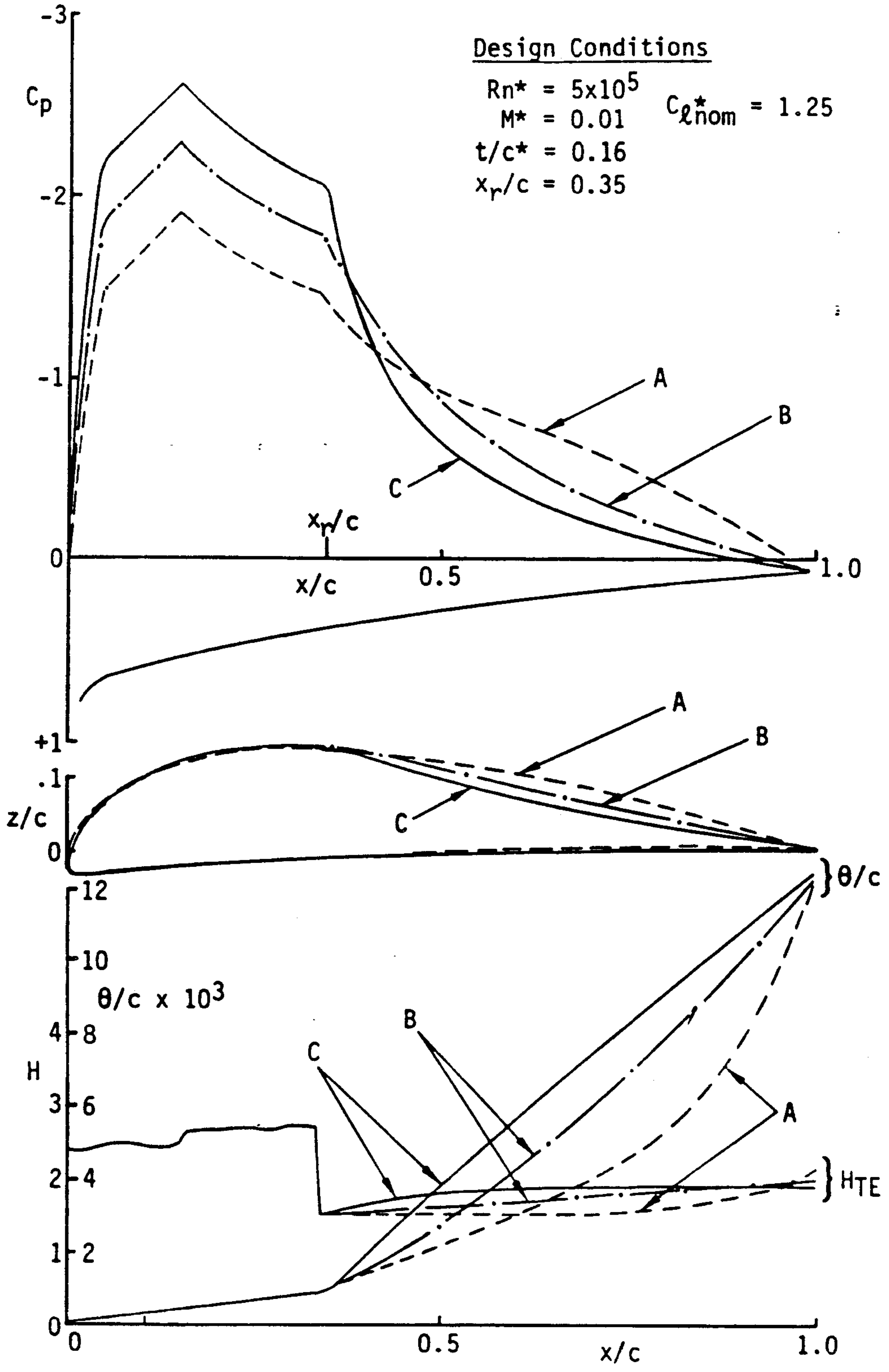
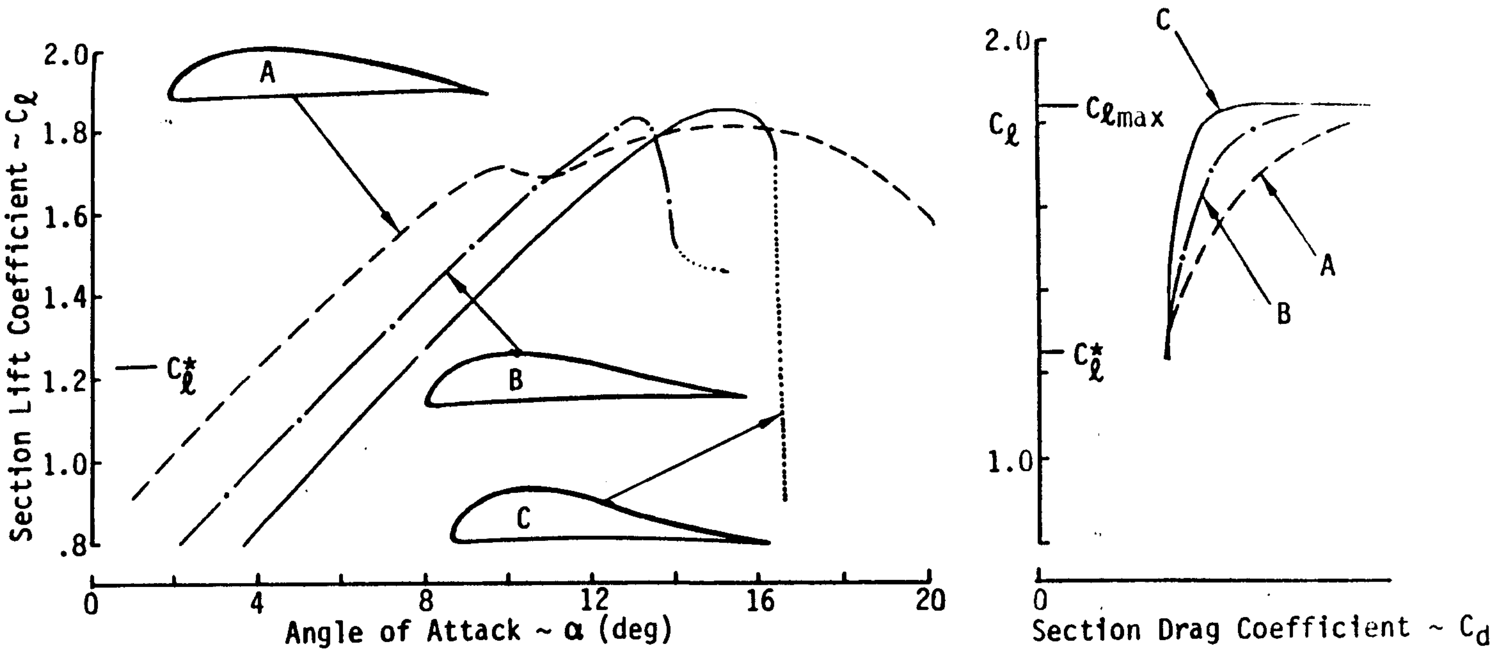


FIGURE 11 THREE AIRFOILS DESIGNED FOR THE SAME LIFT COEFFICIENT AT Rn = .5 × 10^6



Design Conditions

$Rn^* = 5 \times 10^5$
$M^* = 0.01$
$t/c^* = 0.16$
$x_r/c = 0.35$
Nom. $C_\ell^* = 1.25$

Air foil	H_r var.	H_{TE_u}	C_ℓ^*	C_m^*	ε^*	$C_{\ell max}$	$C_{d_u}^*$	$C_{f_{r_u}}^*$
A	Exponent	2.20	1.21	-0.152	105	1.79	0.0112	0.0073
B	Linear	1.99	1.22	-0.061	104	1.82	0.0112	0.0056
C	Constant	1.92	1.24	-.004	103	1.83	0.0114	0.0041
	Recovery Region						Squire & Young	Recovery Region

FIGURE 12 AERODYNAMIC CHARACTERISTICS OF AIRFOILS SHOWN IN FIGURE 11

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Wing Design

7.4. WING DESIGN FOR LOW-SUBSONIC AIRCRAFT

The class of aircraft referred to as "low-subsonic" is interpreted as aircraft with an operational flight envelope which does not allow critical compressibility effects to occur up to the Design Diving Speed or Mach number. These aircraft have maximum level flight Mach numbers of less than about M = .6 and dive Mach numbers of up to about M = .7, depending primarily on the wing thickness ratio. Straight wings can be used and the airfoil thickness/chord ratio may be varied within certain limits in order to obtain a favorable interplay between aerodynamic and structural requirements.

The term "straight wing" may be interpreted as a wing with zero sweepback of a spanwise line interconnecting corresponding points at the tip and root sections, and must not be confused with the term "untapered wing". The aerodynamicist will favor a wing with zero sweepback of the quarter-chord line, as this considerably simplifies the aerodynamic analysis. However, the location of the wing root cannot always be freely chosen in view of considerations pertaining to the general layout or structural design. In order to get the airplane balanced it is necessary to find an optimum location of the center of gravity relative to the aerodynamic center. To this end, the aerodynamic center may be shifted backwards or forwards by applying sweepback or sweep forward, respectively, while the center of gravity is shifted over a smaller distance.

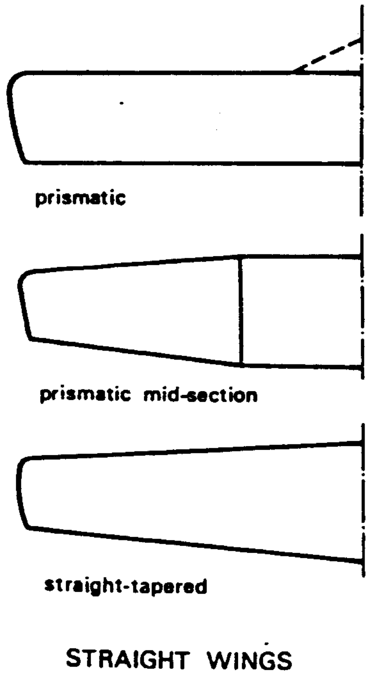


Fig. 7-6. Basic planform shapes for straight wings

7.4.1. Planform

The planform of a wing is defined as the shape of the wing when viewed from directly above, as shown in Fig. 7-6. Planform is directly related to aspect ratio and taper and the main aerodynamic characteristics influenced by planform are the induced drag coefficient and the stalling characteristics.

Great variations in the planform can sometimes be observed in the final design compromise adopted by different airplane designers, even though the design specifications may be almost the same. The choice of the basic wing shape, however, to be made by the individual designer or design team, does not offer quite so much scope as might be expected after observing all the shapes which have been actually adopted on aircraft. Prior to a preliminary design effort, various design offices have gained experience with certain shapes, or may have carried out test programs to investigate aspects of aerodynamic performance, stability and control and structural design. One or more acceptable concepts will emerge from such a program, to be further evaluated during preliminary design.

It is observed that nowadays there are basically three forms of straight wings: the tapered wing, the untapered (rectangular) wing and the wing with a prismatic inner portion and a tapered outboard portion (Fig. 7-6). Tapered wings have been adopted for the majority of aircraft since they offer an efficient solution on account of their low induced drag, high maximum lift, low structural weight and good stowing provisions for the undercarriage. Acceptable stalling characteristics can be obtained, provided the wing is not too sharply tapered.

The untapered wing is attractive from the point of view of manufacture, since only one airfoil contour is involved; this simplifies jigging as there are no compound curvatures. It is aerodynamically inferior to the tapered wing, but may nevertheless be the logical choice for inexpensive private aircraft, where the utilization factor is low and initial cost and cheapness of components are important. Untapered wings are well suited for the application of efficient full-span flaps, where the structural complication is outweighed by the relative simplicity of constant-chord flap segments. Untapered cantilever wings are generally of relatively low aspect ratio to save weight, but braced wings may have a high aspect ratio in spite of the absence of any taper (e.g. Short Skyvan). Wings with a prismatic inboard section have good aerodynamic characteristics and offer some advantages for the structural design and manufacturing of the mid-section, particularly in the case of twin-engine aircraft with wing-mounted nacelles.

7.4.2. Aspect ratio

The aspect ratio denotes the ratio of the wing span to the mean geometric chord. For a given wing area, it provides a direct measure for the wing span:

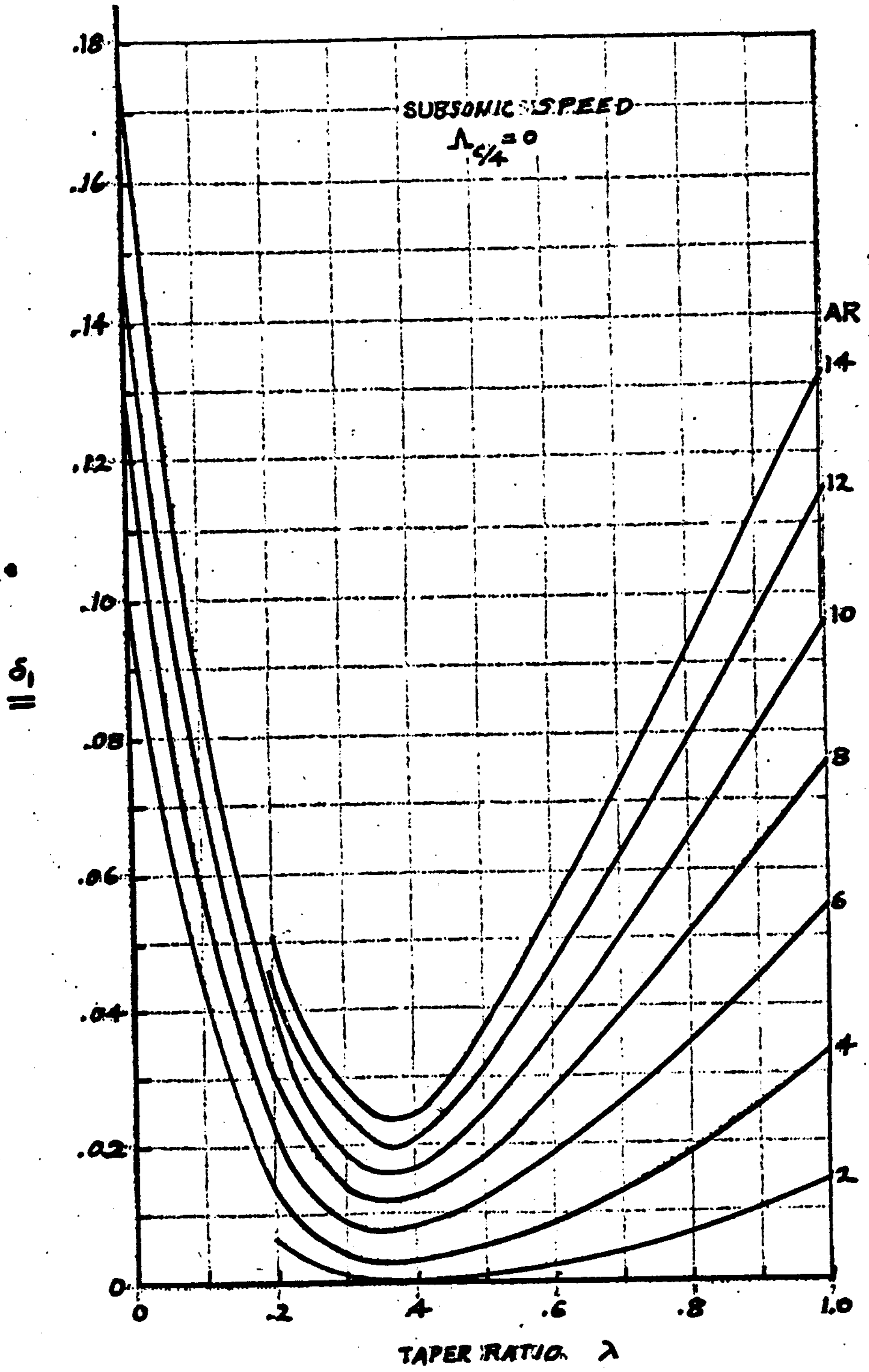
$$b = \sqrt{AS} \tag{7-24}$$

Instead of the aspect ratio, which determines the vortex-induced drag coefficient, use is sometimes made of the span loading, which is related to A and the wing loading:

$$\frac{W}{b^2} = \frac{W/S}{A} \tag{7-25}$$

It can readily be shown that the span loading is a direct measure for the vortex-induced drag as a fraction of the weight, if the dynamic pressure is fixed. The span loading will therefore be a good criterion in design studies where the restricted field length imposes a limit on the stalling speed and the aim is to lift as high a takeoff weight as possible under the adverse condition of engine failure. This criterion will be used to make an initial choice of A and the span for jet-propelled transports.

The climb requirements for propeller transports work out slightly differently. It can be shown that the parameter



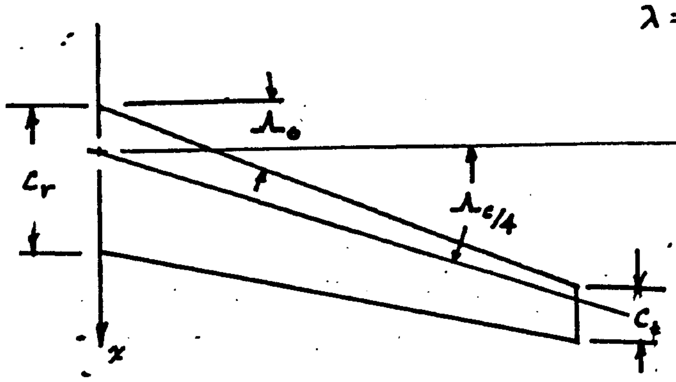
$$C_{D_i} = \frac{C_L^2}{\pi AR}(1 + \delta_1 \delta_2)$$

For planar wings without twist.

$$\therefore k_w = 1 + \delta_1 \delta_2$$

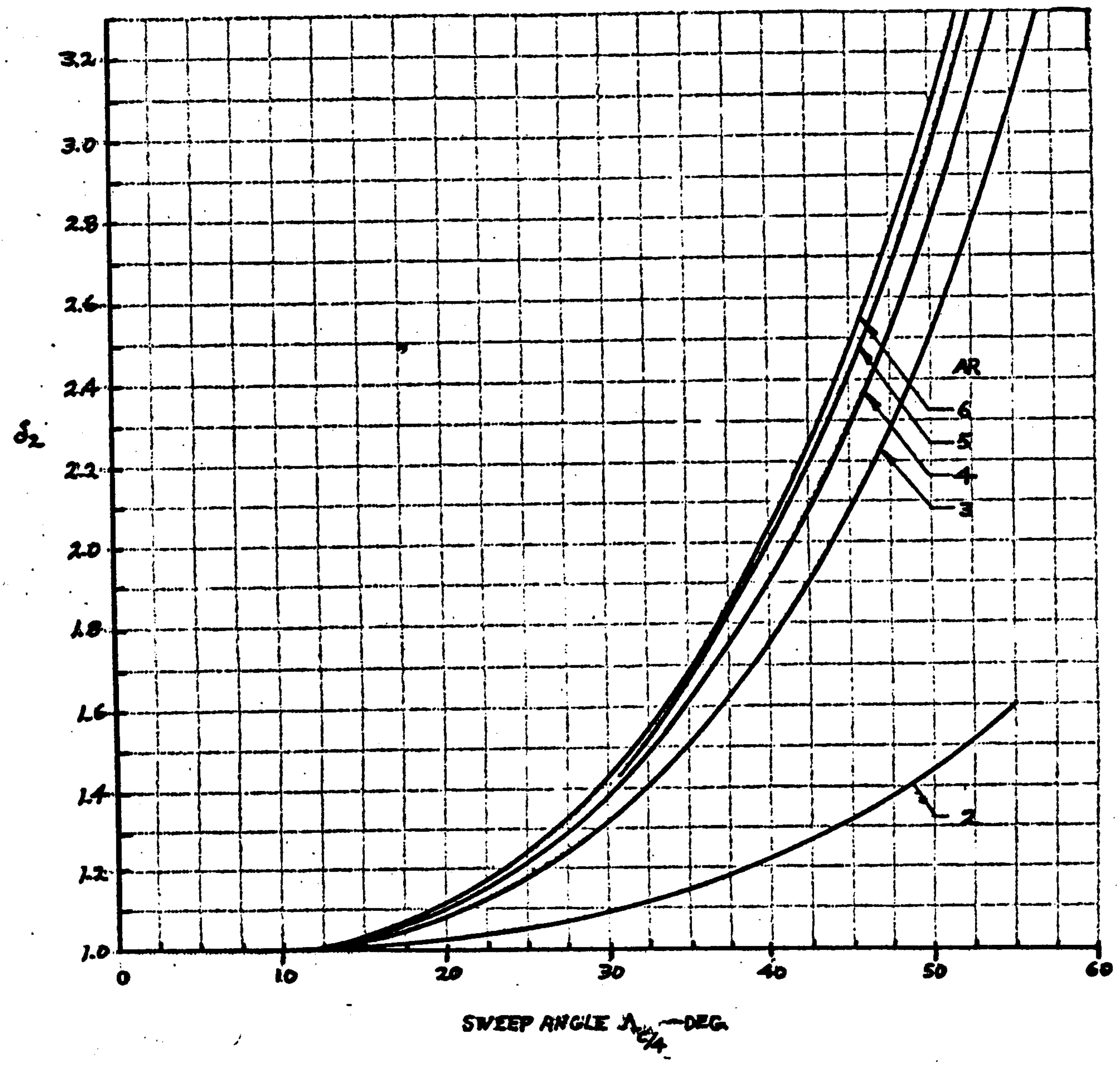
$\delta_2 = 1.0$ when $\Lambda_{c/4} = 0$

λ = taper ratio = c_t/c_r



$$\tan\Lambda_o = \tan\Lambda_{c/4} + \frac{1}{AR}\left[\frac{1-\lambda}{1+\lambda}\right]$$

(Aspect Ratio = b^2/S



Sweep Angle Correction Factor

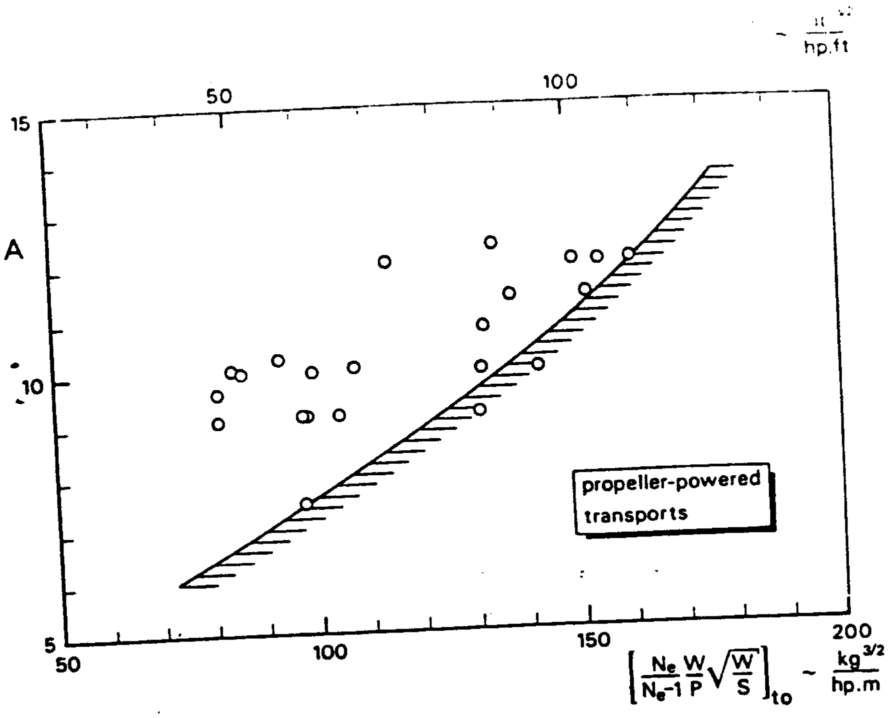


Fig. 7-7. Recommended lower limit for the aspect ratio of propeller transports.

$$\frac{N_e - 1}{N_e}\frac{P_{to}}{W_{to}} \Big/ \sqrt{\frac{W_{to}}{C_{L_{max}}\, S}}$$ N_e = # of engines

is of primary importance here. The reciprocal value of this factor is shown in Fig. 7-7 - where variations in C_L-max are ignored - which may be used as an indication for the minimum acceptable value of A.

It is recognized that for wing loadings satisfying (7-7) the maximum L/D ratio and the range for given cruising speed are quite sensitive to A. Good range performance may thus be obtained for a highly loaded, high aspect ratio wing, but an efficient flap system will be required to ensure acceptable stalling speeds in this case. The concept of a high aspect ratio wing is therefore a logical one for transport aircraft where the emphasis lies on high cruising efficiency; however, sophisticated high-lift devices are inherent to this design concept.

For light aircraft, the wing loading is usually fairly low, the complication of a sophisticated high-lift system generally being considered as undesirable by most manufacturers. Consequently, the optimum cruising speed for long-range flight may be too slow to make it an attractive speed, particularly when the altitude is limited to some 10,000 ft (3,000 m). These aircraft are usually flown at the maximum cruise rating and it can readily be shown that a large increase in A results in a relatively modest gain in speed and range. For example, increasing the aspect ratio of an aircraft with W = 3,300 lb (1,500 kg), S = 160 sq.ft (15 m^2), P = 180 hp and C_{D_o} = .025 from 6 to 10 results in a speed increment at sea level from 128 knots (238 km/h) to 134 knots (248 km/h), a gain of only 4%.

A high aspect ratio may result in a low drag in the landing configuration, which tends to flatten the approach glide, makes judgment of the landing point more difficult to the pilot and gives the aircraft a tendency to "float" after the landing flare. In addition, a high aspect ratio wing does not favor good maneuverability in roll due to its large damping and the reduced effectiveness of the small-chord ailerons (Ref. 7-96).

In conclusion, moderate aspect ratios between 7 and 9 are usually applied for twin-engine general aviation aircraft; for single-engine aircraft these values are usually somewhat lower, e.g. between 5.5

and 8.

7.4.3. Thickness ratio

The desirable high aspect ratio for low-speed transport aircraft can be achieved only if sufficient structure height is available at the wing root, where the bending moment during flight is maximum. In this connection use is often made of the cantilever or overhang ratio, which is defined as the structural wing semi-span, divided by the maximum root thickness. Expressions for the wing weight show that the structural wing weight fraction increases linearly with increasing cantilever ratio, provided all other parameters remain constant.
The cantilever ratio is plotted in Fig. 7-8 for aircraft of various weight categories. Transport aircraft usually have values of between 18 and 22; ratios in excess of 25 are rare, even in the case of supersonic transport aircraft. A rather lower value is found for trainers, probably because of the high maneuver load factors for which their wings must be designed.

The favorable trend of low induced drag of a high aspect ratio wing is partly cancelled out by a profile drag increment if the thickness ratio is allowed to increase in proportion to A. Maximum lift is also affected by the thickness ratio, as shown by Fig. 7-9. The trend for basic wing sections is readily explained by recalling that for thin wings the leading edge type of stall dominates, while for thick sections the trailing edge stall is predominant. The highest maximum lift of conventional, standard NACA airfoil sections is achieved for thickness ratios of 12 to 15 percent chord, where a combined stall will be observed. Recent developments in sections for low-speed aircraft show that higher maximum lift coefficients can be obtained with special sections having a thickness ratio of about 17 percent. The maximum lift coefficient with trailing-edge flaps deflected has a relatively flat maximum for

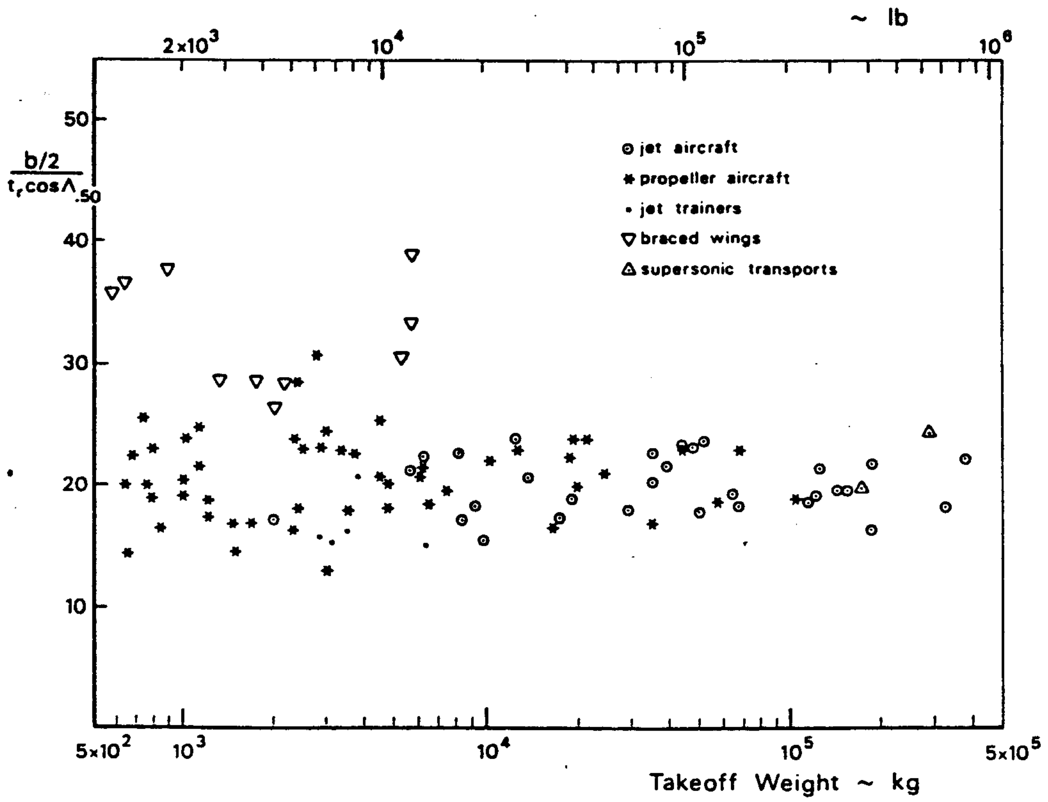


Fig. 7-8. Cantilever ratio vs. aircraft size

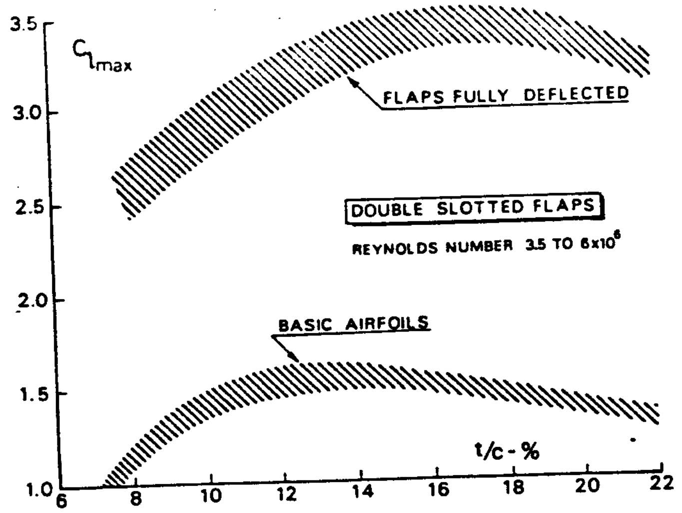


Fig. 7-9. Trends of best maximum lift values of NACA sections vs. thickness ratio.

thickness ratios between 15 and 20 percent.

It is concluded that the root section thickness of transport aircraft should be chosen such that a good cantilever ratio is obtained. For a given taper ratio this implies that the thickness ratio at the root will increase proportionally to the aspect ratio. A thickness ratio of between 15 and 20 percent is in the interest of good performance when using relatively simple trailing-edge high-lift devices and provides adequate room for retracting the undercarriage. Thickness ratios above 20 percent may show diminishing returns due to the increasing profile drag and the relatively low maximum lift and this, in turn, limits the aspect ratio to a maximum of approximately 13 for cantilever wings. For aircraft cruising at Mach numbers above .5 a check should be made to ensure that up to the dive Mach number, which is about .1 above the cruising speed, the flow is essentially subcritical. This condition imposes a limit on the thickness ratio, which is affected to some extent by the airfoil section shape. A method for estimating the critical Mach number is given in Section 7.5.1d. Light aircraft wings have lower aspect ratios and the best root sections are approximately 15 percent thick.

Tip sections (without flaps) should be between 10 and 15 percent in order to attain a high maximum lift. This reduction relative to the root is also in favor of low structural weight. A minimum practicable thickness should be present on light aircraft in order to provide adequate room for control system elements.

7.4.4. Wing taper

The taper ratio λ has a great effect on the spanwise lift distribution. The spanwise position of the center of pressure of a half wing moves in the direction of the wing root as λ decreases and the root bending moment due to lift decreases accordingly. Since the structural height of the wing root also increases - for a given wing area, span and section shape - a highly tapered wing can be built lighter and with much more torsional rigidity than a rectangular wing. In the case of small aircraft, a practical lower limit to λ is imposed by the structural height required at the tip to provide room for the ailerons and their control elements.

The taper ratio is a dominant parameter in controlling the spanwise stall progression, as it has a large effect on the spanwise lift coefficient distribution (Fig. 7-10).

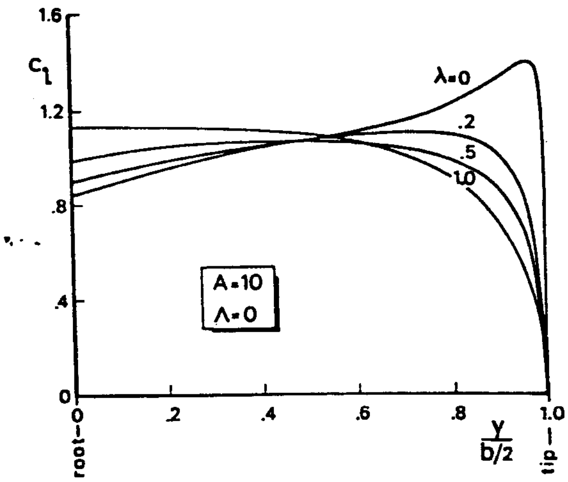


Fig. 7-10. Lift distribution at $C_L = 1$ for straight wings with various taper ratios

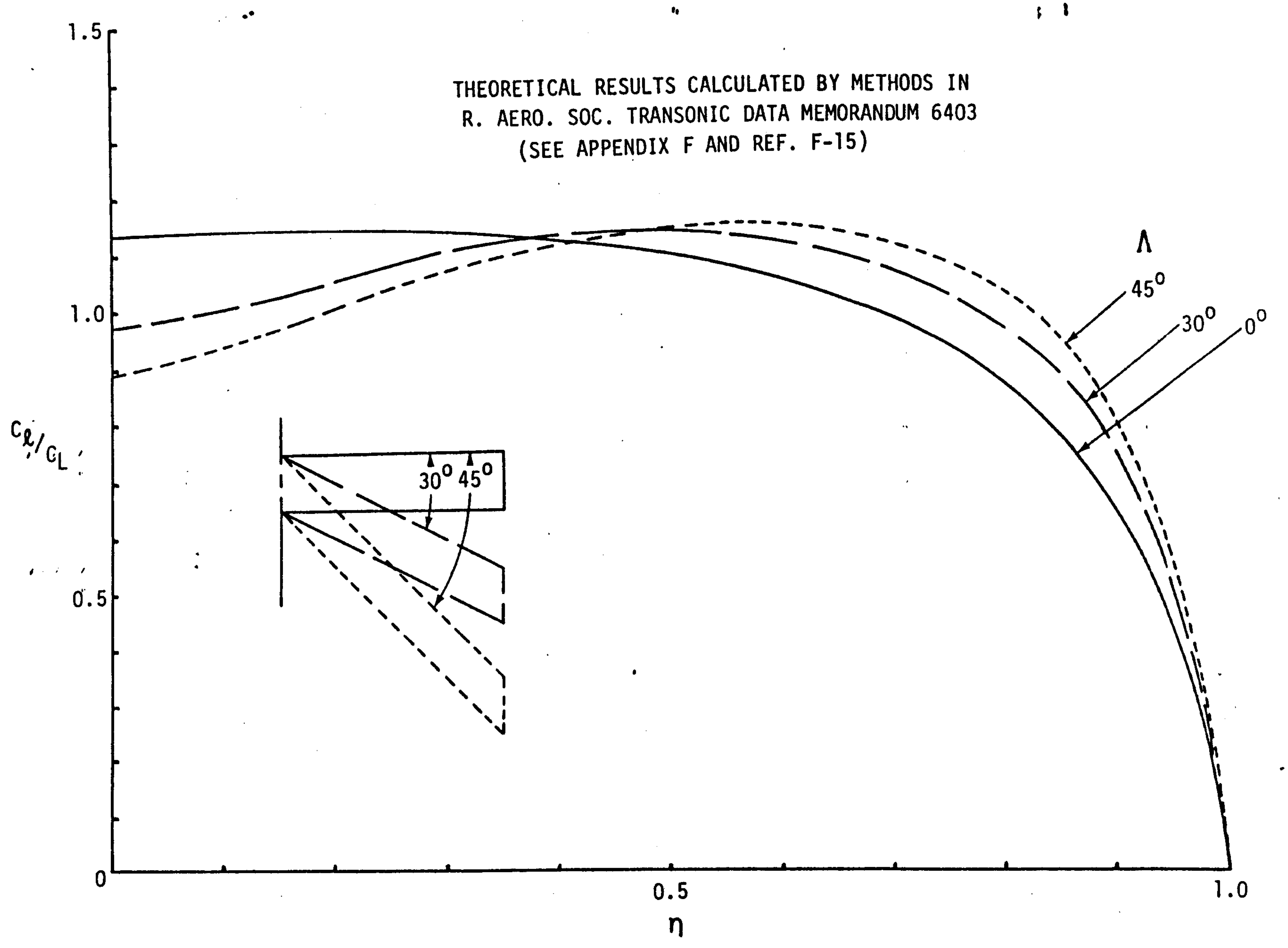


FIGURE [redacted] VARIATION IN SPAN LOADING WITH SWEEP FOR A RECTANGULAR WING OF FIXED ASPECT RATIO.

Compare with Fig. 7-10

A first-order approximation of the location where the c_ℓ-distribution is at its peak is:

$$\eta = 1 - \lambda \qquad (7\text{-}26)$$

for untwisted, straight-tapered wings. Hence, a wing with constant section properties and a taper ratio of .4 will tend to stall first at 60 percent semi-span, fairly close to the inner part of the aileron. If a wing is tapered sharply, there will also be a notable reduction in the maximum lift coefficient near the tip due to the locally reduced Reynolds number, thus aggravating the tendency towards early tip stall. Although precautions can be taken to shift the initial separation point inboard by means of section shape variation and twist, the amount of taper has a definite limit. Realizing that the vortex-induced drag of a tapered wing is minimum for a taper ratio of about .4 and is insensitive to relatively large deviations from this value, it is concluded that for straight wings taper ratios appreciably below .4 are of little use (Fig. 7-11).

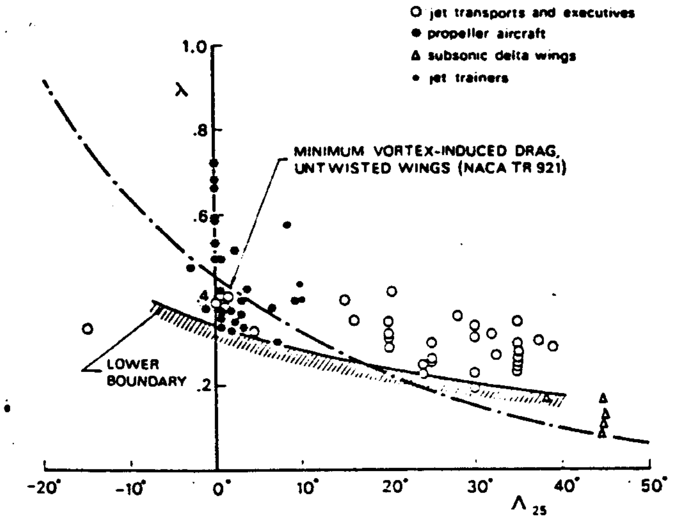


Fig. 7-11. Taper ratics of straight, swept and delta wings.

Once the taper is chosen, the wing geometry is known, provided the area and aspect ratio have also been selected. For straight-tapered wings the tip and root chords are given by:

$$c_r = \frac{2}{1 + \lambda} \frac{S}{b} \qquad (7\text{-}27)$$

$$c_t = \lambda \, c_r \qquad (7\text{-}28)$$

and the plan view of the wing may be provisionally drawn.

It will be noted that wings with taper both in planform and in the sectional thickness ratio, airfoil sections between fairing stations will be slighly distorted when linear lofting is used for structural simplicity. A tapered wing with respectively 18 percent and 12 percent thickness ratios at the root and tip does not have a 15 percent thickness ratio at the section midway between root and tip, but at a station closer to the tip.

7.4.5. Airfoil selection

In selecting the airfoil sections the designer must give consideration to several general requirements.

1. The basic airfoil must have a low profile drag coefficient for the range of lift coefficients used in cruising flight.
2. For the inboard sections with flaps extended, the drag must be low in high lift conditions, particularly during the take-off climb.
3. The tip section should have a fairly high maximum lift coefficient and gradual stalling characteristics.
4. The inboard wing section should have high maximum lift with flaps extended.
5. The critical Mach number should be sufficiently high to ensure that critical compressibility effects are avoided in the case of aircraft reaching dive Mach numbers of approximately .65.
6. The pitching moment coefficient should be of low to moderate magnitude to prevent a high trim drag and torsional moments at maximum dynamic pressure.
7. The aerodynamic characteristics should not be extremely sensitive to manufacturing variations in the wing shape, contaminations and dirt, etc.
8. The wing sections should have the largest possible thickness ratio in the interest of low structural weight. Sufficient internal space must be provided for

fuel, main gear, mechanical controls and possibly other components.
All these requirements cannot be satisfied by one single airfoil. Spanwise variation of the sectional shape and some measure of compromise will therefore generally be accepted.

For low-subsonic aircraft the selection is usually made from NACA standard sections, to which modifications may be made if necessary, usually during the stage of detailed aerodynamic design. The effect of systematic variations in the profile shape has been the subject of thorough investigations by the NACA, resulting in many series of satisfactory airfoil sections. The relevant findings have been presented in the form of very complete information. Refs. 7-5 and 7-51 in particular are very useful tools for the designer, and the use of this data has greatly simplified the choice of a suitable airfoil for conventional aircraft. The observations made in Ref. 7-5 (Chapter 7) are fairly complete as far as wing sections are concerned, while Ref. 7-96 gives a systematic treatment of the effect of airfoil variation on stalling characteristics, using the successful method or R.F. Andersen (Ref. 7-70). A survey will follow of the most commonly used NACA sections , examples of which are shown in Fig. 7-12.

a. The NACA four-digit wing sections basically constitute a synthesis of early Göttingen and Clark Y sections, empirically developed on a basis of pre-war experimental data. Both the thickness distributions and the mean lines are defined in the form of polynomials; the sections have a near-elliptic shape. The maximum camber is at approximately the mid-chord position.
Although the sections in the 4-digit series are by no means low-drag profiles, the drag increase with lift is fairly gradual. The cambered sections have relatively high maximum lift and the stalling is fairly docile. These properties have marked the 2412 and 4412 sections, for example, as being suitable tip sections for the wings of light aircraft and tailplanes. In view of the gradual changes in drag and pitching moment with lift, the 4-digit sections are frequently used for light trainers, which often fly in different conditions.
Recent experiments with 16 percent thick sections have shown remarkable c_ℓ-max figures of up to about 2.1 for a basic 6716 section and 4.0 for a 4416 section with single slotted flaps, deflected 30° (Ref. 7-64).

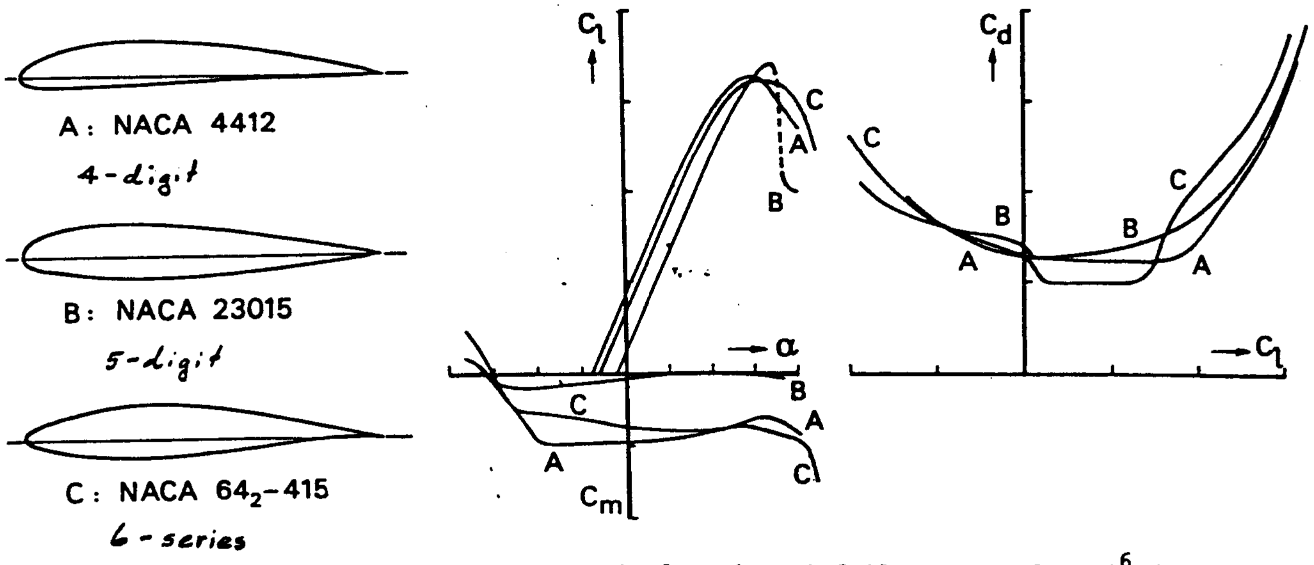


Fig. 7-12. Characteristics of NACA standard series airfoils at $R = 6 \times 10^6$ (Ref. 7-5).

b. The NACA 5-digit wing sections have the same thickness distribution as the 4-digit series, but the mean lines are different, having their maximum ordinate further forward. The well-known 230-series airfoils have the maximum camber at the .15 chord point. These sections have the highest maximum lift of the standard NACA sections, but the stalling behavior is not particularly favorable and rather sensitive to scale effects. For wings where high lift performance is a prerequisite, the 230-series sections have been frequently used, sometimes combined with a 4-digit section at the tip.

c. The NACA 6-series ("laminar flow") of wing sections is the outcome of a succession of attempts to design airfoils by (approximate) theoretical methods, aimed at achieving low profile drag in a limited range of lift coefficients: the "sag" or "bucket" in the low-drag range. The laminar boundary layer over the forward part of the section is stabilized by avoiding pressure peaks, keeping the local velocities low and applying a favorable pressure gradient over the forward part of the upper surface. The extent of laminar layer is limited by the separation of the turbulent boundary layer over the rear part. The low supervelocities on these airfoils also favor the attainment of a high critical Mach number.

Due to the relatively sharp nose of thin laminar flow sections, their maximum lift is notably below that of the 4- and 5-digit series, although the difference for the thicker cambered sections is negligible; these sections also exhibit a docile stall. The profile drag, although very low under ideal conditions, is sensitive to surface roughness, excrescences and contaminations. Special structures are therefore needed to maintain the laminar flow, and on practical wing constructions of transport aircraft the potentially large extent of laminar layer will not normally be realized.

A great advantage of the 6-series is that sectional properties have been tested extensively and reported systematically. The designer is thus provided with a tool for establishing the best sectional shape by systematically varying the shape parameters. A modification to the standard series is the A-series* (Ref. 7-52) in which the sharp trailing edge angle is replaced by a larger one, resulting from straight contours which run from 80 percent chord backwards. *eg 63A212

Having decided on the series of sections to be used, the designer will have to choose the various shape parameters.

THICKNESS/CHORD RATIO

LOCATION OF MAXIMUM THICKNESS: the further back this point is chosen, the lower the minimum profile drag (of smooth profiles) and the higher the critical Mach number at the design c_ℓ. This works out at the expense of c_ℓ-max and profile drag at high lift. For these reasons, the 63- and 64-sections are the most popular amongst the 6-series airfoils.

(MAXIMUM) CAMBER: determines the angle of attack for zero lift, the pitching moment coefficient, the lift coefficient for minimum profile drag and c_ℓ-max. A large camber is in the interest of high c_ℓ-max, but the tail load required to trim the aerodynamic pitching moment may cause too much extra drag. The camber is usually chosen so that in normal cruising flight the section operates at its design c_ℓ. Little camber is used on trainers, where a requirement exists for acceptable characteristics in inverted flight.

SHAPE OF THE MEAN LINE: a forward location of the point of maximum camber results in a high c_ℓ-max with a leading-edge type of stall at normal thickness ratios. A lower c_ℓ-max and a more gradual stall are obtained when the maximum camber is further back.

7.4.6. Stalling characteristics and wing twist

The following observations, originally

made in Ref. 7-71, are considered as a good starting point for achieving the desirable stalling characteristics of straight wings, specified in Section 7.3.:

1. The point of initial stalling should be sufficiently far inboard, the best location being at about 40 percent semi-span from the root.
2. Stall progression should be more rapid to the inboard than to the outboard sections.
3. The margin in c_ℓ at 70 percent of the semi-span from the root - approximately at the inboard end of the aileron - should be at least .1 in the condition where separation occurs first.

Considering the practical measures available to produce the desirable characteristics, it is possible on straight wings of moderate to high aspect ratios to choose a suitable spanwise variation of the local and maximum lift coefficient. An example is shown in Fig. 7-13. The designer is also

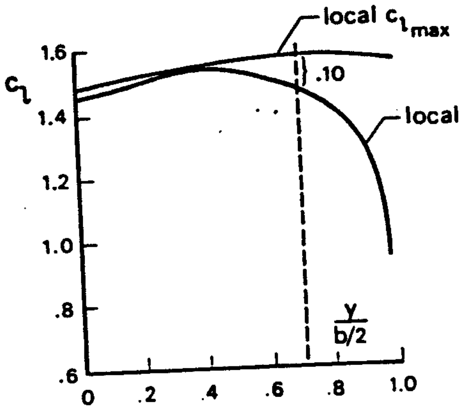


Fig. 7-13. Calculated spanwise lift distribution at high incidence (flaps ups)

concerned with the abruptness of the stall progression at slightly higher angles of attack and he may try to gain an impression of the chordwise stall progression by using Fig. 7-5 and the measured sectional characteristics.

The spanwise lift distribution is influenced primarily by the wing planform and wing twist (washout). The type of airfoil section has little effect in the linear range of incidences in view of the generally small spanwise variations in the lift-curve slope. The curve of local c_ℓ-max is determined exclusively by the local sectional shape. The effect of variations in the taper ratio, aspect ratio and washout on stalling characteristics, C_L-max and induced drag may therefore be investigated without making any decision on the section to be used, which saves a lot of work.

The large number of wing shapes to be studied may be reduced by making several restrictions, for example:

1. Although the aspect ratio is important from aircraft performance considerations, its effect on the stalling characteristics is generally small.
2. An aerodynamic washout of more than about five degrees results in unacceptably large induced drag increments of the order of 5 to 10 counts.
3. Low taper ratios may be used only on wings with thick root sections of the 4- and 5-digit series and tip thickness ratios of about 12 percent.
4. The effect of wing taper on C_L-max may be much larger for 5-digit series airfoils than for 4- and 6-series, particularly if the Reynolds number at the tip is below 2 million.

There is no chain of logic to show how the designer will arrive at a (provisional) solution. The basic wing shape will finally be developed after many hours of wind tunnel work. Modifications to the basic standard section may appear desirable and during flight tests stall control devices may prove unavoidable, even though theoretical predictions and wind tunnel tests have not shown any deficiencies.

It should also be noted that most theoretical methods do not take account of any wing/fuselage interference effects. For high-wing aircraft, where the flow interaction is confined mainly to the less critical lower surface, these effects are small. The low-wing position introduces the largest interference effects, but these can usually be controlled by local fuselage contour modifications and/or adequate root fairings. Established theoretical methods for predicting the characteristics of faired wing/fuselage combinations are not available and the final geometry of these fairings is determined by wind tunnel or flight experiments. Propeller slipstream effects near maximum lift generally result in unstalling the wing in the areas directly influenced by the slipstream. In flight tests aimed at establishing stalling speeds these effects are small as a power-off condition is then required. Power-on stalls may, however, cause a completely different type of behavior, depending on the configuration.

7.5. WING DESIGN FOR HIGH-SUBSONIC AIRCRAFT

In the aerodynamic design of high-performance aircraft, emphasis is placed on speed as a major factor contributing to the economy and operational suitability of the conceptual design. If we assume for the moment that the type of powerplant to be installed and its rating are fixed, the main factor in determining the speed is the drag area in the case of relatively slow transports and light aircraft, as shown in Fig. 7-14. For a given flying al-

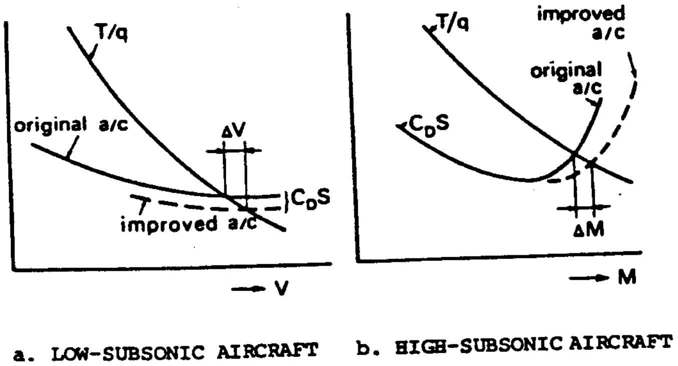


Fig. 7-14. The basic high-speed design problems of improving low-subsonic and high-subsonic aircraft.

titude any improvement in the speed ΔV may be obtained by a reduction of the drag area C_DS - for example by minimizing the wetted area, streamlining, reducing interference, optimizing the wing loading and aspect ratio, etc.

For high-subsonic aircraft these aims are still present but, contrary to the situation with low-speed aircraft, the region of compressible flow is penetrated intentionally in order to attain as high a cruising speed as possible. The problems of aerodynamic design in these aircraft mainly relate to attaining a high critical Mach number, avoiding undesirable flight characteristics at off-design conditions (maneuvering, gustiness, speed overshoots) and providing good low-speed characteristics of the sweptback wing.

The basic opportunities for attaining a high cruise Mach number are the adoption of sweepback (or sweep forward), reduction of the thickness/chord ratio, design of improved airfoil sections and optimum distribution of spanwise camber and twist.

The use of a moderate wing loading and aspect ratio is of some help, but may conflict with other objectives of performance optimization. The application of wing/fuselage blending, fairings and anti-shock bodies may be considered, provided they do not conflict with the structural and layout design.

The aerodynamic design problems are by no means different in the case of a new aircraft which is not required to fly faster than the aircraft to be replaced. The advanced technology may then be used to increase the thickness and/or aspect ratio, or to reduce the angle of sweepback.

This section will be confined to the contribution of the wing only; compressibility effects in other areas, such as the nacelle/airframe junction, may have an appreciable effect.

7.5.1. Wing sections at high-subsonic speeds

a. Subcritical speeds.

In the subcritical region the flow around an airfoil in two-dimensional flow is subsonic throughout. The effects of compressibility on the pressure distribution are well described by potential flow methods, known as the Prandtl-Glauert correction,

$$C_p = \frac{C_{p_i}}{\sqrt{1 - M_\infty^2}} \qquad (7\text{-}29)$$

or more accurate approximations, e.g. the von Karman-Tsien relation. In (7-29) C_p is the pressure coefficient,

$$C_p = \frac{p - p_\infty}{q_\infty} \qquad (7\text{-}30)$$

and the subscript i refers to the incompressible situation ($M_\infty << 1$).
Fig. 7-15 shows both the effect of sub-

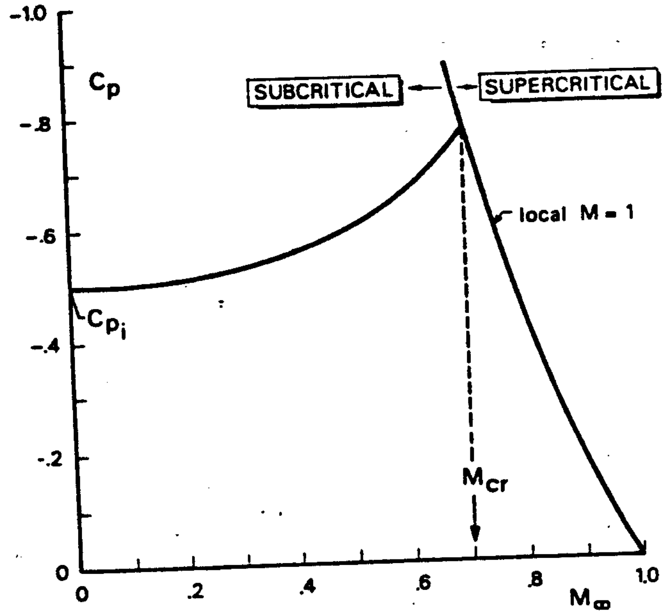


Fig. 7-15. Variation of the pressure coefficient with Mach number and determination of the critical Mach number

critical compressibility on a local pressure coefficient and the boundary of the subcritical region, which is defined by

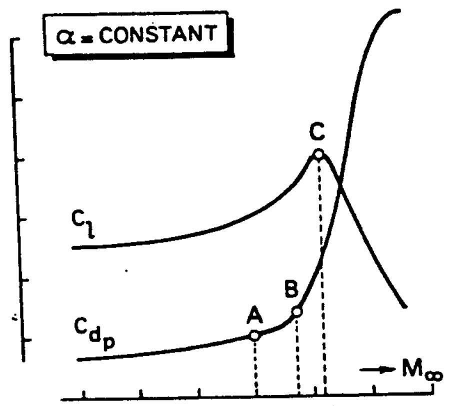
$$C_{p_{cr}} = \frac{2}{\gamma M_\infty^2}\left[\left\{\frac{2 + (\gamma-1)\, M_\infty^2}{\gamma + 1}\right\}^{\gamma/(\gamma-1)} - 1\right] \qquad (7\text{-}31)$$

This equation can be derived by using the Bernoulli equation for compressible flow and substituting M = 1 for the local velocity.
The critical Mach number of an airfoil section is defined as the free stream Mach number for which sonic flow is reached at the point of minimum pressure. Provided the value of C_p in Fig. 7-15 refers to this pressure, the critical Mach number is defined by the intersection of both curves, assuming a constant angle of attack. The critical Mach number is thus easily determined from the low-speed pressure distribution, and for NACA standard airfoils this method is used to compute M_{cr} (Ref. 7-51).
Fig. 7-16 shows that the effects of subcritical compressibility on the profile drag and pitching moment coefficients at constant angle of attack are small, while the lift coefficient - and hence the lift-curve slope - are affected in a similar fashion to C_p.

b. Supercritical speeds.
In the case of positive α, regions of supersonic flow will appear on the upper surface when M_{cr} is exceeded. As soon as this region terminates in a shock wave of appreciable strength, which thickens the boundary layer, the drag increases noticeably. The drag-divergence (or drag-critical) Mach number is defined as M_∞ for which:

$$\Delta c_{d_p} = .002 \qquad (7\text{-}32)$$

or alternatively, according to the NACA nomenclature:



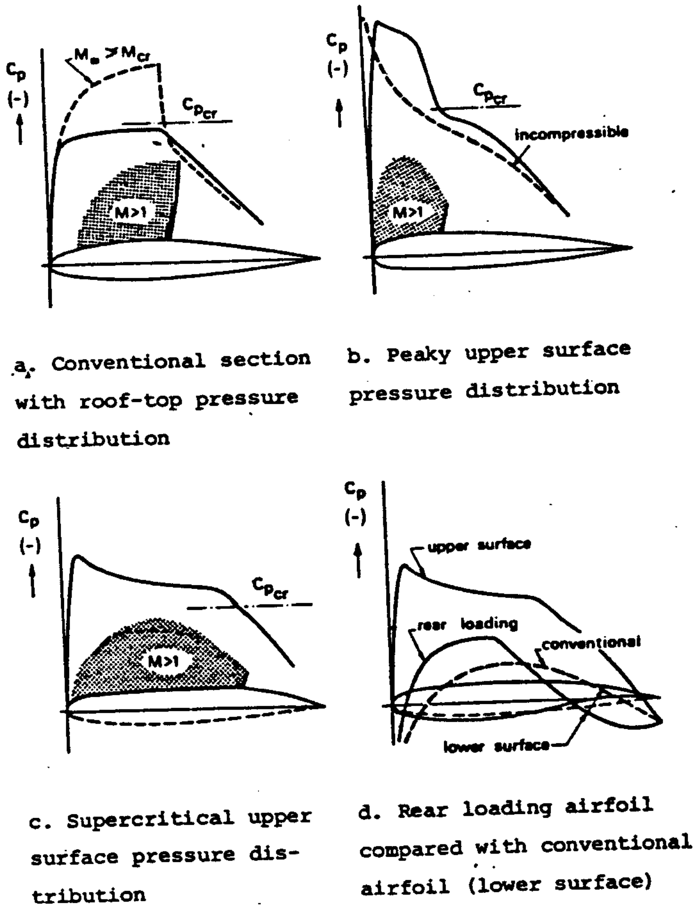
A: critical pressure - M_{cr}
B: drag-divergence - $M_{cr_D} = M_{DD}$
C: lift-divergence - M_{cr_L}

Fig. 7-16. Section lift and drag coefficients vs. Mach number

$$dc_{d_p}/dM_\infty = .10 \qquad (7\text{-}33)$$

An increase in M_∞ above this speed (point B in Fig. 7-16) results in a progressive rise in drag. The lift continues to increase until - at point C - a shock appears on the lower surface of the section. At this point - the lift-critical or lift-divergence Mach number - the lift coefficient diverges from its previous trend. As from this point onwards, dc_ℓ/dM_∞ is negative, a flight regime of longitudinal instability ("tuck under") becomes manifest. This must be neutralized by an artificial stabilization system ("Mach trimmer") or by aerodynamic means.
The presence of strong shock waves may lead to separation of the flow and large pressure fluctuations, experienced as an excitation of the wing. This phenomenon, referred to as buffet, constitutes a limitation to the operational flight regime.

c. Trends in high-speed section design.
Much work is being devoted to the development of airfoil sections, the objective being to increase the thickness ratio for given design conditions (drag-divergence Mach number and lift coefficient) and to improve the off-design characteristics. Early attempts in this direction were based on designing for low supervelocities in order to postpone supercritical flow to high Mach numbers. They were followed by designs where regions of local supersonic flow were admitted on the forward part of the airfoil, terminating in near-isentropic and shock free compression. These designs allow greater thickness ratios to be used for the same design Mach number and c_ℓ.
The aerodynamic analysis of mixed (subsonic and supersonic) flows is only possible with sophisticated methods, and a certain amount of empiricism is still common practice here. Several aspects of existing aerodynamic design concepts will be further explained, using the examples in Fig. 7-17. It is not suggested here that in modern section design a choice must be



a. Conventional section with roof-top pressure distribution

b. Peaky upper surface pressure distribution

c. Supercritical upper surface pressure distribution

d. Rear loading airfoil compared with conventional airfoil (lower surface)

Fig. 7-17. Aerodynamic design concepts for high-subsonic airfoil sections in the design condition

made from the various concepts; a mixture of these will be present in a practical wing.

ROOF-TOP PRESSURE DISTRIBUTIONS have a gradually changing or approximately constant upper surface pressure over the forward part of the section, which delays the critical Mach number by virtue of a uniform velocity at the design condition (Fig. 7-17a). Slightly above this speed large regions of supersonic flow will appear and the associated suction forces then occur near the crest* of the airfoil or behind it. The NACA 6-series have this type of pressure distribution at subcritical speed for a limited region of c_ℓ values. Since the pressure distribution is designed pri-

*The highest point of the airfoil relative to the free flow direction.

marily with the aim of obtaining low supervelocities rather than special high-speed characteristics, a strong shock and a rapid drag rise occur soon after M_{cr} is exceeded.
The drag-divergence Mach number of an airfoil section with a roof-top pressure distribution can be increased by extending the roof-top further back. For example, a crest at about 60 percent chord results in a ΔM_{cr_D} value of about .04 relative to a section with the crest at 30 percent. This effect cannot be exploited too far since the boundary layer may not be able to cope with the adverse pressure gradient aft of the roof-top without separating. The wing of the Aerospatiale Caravelle, originating from about 1950 is based on the 65-series airfoils; the operational speeds of this aircraft are relatively low, and allow a reasonable thickness ratio of 12 percent.

A PEAKY PRESSURE DISTRIBUTION (Fig. 7-17b), pioneered by Piercy at the NPL, and by others, intentionally creates supersonic velocities and suction forces close to the leading edge. The airfoil nose is carefully designed so that near-isentropic compression and a weak shock are obtained. The suction forces have a large forward component and the drag rise is postponed to high speeds. As compared with conventional sections of the same thickness ratio, the value of M_{cr_D} is approximately .03 to .05 higher and the off-design behavior is improved. This type of airfoil has been used on the BAC 1-11, VC-10 and DC-9 aircraft. The technique employed in designing peaky airfoils was highly empirical.

Recent advances in high-speed airfoil development have resulted in SUPERCRITICAL SECTIONS - first proposed by R.T. Whitcomb - which have a relatively flat upper surface contour (Fig. 7-17c). With these sections a much greater extent of shock-free supersonic flow can be created than in the peaky design. The amount of flattening of the upper surface is limited by the pressure rise which the boundary layer can accept without separation. An extensive NASA program has been conducted to investigate the potentials of this wing technology (Ref. 7-39). Large gains in drag-divergence Mach number and off-design performance are claimed if rear loading* is also used in designing supercritical airfoils.

*REAR LOADING (Fig. 7-17d) is a method for improving high-speed performance by generating lift at the rear part of the airfoil, mainly by pronounced camber of the lower surface.
The effect of rear loading may be explained in different ways. For a given thickness ratio and c_ℓ the supervelocities at the upper surface can be reduced, and M_{cr_D} can be increased, by generating higher pressures at the lower surface. Alternatively, if an airfoil is considered for which the upper surface has a critical flow condition, the rear part is contoured in such a way that a higher lift is generated, maintaining the same thickness ratio and M_{cr_D}. Finally, for a given M_{cr_D} and c_ℓ the front part of the airfoil may be beefed up until near-sonic flow is created at the lower surface, while the associated suction forces are cancelled out by high pressures near the trailing edge. Sections of high thickness ratio can thus be obtained for given design conditions.
The extent to which the rear loading can be accommodated is limited by the nose-down pitching moment and trim drag associated with the rear location of the center of pressure. It may also prove difficult to install an effective flap system in the sharp, cambered rear part of such a section. The European Airbus A-300 wing is an example of a wing design with a limited amount of rear loading (Ref. 7-28).

d. Criteria for section characteristics in design and off-design conditions.

The conditions to be considered in selecting airfoil sections are illustrated in Fig. 7-18. The sections should be selected

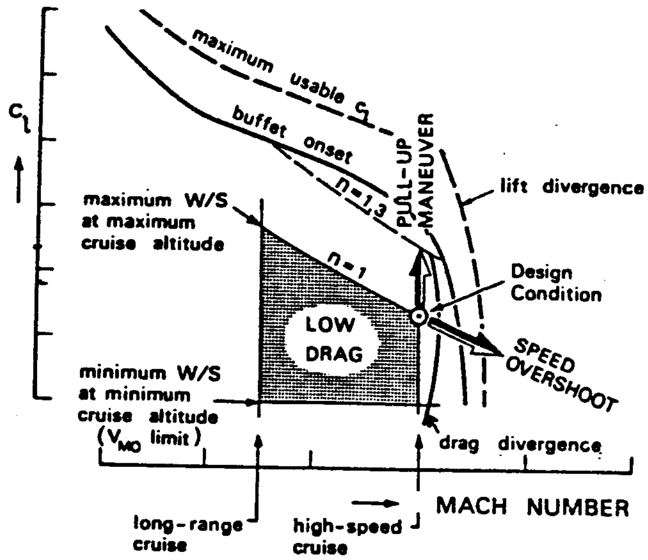


Fig. 7-18. Criteria for the selection of high-speed sections

primarily in order to achieve low drag at the high-speed cruise Mach number and the highest lift coefficient relevant to this speed. This point is labeled Design Condition in the figure. The lift coefficient in this condition corresponds with the maximum cruise altitude specified for the aircraft and the highest wing loading anticipated for cruising flight at that altitude.

The drag characteristics of an airfoil section selected on this basis will generally also be satisfactory at lower lift coefficients and reduced Mach numbers (shaded area in the figure) which correspond to lower cruising altitudes and speeds, down to the long-range cruise Mach number. This region of cruising conditions must be specified for any high-speed aircraft design in order to check the drag of the profile sections. Off-design conditions outside this region result from aircraft maneuvering and gustiness and may be associated with alleviated equirements relating to drag. A distinction may be made here between two types of maneuvers: overshoots in speed to increased Mach numbers, without significant change in c_ℓ, and pull-ups or turning flights with increased c_ℓ at constant Mach number. The overshoot in speed has to be demonstrated in certification flight testing to show compliance with requirements relating to stability and maneuvering characteristics up to the Dive Mach number M_D. In this Mach number region the interaction between shocks of increasing strength and the boundary layer will result in flow separations. A rapid rise in drag beyond the design Mach number caused by these effects may be conducive to smaller overshoots in speed, but a rapid lift divergence may be unacceptable as it can adversely affect stability. Shock-induced separations and shock waves hitting the tailplane cause buffeting vibrations, which should remain sufficiently mild in the dive and the ensuing pull-up maneuver to restore the normal flight condition. Buffet will also occur if c_ℓ is increased for constant M during pull-up or turning maneuvers and in gusty weather. Buffet onset will be experienced as a light vibration when the boundary layer starts to separate. The violence of the excitation will subsequently increase with the angle of attack through conditions of light and moderate to heavy buffet - a condition which defines the upper limit to the range of useful lift coefficients at high speeds. It is generally acknowledged, though not considered as a formal requirement, that for transport aircraft during cruising flight a maneuver load factor of at least 1.3 g must be available without any buffeting vibrations occurring. It is sometimes additionally laid down that load factors of up to 1.6 or normal gust velocities up to 12.5 m/s (41 ft/s) and a wavelength of 33 m (110 ft) must be covered with only a moderate amount of buffet, this condition being taken as the maximum penetration of the buffet regime acceptable for civil operations.

An important additional requirement for the airfoil section is therefore that it should be capable of producing lift without flow separation up to 1.3 times the design c_ℓ. Cruising performance may be seriously impaired if the buffet boundaries do not permit this reserve in c_ℓ in the design condition. A reduced cruising altitude is generally unfavorable and since the optimum cruise wing loading is high for high-speed transports much attention is devoted in the design stage to the development of section and wing shapes which are suitable both at design and off-design lift coefficients providing generous buffet margins.

It should be noted that the criteria presented here are basically applicable to the three-dimensional wing, or rather the complete aircraft. It will be shown, however, that for high aspect ratio wings these requirements can be transformed into airfoil selection criteria.

e. Thickness ratio and drag-divergence Mach number.

An objective of high-speed airfoil design is to obtain a section with the highest possible thickness for a specified combination of M_∞ and c_ℓ. In view of the very complex character of mixed flow, concise methods for making predictions of aerodynamic characteristics, such as drag- and lift-divergence Mach numbers, are not available. Faced with this problem, the preliminary design engineer will have to consult an aeronautical department or establishment equipped with facilities to tackle the problems of supercritical flow.

An estimation of the permissible thickness ratio of a section for a given value of M_{cr_D} may be obtained from available design charts or from the following suggested approach. Sectional data at low speeds ($M_\infty << 1$) indicate that the lowest pressure coefficient of symmetrical sections at zero lift is approximately:

$$\left(C_{p_i}\right)_{min} = \text{constant} \cdot \left(\frac{t}{c}\right)^{1.5} \qquad (7\text{-}34)$$

where the constant is determined by the thickness distribution. Using (7-29) and (7-31) it is easily found that for a given critical Mach number the permissible thickness is represented by:

$$t/c = \text{constant}\left\{\frac{2}{\gamma M_{cr}^2}\left[1-\left\{\frac{2+(\gamma-1)M_{cr}^2}{\gamma+1}\right\}^{\gamma/(\gamma-1)}\right]\cdot\sqrt{1-M_{cr}^2}\right\}^{2/3} \qquad (7\text{-}35)$$

For conventional sections, this expression gives a good result if the constant is taken equal to .24. More advanced sections may also be tackled by introducing a Mach number M^*, which represents the extent of supersonic flow at the condition of drag divergence. Substituting M^* into the compressible Bernoulli equation and into (7-30), we find for symmetrical sections at zero lift:

$$t/c = .30\left\{\left[1-\left\{\frac{5+M^2}{5+(M^*)^2}\right\}^{3.5}\right]\frac{\sqrt{1-M^2}}{M^2}\right\}^{2/3} \qquad (7\text{-}36)$$

In this equation M denotes the design (drag-critical) Mach number for which the airfoil is to be designed.

The factor M^* in (7-36) has no physical meaning and is merely a figure defining the aerodynamic sophistication employed to obtain supercritical flow at the design condition. Good results are obtained by taking:

$$\left.\begin{array}{l} M^* = 1.0\text{, conventional airfoils; maximum} \\ \quad t/c \text{ at about .30c} \\ M^* = 1.05\text{, high-speed (peaky) airfoils,} \\ \quad 1960\text{-}1970 \text{ technology} \\ M^* = 1.12 \text{ to } 1.15\text{, supercritical airfoils} \end{array}\right\} \qquad (7\text{-}37)$$

It is difficult to make adequate allowance for the effects of airfoil camber and lift. Provided the airfoil operates at the design c_ℓ, it is possible to use an approximation by reducing M^* in (7-36) by .25 times the design c_ℓ for c_ℓ up to .7.

7.5.2. Wing design for high speeds

a. Simple wing sweep theory and its limitations.

Sweeping back the wing postpones the effects of critical compressibility to a certain extent, an effect which can be explained by the "simple sweep concept". This assumes an infinitely long sheared wing (Fig. 7-19a), for which the supervelocities and the pressure distribution are determined solely by the velocity com-

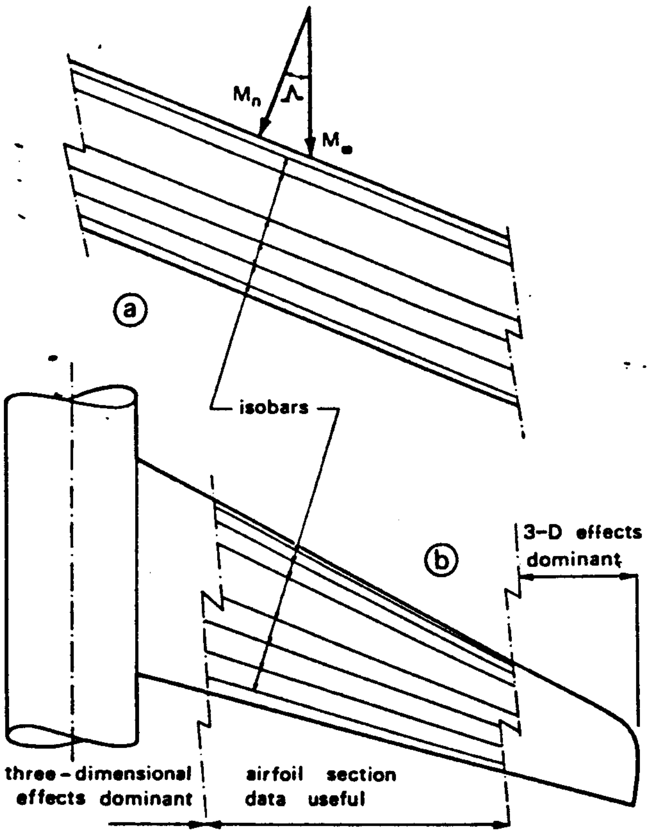


Fig. 7-19. The simple wing sweep theory for infinitely long and high aspect ratio wings

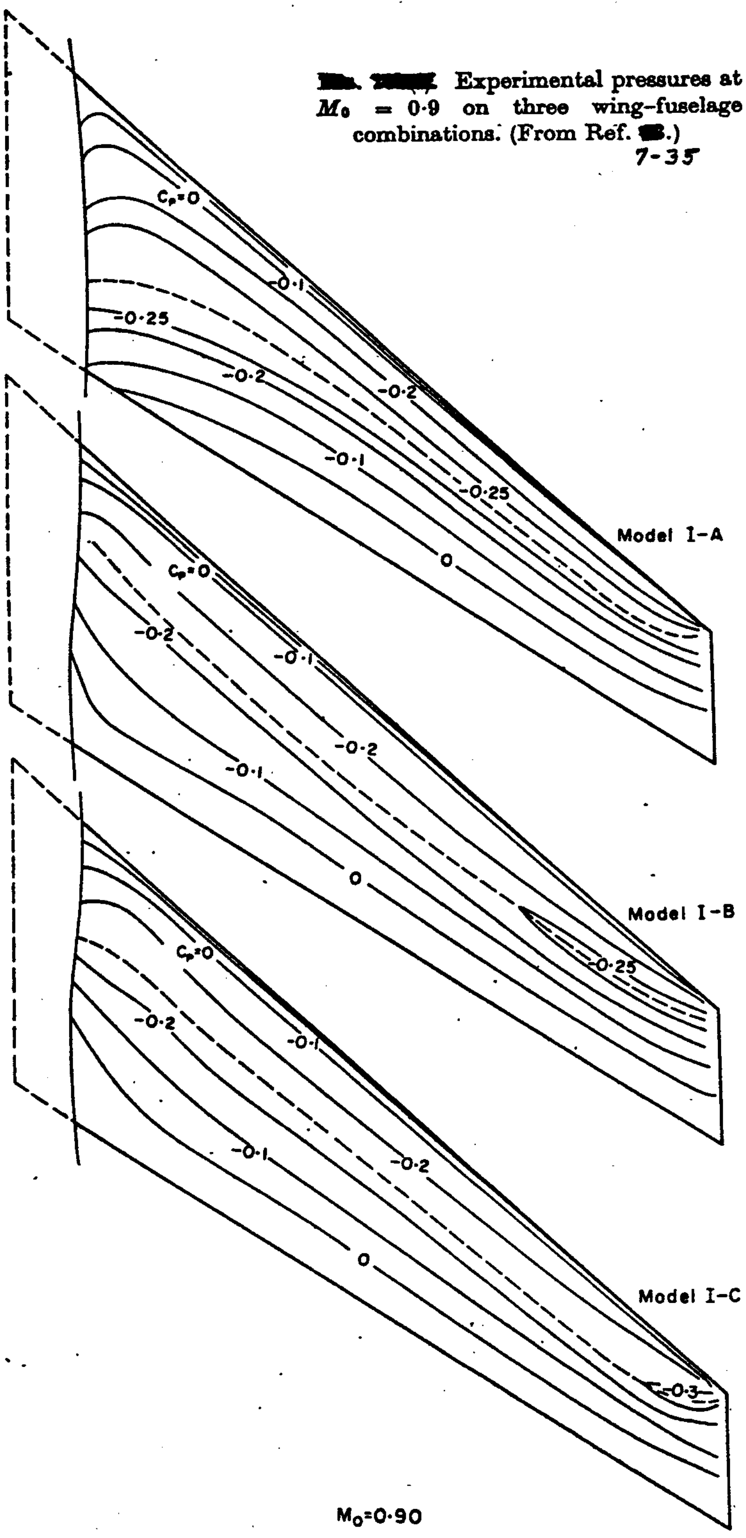
ponent normal to the leading edge, $M_n = M_\infty \cos\Lambda$. The section normal to the leading edge is thus the relevant shape to be considered. The simple sweep concept yields the following relationships.

1. The thickness ratio and effective angle of attack of the normal section are greater than those of the streamwise section by an amount equal to $1/\cos\Lambda$.
2. The wing lift coefficient C_L is equal to $c_\ell (\cos\Lambda)^2$, where c_ℓ is the normal section lift coefficient.
3. The critical Mach number of the wing is $M_{cr}/\cos\Lambda$, where M_{cr} is the normal section's critical Mach number at the normal section's c_ℓ.

These relationships show that potentially the critical compressibility effects may be postponed to a free stream Mach number which is increased relative to that for a straight wing by a factor $(\cos\Lambda)^{-1}$. The normal section shape, however, has to be designed to cope with a design value of c_ℓ which is greater than the wing lift coefficient by a factor $(\cos\Lambda)^{-2}$. For example, a wing sweptback by 35 degrees, and operating at $M = .85$ and $C_L = .4$, will have normal sections designed for operation at $M = .7$ and $c_\ell = .6$.

In early applications of sweptback wings it became apparent that the gain actually obtained in M_{cr} was smaller than predictions based on the simple sweep theory had indicated; in effect a factor $(\cos\Lambda)^{-1/2}$ was achieved, rather than $(\cos\Lambda)^{-1}$. The reasons for this observation stem from the finite span and the detrimental effects of fuselage and nacelles. For an untwisted sweptback wing with constant section shapes the lift is concentrated towards the tip and the outboard sections work at relatively high c_ℓ values, resulting in a local reduction in M_{cr}. In addition, root and tip effects decline the isobars in a direction normal to the flow, so that the sweep effect is, in fact, reduced. The points of lowest pressure will be shifted backwards near the root and forwards near the tip. The natural curved path of the flow over a swept wing is hampered by the fuselage and nacelles, if any, and this results in a system of expansion waves, terminating in a spanwise shock. The various effects combine to form a complex pattern of shocks and expansion waves, appearing first on the outboard wing and next on the rear and front part of the inner wing. The complete picture is sometimes referred to as a λ-shock. The techniques developed to restore the full effect of wing sweep are therefore based on eliminating these detrimental effects by reshaping the wing and bodies in the areas affected (Fig. 7-19b). The mid-section of a high aspect ratio wing, however, is only slightly affected by three-dimensional effects, provided suitable measures have been taken to counteract the root and tip effects. Two-dimensional section shapes and data may therefore be used to define this part of the wing.

[redacted]. [redacted] Experimental pressures at $M_0 = 0\cdot9$ on three wing-fuselage combinations. (From Ref. [redacted].)
7-35



. . . . . . optimization of the wing and drag-divergence Mach number.

The objective of high-speed wing design is to obtain a pattern of approximately straight isobars swept back at an angle at least equal to the wing sweepback angle, the upper surface generally being critical for the drag divergence*. If this aim is achieved, the flow will be approximately two-dimensional and the drag-divergence will occur at the same Mach number everywhere along the span. A detailed examination of the very complex optimization procedures of this type is outside the scope of this *note*, but it is considered appropriate to mention some of the measures which may be taken, although not all of them are required for each design.

1. The sweep angle and thickness ratio between approximately 30 and 80 percent semi-span from the root are based on a pressure distribution obtained from the simple sweep concept, *corrected for taper.*#

2. The points of maximum thickness at the root and tip are shifted forwards and backwards, respectively. Streamwise tips are used on the BAC VC-10.

3. The lift on the inboard wing is increased by a negative twist (washout). The example in Fig. 7-20 shows a linear lofted twist compared with the more complex twist distribution required to increase the critical Mach number.

4. For low-wing designs the pressures tend to be increased at the lower surface. This may be turned to advantage by locally thickening the lower part of the section and bending the nose of the root section slightly upwards. This results in a root section with negative camber which is a few percent thicker than the outboard wing (Fig. 7-20).

5. The sweep angle near the root section may be increased by introducing a kink in the leading edge (Fig. 7-21). Incidentally, the kink in the trailing edge, observed

*Wings with rear loading sections may form an exception in view of the near-critical conditions at the lower surface

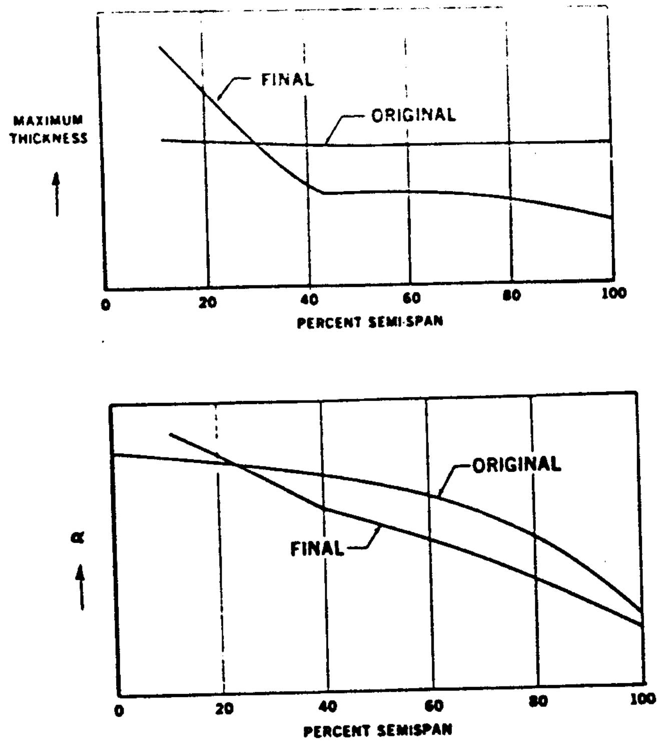


Fig. 7-20. Typical spanwise thickness and twist angle distribution before and after aerodynamic optimization (Ref. 7-24).

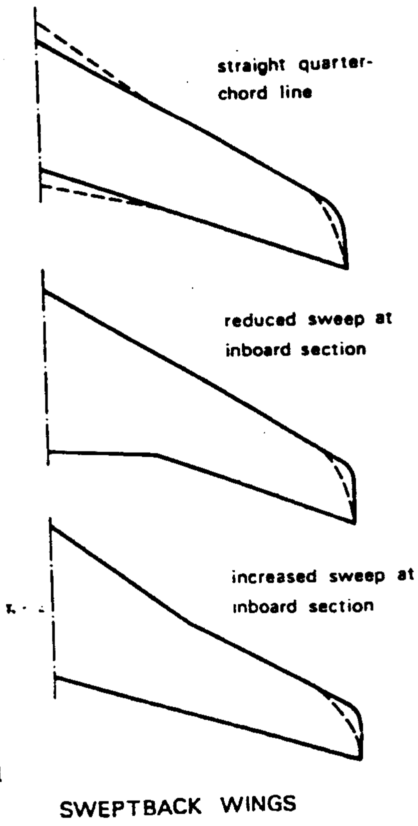


Fig. 7-21. Sweptback wing planforms.

#Locke Method, pp 21-22, ref. 7-36

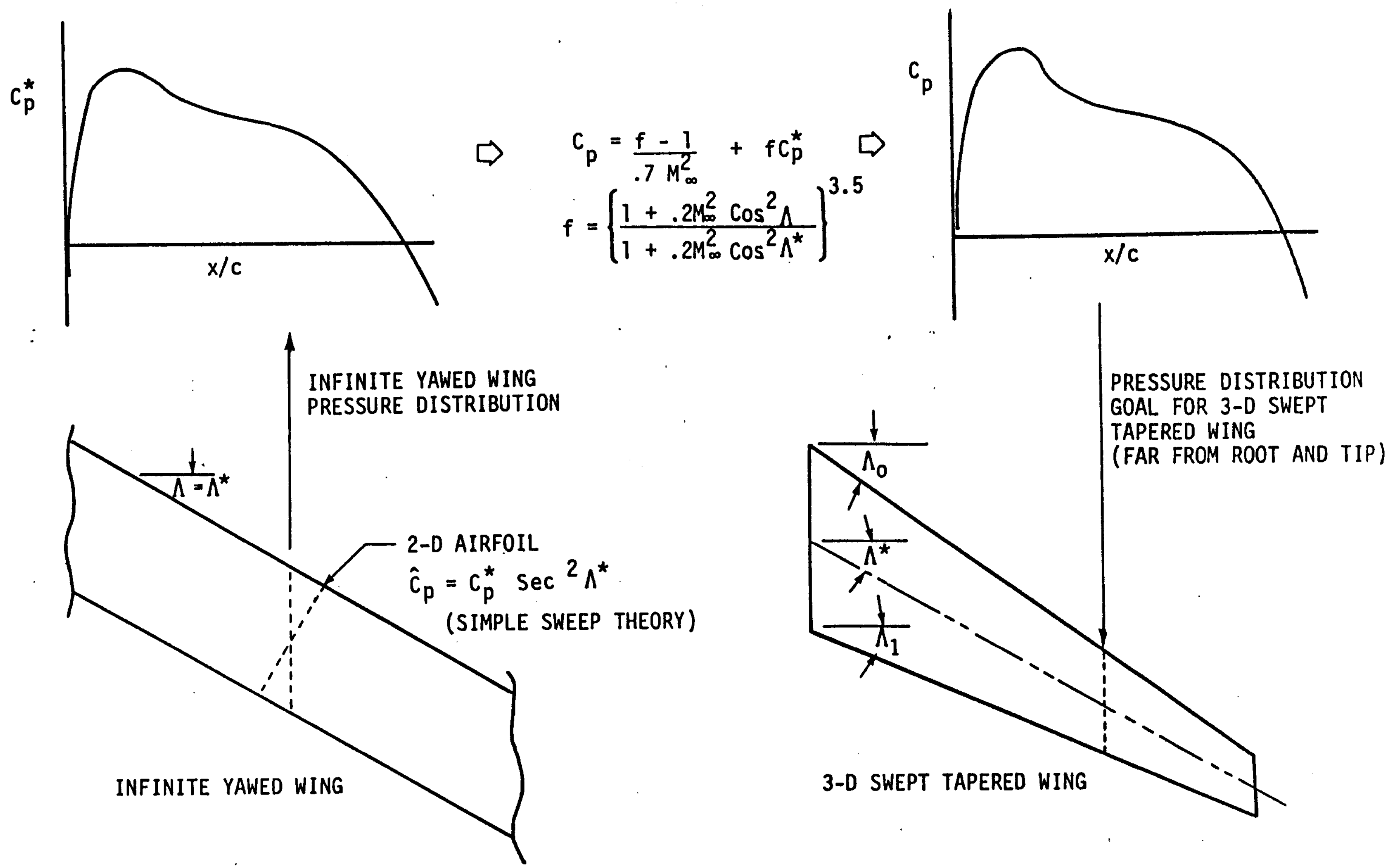


FIG. B-3 APPLICATION OF LOCK'S APPROXIMATE EQUIVALENCE METHOD

8.22

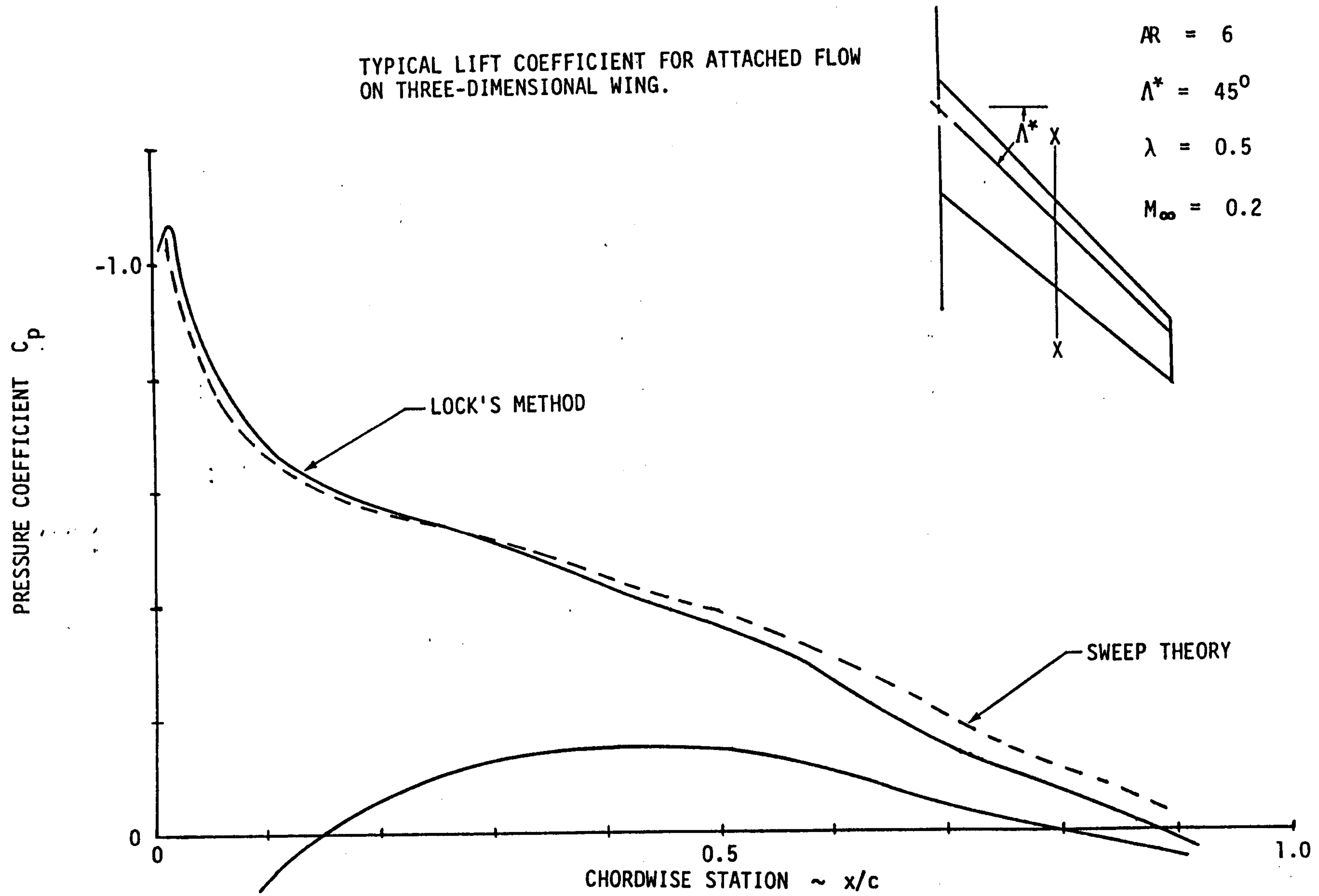


FIGURE 35. COMPARISON OF CHORDWISE LOADING ON A PLAIN TAPERED SWEPT WING USING SIMPLE SWEEP THEORY AND LOCK'S METHOD.

8.23

in many high-subsonic transport, is used to provide internal space for retracting the undercarriage.

6. Adequate wing/fuselage fairings must be incorporated as a means of obtaining a good area distribution.

The cylindrical fuselage is generally maintained in the interest of interior layout but if the speed is increased above, say, $M_\infty = .90$, wing/fuselage blending is generally regarded as unavoidable.

Some examples of swept wing planforms are shown in Fig. 7-21. These wings are usually composed of at least two sections, with linear lofted contours between the intermediate profiles. This sophistication in aerodynamic design results in compound curvatures, giving rise to complications in the structural design and the manufacturing process. An early example of an aerodynamically efficient design, the crescent wing (Handley Page Victor) has never been widely adopted, probably due to its structural complexity.

For the purpose of preliminary wing optimization it may be desirable to derive the combinations of $\overline{t/c}$ and $\Lambda_{.25}$ required to achieve a specified drag-divergence Mach number. Equation 7-36 may be modified by substitution of $M\cos\Lambda_{.25}$ and $\overline{t/c}$ x $(\cos\Lambda)^{-1}$ where M and t/c are used. For wings with symmetrical sections at zero lift we then find:

$$\frac{\bar{t}}{c} = \frac{.30}{M}\left\{(M\cos\Lambda_{.25})^{-1} - M\cos\Lambda_{.25}\right\}^{1/3} \times \left[1 - \left\{\frac{5 + (M\cos\Lambda_{.25})^2}{5 + (M^*)^2}\right\}^{3.5}\right]^{2/3} \tag{7-38}$$

Here M refers to the drag-divergence Mach number, defined by $\Delta C_{D_p} = .002$, and $\overline{t/c}$ is the thickness ratio at about 50 percent semi-span from the root. The values for M^* are given by (7-37) for zero lift and should be reduced by $.25C_L\,(\cos\Lambda_{.25})^{-2}$ to account for lift and camber at the design condition.

The drag-critical Mach number to be selected depends on the amount of compressibility drag the designer is prepared to accept during cruising flight. It is fair to assume that in high-speed flight over relatively short distances a penalty of up to some 20 to 30 counts will be acceptable. In this case the drag-divergence Mach number may be taken as being equal to or slightly below the high-speed Mach number.

For example, if a high-speed cruising flight Mach number of .8 is specified at $C_L = .3$, we may assume for a straight wing with a peaky-type airfoil $M^* = 1.05 - .25(.3) = .975$. Substitution of this value, together with $M = .8$ and $\Lambda = 0$ into (7-38), results in a permissible thickness ratio of only 9.7 percent. For a sweep angle of 30° we obtain $M^* = .95$, and a permissible thickness ratio of 12 percent. This would be increased to 15.3 percent if an advanced airfoil could be designed with $M^* = 1.20$ at zero lift. This may be compared with the allowable 11 percent thickness for a conventional airfoil with the maximum thickness at about 30 percent chord.

The permissible thickness of a wing may well be determined by the requirement of freedom from buffet. As the prediction of buffet boundaries is as yet based on wind tunnel measurements; theoretical design methods for matching a wing to a specified buffet boundary cannot be presented.

3.0 WING-BODY DESIGN PROCESS

3.1 Design Constraints

3.1.1 Parametric Constraints

The designer will either decide on, or be supplied with, the following constraints:

- o S_w - wing area (trapezoid),
- o b - wing span (trapezoid),
- o $\Lambda_{effective}$ - sweep of shock line, or
- o $\Lambda_{c/4}$ - quarter chord sweep (trapezoid),
- o λ - taper ratio, tip chord/root chord (trapezoid),
- o t/c - spanwise thickness/chord (excluding yehudi),
- o outboard section - either 2-D airfoil or wing section,
- o body station of wing mean aerodynamic center,
- o cruise lift coefficient and span loading,
- o cruise Mach number.

For transonic cruise speeds, the additional constraints apply:

- o total area distribution
- o lift compensation

The wing trapezoid is used for these parameters. This is because the span load is a design constraint, and any planform variations to the trapezoid should be used to modify the wing section lift coefficient, not the span loading. Changes in span loading would change the induced drag, and the selection of design constraints should have assumed the most favorable span loading by trading induced drag against structural weight.

AD 1846D

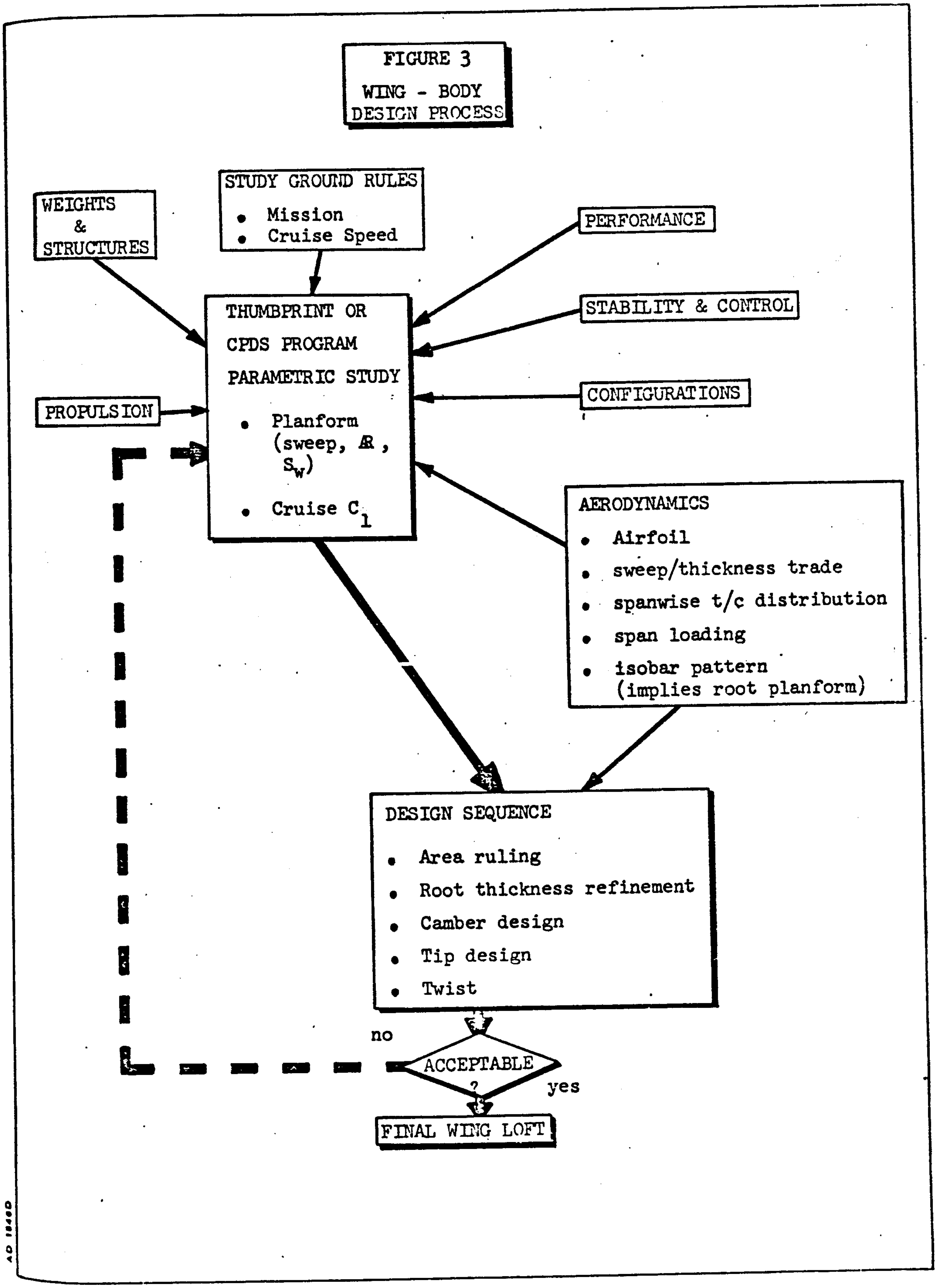
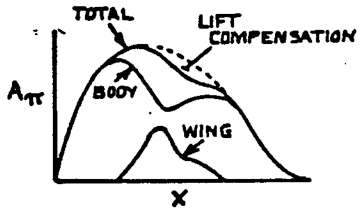
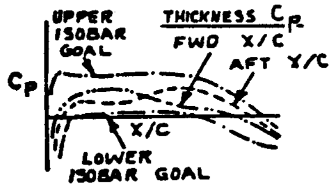


FIGURE 3
WING - BODY
DESIGN PROCESS
WEIGHTS
&
STRUCTURES
STUDY GROUND RULES
• Mission
• Cruise Speed
PERFORMANCE
STABILITY & CONTROL
CONFIGURATIONS
THUMBPRINT OR
CPDS PROGRAM
PARAMETRIC STUDY
• Planform
(sweep, Æ,
Sw)
• Cruise Cl
PROPULSION
AERODYNAMICS
• Airfoil
• sweep/thickness trade
• spanwise t/c distribution
• span loading
• isobar pattern
(implies root planform)
DESIGN SEQUENCE
• Area ruling
• Root thickness refinement
• Camber design
• Tip design
• Twist
no
ACCEPTABLE
?
yes
FINAL WING LOFT

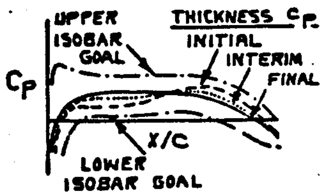
TABLE I
WING-BODY DESIGN SEQUENCE



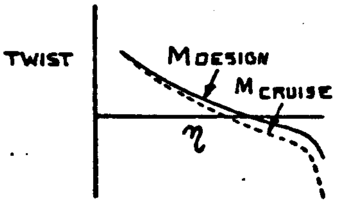
BODY DESIGN. For subsonic designs, the body geometry will be predefined. For transonic designs, calculate the wing normal area using initial airfoil sections, then design the fuselage cross section to match the total area distribution, allowing for lift compensation.



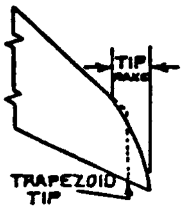
ISOBAR GOAL AND INITIAL THICKNESS EVALUATION. Choose the upper isobar goal. Use design program to evaluate various locations for maximum thickness in the wing root. Choose the most aft location that can meet isobar and section C_l design goals, as well as fuselage indentation criteria.



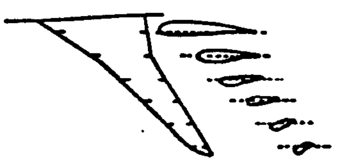
THICKNESS FORM REFINEMENT AND CAMBER DESIGN. Modify t/c forms until desired thickness pressures (from desired isobars) are achieved. Calculate camber shape to obtain desired isobars using the final thickness form.



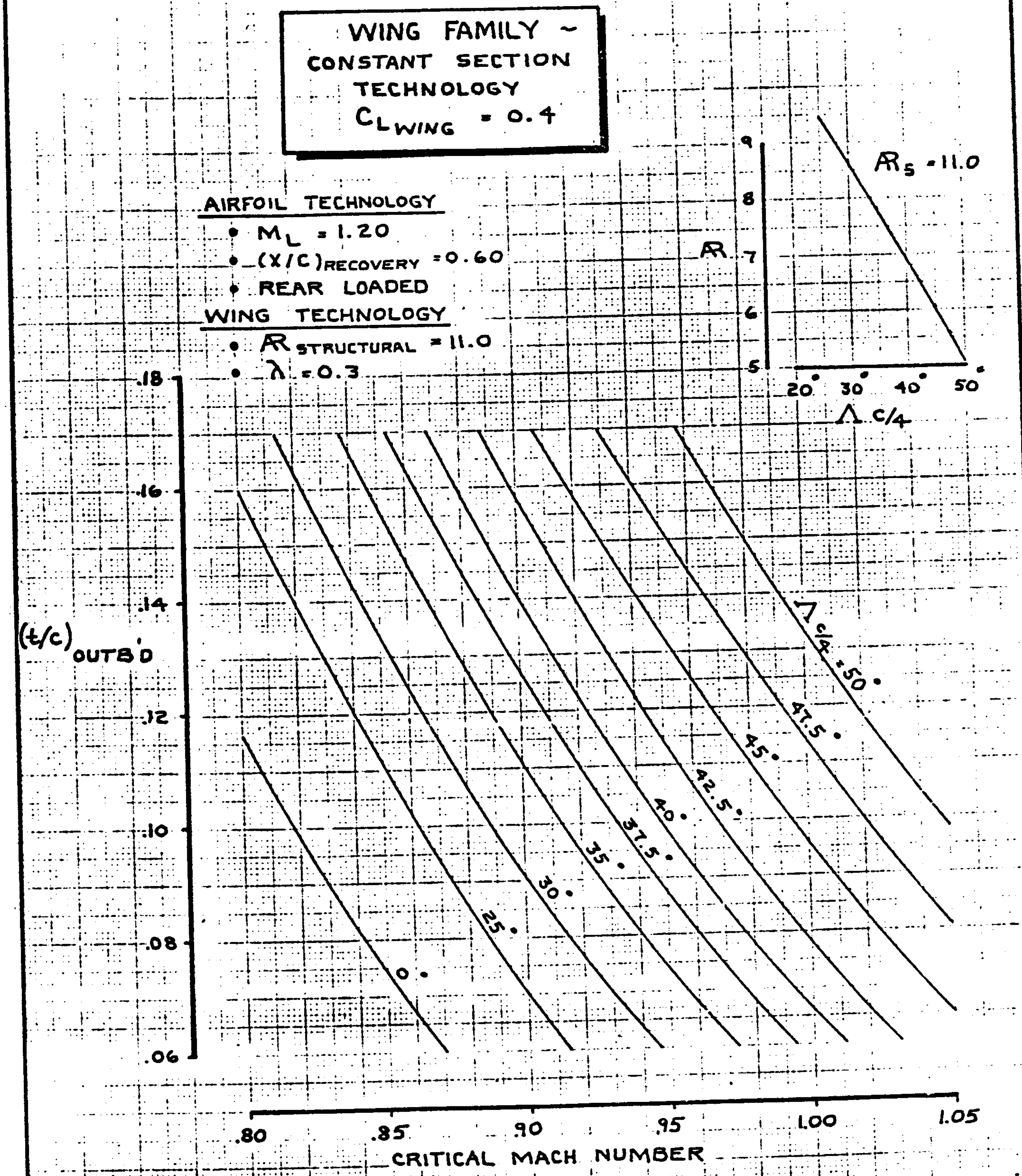
TWIST DESIGN. Input desired span loading and final airfoil sections to the design program at the cruise Mach number to calculate the required section twist. This twist will be used with the final sections.



TIP DESIGN. Apply an empirical tip rake rule, local thinning and section changes to generate the actual tip geometry.

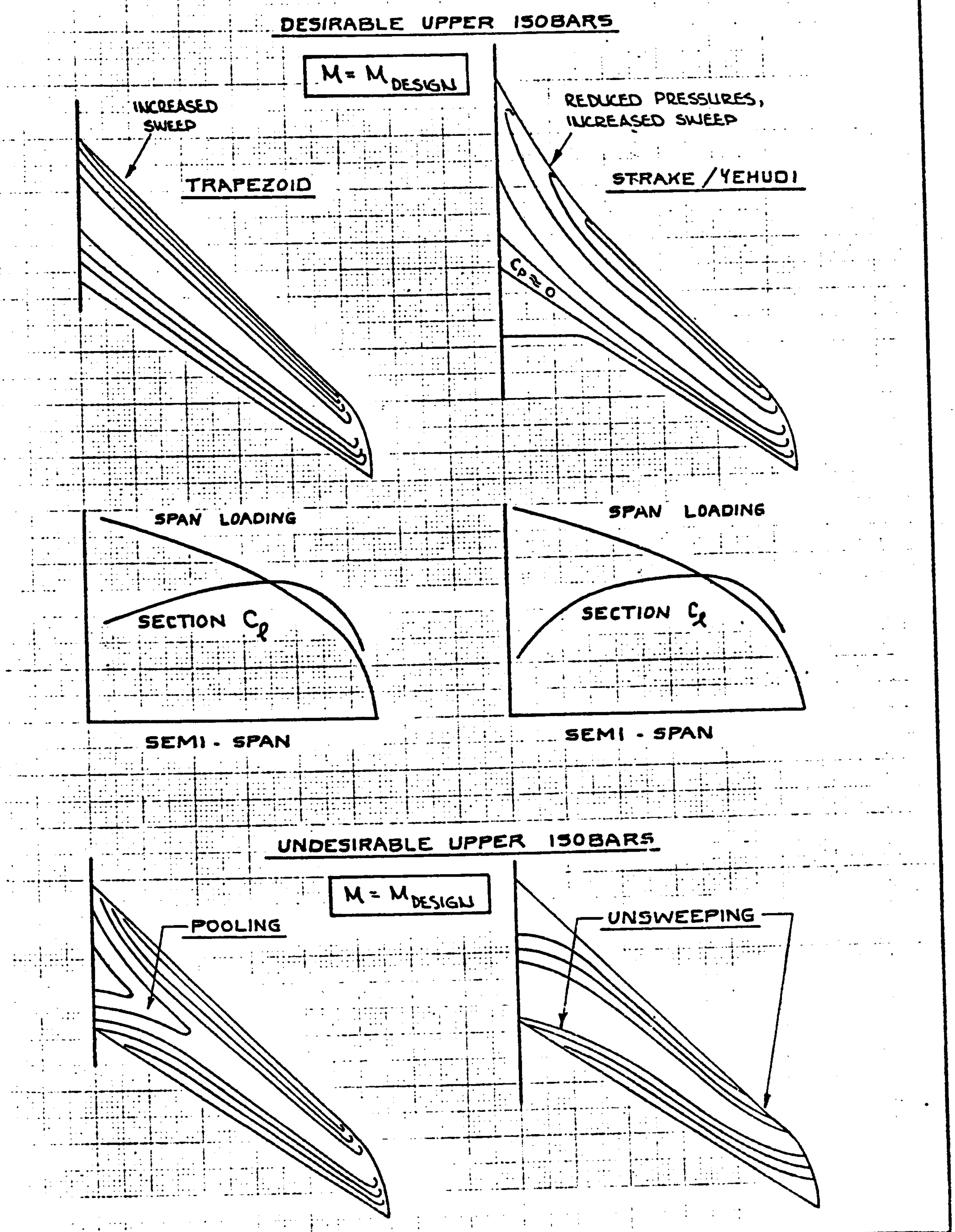


FINAL WING LOFT. Combine sections and twist to define the final wing geometry.



CALC	GILLETTE	3-9-72	REVISED	DATE	WING FAMILY TRADES FOR CONSTANT SECTION TECHNOLOGY
CHECK					FIG. A
APR					PAGE
APR					THE BOEING COMPANY
	SAW	3-7-72			

K-E



CALC	GILLETTE	4-19-71	REVISED	DATE	UPPER SURFACE ISOBAR PATTERNS AT DESIGN MACH NUMBER	FIG. 9
CHECK						
APR					THE BOEING COMPANY	PAGE
APR						

COMPARISON OF 2-D AIRFOIL & DERIVED 3-D WING PRESSURE DISTRIBUTIONS

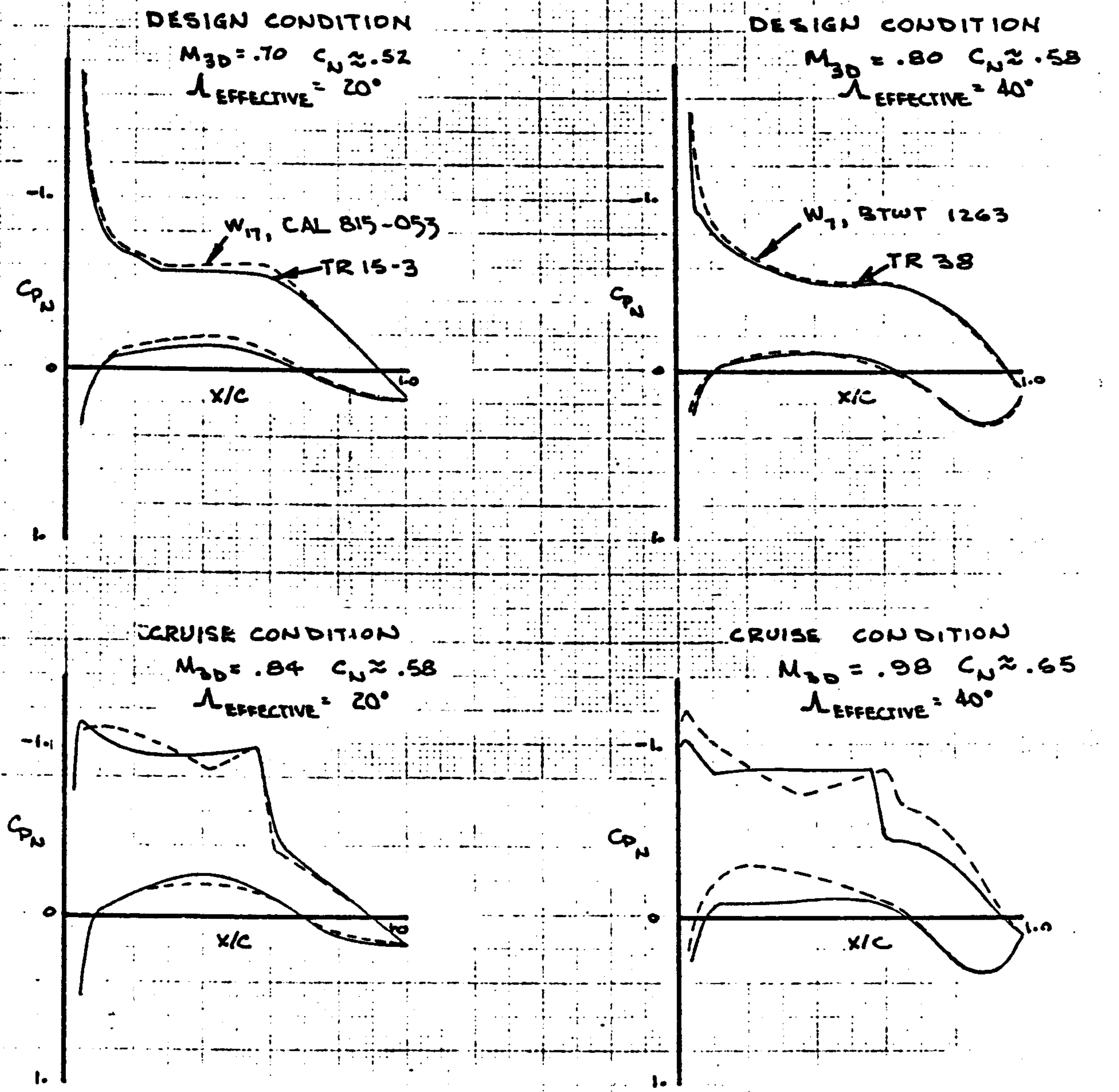
SIMPLE SWEEP:

$$C_{P_N} = C_{P_{3D}} / \cos^2\Lambda$$

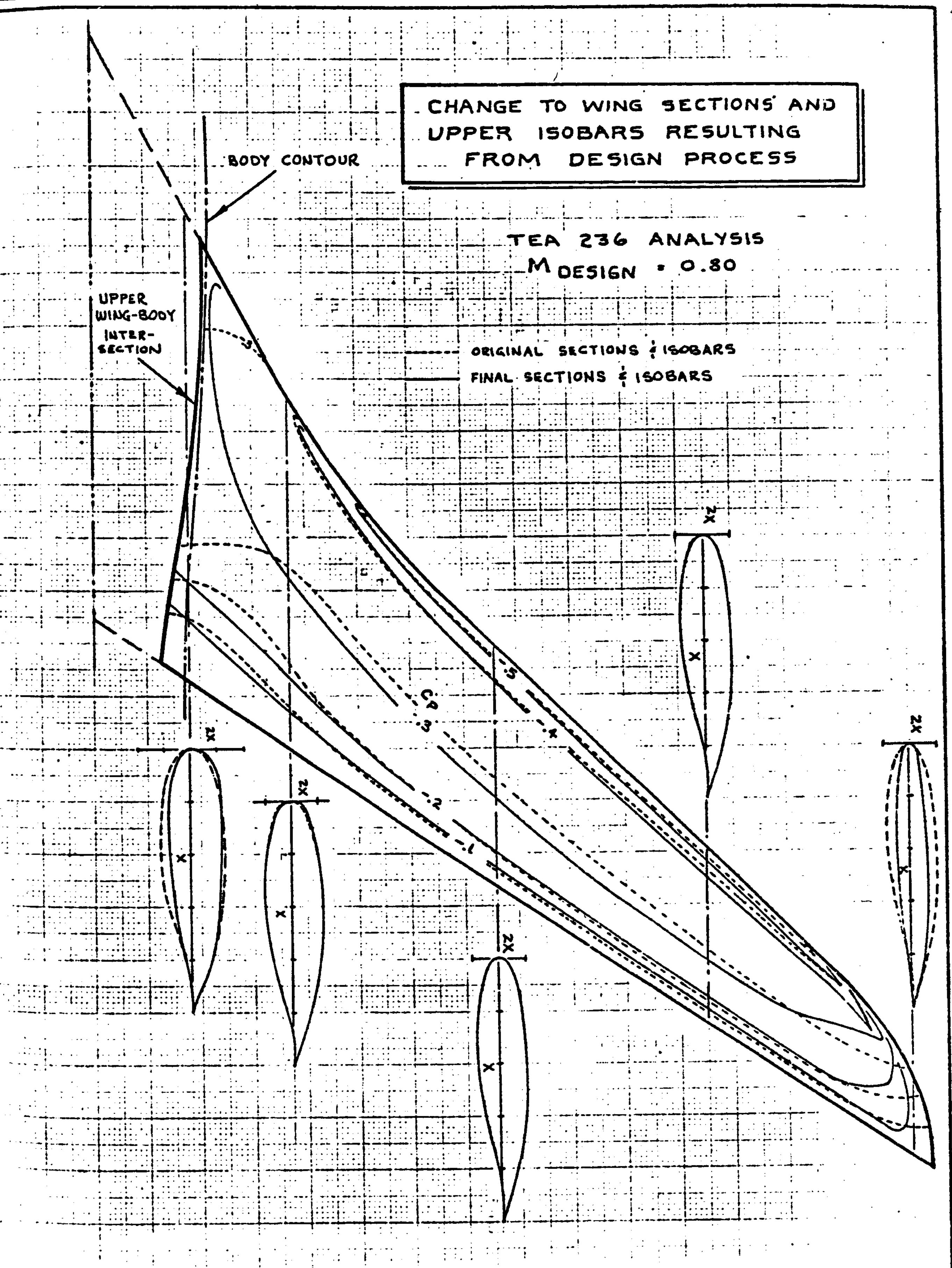
$$C_N = C_{N_{3D}} / \cos^2\Lambda$$

Λ = SHOCK SWEEP ANGLE

CONVENTIONAL TRANSPORT | TRANSONIC TRANSPORT



CALC	GILLETTE	8·2·71	REVISED	DATE	2-D AIRFOIL & 3-D WING PRESSURE COMPARISONS USING SIMPLE SWEEP	FIG. 13
CHECK						
APR						
APR					THE BOEING COMPANY	PAGE



CALC	GILLETTE	4-21-71	REVISED	DATE	WING SECTION CHANGES RESULTING FROM DESIGN SEQUENCE	
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APR					THE BOEING COMPANY	PAGE
APR						

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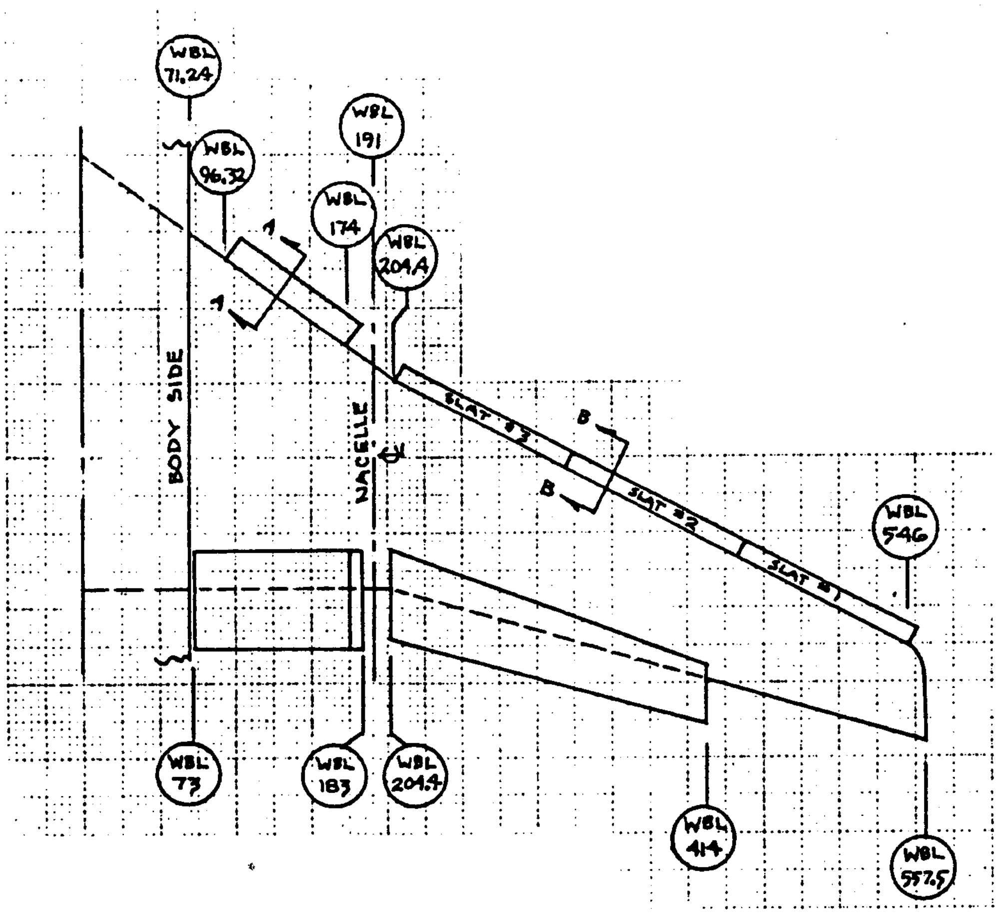
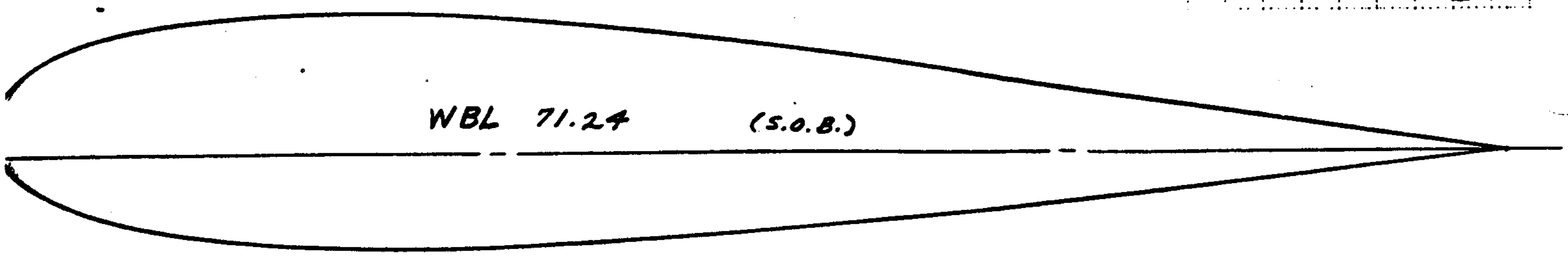
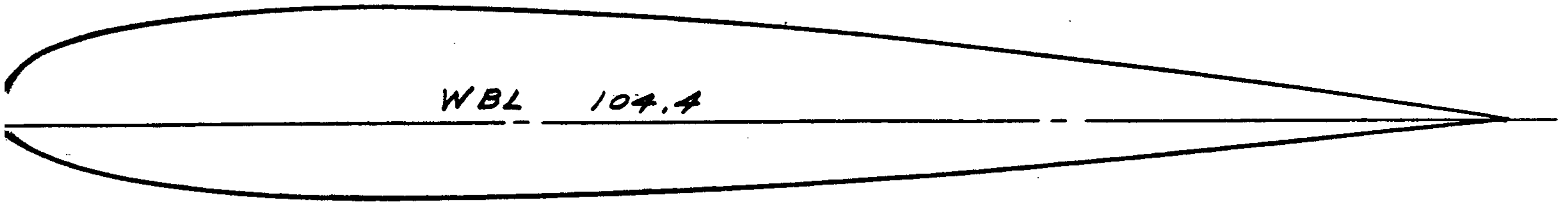
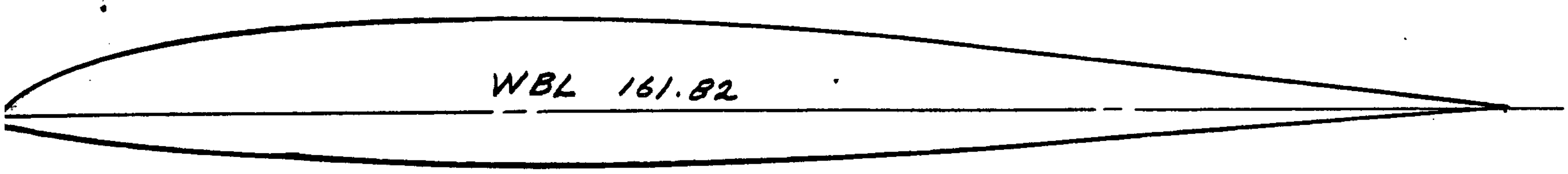
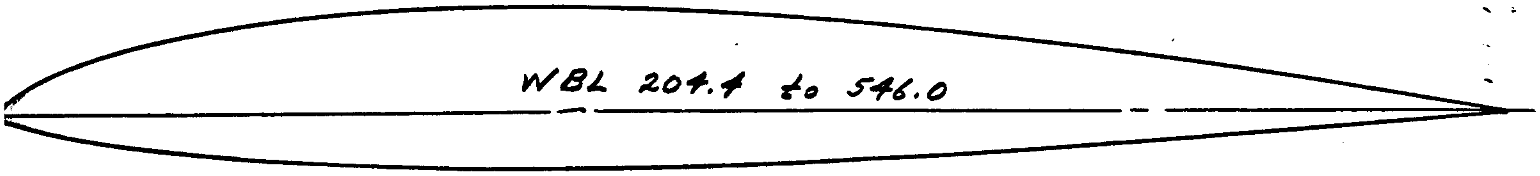
K·E ALBANENE 195L 8.5 G-6000

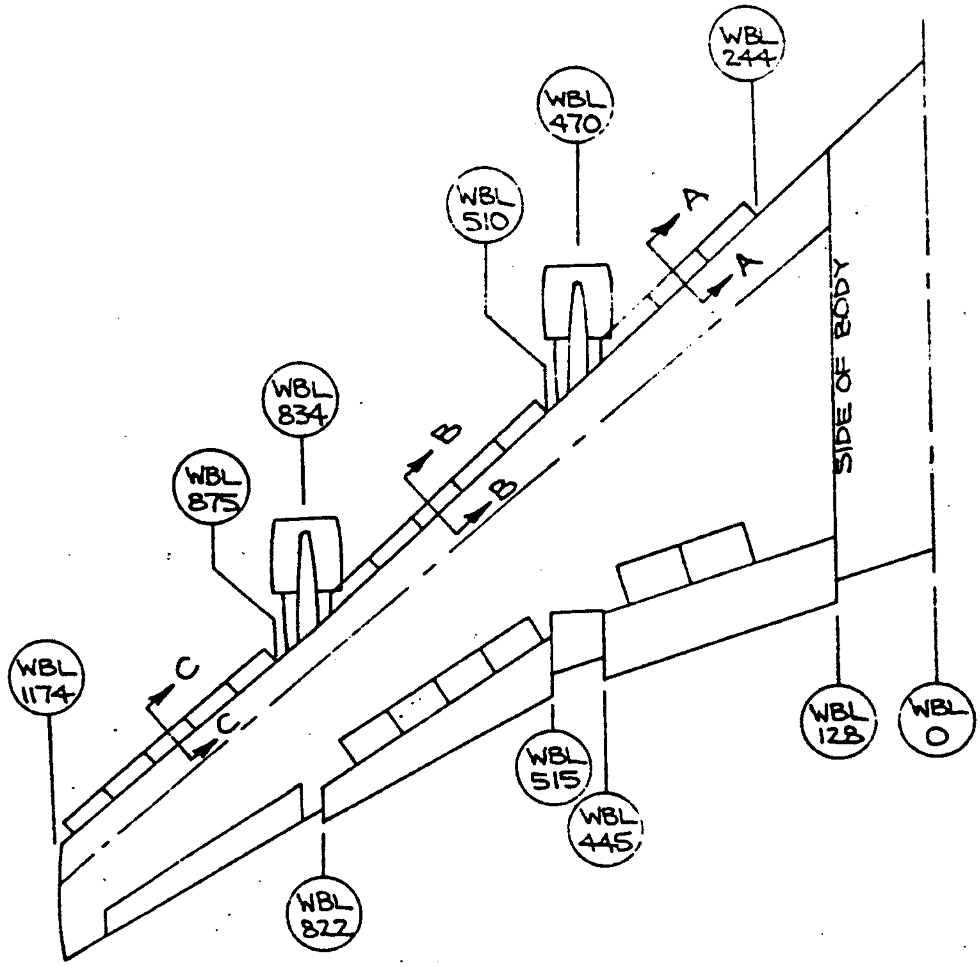

WBL 71.24
WBL 96.32
WBL 191
WBL 174
WBL 204.4
A
A
BODY SIDE
NACELLE
SLAT #3
SLAT #2
SLAT #1
B
B
WBL 546
WBL 73
WBL 183
WBL 204.4
WBL 414
WBL 557.5


WBL 71.24 (S.O.B.)


WBL 104.4


WBL 161.82


WBL 204.4 to 546.0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WBL 834
WBL 875
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A
A
B
B
C
C
SIDE OF BODY
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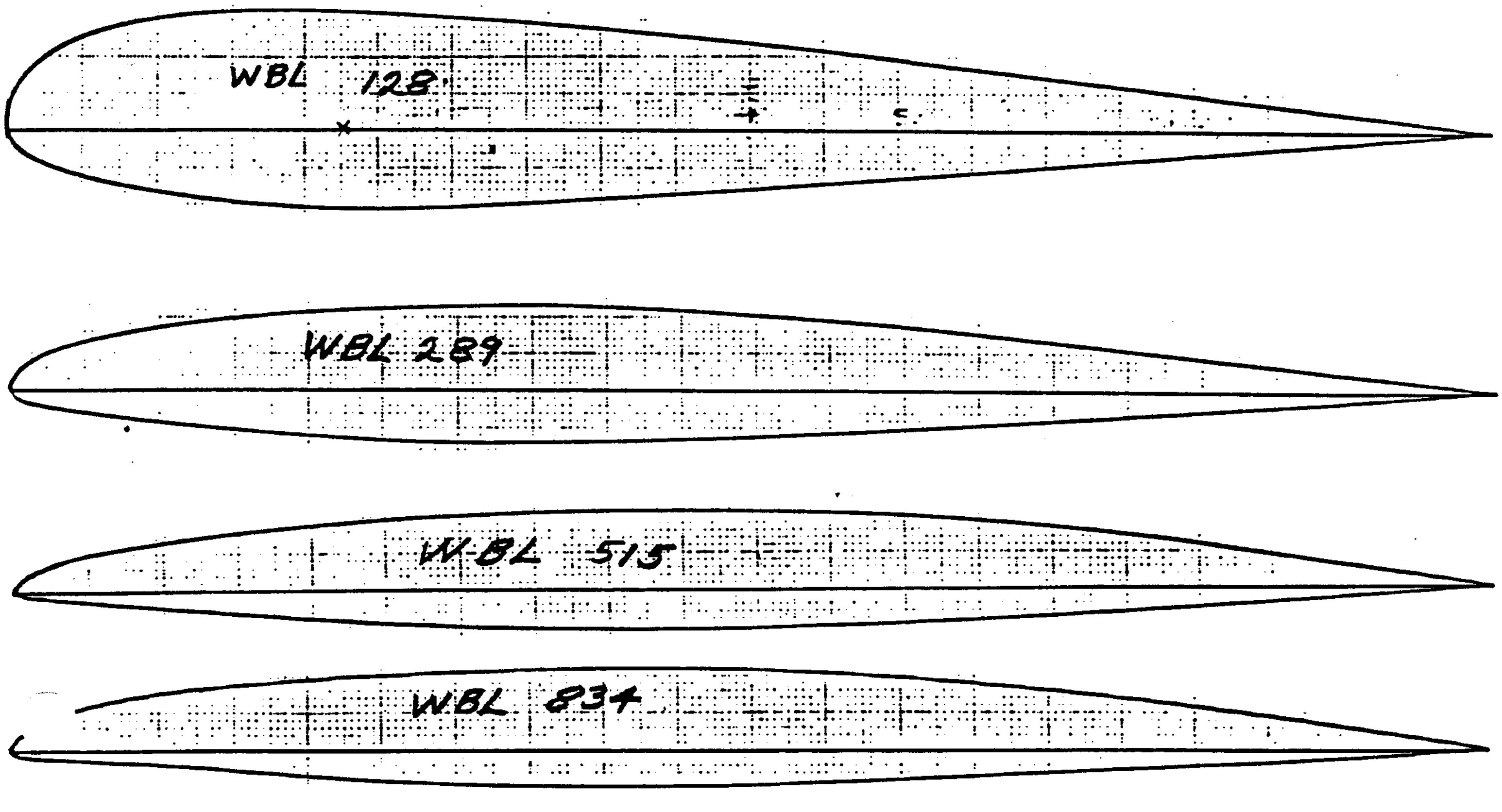
WBL 0
WBL 515
WBL 445
WBL 822


WBL 128
WBL 289
WBL 515
WBL 834

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JET AIRCRAFT

	AIRCRAFT TYPE	1st flight prototype	Aspect ratio A	Taper ratio λ	Sweep angle $\Lambda_{.25}$ deg.	Geom. twist ϵ_g deg.	Dihedral Γ deg.	Profile type and streamwise thickness root %	tip %	$(\overline{t/c})$ %	V_{MO}*** km/h EAS	M_{MO}	V_D km/h EAS	M_D	Flap type* T.E./L.E.	$(\overline{c_f/c})$ streamwise	b_f/b %	Flap angle takeoff deg.	Flap angle landing deg.	$C_{L_{max}}$ (flight test) takeoff	$C_{L_{max}}$ (flight test) landing
JET TRANSPORT AIRCRAFT	Yakovlev YAK 40	1966	9	.396	0	-	4°30'	15	10	12,5		.70	-	-	P	31.5	67	-	-	-	2.10
	VFW-Fokker 614	1971	7.22	.402	15	3°45'	3°	63_2A-015	65_1A-012	13,5	528	.650	615	.740	F1	31	69	20	40	2.12	2.37
	Grumman Gulfstream II	1966	5.97	.370	25	-	3	a=.4 mod.	a=.5 mod.	-	680	.860	-	-	F1	30	73	20	40	-	1.80?
	Fokker-VFW F 28	1967	7.27	.355	16	-	2°30'	13 Naca-6 series 13.2	10	13	611	.750	-	.830	F1-2	32	69.5	25	42	2.16	2.53
	BAC One-Eleven Srs 200/400	1963	8.00	.321	20	-	2	12.5	11	11,8	652*	.780	760	.860	F1	30	75	18	45	1.86	2.40
	McDonnell-Douglas DC-9 srs 10	1965	8.56	.246	24	-4	3	13.65	9.8	11,65	630	.840	-	.890	S2	36	67	15	50	-	2.40
	McDonnell-Douglas DC-9 srs 30	1966	8.72	.226	24	-4	3	13.65	9.6	11,6	630	.840	-	.890	S3/I	36	67	15	50	2.45	2.98
	Tupolev Tu 134/134A	1964	7.42	.287	35	-	-1°30'	-	-	-	-	.817	-	-	S2	23	60.5	25	30	1.51	1.67
	Boeing 737 srs 100/200	1967	8.83	.251	25	-	6	14	11.5	12,9	648	.840	722	.890	F3/I,II	29	74	15?	40	2.20	3.05
	Aérospatiale Caravelle	1955	8.02	.354	20	-2	3	65_1-212	65_1-212	12	556	.810	-	.870	S1	27	66	10	45	1.74	2.10
	Hawker Siddeley Trident 2E	1967	6.57	.240	35	-5	3	11.5	-	9,8	630	.880	732	.950	S2/I	28	69	20	50	1.93	2.40
	Boeing 727 srs 100/200	1963	7.67	.323	32	-	3	13	9	11	722	.900	-	.950	F3/1,II	30	74	25	40	2.35	2.75
	Tupolev Tu 154	1968	8.14	.250	35	-	-	-	-	-	575	.900	-	-	S3/I	31	75	-	-	2.05	2.70
	Boeing 707/720	1957/60	7.11	.293	35	-	7	12.5	9	10	710*	.900	787	.950	F1/I,II	27	67	15	50	1.76	1.94/2.06
	McDonnell-Douglas DC-8 srs 10,50,61	1958/60/66	7.30	.244	30	-	6°30'	12	10.16	11,1	630	.880	751	.950	S2/II	30	76	25	50	1.87	2.03
	McDonnell-Douglas DC-8 srs 62/63	1966/67	7.65	.194	30	-	6°30'	-	-	-	-	-	770	.950	S2/II	30	73	23	50	1.87	2.05
	BAC VC-10 srs 1100/Super VC-10	1962/64	7.49	.273	32°30'	-	3	12.5	9.75	10	562	.860	704	.940	F1/I	29	62	-	45	-	2.27
	Lockheed L-300 Starlifter	1963	7.90	.350	25	-5°30'	-1°26'	0013 mod.	0010 mod.	11,5	-	-	760*	.890	S2/I	32	66	-	-	-	2.32
	Ilyushin IL 62	1963	6.675	.262	35	-	-	-	-	-	-	-	-	-	S2	24	72	-	30	-	2.36
	BAC Three-Eleven	1972	8.00	.255	25	-	-	12.5	10	14,2	656*	.840	-	-	F1/I	-	75	22	45	-	3.19
	A 300B	1972	8.60	.364	28	-	-	15	-	13,5	668	.840	-	-	F2/I	32.5	82	15	35	2.70	3.00
	Lockheed 1011	1970	7.16	.296	35	-3°40'	7°31'/5°30'	12.4	9	10,7	-	.858	-	.950	S2/I	26	77	22	42	2.46	2.65
	McDonnell-Douglas DC-10 srs 10	1970	6.90	.250	35	-	-	12.5	10	11	-	.880	-	-	S2/I	25?	77.5	25	50	2.34	2.73
	McDonnell-Douglas DC-10 srs 30	1971	7.21	.230	35	-	-	12.2	8.4	11	-	.880	-	-	S2/I	25?	77.5	15	55	2.15	2.98
	Boeing 747/747B	1969/70	6.96	.309	37°30'	-3°30'	7	13.44	8	9,4	695	.920	824	.970	S3/II	30	70	20	30	1.89	2.55
	Lockheed L500 Galaxy	1968	7.75	.256	25	-	-5° 6'	12.5	9.7	11,5	648*	.825	745/726*	.875	F1/I	24	72	30	40	2.21	2.63
JET EXECUTIVE AIRCRAFT	Potez-Air Fouga CM170 Magister	1952	7.42	.400	9°50'	-	0	64-219	64-212	15,5	700	-	740	.820	S1	25	59.5	15	40	-	-
	Aermacchi MB 326	1957	5.26	.600	8°22'	-	2°55'	63A-213.7	63A-212	12,9	806	-	-	.800	S1	25	61	-	-	-	1.70
	Cessna Model 500 Citation	1969	7.45	.390	1°10'	-	-	23014 mod.	23012	13	531*	.700	-	-	S1	25	56	20	-	1.55	1.72
	SNIAS 600 Corvette	1970	7.45	.447	20° 6'	-1°13'	3° 6'	13.65	11.50	12,6	657**	.77	695	.82	S2	31	73	15	40	-	2.40
	Cessna 318 (A/T37)	1954	6.20	.681	≈ 0	-2°38'	3	2418 mod.	2412 mod.	16	816**	.705**	843	-	S1	24	53	-	-	-	1.93
	Gates Lear Jet Model 25	1969	5.02	.510	13°	-	2°30'	64A 109	64A 109	9	617	.765	833	.850	S1	28	61	20	40	1.37	1.37
	Aerocommander Jet Commander	1963	6.19	.333	4°37'	-2	2	64_1-212	64_1-212	12	667	.800	833	.890	P	25	60	20	60	-	2.00
	Dassault Minifalcon 10	1970	7.11	.263	≈27°45'	-	-	-	-	-	-	.87	-	-	S2/I	31.5	67	15	55	2.04	2.35
	MBB 320 Hansa Jet	1964	6.00	.333	-15	-	6	65A-1.5-13	63A-1.8-11	12	825	-	700	.830	S2	22.5	67	-	-	1.61	1.91
	Piaggio Douglas PD 808	1964	6.25	.410	1°50'	-	3	DES0010-1.1-40/11° mod.	DES0008-1.1-40/9° mod.	9	852**		788	.850	S1	30	61	-	-	-	1.71
	North Am. Rockwell Sabre Liner	1958	5.77	.313	28°33'	-	-			-	617	.765	-	.850	S1/I			-	-	-	2.37
	Hawker Siddeley HS.125 srs 400B	1962	6.25	.300	20	-2°30'	2	14	11	12,5	528	.755	685	.825	S2	29	65	15	50	1.84	2.19
	Dassault Fan Jet Falcon 20.F	1963	6.40	.312	30	-	2	10.5	8	9,3	685**	.850	-	-	S1/II	25	61	10	40	-	2.32
	Lockheed Jetstar	1957	5.47	.335	30	-2	2	63A 112	63A 309	10,5	710**	.870	787	.90	S2/II	21	69.5	-	-	-	1.67

Table 7-1. Wing design data

*CAS or IAS

**maximum level flight speed

***V_{NE} is taken equal to V_{MO}

*P - plain or split flap

S - slotted flap; 1, 2 or 3: number of slots

F - Fowler flap

I - slat

II - droop nose, leading edge flap or slot

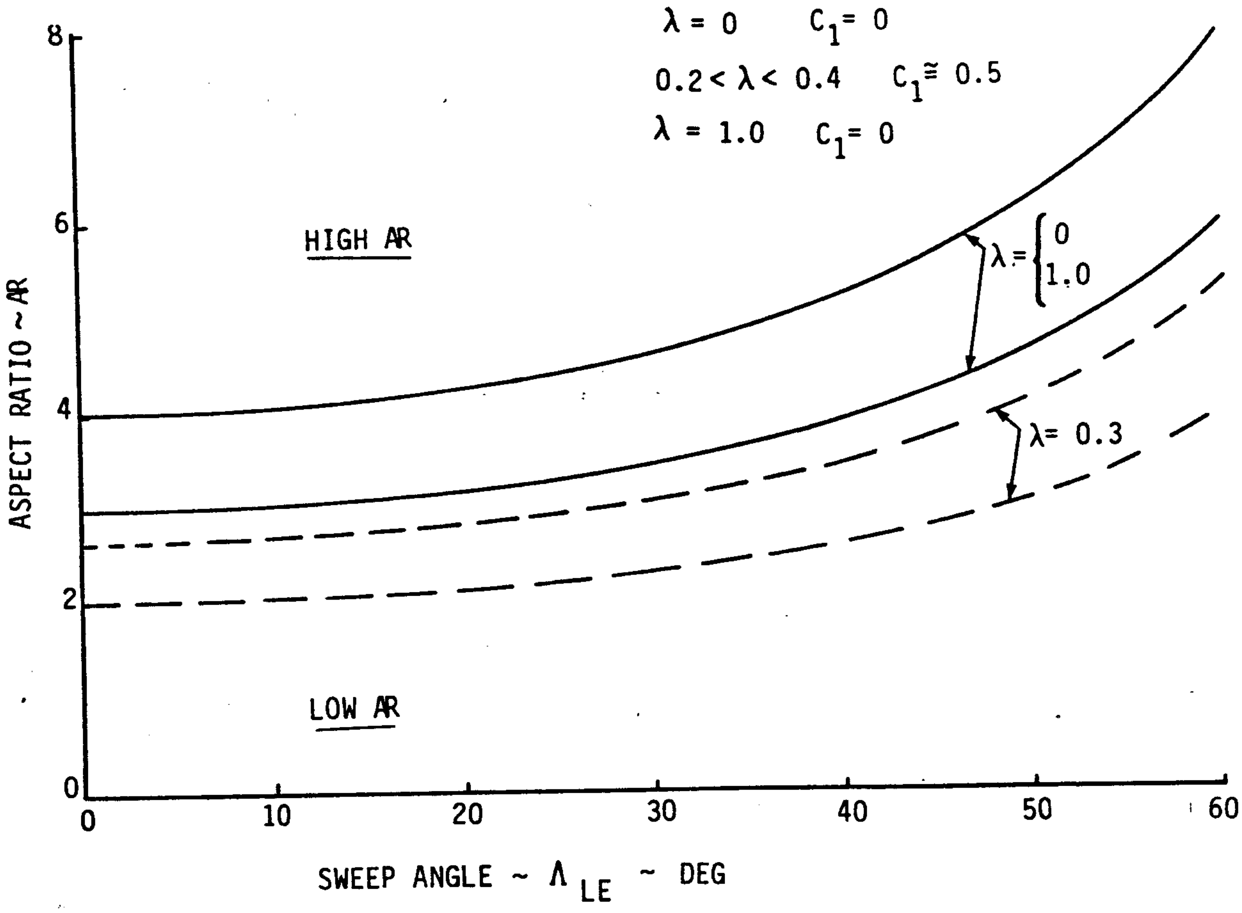
8.39

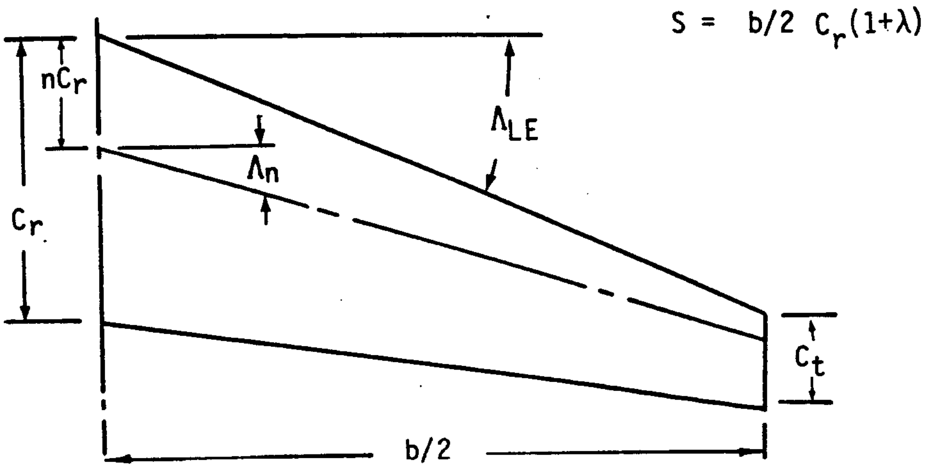
AIRCRAFT TYPE	1st flight prototype	Aspect Ratio A	Taper Ratio λ	Sweep angle $\Lambda_{.25}$ deg.	Geom. twist ϵ_g deg.	Dihedral Γ deg.	Profile type and streamwise thickness root %	tip %	$(\overline{t/c})$ %	V_{MO} (EAS) km/h	M_{MO}	V_D (EAS) km/h	M_D	flap type T.E./L.E.	$(\overline{c_f/c})$ stream-wise %	b_f/b %	Flapangle takeoff deg.	Flapangle landing deg.	$C_{L_{max}}$ flight test takeoff	$C_{L_{max}}$ flight test landing
TURBOPROP TRANSPORT AIRCRAFT																				
LET L-410 Turbolet	1969	9.30	.500	0	$-2^{\circ}30'$	$1^{\circ}45'$	63A 418	63A 412	15	385	-	518	-	S-2	29	66	-	45-70	-	2.45
Short Skyvan Srs 3	1963	11.30	1.0	0	0	$2^{\circ}\ 2'$	63A ars	63A ars	14	402	-	445	-	S-1	30	69	18/8	20/45	2.07	2.72
IAI Arava	1969	10.00	1.0	0	-	$1^{\circ}30'$	63 (215) A 417 (mod.)		17	350	-	466	-	S-2	-	79	-	-	-	2.45
Beriev Be 30	1967	9.04	.475	0	-	-	-	-		480	-	-	-	S-2	38.4	62.5	-	-	-	-
DHC 6 Twin Otter Srs 300	1965	10.05	1.0	0	0	3°	6A ars meanline;0016 (mod) thickness distribution		16	-	-	-	-	S-2	27	96.3	-	-	2.2	2.55
Handley Page Jetstream	1967	10.01	.333	$0^{\circ}34'$	-2°	7°	63A 418	63A 412	15	398	≈ .5	518	≈ .7	S-2	30	61	20	60	1.93	2.35
Swearingen Metro	1969	7.71	.400	$0^{\circ}54'$	-2°	5°	65_2A 215	64_2A 415	15	422	-	-	-	S-2	20	62	-	-	-	-
SNIAS (NORD) 262 Fregat	1962	8.72	.580	0	-	3°	23016 mod.	23012 mod.	14	386	-	496	-	S-1	24.8	66	15	25/35	1.55	2.21
DHC-7 STOL	1975	10.00	.400	$3^{\circ}13'$	-	$4^{\circ}30'$	63A 418	63A 415	-	444	-	-	-	S-2	36	80	25	25/60	2.70	3.15
HP Dart Herald srs 200	1958	10.20	.521	$0^{\circ}23'$	-	4°	23016.5	4412	14.3	443	-	494	-	F-1	32	66	5°	-	-	2.80
Fokker VFW F 27 Friendship	1955	12.00	.400	$1^{\circ}13'$	-2°	$2^{\circ}30'$	64_4-421 mod.	64_2-415 mod.	18	421	-	533	-	S-1	31.3	69	16.5	$26^{\circ}20'/40$	2.435	2.943
Hawker Siddeley 748 Srs 2A	1960	11.97	.386	$2^{\circ}54'$	-	7°	23018	4412	15	417	-	528	-	F-1	31.5	66.5	15	-	2.11	2.88
Antonov AN 24V srs II	1960	11.77	-	-	-	-	-	-	-	450	-	-	-	F-1	18	64.7	-	-	?	2.747
NAMC YS-11A	1962	10.81	.340	$3^{\circ}11'$	-	$6^{\circ}19'$	64A 218 mod.	64A 412 mod.	15	245	.475	546	.601	F-1	30	65	-	-	-	2.70
Breguet 941C STOL	1961	6.56	.530	0	-	4°	63A 416	63A 416	16	-	-	-	-	S-2	25	100	$45^{\circ}/30^{\circ}$	$98^{\circ}/65^{\circ}$	-	-
Vickers Vanguard	1959	9.10	.380	-	-	6°	63-X15	63-X13	14	645	-	717	-	F-1	31	64	-	40	-	2.46
Lockheed L188 Electra	1957	7.50	.40	-	-	6°	0014-1.10	0012-1.10	13	675	-	756	.711	F-1	32	65	-	40	-	2.54
Antonov AN 10	1957	12.03	-	-	-	-	-	-	-	-	-	-	-	S-2	24	65	-	-	-	-
Ilyushin IL-18	1957	10.00	.333	-	-	-	-	-	14	-	-	-	-	S-2	-	63	-	-	-	-
Lockheed L100 Hercules	1954	10.07	.513	-	-3°	$2^{\circ}30'$	64A 318	64A 412	15	550	.660	611	.710	F-1	30	70	18	36	2.17	2.28
Bristol Britannia	1951	9.53	.300	$7^{\circ}\ 3'$	$2^{\circ}24'$	3°	25017	4413	15	533	.650	593	.700	S-2	32	67	15	45	-	2.51
Tupolev Tu-114	1957	8.39	.350	35	-	-	-	-	-	-	-	-	-	F1	-	-	-	-	-	-
Antonov AN 22	1965	12.02	.360	$2^{\circ}51'$	-	$-2^{\circ}30'$	-	-	-	740	-	-	-	S-2	22	62.7	-	-	-	-
										V_{max}(TAS) km/h		V_D(TAS) km/h								
PISTON ENGINED GENERAL AVIATION AIRCRAFT																				
Beagle Pup B.121	1967	8.04	.550	$2^{\circ}56'$	-2°	$6^{\circ}30'$	63_2-615	63_2-615	15	259	-	314	-	S-1	21	59	10	40	1.60	1.87
Beagle B.206	1961	10.00	.400	≈0	$-1^{\circ}12'$	5°	23015	4412 mod.	13.5	372	-	483	-	S-2	28	50.5	20	-	-	2.10
Beechcraft Queen Air Model 65	1958	7.51	≈ .42	0	$-4^{\circ}48'$	7°	23018	23012	15	384	-	433	-	S-1	-	-	65%	100%	1.22	1.75
Beechcraft Musketeer	1961	7.50	1.0	0	-2°	$6^{\circ}30'$	63_2A 415	63_2A 415	15	235	-	-	-	S-1	-	-	-	-	-	1.80
Beechcraft Bonanza V35A-TC	1945	6.10	.457	0	-3°	6°	23016.5	23012 mod.	14.3	402	-	-	-	S-1	22	54.4	20	30	-	1.85
Beechcraft Baron D 55	1960	7.16	.410	0	-5°	6°	23017.5	23010.5	14	390	-	-	-	S-1	23	48.7	-	-	-	1.76
Bölkow Bo 208C Junior	1962	6.90	1.0	-3°	-	1°	23009 mod.	23009 mod.	9	230	-	283	-	P-1	22	65.5	-	-	-	-
Messerschmitt-Bölkow Bo 209 Monsun	1969	6.90	.650	$1^{\circ}24'$	-	$2^{\circ}30'$	64-215	64-212	13.5	274	-	320	-	-	23.5	69.5	-	-	-	1.99
Cessna Model 150	1957	7.00	.687	0	-1°	1°	2412	2412→ symm.	12	196	-	261	-	S-1	27.5	61.5	-	40	-	1.73
Cessna Model 172	1956	7.52	.672	0	-3°	$1^{\circ}44'$	2412	2412→ symm.	12	224	-	280	-	S-1	32.9	46.1	20	40	-	2.10
Cessna T 210 Centurion	1965	7.66	.726	0	-3	$1^{\circ}30'$	64_2A215(a=.5)	64_1A412 (a=.5)	13.5	322	-	-	-	S-1	-	65.0	-	40	-	1.87
Cessna Mod. 337 Super Skymaster	1961	7.18	.650	0	-2	3	2412	2409	10.5	322	-	-	-	S-1	-	70.5	20.	40	-	1.82
Cessna Model 401/402	1965	7.50	.679	0	$-2^{\circ}30'$	$0/5^{\circ}$	23018	23009	14	-	-	428	-	-	27.5	-	-	-	-	2.00
Dornier DO27	1955	7.'2	1.0	0	0	0	23018	23018	18	-	-	-	-	S-2/I	33	71	26	42	-	-
Dornier DO28 Skyservant	1959	8.50	1.0	0	0	$1^{\circ}30'$	23018	23018	18	-	-	-	-	S-2/I	33	71	-	-	-	2.36
Helio H-295 Super Courier STOL	1965	6.58	1.0	0	0	1°	23012	23012	12	269	-	304	-	S-1/I	27	74	-	-	-	6.42
Piper PA 23 Aztec D	1961	6.80	1.0	0	$-1^{\circ}12'$	5°	USA 35B mod.	USA 35B mod.	14	348	-	446	-	P-1	25	50	-	50	-	2.12
Piper PA 24-260 Comanche C	1956	7.28	.460	$-2^{\circ}30'$	-3°	5°	64_2A 215	64_2A 215	15	314	-	365	-	S-1	18	60	-	-	-	1.71
Piper PA 28-140C Cherokee	1960	5.63	1.0	0	-	7°	65_2-415	65_2-415	15	245	-	-	-	S-1	17	57	10-25	50	-	1.74
Piper PA 31 Navajo	1964	7.25	.372	$0^{\circ}30'$	$-2^{\circ}30'$	5°	63_2-415	63_1-212	13.5	365	-	-	-	S-1	-	65.6	-	-	-	2.15
SNIAS GY-80 Horizon	1962	7.20	.570		$-2^{\circ}30'$	7°	63-517	4413.6	15.3	-	-	-	-	F-1	-	70	-	-	-	-

Table 7-1 (continued)

8.40

$$\frac{3}{(1+C_1)\cos\Lambda_{LE}} < AR_{intermed} < \frac{4}{(1+C_1)\cos\Lambda_{LE}}$$

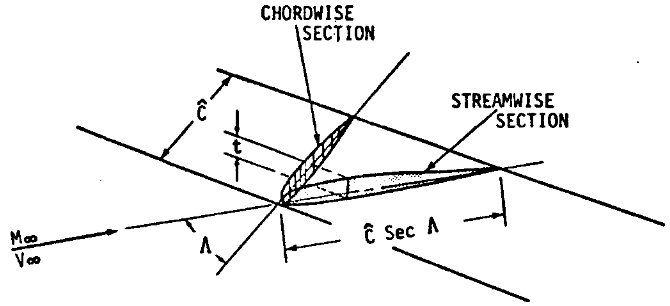




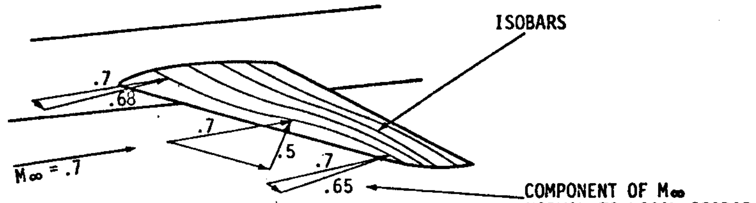
$$\tan\Lambda_n = \tan\Lambda_{LE} - \frac{4n}{AR}\left[\frac{1-\lambda}{1+\lambda}\right], \quad \lambda = \frac{C_t}{C_r}$$

FIGURE A-1 DEFINITION OF ASPECT RATIO RANGES

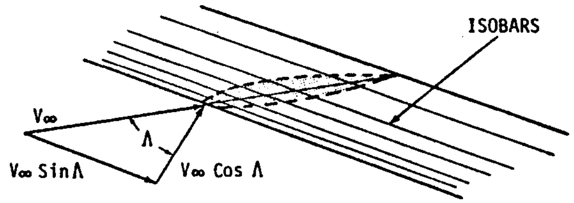
FIG. B-2 SIMPLE SWEEP THEORY



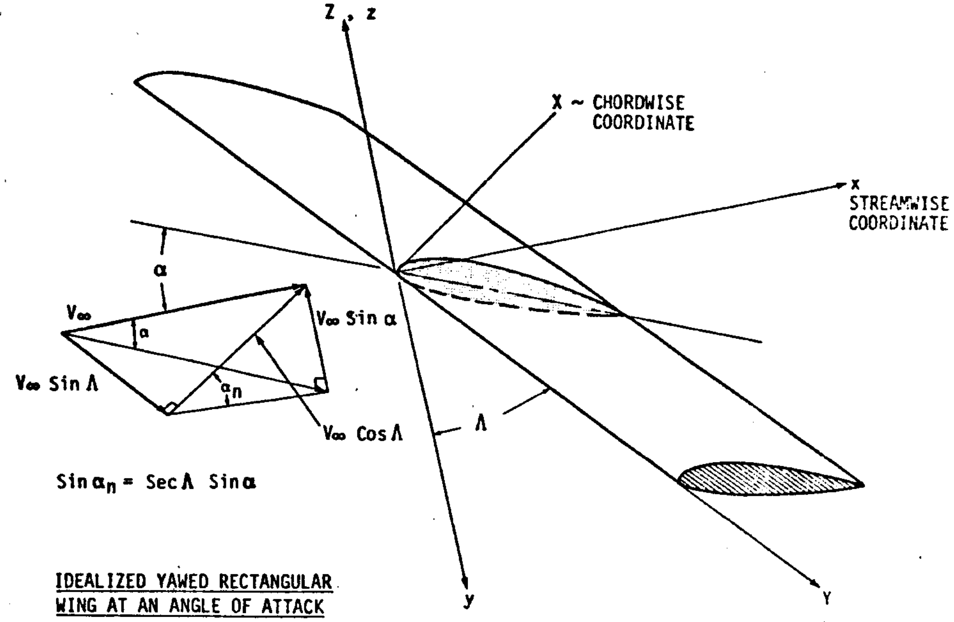
YAWED WING $\alpha = 0^{o}$



SWEEP EFFECT ON A REAL WING



IDEALIZED SWEPT WING $\alpha = 0^{o}$



$\sin\alpha_n = \sec\Lambda \ \sin\alpha$

IDEALIZED YAWED RECTANGULAR WING AT AN ANGLE OF ATTACK

$$C_p(X) = \frac{\Delta P}{q(X)} = C_{p_{2D}} \qquad C_p(x) = \frac{\Delta P}{q(x)} = C_{p_{3D}}$$

$q \equiv \frac{1}{2}\rho V^2$

$$V(X) = V(x)\cos\Lambda$$

$$q(X) = q(x)\cos^2\Lambda$$

FOR SMALL α

$$C_p(x) = C_p(X)\cos^2\Lambda$$

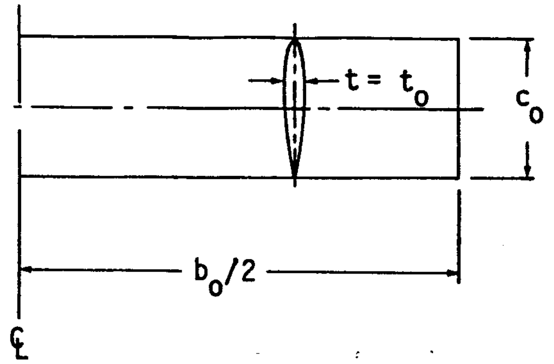
OR

$$\boxed{C_{p_{3D}} = C_{p_{2D}}\cos^2\Lambda}$$

SIMPLE SWEEP THEORY

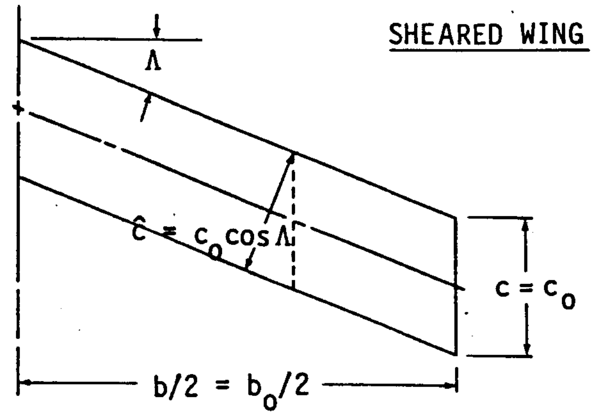
~~731~~ 8.42

BASIC WING



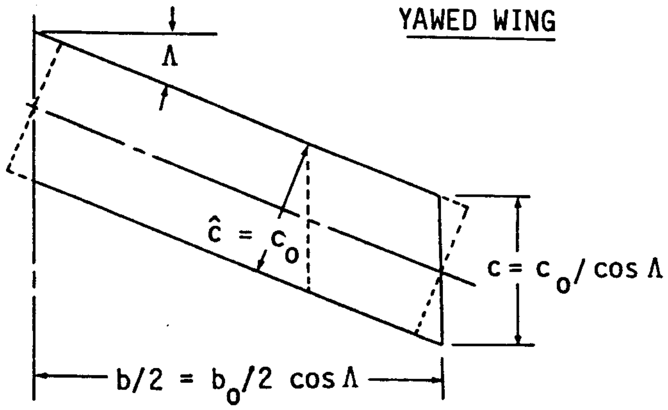
$$S_o = b_o c_o$$
$$\mathcal{R}_o = b_o/c_o = b_o^2/S_o$$
$$\delta = t/c = t_o/c_o = \delta_o$$

SHEARED WING



$$S = bc = S_o$$
$$\mathcal{R} = b/c = \mathcal{R}_o$$
$$\delta = t/c = \delta_o$$
$$\hat{\delta} = t/\hat{c} = \delta_o/\cos\Lambda$$

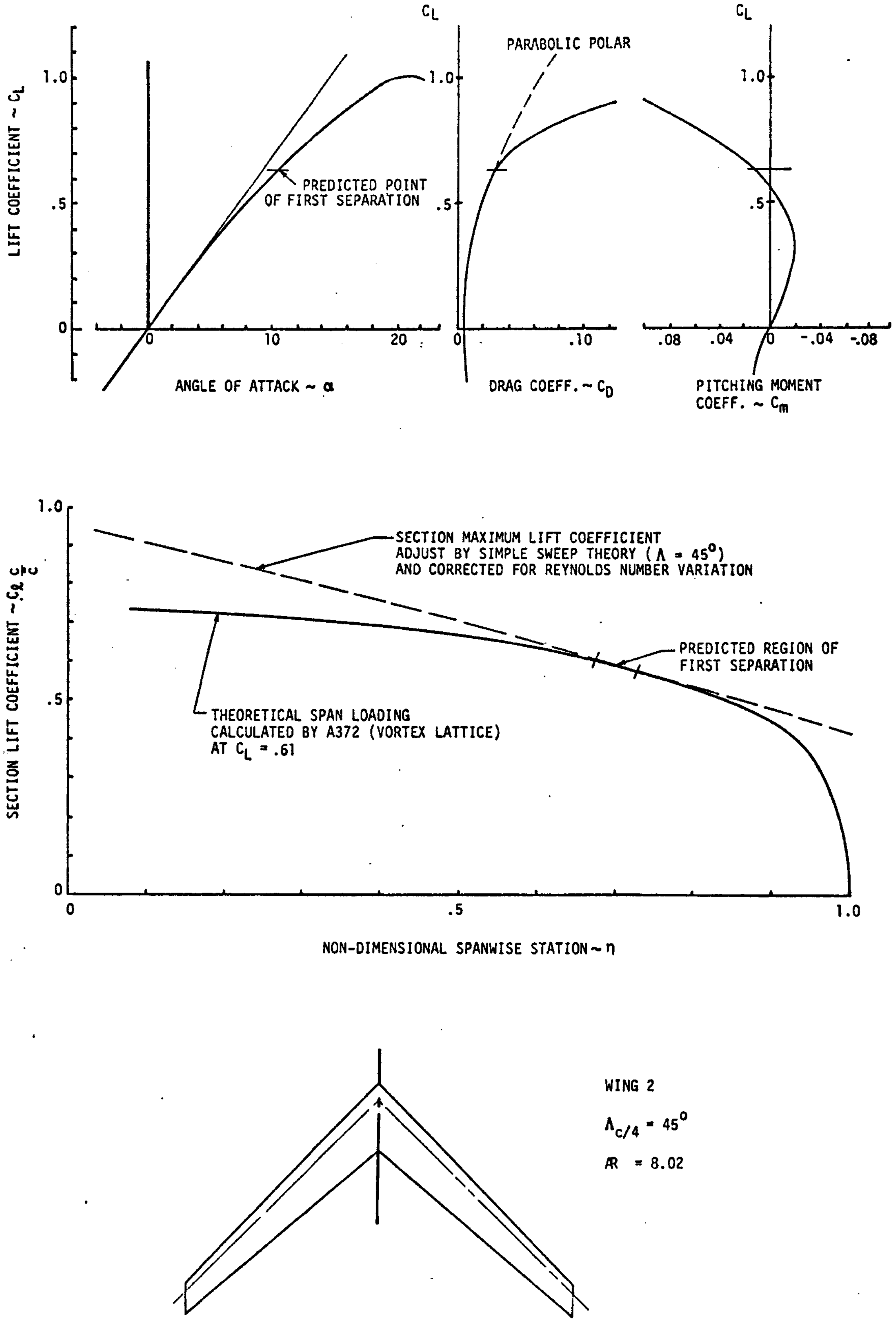
YAWED WING



$$S = 2(b/2\ \cos\Lambda)(c) = S_o$$
$$\mathcal{R} = b/c \quad \mathcal{R}_o \cos^2\Lambda$$
$$\delta = t/c = \delta_o/\cos\Lambda$$
$$\hat{\delta} = t/\hat{c} = \delta_o$$

FIGURE B-1 EQUIVALENT SWEPT AND STRAIGHT WINGS

FIG. 37 CRITICAL SECTION ANALYSIS ON A PLAIN SWEPT WING



8.44
~~7.33~~

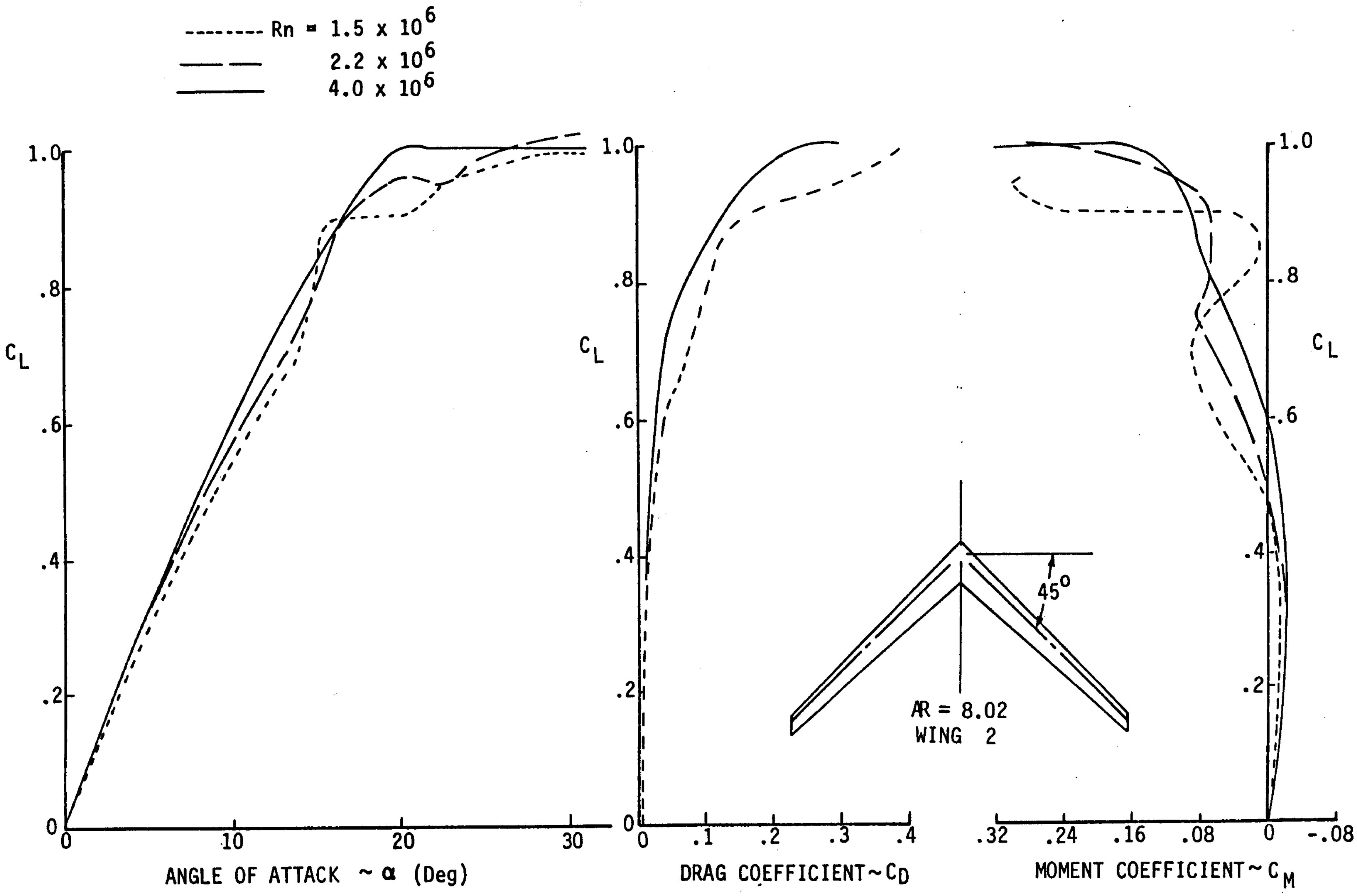


FIGURE 12 EXPERIMENTAL AERODYNAMIC CHARACTERISTICS FOR WING 2

8.45
~~7.34~~

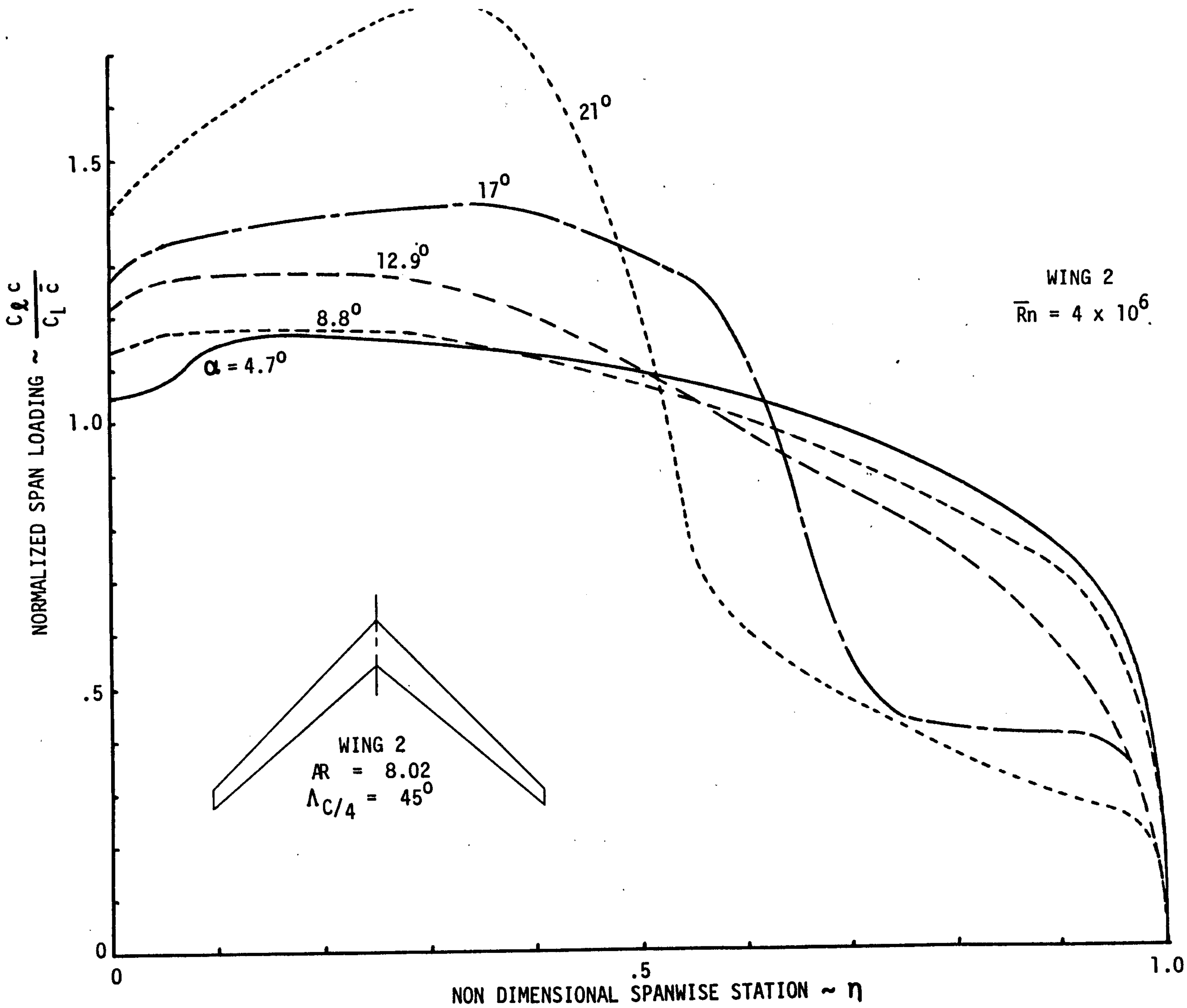


FIGURE 16 EXPERIMENTAL SPAN LOADINGS FOR WING 2. (TABLE 2)

8.46
7.35

WING 2 $R_n = 4 \times 10^6$

NACA 63_1-012 SECTION (STREAMWISE)

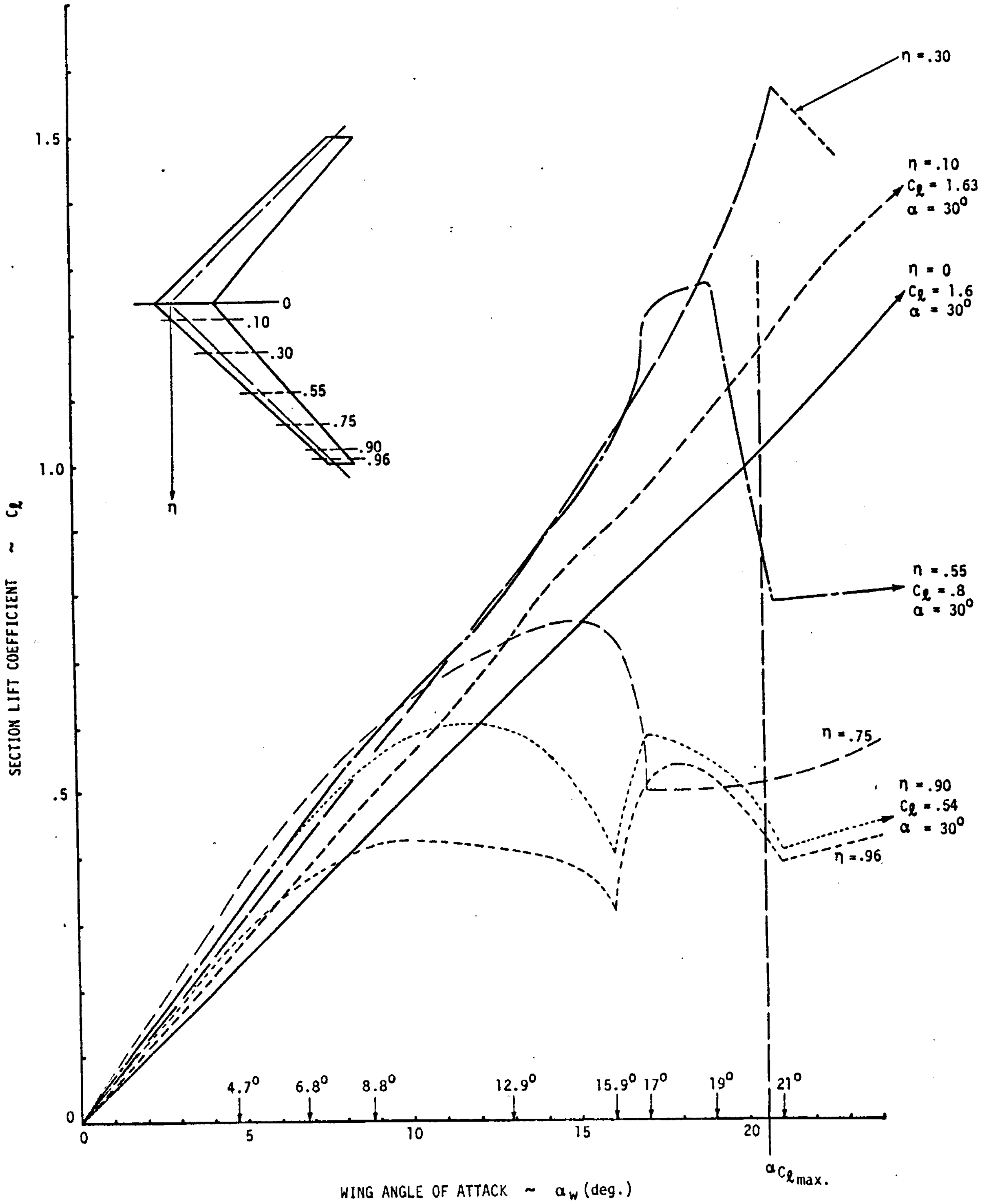


FIG. 18 SECTION LIFT CURVES FROM WING 2

8.47
~~7.36~~

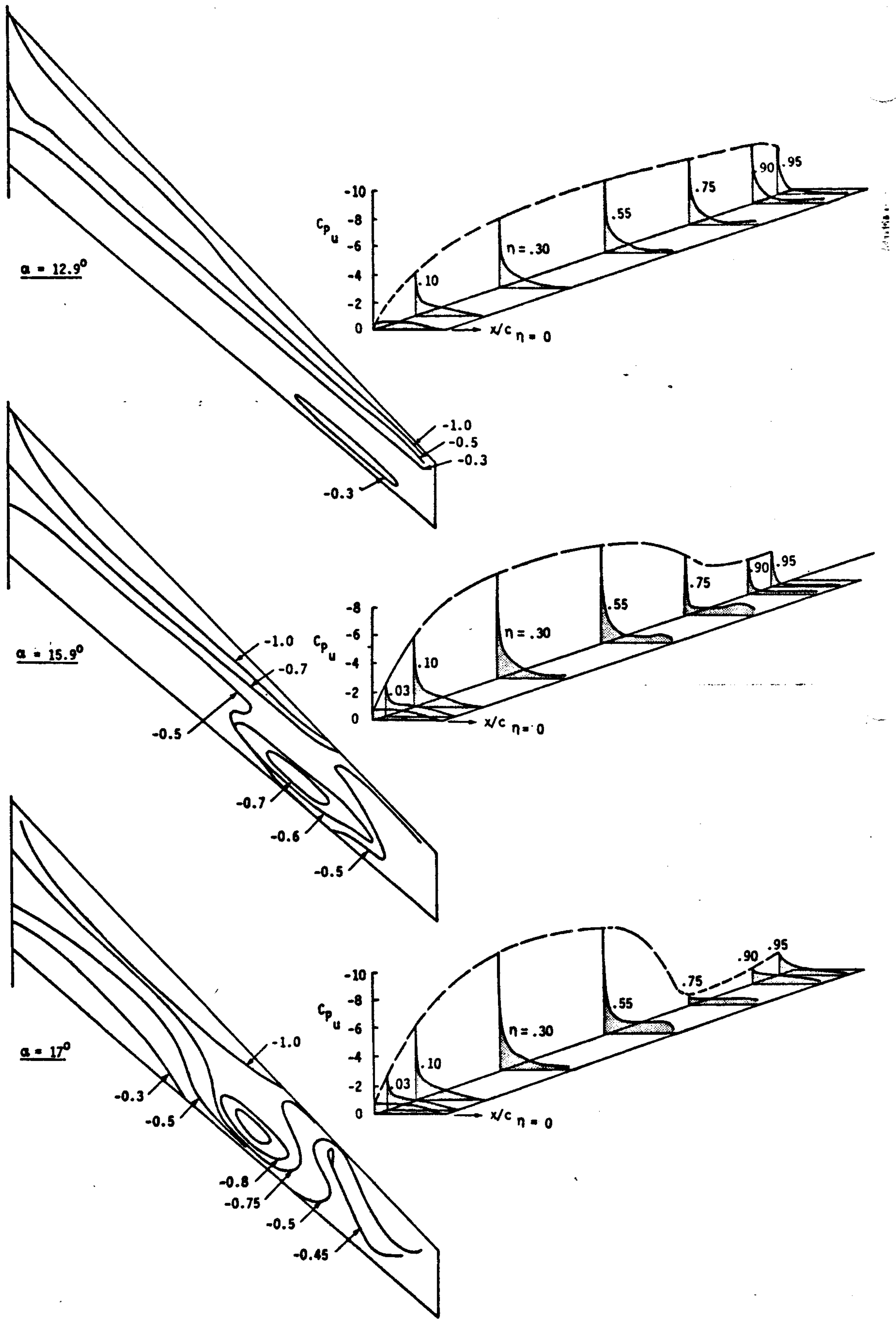


FIGURE 17 EXPERIMENTAL PRESSURE DISTRIBUTIONS
ON WING 2 (TABLE 2) AT HIGH ANGLES OF ATTACK

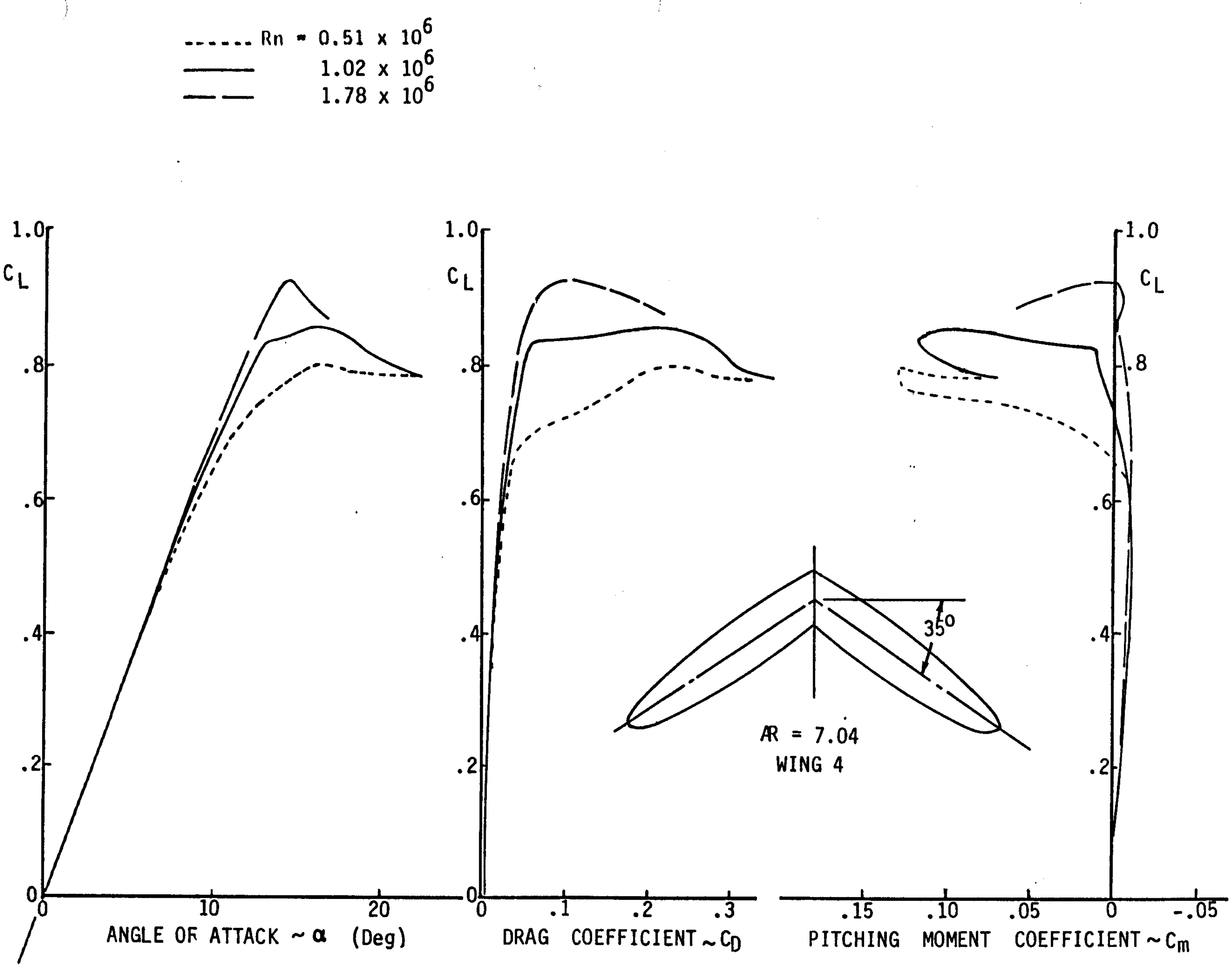


FIGURE 13 EXPERIMENTAL CHARACTERISTICS FOR WING 4.

8.49
7.38

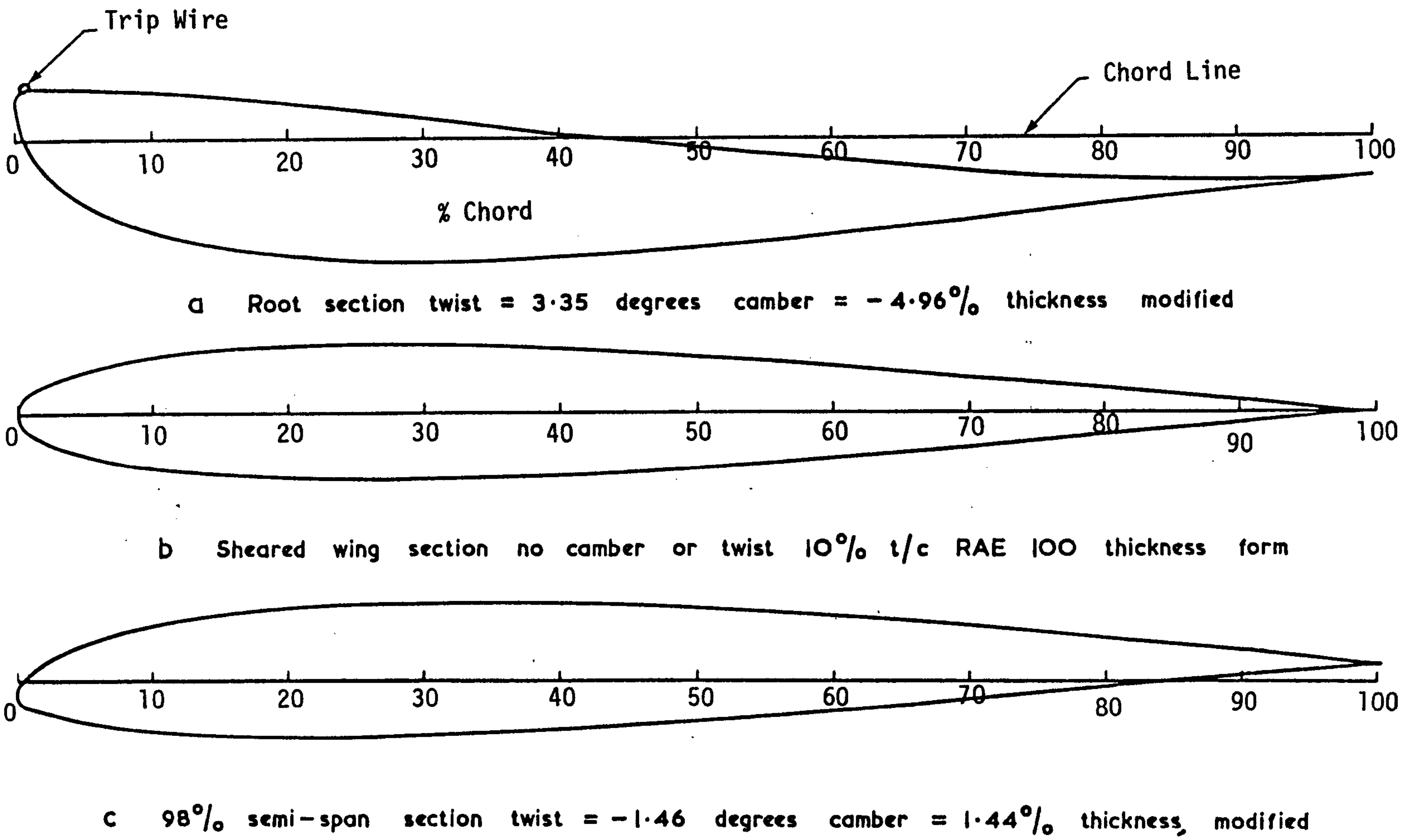
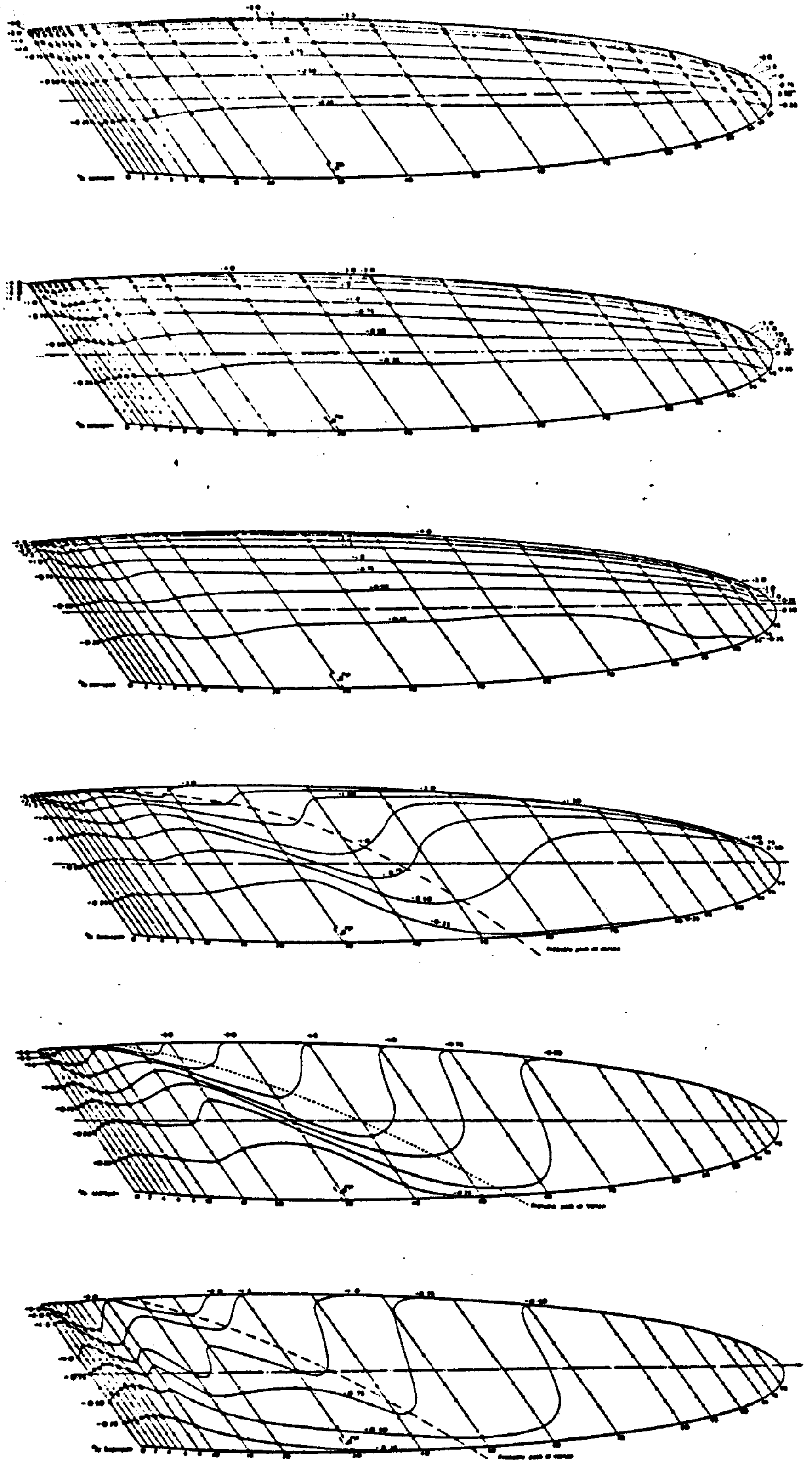


FIGURE 19 WING SECTIONS AT ROOT, MID-SEMI SPAN, AND NEAR THE TIP OF WING 4



$C_L = 0.68$
$\alpha = 9.77^\circ$

$C_L = .076$
$\alpha = 11.33^\circ$

$C_L = 0.81$
$\alpha = 12.30^\circ$

$C_L = 0.82$
$\alpha = 13.56^\circ$

$C_L = 0.82$
$\alpha = 15^\circ$

$C_L = 0.85$
$\alpha = 17^\circ$

FIGURE 20. PRESSURE DISTRIBUTIONS ON WING 4 AT A REYNOLDS NUMBER OF 1.02×10^6

Note: Wire ON

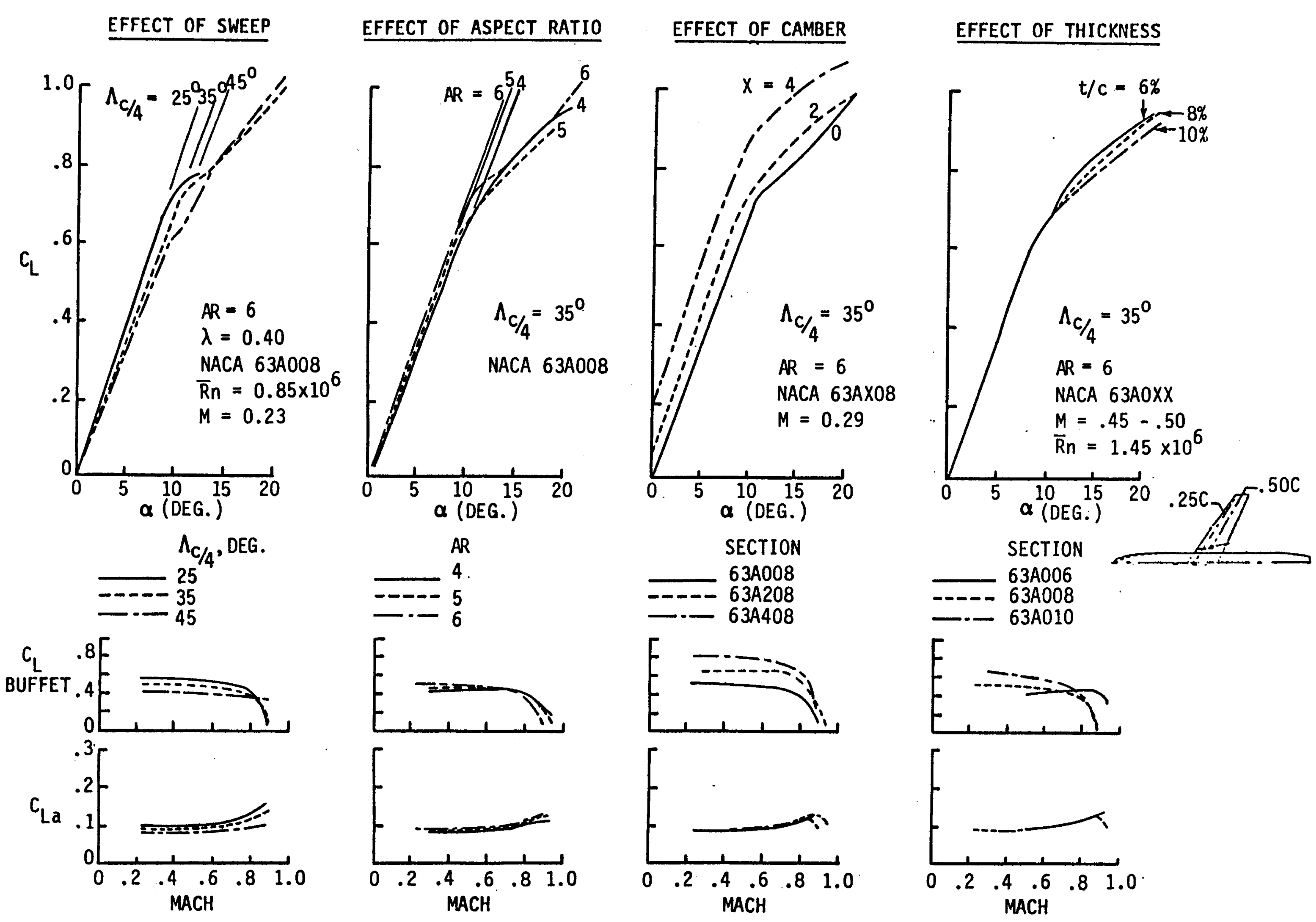
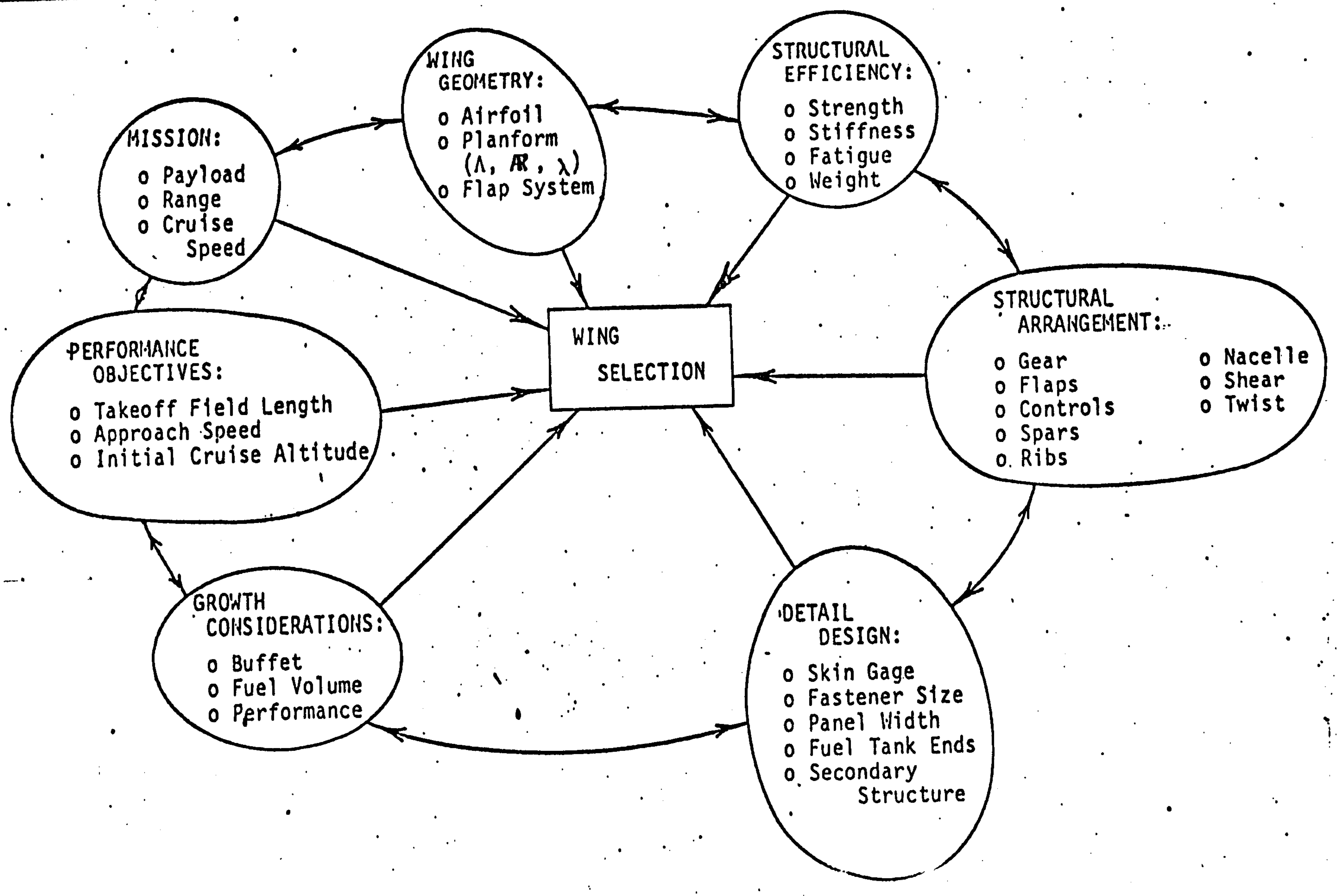


FIGURE 11 MACH NUMBER EFFECTS ON THE AERODYNAMIC CHARACTERISTICS OF A FAMILY OF SWEPT WINGS

8.52
~~7.41~~

TYPICAL WING DESIGN ELEMENTS

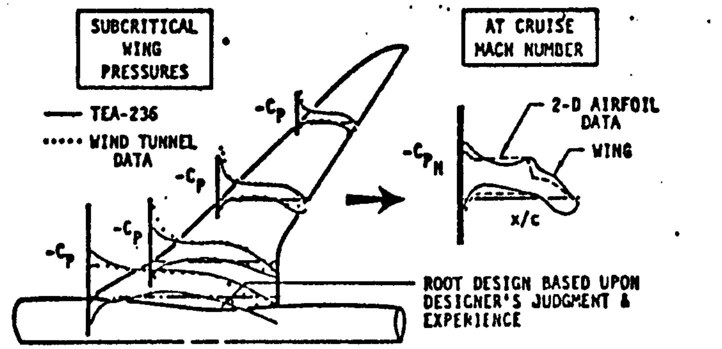


WING DESIGN PROCEDURE CONCEPTS

1. CURRENT DESIGN PRACTICE

- WING PLANFORM AND SWEEP SELECTED.
- SELECT OUTBOARD AIRFOIL FROM 2-D TESTS AND t/c FROM TRADE DATA.
- ESTABLISH DESIRED SUBCRITICAL, UPPER SURFACE ISOBAR PATTERN; AND "1-G" SPAN LOADING.
- CALCULATE USING TEA-236: ROOT AND TIP AIRFOIL SECTIONS; TWIST AND CAMBER DISTRIBUTIONS. LOFT BASED ON STRAIGHT TRAILING EDGE.
- TEST IN WIND TUNNEL, TAILOR, RECYCLE.

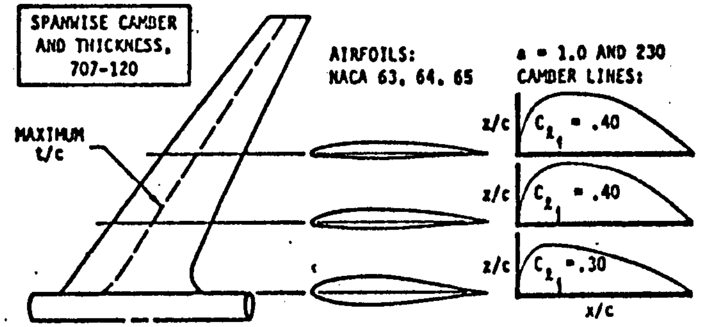
ATP
7X7 MR
BOEING/AERITALIA
7X7 LR
747 PR. DEV.
DED. AIR FREIGHTER



2. RECENT DESIGN PROCEDURE

- SELECT NACA AIRFOILS AND CAMBER FORMS VERSUS SPAN. $C_{\ell_i} \approx C_L$ CRUISE.
- SELECT OUTBOARD t/c AND Λ FOR CRUISE M_{DD}.
- TWIST DESIGNED FOR LOWEST CRUISE DRAG AND ROOT BENDING MOMENTS OR STALL PATTERN.
- WIND TUNNEL FORCE TESTS (AND WAKE SURVEYS VERSUS SPAN).
- CHANGE ONE DESIGN FEATURE AND RE-TEST. TAILOR.

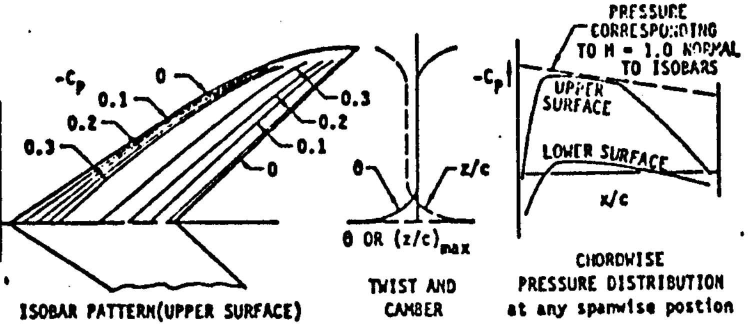
B47
B52
707
727
737
747



3. R.A.E. DESIGN PROCEDURE

- SELECT OUTBOARD AIRFOIL DATA FROM 2-D TESTS (USING PEARCEY DESIGNS AND CONCEPTS).
- ESTABLISH DESIRED SUBCRITICAL UPPER SURFACE ISOBAR PATTERN AND SPAN LOADING.
- CALCULATE CAMBER AND TWIST FOR WING ALONE CASE (USING KÜCHEMANN'S LIFTING SURFACE THEORY).
- TEST IN WIND TUNNEL, TAILOR AND RECYCLE.

VC-10
BAC-111
TRIDENT



SYM

8.54
7.42

IX. Preliminary Design III - Weight & Structures (3 hours)

A. Square-Cube Law

B. Weight Eqns.

C. Structural Concepts

D. Material Properties

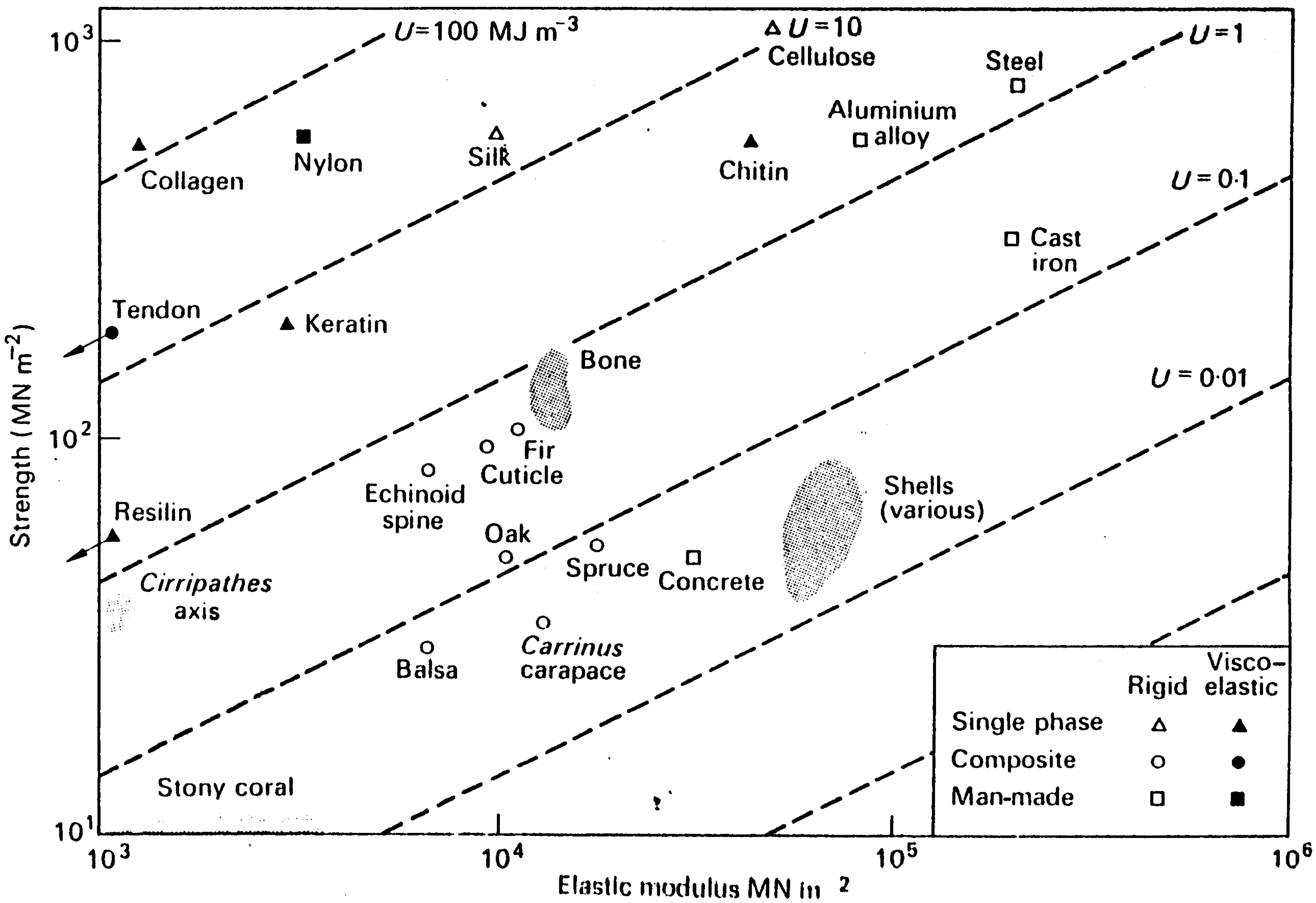


Fig. 2.4 Energy absorption U of some materials.

9.02

20.2.2 Conventional Metal Aircraft - Light Utility Aircraft:

The following equations, developed by Mr. Robert Anderson of the Air Force Flight Dynamics Laboratory are recommended for the low to moderate performance (up to about 300 knots) light utility aircraft. The weight equations give the component weight in pounds.

1.0 Structure:

1.1 Wing:

$$Wt = 96.948\left[\left(\frac{W_{TO}N}{10^5}\right)^{.65}\left(\frac{A}{\cos\Lambda_{1/4}}\right)^{.57}\left(\frac{S_W}{100}\right)^{.61}\left(\frac{1+\lambda}{2t/c}\right)^{.36}\left(1+\frac{V_e}{500}\right)^{.5}\right]^{.993} \qquad (20\text{-}69)$$

where W_{TO} = take-off weight in lbs

N = ultimate load factor (1.5 times limit load factor)

A = aspect ratio

$\Lambda_{1/4}$ = wing quarter chord sweep

S_W = wing area in ft^2

λ = taper ratio

t/c = maximum thickness ratio

V_e = equivalent max airspeed at SL in knots

1.2 Fuselage:

$$Wt = 200\left[\left(\frac{W_{TO}N}{10^5}\right)^{.286}\left(\frac{L}{10}\right)^{.857}\left(\frac{W+D}{10}\right)\left(\frac{Ve}{100}\right)^{.338}\right]^{1.1} \qquad (20\text{-}70)$$

where L = fuselage length in feet

W = fuselage max width in feet

D = fuselage max depth in feet

1.3 Horizontal Tail:

$$Wt = 127\left[\left(\frac{W_{TO}N}{10^5}\right)^{.87}\left(\frac{S_H}{100}\right)^{1.2}\left(\frac{\ell_T}{10}\right)^{.483}\left(\frac{b_H}{t_{HR}}\right)^{.5}\right]^{.458} \qquad (20\text{-}71)$$

where S_H = horizontal tail area in ft^2

ℓ_T = distance from wing 1/4 MAC to tail 1/4 MAC

b_H = horizontal tail span in feet

t_{HR} = horizontal tail max root thickness in inches

1.4 Vertical Tail:

$$Wt = 98.5\left[\left(\frac{W_{TO}N}{10^5}\right)^{.87}\left(\frac{S_V}{100}\right)^{1.2}\left(\frac{b_V}{t_{VR}}\right)^{.5}\right] \qquad (20\text{-}72)$$

where S_V = vertical tail area in ft^2

b_V = vertical tail span in ft

t_{VR} = vertical tail max root thickness in inches

1.5 Landing Gear:

$$Wt = .054\,(L_{LG})^{.501}\,(W_{Land}\,N_{Land})^{.684} \qquad (20\text{-}73)$$

where L_{LG} = length of main landing gear strut in inches

W_{Land} = landing weight (if unknown, use W_{TO} - 60 percent fuel)

N_{Land} = ultimate load factor at W_{Land}

2.0 Propulsion:

2.1 Total Installed Propulsion Unit Weight Less Fuel System:

This includes mounting and air induction weight.

$$Wt = 2.575\,(W_{ENG})^{.922}\,(N_E) \qquad (20\text{-}74)$$

where W_{ENG} = bare engine weight

N_E = number of engines

2.2 Fuel System:

This includes fuel pumps, lines, and tanks.

$$W_{FS} = 2.49\left[(F_G)^{.6}\left(\frac{1}{1 + Int}\right)^{.3}(N_T)^{.2}(N_E)^{.13}\right]^{1.21} \qquad (20\text{-}75)$$

where F_G = total fuel in gallons

Int = percent of fuel tanks that are integral

N_T = number of separate fuel tanks

3.0 Surface Controls:

For powered surface control systems, use

$$Wt = 1.08\,(W_{TO})^{.7} \qquad (20\text{-}76)$$

For unpowered surface control systems, use

9.04

5.0 Electrical System:

The weight prediction relationships are expressed in terms of the total weight of the fuel system and the electronics system - the primary users of electrical power on the aircraft.

$$Wt = 426 \left(\frac{W_{FS} + W_{TRON}}{1000} \right)^{.51} \tag{20-78}$$

where W_{FS} = fuel system weight in lfs, equation 20-75

W_{TRON} = weight of installed electronics in lbs, equation 20-81

6.0 Furnishings:

The weight expression for the crew seats is

$$Wt = 34.5\,(N_{CR})\,(q)^{.25} \tag{20-79}$$

where N_{CR} = number of crew

q = maximum dynamic pressure, psf

7.0 Air Conditioning and Anti-Icing:

If the aircraft has air conditioning and anti-icing, the following expression can be used to estimate the weight of this equipment:

$$Wt = .265\,(W_{TO})^{.52}\,(N_{CR} + N_{PASS})^{.68}\,(W_{TRON})^{.17}\,(M_E)^{.08} \tag{20-80}$$

where N_{PASS} = number of passengers

N_{CR} = number of crew

W_{TRON} = weight of installed electronics in lbs, see equation 20-81

M_E = equivalent max Mach number at sea level

8.0 Electronics (Avionics):

The total installed weight of the avionics equipment is

$$W_{TRON} = 2.117\,(W_{AU})^{.933} \tag{20-81}$$

where W_{AU} = bare avionics equipment weight (uninstalled)

9.05

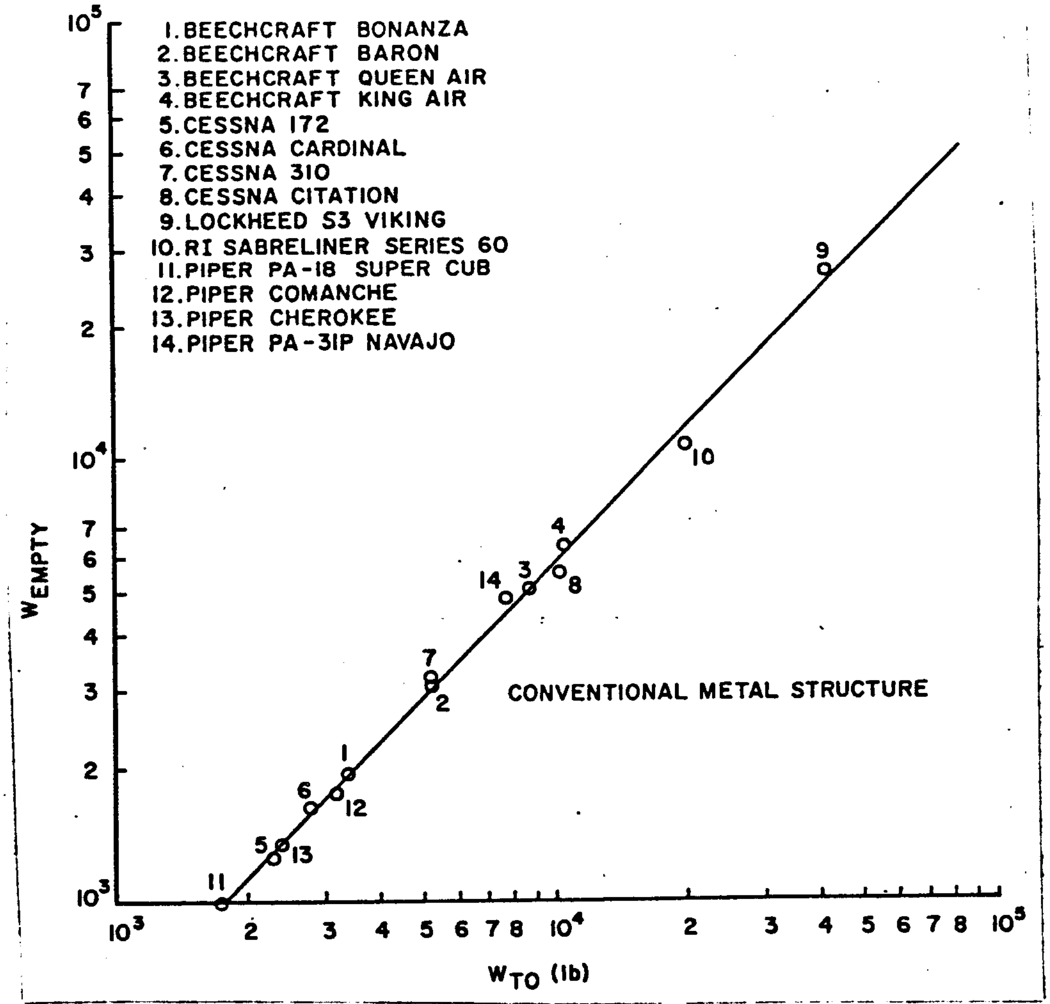


Fig. 5.3 Trend of Empty Weight vs. W_{TO} for Light Civil Aircraft

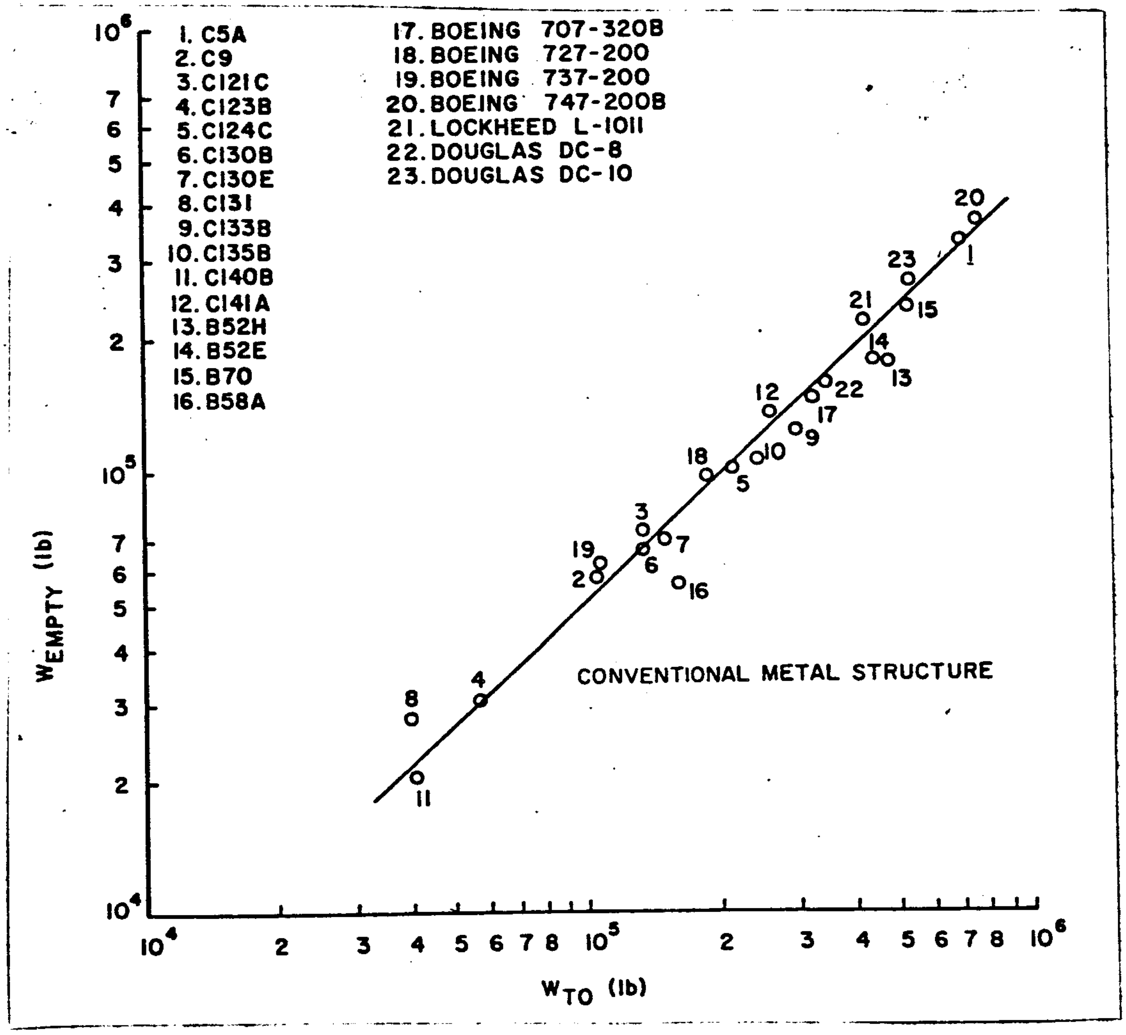


Fig. 5.2 Trend of Empty Weight vs. W_{TO} for Bomber and Transport Aircraft

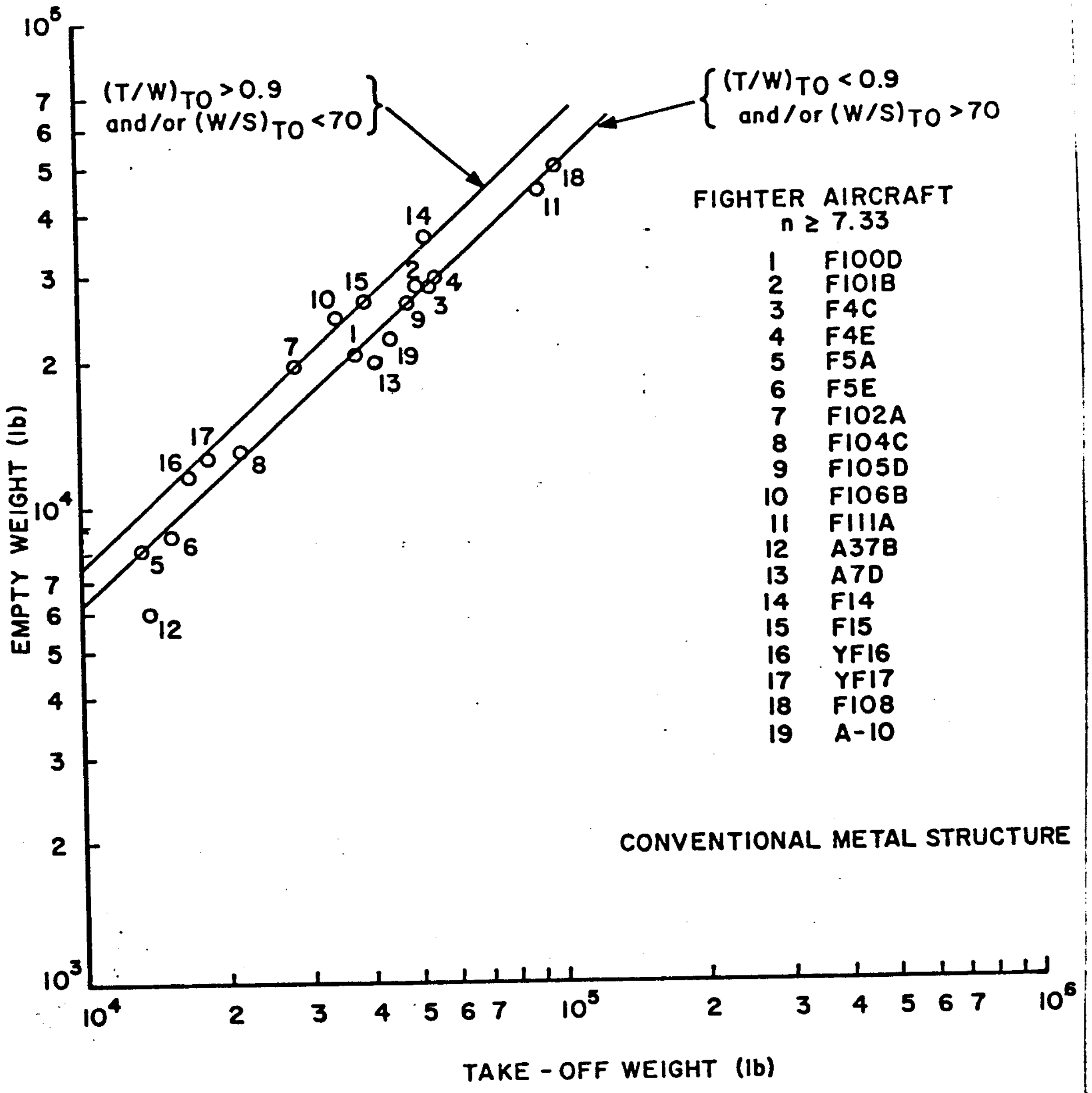
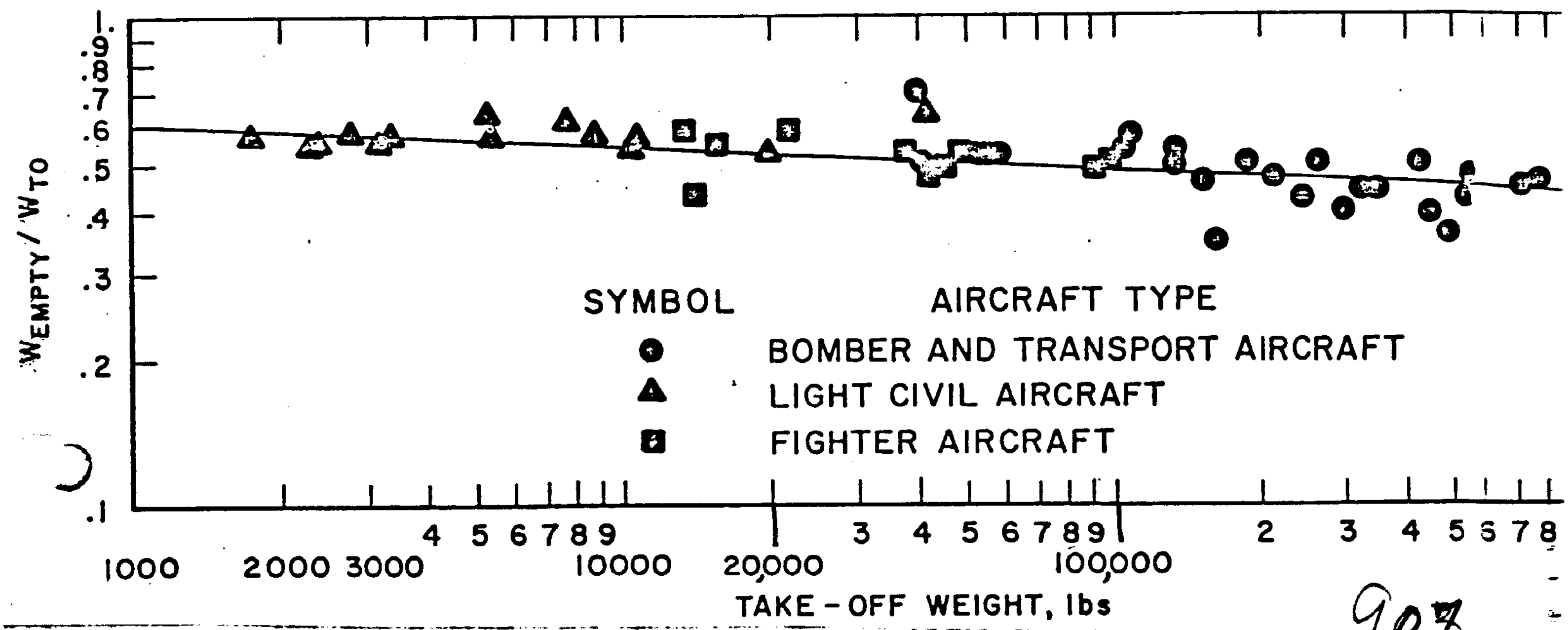


Fig. 5.1 Trend of Empty Weight vs. W_{TO} for Fighters



9.03

Breakdown of All-up weight

Item	Sailpane	Light and Executive Aircraft	Subsonic Transport	Supersonic Transport	Supersonic Fighter*	Supersonic Bomber*
	Approximate Percentage All-up weight					
Powerplant	—	23	18	15	20	13
Fuel } Disposable load	—	(10–15)	(22–31)	(45–50)	(25–30)	(50–55)†
		35	40	55	35	58†
Payload }	36	(20–25)	(9–18)	(5–10)	(5–10)	(3–8)
Structure	60	30	28	21	32	20 (17 low level)
Equipment and Services	4	12	14	9	13	9
Total %	100	100	100	100	100	100

* Unconfirmed estimates.
† Add 3 per cent of AUW for low-level aircraft.

TABLE 12.2

Breakdown of Structure Weight

Structural Component	Sailplane	Light and Executive Aircraft	Subsonic Transport	Supersonic Transport	Supersonic Fighter*	Supersonic Bomber*
	Approximate Percentage All-up weight					
Wing	30	13·5	11	8	12	7 (4 low level)
Fuselage	25	10·5	9	7	12	7
Stabilizers	3	2	3	2	4	2
Undercarriage	2 (+ballast)	4	5	4	4	4
Total %	60	30	28	21	32	20 (17 low level)

* Unconfirmed estimates: for variable sweep add 20 per cent of wing weight, *i.e.*, 2 to 3 per cent of AUW and a further 1 to 2 per cent AUW for additional services, in Table 12.1. Fuel weight may be correspondingly reduced.

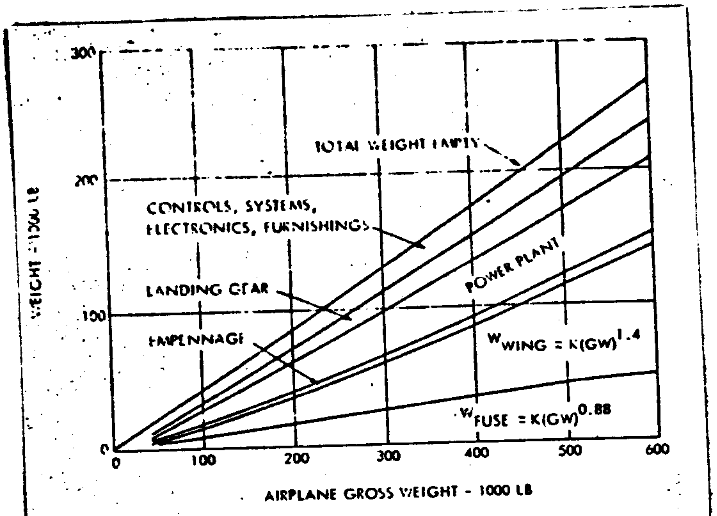


Fig. 11 Component weight variation with gross weight.

9,0

AIRPLANE CATEGORY AND TYPE		MTOW	WING GROUP		TAIL GROUP		FUSELAGE GROUP		LANDING GEAR		SURFACE CONTROLS		NACELLE GROUP	
		10^3 lb	10^3 lb	%	10^3 lb	%	10^3 lb	%	10^3 lb	%	10^3 lb	%	10^3 lb	%
LIGHT SINGLE RECIPROCATING	Cessna - 150A	1.50	0.213	14.2	0.041	2.73	0.166	11.1	0.106	7.07	0.031	2.07	0.024	1.60
	- 172B	2.20	0.236	10.7	0.061	2.77	0.253	11.5	0.122	5.55	0.031	1.41	0.031	1.41
	- 180D	2.65	0.254	9.58	0.059	2.23	0.270	10.2	0.119	4.49	0.036	1.36	0.037	1.40
	- 182D	2.65	0.254	9.58	0.061	2.30	0.273	10.3	0.136	5.13	0.036	1.36	0.036	1.36
	- 185	3.20	0.266	8.31	0.071	2.22	0.290	9.06	0.132	4.13	0.036	1.13	0.041	1.28
	- 210	2.90	0.261	9.0	0.071	2.45	0.316	10.90	0.207	7.14	0.044	1.52	0.031	1.07
	Beechcraft J-35	2.90	0.379	13.1	0.058	2.00	0.200	6.90	0.205	7.07	0.056	1.93	0.062	2.14
	Saab Safir	2.66	0.276	10.4	0.060	2.26	0.386	14.5	0.119	4.47	**	-	**	-
LIGHT TWINS RECIPROCATING	Cessna C-310	4.83	0.454	9.40	0.118	2.44	0.319	6.60	0.263	5.45	0.066	1.37	0.129	2.67
	Beechcraft G-50	7.15	0.656	9.17	0.156	2.18	0.495	6.92	0.447	6.25	0.120	1.68	0.261	3.65
	-65	7.37	0.670	9.09	0.153	2.08	0.601	8.15	0.444	6.02	0.132	1.79	0.285	3.87
	-95	4.00	0.458	11.5	0.079	1.98	0.276	6.90	0.218	5.45	0.073	1.83	0.180	4.50
	L-18S	8.75	0.858	9.81	0.177	2.02	0.733	8.38	0.560	6.40	0.115*	1.31	0.311	3.55
	E-18S	9.70	0.874	9.01	0.180	1.86	0.768	7.92	0.585	6.03	0.115	1.19	0.331	3.41
	De Havilland Dove	8.80	0.930	10.6	0.196	2.23	0.745	8.47	0.391	4.44	**	-	0.220*	2.50
JET TRAINERS	Cessna T-37	6.44	0.531	8.24	0.128	1.99	0.839	13.0	0.330	5.12	0.154	2.39	-	-
	Fouga Magister	6.28	1.089	17.3	0.165	2.63	0.743	11.8	0.459	7.31	0.260	4.14	-	-
	Canadair CL-41	6.50	0.892	13.7	0.201	3.09	0.955	14.7	0.318	4.89	0.172	2.65	0.040	0.62
JET EXECUTIVES	H. Siddeley - 125	21.200	1.968	9.28	0.608	2.87	1.628	7.68	0.659	3.11	0.217	1.02	**	-
	Jet Commander 1121	16.000	1.322	8.26	0.425	2.66	1.622	10.1	0.443	2.76	0.223	1.39	0.35	2.19
	N.Am.Sabreliner	16.700	1.753	10.5	0.297	1.78	2.014	12.1	0.728	4.36	0.344	2.06	0.315	1.89
	Lockheed Jetstar	30.680	2.827	9.21	0.879	2.87	3.491	11.4	1.061	3.46	0.768	2.50	0.792	2.58

* estimated ** included in other items

Table 8-5. Weight breakdown of the structure group weight

PROPELLER TRANSPORTS

AIRPLANE CATEGORY AND TYPE			MTOW	WING GROUP		TAIL GROUP		FUSELAGE GROUP		LANDING GEAR		SURFACE CONTROLS		NACELLE GROUP	
			10^3 lb	10^3 lb	%	10^3 lb	%	10^3 lb	%	10^3 lb	%	10^3 lb	%	10^3 lb	%
RECIPROCATING	2 ENGINES	De Havilland DHC-4	24.000	2.925	12.2	0.790	3.29	2.849	11.9	1.23	5.13	0.326	1.36	0.781	3.25
		Saab Scandia	35.273	4.195	11.9	0.584	1.66	2.773	7.86	1.841	5.22	0.369	1.05	1.479	4.19
		H. Page Herald	37.500	4.365	11.6	0.987	2.63	2.986	7.96	1.625	4.33	0.364	0.97	0.830	2.21
		S.A. Twin Pioneer	14.600	2.121	14.5	0.576	3.95	1.381	9.46	0.703	4.82	0.300	2.05	0.230	1.58
		Canadair CL-21	32.500	3.99	12.3	1.055	3.25	3.260	10.0	1.609	4.95	0.371	1.14	1.29	3.97
	4 ENGINES	Douglas DC-6B	81.500	7.506	9.21	1.406	1.73	5.471	6.71	4.165	5.11	1.052	1.29	2.871	3.52
		DC-7C	143.000	11.100	7.76	1.900	1.33	8.450	5.91	5.130	3.59	1.215	0.85	4.130	2.89
		Lockheed L-749	102.072	11.102	10.9	2.059	2.02	7.407	7.26	4.782	4.68	1.488	1.46	3.869	3.79
		L-1049	137.500	11.542	8.39	2.604	1.89	12.839	9.34	5.422	3.94	1.685	1.23	4.420	3.21
TURBOPROPELLER	2 ENGINES	Nord 262	23.050	2.698	11.7	0.805	3.49	3.675	15.9	1.085	4.71	0.408	1.77	0.236	1.02
		Fokker F-27/100	39.000	4.408	11.3	0.977	2.51	4.122	10.6	1.940	4.97	0.613	1.57	0.628	1.61
		F-27/200	43.500	4.505	10.4	1.501	2.42	4.303	2.89	1.825	4.20	0.620	1.43	0.667	1.53
		F-27/500	45.000	4.510	10.0	1.060	2.35	5.142	11.4	1.865	4.14	0.626	1.39	0.668	1.48
		Grumman Gulfstream	33.600	3.735	11.2	0.874	2.60	3.718	11.1	1.207	3.59	0.461	1.37	1.136	3.38
		Short Skyvan	12.500	1.220	9.76	0.374	2.99	2.154	17.2	0.466	3.73	0.265	2.12	0.254	2.03
	4 ENGINES	Bréguet 941	58.421	4.096	7.01	1.387	2.37	6.481	11.1	2.626	4.94	1.056	1.81	***	-
		H.S. Argosy	82.000	10.800	13.2	1.300	1.59	11.100*	13.5	3.180	3.88	**	-	1.200	1.46
		Vickers Viscount 810	69.000	6.25	9.06	1.245	1.80	6.900	10.0	2.469	3.58	0.824	1.19	1.810	2.62
		Bristol Brit. 300	155.000	13.433	8.67	3.202	2.07	11.100	7.16	5.785	3.73	1.221	0.79	4.930	3.18
		Brit. 320	184.523	14.199	7.69	3.221	1.75	11.750	6.38	6.500	3.52	2.048	1.11	7.350	3.98
		Canadair CL-44C	205.000	15.710	7.66	3.749	1.83	20.524	10.0	7.083	3.46	2.146	1.05	6.834	3.33
		CL-44D	205.000	15.588	7.60	3.540	1.73	16.047	7.83	7.300	3.56	1.830	0.89	6.043	2.95
		Lockheed Electra	106.700	7.670	7.19	1.924	1.80	9.954	9.33	3.817	3.58	***	-	4.417	4.14
		C-130E	151.522	11.697	7.72	3.425	2.26	14.340	9.46	5.341	3.53	1.702	1.12	2.675	1.77
		C-133A	275.000	27.403	9.96	6.011	2.19	30.940	11.3	10.635	3.87	1.804	0.65	3.512	1.28

* tail booms (2,360 lb) included ** included in other items *** no data available

Table 8-5. (Continued)

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	AIRPLANE CATEGORY AND TYPE	MTOW	WING GROUP		TAIL GROUP		FUSELAGE GROUP		LANDING GEAR		SURFACE CONTROLS		NACELLE GROUP	
		10^3 lb	10^3 lb	%	10^3 lb	%	10^3 lb	%	10^3 lb	%	10^3 lb	%	10^3 lb	%
2 ENGINES	VFW-Fokker 614	40.981	5.767	14.1	1.121	2.74	5.233	12.8	1.620	3.45	0.745	1.82	0.971	2.37
	Fokker-VFW F-28/1000	65.000	7.330	11.3	1.632	2.46	7.043	10.8	2.759	4.24	1.387	2.13	0.834	1.28
	F-28/2000	65.000	7.347	11.3	1.632	2.46	7.649	11.8	2.759	4.24	1.400	2.15	0.834	1.28
	F-28/5000	70.800	8.223	11.6	1.632	2.31	7.043	9.95	2.759	3.90	1.665	2.35	0.849	1.20
	F-28/6000	70.800	8.244	11.6	1.632	2.31	7.649	10.8	2.789	3.94	1.674	2.36	0.849	1.20
	BAC 1-11/300	87.000	9.643	11.1	2.369	2.72	9.713	11.2	2.865	3.29	1.481	1.76	**	-
	1-11/400	87.000	9.670	11.1	2.419	2.78	9.743	11.3	2.899	3.33	1.207	1.39	**	-
	Mc D. Douglas DC-9/10	91.500	9.470	10.3	2.630	2.87	11.206	12.2	3.660	4.00	1.264	1.38	1.417	1.55
	Boeing 737-100M	97.800	9.968	10.2	2.700	2.76	12.380	12.7	3.687	3.77	1.589	1.62	***	-
	737-200	100.000	10.613	10.6	2.718	2.72	12.108	12.1	4.354	4.35	2.348	2.35	1.392	1.39
	Aerospat. Caravelle XIR	110.230	14.735	13.4	1.957	1.77	11.570	10.5	5.110	4.63	2.063	1.87	1.581	1.43
	Airbus A300B/2	304.000	44.131	14.5	5.941	1.95	35.820	11.8	13.611	4.47	5.808	1.94	7.029	2.32
3 ENGINES	H. Siddeley 121-IC	115.000	12.600	11.0	3.225	2.80	12.469	10.8	4.413	3.84	1.792	1.56	**	-
	121-IE	134.000	13.462	10.0	3.341	2.49	13.328	9.95	5.073	3.79	1.689	1.26	**	-
	Boeing 727-100	161.000	17.764	11.0	4.133	2.57	17.681	10.9	7.211	4.48	2.996	1.86	3.864*	2.40
	727-100C	160.000	17.492	10.9	4.142	2.59	20.044	12.5	6.860	4.29	2.957	1.85	3.839	2.40
4 ENGINES	Boeing KC-135	297.000	25.251	8.50	5.074	1.71	18.867	6.35	10.180	3.43	2.044	0.69	2.575	0.87
	707-121	246.000	24.024	9.76	5.151	2.09	20.061	8.15	9.763	3.97	2.044	0.83	4.629	1.89
	707-320	311.000	29.762	9.57	5.511	1.77	21.650	6.96	12.700	4.08	2.400*	0.77	4.437	1.45
	707-320C	330.000	32.255	9.77	6.165	1.87	26.937	8.16	12.737	3.86	3.052	0.92	4.133	1.27
	707-321	301.000	28.647	9.52	6.004	1.99	22.129	7.35	11.122	3.70	2.408	0.80	5.119	1.70
	720-022	203.000	22.850	11.3	5.230	2.58	19.035	9.38	8.110	4.00	2.430	1.21	4.510	2.22
	747-100	710.000	86.402	12.2	11.850	1.67	71.845	10.1	31.427	4.43	6.982	0.98	10.021	1.41
	747-200B	775.000	92.542	11.9	11.842	1.53	72.053	9.30	32.693	4.22	7.073	0.91	10.136	1.31
	Mc D. Douglas DC-8-10	273.000	26.235	9.61	4.740	1.74	21.495	7.87	10.185	3.73	2.000*	0.73	3.515	1.28
	DC-8-55	328.000	34.759	10.6	4.889	1.49	22.248	6.78	11.255	3.43	2.253	0.69	4.615	1.43
	BAC VC-10-1101	312.000	34.672	11.1	6.958	2.23	25.113	8.05	10.489	3.36	***	-	**	-
	G. Dynamics 880	184.500	17.669	9.58	4.247	2.30	13.699	7.42	6.203	3.36	***	-	3.685	2.00
	990	253.000	26.871	10.6	5.326	2.11	16.673	6.59	8.718	3.44	***	-	6.772	2.68

* estimated ** included in other items *** no data available

Table 8-5. (Continued)

taken as the larger of the maximum positive gust or the maneuver load factor for the applicable weight at the most critical flight altitude (approximately 20,000 ft for pressurized transports). For further details see Appendix C.

b. Wing group.

A reasonably accurate wing weight estimate can be made in preliminary design as the loads on the wing are fairly well known at the design stage. Usually the bending moment in flight is assumed to be decisive for most of the primary structure. For a certain category of high-speed aircraft, however, torsional stiffness requirements may become dominant and the extra structure weight required to safeguard against flutter may amount to as much as 20% of the wing weight. The location of the inertia axis of the wing plus wing-mounted engines is of importance. A fairly large portion is also made up of secondary structure and non-optimum penalties, such as joints, non-tapered skin, undercarriage attachments, etc.

The derivation of a typical wing weight prediction method is explained in Ref. 8-101, the results of which are summarized in Appendix C. If sufficient data are not available to apply this method, the following simplified approximation can be used for civil airplanes with Al-alloy cantilever wings. The following basic expression is valid for the case of a wing-mounted retractable undercarriage, but not for wing-mounted engines:

$$\frac{W_w}{W_G} = k_w b_s^{.75}\left\{1+\sqrt{\frac{b_{ref}}{b_s}}\right\} n_{ult}^{.55}\left(\frac{b_s/t_r}{W_G/S}\right)^{.30} \qquad (8\text{-}12)$$

where b_{ref}= 6.25 ft or 1.905 m for b_s in ft or m, respectively, while $b_s = b/\cos\Lambda_{\frac{1}{2}}$, the structural wing span. The factor of proportionality is as follows:

Light aircraft, $W_{to} \leq 12{,}500$ lb (5670 kg):
$k_w = 1.25\times10^{-3}$; $W_G \equiv$ MTOW in lb, b_s in ft, S in ft^2, W_w in lb.
$k_w = 4.90\times10^{-3}$; $W_G \equiv$ MTOW in kg, b_s in m, S in m^2, W_w in kg.

APPENDIX F

WEIGHTS

Three supplementary topics are discussed in this appendix. First some simple geometric consequences of the "square-cube law" are listed in Table 14. The table is self explanatory. Next, several well known existing statistical laws of sailplane weight variation with their geometric parameters are listed in Table 15. A comparison of these laws for some typical numerical values of the required coefficients is shown in Figure 33. Finally, several laws for the variation in wing weight with wing span, etc. derived by methods indicated in Shanley's book (114) are listed in Table 16.

A few comments on statistically derived weight laws need to be made in light of the results of the weight equation analysis presented in the Analysis chapter of this study:

1. One needs a good sized data sample covering the entire range of anticipated sizes and weights.

2. There is great difficulty in determining the actual ultimate load factor (n). This parameter has a large influence on weight and there may be wide divergence between design u.l.f., actual u.l.f. and proof load factor.

3. There is often substantial variation in internal structural arrangements (e.g. number and placement of spars) for vehicles with generally similar missions and external shape. Thus strict

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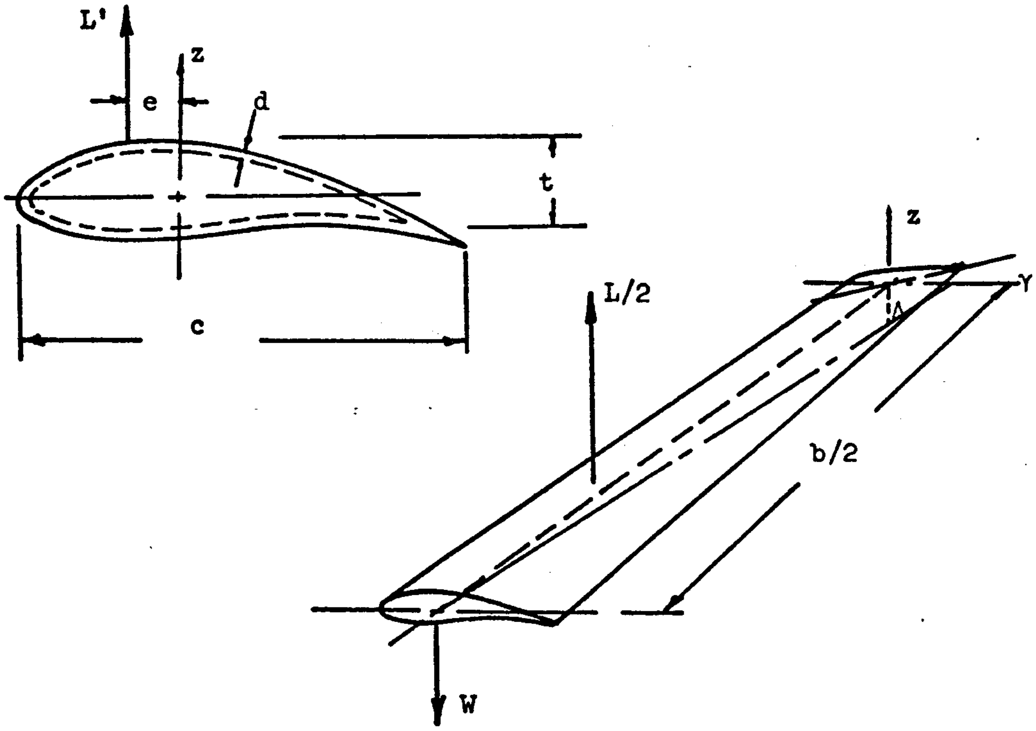
comparison by simplistic rules is tenuous.

4. Vehicle weight and size is not independent of aerodynamic and operational factors.

TABLE 13. Square-Cube Law Comparison



	Case I	Case II
Wing Loading: W/S	constant	variable
Aspect Ratio: $AR = b^2/S$	constant	constant
Thickness/Chord: t/c	constant	constant
Lift Coefficient: C_L	constant	variable constant
Stress Level: σ	constant	constant

M = bending moment = $\sigma I/(t/2)$ $\qquad I \sim t^3 c$

T = torque = $L\,e/2 \sim W\,c/2$

γ = angular deflection $\sim Tb/GJ$ $\qquad J \sim 4(tc)^2 d/2(t+c)$

δ = tip bending deflection $\sim M\,b/EI$

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TABLE 13. continued

	Case I	Case II	
		t/c constant	t/c variable
Weight: W_2/W_1	A	A	A
Wing Span: b_2/b_1	$A^{1/2}$	$A^{1/3}$	$A^{1/3}$
Wing Chord: $\bar{c}_2/\bar{c}_1$	$A^{1/2}$	$A^{1/3}$	$A^{1/3}$
Wing Area: S_2/S_1	A	$A^{2/3}$	$A^{2/3}$
Wing Thickness: t_2/t_1	$A^{1/2}$	$A^{1/3}$	$A^{4/9}$
Thickness Ratio: $(t/c)_2/(t/c)_2$	1	1	$A^{1/9}$
Bending Moment: M_2/M_1	$A^{3/2}$	A	$A^{7/6}$
Torque: T_2/T_1	$A^{3/2}$	$A^{4/3}$	$A^{4/3}$
Bending Deflection:	1	1	$A^{-1/6}$
Twist:	1	$A^{1/3}$	1
Spar Weight: W_{sp_2}/W_{sp_1}	$A^{3/2}$	A	$A^{10/9}$
Performance Consequences:	No Rn scale	No Rn scale	Turbulent flat plate
Optimum Speed: V_2/V_1	1	$A^{1/6}$	$A^{11/57}$
Optimum Lift-Drag Ratio:	1	1	$A^{1/19}$
Power Required: P_2/P_1	A	$A^{7/6}$	$A^{65/57}$

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TABLE 14. Statistical Weight Scale Rules

1. Greenwalt (Insects, Birds, Bats) (Ref. 5)

$$W = C_1 b^3 = C_2 S^{3/2}$$

$$W_w = \text{wing weight} = C_3 S^{5/3}$$

2. Carmichael/Stender (Wooden Sailplanes)

$$W = U + n[\, C_1 + C_2 b^3 / \, (t/c)^{1/3}\, AR^{1/2}\,]$$

3. Wilkenson (Wooden Sailplanes)

$$W = U + [\, C_1 + b(C_2 - C_3 AR)\,]$$

4. Stender (Wood and Metal Sailplanes) (Ref. 111)

$$W = U + C_e\, (nSb^3)^{3/8}$$

$$W_w = \text{wing wt.} = C_w[n(W - W_w)\, \frac{(1 - c_t/c_r)}{2}\, b^2\, AR^{1/2}\, (t/c)^{-1/2}]^{0.64}$$

5. Morélli (Wooden Sailplanes) (Ref. 110)

$$W = U + C_2 + C_1(C_3 + C_4)nb^4 + C_1(C_2 + U)nbAR + (C_3 + C_4)b^3/AR$$

6. Cone (Wooden Sailplanes) (Ref. 117)

$$W = U + C_1 S + C_2 b^3 + C_3 b$$

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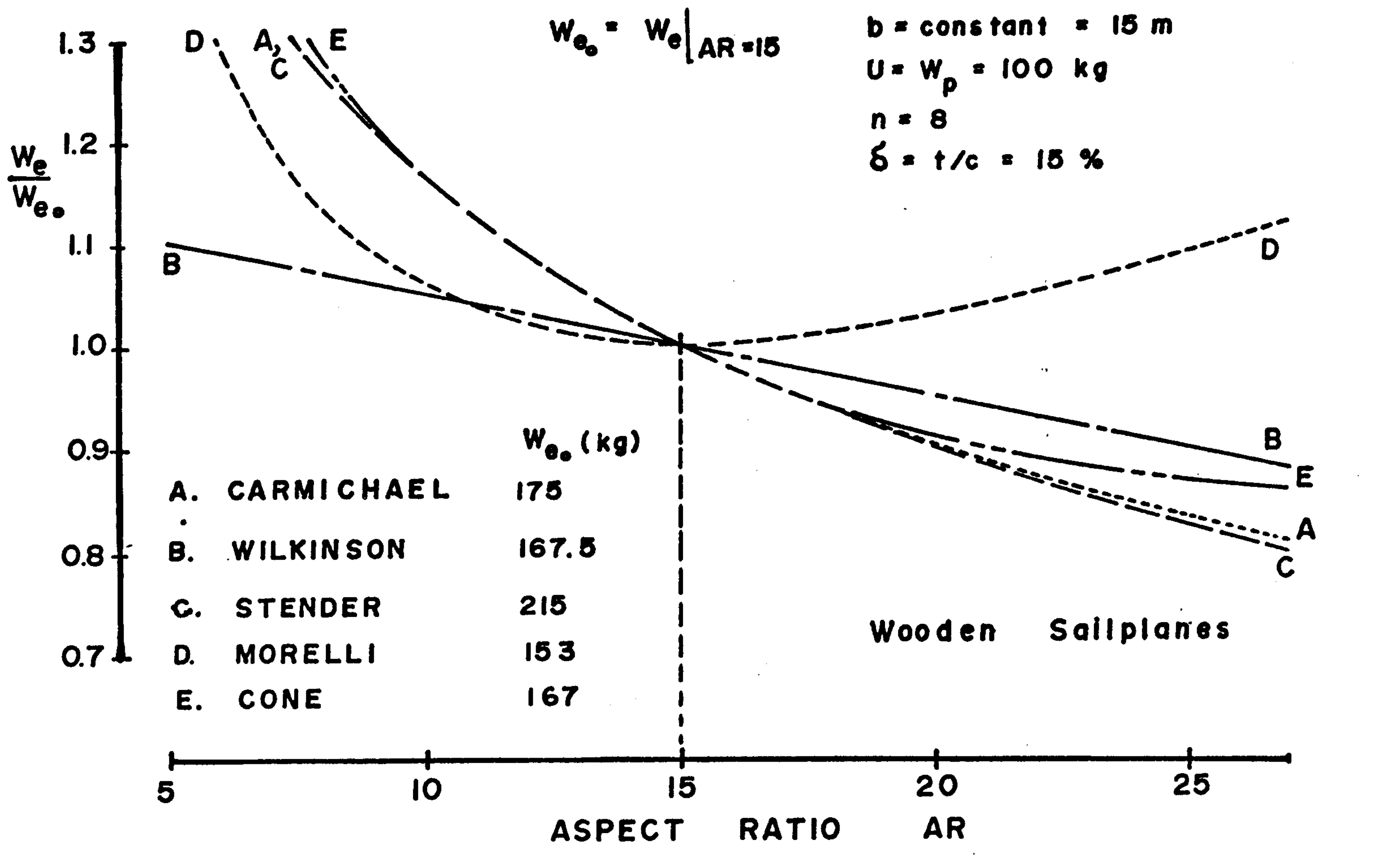

FIGURE 33. COMPARISON OF SEVERAL STATISTICAL WEIGHT SCALE LAWS FOR SAILPLANES

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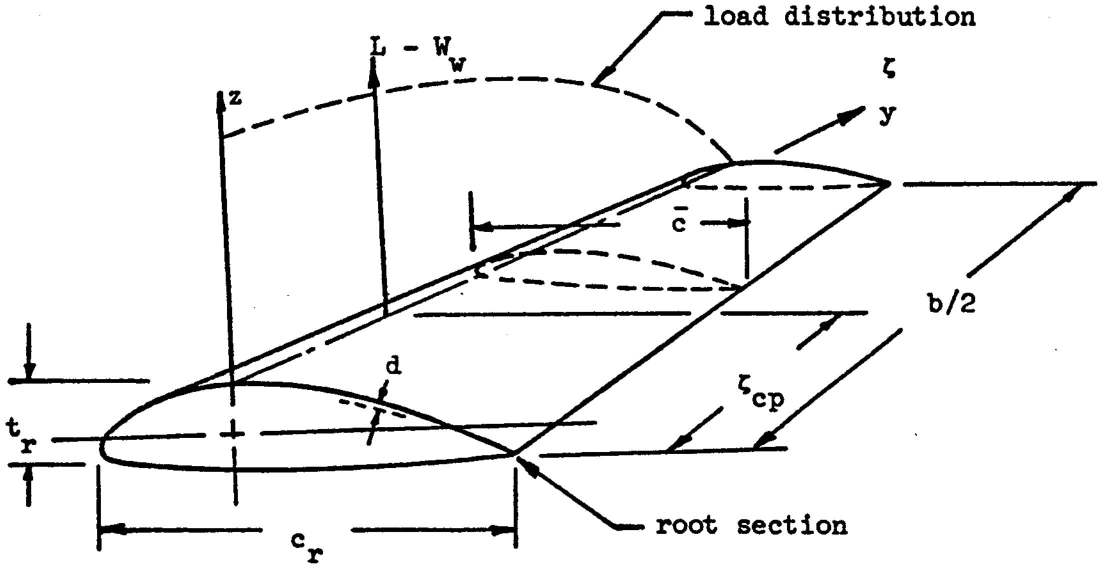
TABLE 15. Wing Weight Scaling Equations



$W = U + W_w$ = gross weight

W_w = wing weight = $W_{spar} + W_{fairing} = C_w W_{spar}$ $\quad c_w = 1 + (W_f/W_{sp})$

$W_{spar} = \rho C_v \, b \, \bar{c} \, \bar{t}$ $\quad$ where $C_v = \bar{p} \, \bar{d} / \bar{c} \, \bar{t}$

$\bar{p}$ = avg. perimeter of spar box

ρ = avg. density of spar structure

I. Bending Strength Limited Wing

$\sigma_{ult} > \sigma_r = M_r \, t_r \, / \, 2I_r$ (stress)

$M_r = (L - W_w) \, \zeta_{cp} \, (b/2)/2$ (moment)

$L = nW$ $\quad n' = n + (n - 1)W_w/U$ (load factor)

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TABLE 15. continued

I. Bending Strength Limited wing

$$W = U + W_w > U + C_{w_1}\ (\rho\zeta_{cp}/\sigma_{ult})\ \frac{n'U}{t/c}\ AR\ b$$

where C_w describes the type of construction

II. Bending Deflection Limited Wing

$$W > U + C_{w_2}\ (U/\theta_{max})\ (\rho/E)\ AR^2\ b\ /\ (t/c)^2$$

$\theta_{max} >$ tip deflection at n = 1 / wing span

III. Torsional Strength Limited Wing

$$\tau_{ult} > \tau = T/2Ad \propto T/2(ct)d \qquad \text{(stress)}$$

$$W > U + C_{w_3}\ (T\ b)\ (\rho/\tau_u)\ (t + c)/\ t\ c$$

IV. Torsional Deflection Limited Wing

$$\gamma_{max} < \gamma = T\ b\ /\ 2\ G\ J$$

$$W > U + C_{w_4}\ (T\ b^2/\gamma_{max})\ (\rho/G)\ (t + c)^2/\ (ct)^2$$

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AIAA-80-0858

Subsonic Transport Noise

C. G. Hodge
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AIAA–INTERNATIONAL ANNUAL MEETING AND TECHNICAL DISPLAY GLOBAL TECHNOLOGY 2000

MAY 6-8, 1980/BALTIMORE, MARYLAND

9.19

SUBSONIC TRANSPORT NOISE

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ABSTRACT

The past twenty-five years in commercial jet transport service has seen a significant decrease in individual aircraft noise levels resulting from an aggressively-funded technology program and engine cycle evolution. Continued community and regulatory pressures, coupled with a diminishing rate of technology improvement and a reduced federal noise technology funding, results in concern for the future and a recommendation for aggressive technology development in four areas.

Noise suppression design technology can result in lower noise levels and/or reduced economic and fuel penalties associated with noise reduction. An analysis of noise sources indicates the need for work on a number of noise source components: jet mixing noise, fan noise, core noise, and airframe noise. Cost-effective suppression requires improved understanding of both noise generation and suppression mechanisms.

Improved component noise prediction technology will have the benefits of reducing the penalties associated with design margins that are required to minimize the risk of a certification failure.

Flight testing technology, together with ground-based flight simulation testing, should be aimed at definitive component, spectral, and more detailed information. A national flight-simulation facility is recommended.

Less costly and more precise certification technology is recommended, including ground testing and analytical procedures.

1.0 INTRODUCTION

The purpose of this paper is to present an assessment, from an airplane manufacturer's point of view, of the current situation regarding aircraft noise and the technology that will be needed to provide a quieter community near the turn of the century. The paper represents an update and expansion on material previously presented at the EPA Noise Symposium (Reference 1).

This paper is restricted in scope to subsonic, conventional takeoff-and-landing (CTOL), jet transport airplanes. (References 2, 3, and 4 deal with propeller and prop-fan technology, rotor craft noise, and supersonic cruise vehicle noise, respectively.)

2.0 PROGRESS IN AIRCRAFT NOISE REDUCTION

A look backward at the quarter century of the commercial jet transport service shows that significant progress has been made in aircraft noise reduction. Figure 1 is a plot of 1500 ft. sideline noise level versus total engine thrust for a number of commercial jet transports starting in the 1950's and extending into the 1980's. The initial commercial jets of the 1950's were turbojet powered and had relatively high jet-mixing noise levels. The jet-mixing noise was reduced significantly in the 1960's

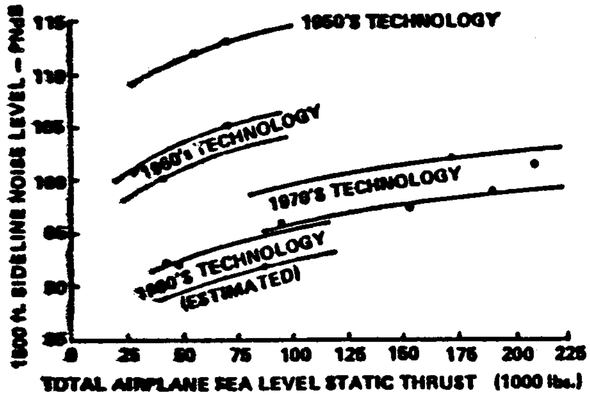

Fig. 1 ENGINE/AIRCRAFT NOISE HISTORY

* Manager, Noise Research

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with the advent of the low-bypass ratio engine, and further in the 1970's when the high-bypass ratio engines were introduced on wide body aircraft. The 1980's are expected to show some additional noise reduction due to technology improvement without a significant change in engine cycle. The figure clearly shows a substantial decrease in noise during the last twenty-five years.

Figure 2 is a cross plot of the information of Figure 1 at 100,000 lbs of thrust. It shows that sideline noise has been reduced with time from the 1950's progressively through the 1980's. However, it is important to realize that the sideline noise reduction resulted largely from a desire to improve performance by increasing bypass ratio.

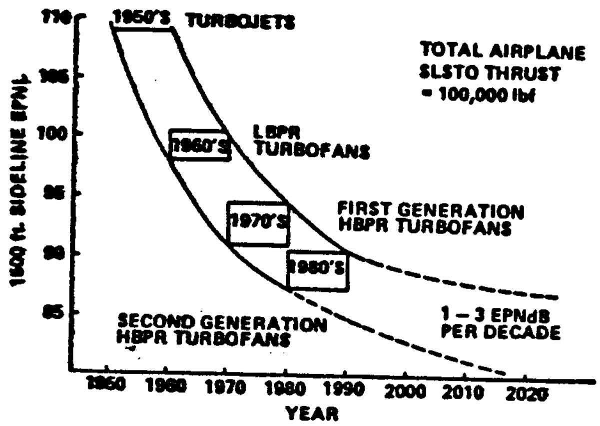

Fig. 2 RATE OF PROGRESS IN SIDELINE NOISE REDUCTION

The progression of engine cycles from turbojet to HBPR turbofan not only reduced jet mixing noise, but also introduced fan noise as an important noise source. The application of acoustic treatment and an increase in the spacing between successive blade rows resulted in effective fan noise suppression. The plot of Figure 3 shows the historical progression to lower noise levels relative to the current FAR 36 Stage 3 certification requirements for new airplanes. Each airplane is represented by three horizontal bars, representing the noise levels at the approach, takeoff, and sideline certification location -- each relative to the corresponding FAR 36 Stage 3 requirement. The black dot represents the amount by which the airplane would exceed or meet the

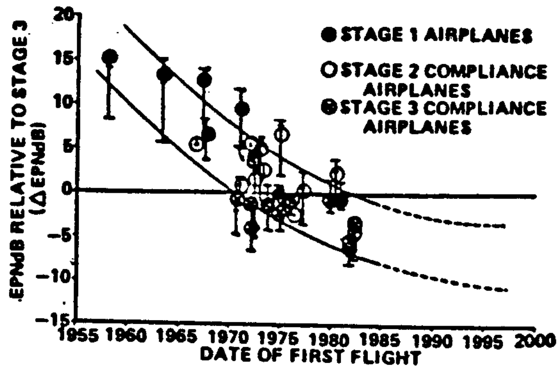

Fig. 3 NOISE REDUCTION TREND vs TIME

Stage 3 regulation after allowance for exceedance tradeoff among the three certification points.

Another way of looking at noise reduction is to examine 100 EPNdB footprint contour areas, combining the takeoff and approach footprints, for airplanes in different time periods. Figure 4 shows long and medium range airplanes and the reduction obtained from the 1950's to the 1970's, as well as short-range airplanes in the 1980's compared to the 1960's. The reduction in exposed area for long and medium range airplanes has been typically 80% to 90% from the 50's to the 70's. For short range airplanes, the reductions have been about 65% from the 1960's to the 1980's.

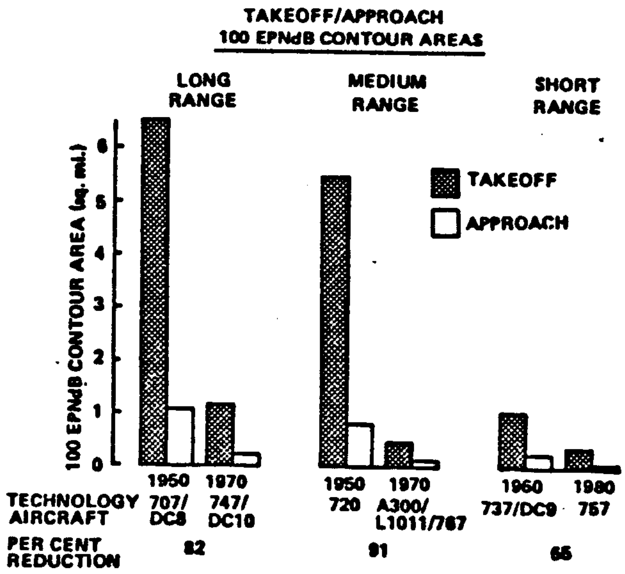

Fig. 4 PROGRESS IN CONTOUR AREA REDUCTION JET AIRCRAFT NOISE TECHNOLOGY

The previous three examples show large improvements in subsonic airplane noise. However, a visual examination of Figures 2 and 3 shows that

9.21

the rate of aircraft noise reduction with time seems to be slowing. In other words, improvements are coming at a slower and slower rate despite considerable dollars having been spent on the problem. Extrapolating largely by eye, but recognizing that decreasing rates of improvement are typical as technology matures, we estimate that in the future we can probably expect 1 to 3 EPNdB per decade improvement. Anything as high as 3 EPNdB per decade will require considerable dollars being spent in well conceived and well managed technology development programs.

3.0 REGULATORY PRESSURES

The regulatory pressures in this country have been mounting since the advent of jet airplanes and show no signs of subsiding. Figure 5 shows that there are three types of regulations currently imposed or being considered by the FAA. The first includes noise type certification requirements at approach, takeoff, and sideline for new and derivative airplanes --i.e., certification of a given airplane design prior to airline service. The requirement to certify CTOL airplanes for noise was first imposed in 1969 (Stage 2 levels) and was made significantly more stringent in 1978 when the Stage 3 requirements were introduced. The Stage 3 requirements range from 3 to 8 EPNdB more stringent than Stage 2.

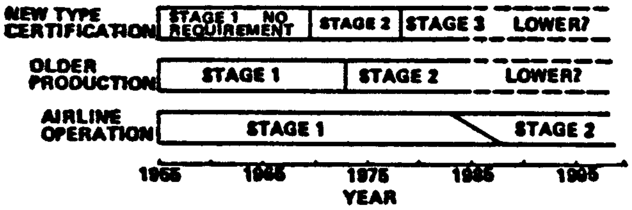

Fig. 5 FEDERAL NOISE REGULATION PROGRESSION

In addition to requirements on new and derivative airplane type certification, Stage 2 requirements have also been imposed on continued production of older domestic aircraft designed and type-certificated prior to 1969; and there is some pressure toward imposing more stringent requirements by the 1985 time period.

The third area of FAA regulation is that of airline aircraft operation. There is a phased compliance schedule at this time to prevent domestic airplanes that do not meet Stage 2 levels from being operated after various dates in the 1983 to 1988 time period.

In addition to the FAA regulations, the International Civil Aviation Organization (ICAO) is in the process of developing a set of recommended standards in the same areas as the FAA requirements. Standards for new designs -- both new models and derivatives -- as well as for continued production of older-type designs, have already been adopted. ICAO adoption by a member nation is similar to an FAR 36 requirement for that nation.

In addition to the federal requirements and international noise standards, individual countries and airports throughout the world in many cases have imposed their own operating restrictions. These restrictions are already placing some economic burden on the airlines and the public by preventing certain airplanes from operating into and out of certain airports, particularly at night.

4.0 FUNDING

Table 1 shows the Boeing Commercial Airplane Company's research and development expenditures in aircraft noise reduction since the inception of the jet age. The figures are in millions of dollars. Over the last twenty-two years Boeing has spent in excess of 100 million dollars in IR&D funds, over 54 million dollars in noise

BOEING AIRPLANE NOISE REDUCTION EXPENDITURES
(MILLIONS OF DOLLARS)

	BOEING "RESEARCH, TECHNOLOGY AND DEMONSTRATION"			
CALENDAR YEAR	BOEING FUNDED	BOEING-PERFORMED GOVERNMENT FUNDED	TOTAL CURRENT DOLLARS	TOTAL BOEING AVERAGE YEARLY STAFFING
1958 – 1964	2.461	.322	2.783	
1965	1.384	.210	1.594	
1966	2.528	.925	3.453	
1967	3.187	3.464	6.651	352
1968	10.957	6.878	17.835	746
1969	8.508	2.873	11.381	746
1970	4.447	1.082	5.529	298
1971	3.594	5.788	9.382	338
1972	6.031	10.918	16.949	636
1973	9.668	8.635	18.303	592
1974	5.866	8.049	13.915	464
1975	7.499	3.232	10.731	340
1976	8.584	.845	9.429	280
1977	11.277	.569	11.846	308
1978	10.631	.407	11.038	296
1979	10.086	.339	10.425	366
TOTAL	108.718 **	54.536	162.254	
1980 (EST)	13.618	.410	14.028	

** ADDITIONAL EXPENDITURES APPROXIMATELY $54 MILLION IN PRODUCTION AIRPLANE NOISE REDUCTION ACTIVITY AND OVER $18 MILLION FOR ACOUSTIC TEST FACILITY CAPITAL EXPENDITURES

Table 1

9.22

reduction activities specifically associated with production airplanes, and over 18 million dollars in expenditures for acoustical test facilities. Furthermore, Boeing has received over 50 million dollars from the U.S. government for work on government-funded noise research and development contracts. The total dollars expended, when capital facilities and production airplane work are included, will by the end of 1980 exceed a quarter of a billion dollars.

The engineering manpower dedicated to noise reduction has fluctuated depending on the company financial position and funding available from the government for research and development; however, it is seen to represent a considerable number of personnel, and is continuing at a high rate.

As pointed out by the EPA in its latest (1978) assessment of federal noise research (Reference 5) and tabulated in Table 2, the total of all federal aircraft noise research, technology, and demonstration spending has decreased in recent years to a level of $15 to 20 million. This contrasts with $40 to 50 million during the most intense phase of the NASA refan program. Boeing is concerned about the decreased government activity in noise research and technology, especially in view of the continuing regulatory and community noise reduction pressure.

FEDERAL SPENDING FOR NOISE RESEARCH, TECHNOLOGY, AND DEMONSTRATION

FISCAL YEAR	MILLIONS OF DOLLARS
1973	46.966
1974	38.228
1975	18.184
1976	16.118
1977	16.840
1978	18.358 (ESTIMATED)

Table 2

The breakdown of fiscal year 1978 federal budget for noise technology research is shown in Table 3. This information is taken from the 1978 Federal Interagency Aviation Noise Research Panel booklet (Reference 6). Federal government spending for research technology and demonstration programs was budgeted to be about 18 million dollars for the fiscal year 1978 with approximately two-thirds in the area of research and

FEDERAL GOVERNMENT NOISE RESEARCH, TECHNOLOGY, AND DEMONSTRATION PROGRAM BUDGET – FY 1978 (MILLIONS OF DOLLARS)

RESEARCH AND TECHNOLOGY PROGRAMS	NASA	OTHER	TOTAL
PROPULSION NOISE	4.460	.418	4.878
ROTOR NOISE	1.272	.410	1.682
INTERIOR NOISE	.683		.683
AIRFRAME NOISE	2.515	.432	2.947
NOISE PREDICTION THEORY	.383	.588	.961
ATMOSPHERIC PROPAGATION AND GROUND EFFECTS	.481		.481
OTHER	.383		.383
TOTAL	10.357	1.710	11.978
DEMONSTRATION PROGRAMS AND SYSTEM STUDIES			
CTOL (SUBSONIC)	2.892	1.730	4.622
STOL	.383		.383
ROTORCRAFT/VTOL	.130		.130
GENERAL AVIATION	1.327		1.327
TOTAL	4.682	1.730	6.382
GRAND TOTAL	14.909	3.449	18.358

Table 3

technology programs, and one-third in the demonstration and systems studies. Boeing feels that the research and technology programs generally have produced more applicable noise technology per dollar than the demonstration programs. The integration of technology into flightworthy and commercially-viable airplane systems is best accomplished in the private sector by airplane and engine manufacturers facing the constraints of regulatory and life-cycle cost considerations.

5.0 TYPES OF AIRPLANES AND ENGINES

As stated above, the scope of this paper deals with conventional takeoff and landing subsonic transports. Looking into the future and recognizing that it is not always possible to foresee breakthroughs that will come to fruition in ten to twenty years, we nevertheless expect that the generations of airplanes for the 1990's and for the early part of the next century will quite likely not be radically different in terms of basic wing-fuselage configuration and type of propulsion system (Reference 7) than the airplanes of the 1970's and 1980's. In other words, we would expect high bypass-ratio conventionally-fueled turbofan engines with single stage fans, perhaps slightly higher in bypass ratio -- i.e., without any radically different concepts, but rather progressive improvements in aerodynamic efficiency, engine durability, and installed weights. Should the energy situation result in new types of fuel for aircraft, the most likely configuration would still be an airbreathing gas turbine. Consistent with these assumptions, the noise technology needs can be identified by considering today's airplanes and projecting the types of technology needed for making such airplanes quieter to meet more stringent noise levels, or for reducing the penalties for meeting today's noise levels. This is the subject of the rest of this paper which will treat the problem in four areas of noise technology : noise suppression design technology, prediction technology, testing technology and certification technology.

6.0 NOISE SUPPRESSION DESIGN TECHNOLOGY.

Design technology for noise suppression is very critical, whether the objective is to make airplanes quieter than the Stage 3 limits, or to minimize the performance and fuel penalties associated with small improvements in noise. Figure 6 is taken from a study made during preliminary design work at Boeing in anticipation of Stage 3 noise requirements. The fraction of gross weight that can be used as payload is reduced as greater takeoff noise reduction is required. The results shown represent nacelle treatment increases, as well as compensating design changes through the airplane, including engine size required to perform the same prescribed payload-range mission. It is important to realize that the curve represents the locus of optimum airplane point designs for the mission, and therefore represents an optimum or minimum cost curve for a fixed technology level.

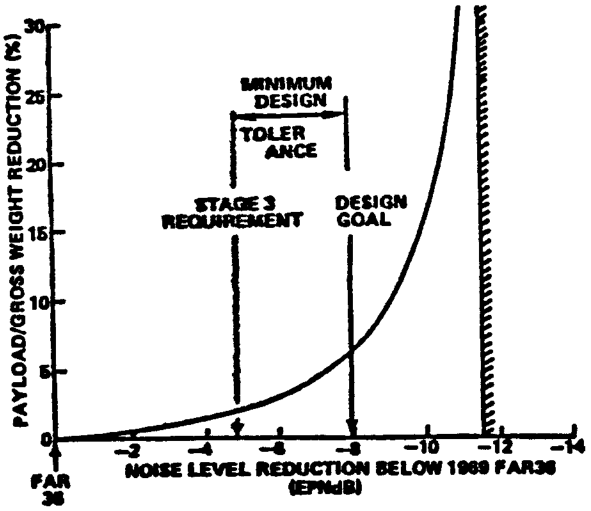

Fig. 6 RELATIONSHIP BETWEEEN NEW TECHNOLOGY AIRPLANE TAKEOFF NOISE REDUCTION AND AIRPLANE PRODUCTIVITY

There are several features of Figure 6 which are characteristic of many noise reduction cost-benefit studies. First, the shape of the curve is such that the cost per unit benefit increases signficantly with increasing noise reduction. Second, a minimum design tolerance must be superimposed on an airplane certification requirement to provide an acceptable risk of non-certification, resulting in a nominal design goal that must be significantly more stringent than the requirement; and there is a definite penalty associated with this tolerance. Third, as cost penalties escalate, a limit is reached where further reduction in payload would tip the scales and result in a non-viable product. Fourth, there is, for a given point in time, and state-of-the-technology, a level beyond which the noise will not be further reduced, irrespective of the cost. In this case the limit happens to be very close to our present design point. In short, the economic viability of an airplane is strongly affected by small differences in required noise levels.

Another example of deterioration of performance with noise reduction for a given state of technology is shown by Figure 7, which shows the increase in fuel consumption caused by noise level reduction. This plot is for the approach

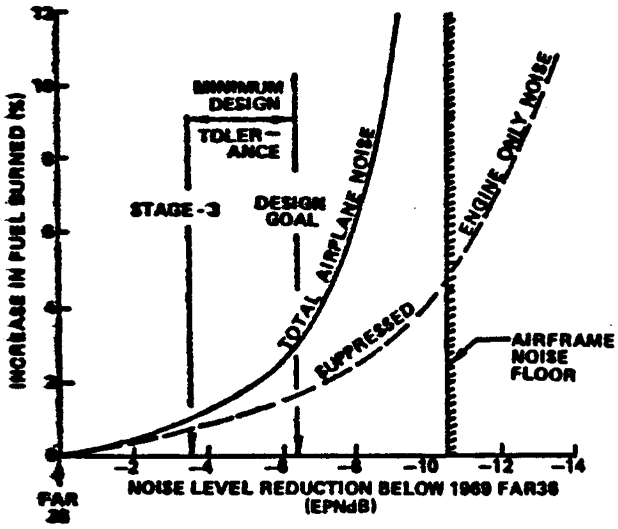

Fig. 7 RELATIONSHIP BETWEEN NEW TECHNOLOGY AIRPLANE APPROACH NOISE REDUCTION AIRPLANE FUEL USAGE

9.24

case as opposed to the previous takeoff case, and we see that as total airplane noise is significantly reduced, the amount of fuel used increases more and more rapidly. The dashed curved line on the chart represents a suppressed engine-only trade curve, or what this penalty would be if the airplane did not have airframe noise. In other words, the airframe noise floor shown represents a real limit of current technology for noise reduction.

The previous two examples illustrate that noise reduction carries significant economic penalties. On the other hand, for the same reason that the performance of an airplane is very sensitive to the required noise levels, moderate (or even small) improvements in noise technology can result in significant fuel savings and improved economic viability of aircraft. This point is illustrated in Figure 8, which is based on the same trade study as Figure 7. The curve illustrates what would happen if we superimpose a 3 dB technology improvement, or what we might expect from a highly funded technology program over ten years. This may not appear to be a very significant noise decrease, but if we are faced with a given noise level requirement and view the technology improvement in terms of fuel savings, there are some very definite benefits to be gained as shown by the curve displayed to the right. Particularly if very low noise level requirements are being considered, we see tremendous potential in fuel savings with correspondingly relative small, although difficult to obtain, decreases in noise levels.

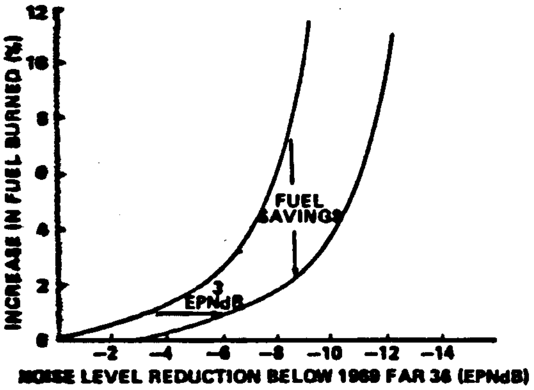

Fig. 8 EFFECT OF NOISE TECHNOLOGY IMPROVEMENT ON FUEL CONSUMPTION AT SAME NOISE LEVEL

The noise from a turbofan-powered commercial transport emanates from a number of different component noise sources (Figure 9): fan noise radiating from the inlet, fan noise from the fan exhaust duct, core noise, turbine noise, jet mixing noise, and airframe noise. In order to examine the implications of future noise reductions, we have taken the projected noise levels of several new airplane designs and examined the question -- what are the engine noise component reductions corresponding to a hypothetical requirement to reduce the noise of these airplanes an additional 5 and 10 EPNdB at takeoff and approach. This is not to say that 5 or 10 EPNdB

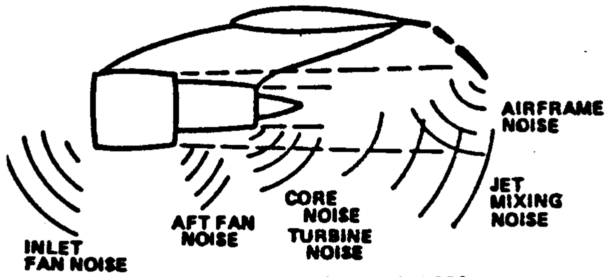

Fig. 9 COMPONENT NOISE SOURCES

below current Stage 3 levels are in any way reasonable goals; rather the intent is to examine the implications of such assumed reductions in terms of component noise reduction requirements. Shown in Table 4 are component reductions for these airplanes corresponding to 5 and 10 EPNdB total noise reductions at takeoff and approach. With the exception of turbine noise, extremely formidable requirements are shown in a number of recognized components: aft fan noise, conventional inlet fan noise, inlet buzzsaw noise, jet noise, core noise, and airframe noise. In other words, the noise reduction design problem has become one of dealing simultaneously with a number of components, as opposed

TARGET COMPONENT NOISE REDUCTIONS FOR 5 AND 10 EPNdB AIRPLANE NOISE REDUCTIONS

ΔEPNdB	TAKEOFF		APPROACH	
TOTAL (GOAL)	5	10	5	10
AFT FAN	5	11	7	14
INLET FAN	0	3	7	13
JET	8	10	0	5
CORE	4	10	1	6
INLET BUZZSAW	1	7	0	0
TURBINE	0	0	0	3
AIRFRAME	0	3	8	12

Table 4

9.25

to coming up with an invention that merely reduces one component of the noise a significant amount, as with jet noise back in the 1950's. A partial list of the important areas of design technology, organized by component, is shown in Table 5, together with corresponding areas of prediction and measurement technology.

IMPORTANT AREAS OF RESEARCH

	COMMUNITY NOISE	ENGINE SYSTEM NOISE	JET/CORE NOISE	TURBO-MACHINERY NOISE	LINING AND DUCT ACOUSTICS	AIRFRAME
GOALS	AIRPLANE AND AIRPORT GOALS WITH REALISTIC SCHEDULES AND COSTS	COMPONENT GOALS CORRESPONDING TO TOTAL GOALS				
SUPPRESSION	FLIGHT PROCEDURES AND TRAFFIC CONTROL	CYCLE DESIGNED FOR LOW NOISE	LSPR AND HPR JET SUPPRESSION CONCEPTS CORE NOISE SOURCE SUPPRESSION	QUIET FAN DESIGN	ADVANCED LINING & OTHER SUPPRESSION CONCEPTS	QUIET FLAP SYSTEMS
MEASURE-MENT	REPEAT-ABILITY	COMPONENT SEPARATION IN GROUND AND FLIGHT TESTING	FLIGHT NOISE SIMULATION FOR LOW-NOISE JETS	INFLOW SIMULATION FOR TESTING	MODE MEASURE-MENT	QUIET WIND TUNNEL
PREDICTION	IMPROVEMENTS IN ANOPP. ANALYTICAL MODELS BASED ON ACOUSTICALLY-IMPORTANT PHYSICS, AND TIED TO EXPERIMENTS					

Table 5

As previously shown, jet mixing noise reduction has historically resulted from basic engine cycle modifications. For a given thrust level, air flow rates have increased and jet velocities have decreased, resulting in reduced jet mixing noise. Further steps in this direction can, in principle, come from further increases in bypass ratio, or introduction of ambient air during takeoff with variable-geometry hardware such as a variable-cycle engine or an ejector-suppressor. The performance penalties of variable-geometry concepts represent a formidable obstacle. Internal mixers, which have been used successfully to mix the primary and fan streams on several models of the low-bypass JT8D engine, may also prove viable on higher-bypass engines. The use of non-symmetric nozzle configurations, with advantageous orientation of the non-axisymmetric noise directivity patterns, represent a mechanical design challenge. Multi-element suppressor designs require better understanding of the fluid mechanical behavior from both a steady-state (aerodynamic) and unsteady (turbulence and acoustic) performance standpoint. An example of research aimed at jet suppression technology is described in reference 8 and is shown in Figure 10; in this work a two-tube nozzle configuration exhibited significant "shielding" of the high-velocity jet by the low-velocity jet.

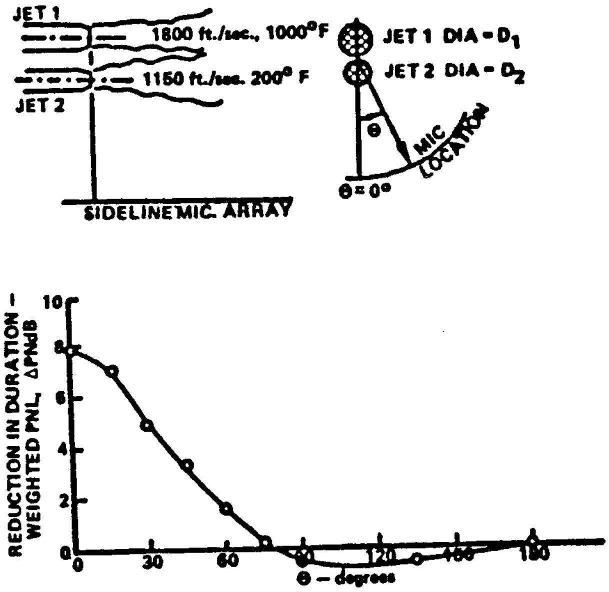

Fig. 10 NOISE RESULTS OF MODEL TEST ON TWO PARALLEL JETS

Fan noise suppression will result from progress in two areas -- basic fan design and duct acoustics. A basic fan design that is significantly quieter will depend on developing knowledge of the fluid-mechanical mechanisms responsible for the noise generation process --e.g., rotor-wake stator interaction; inflow-distortion rotor interaction. Non-axisymmetric duct shapes are a potential source of fan noise reduction, as suggested by the work of Figure 11 and Reference 9. Acoustic lining designed specifically for the acoustic modes generated by the fan has further potential. High Mach number inlets may become economically viable to meet high inlet noise suppression requirements.

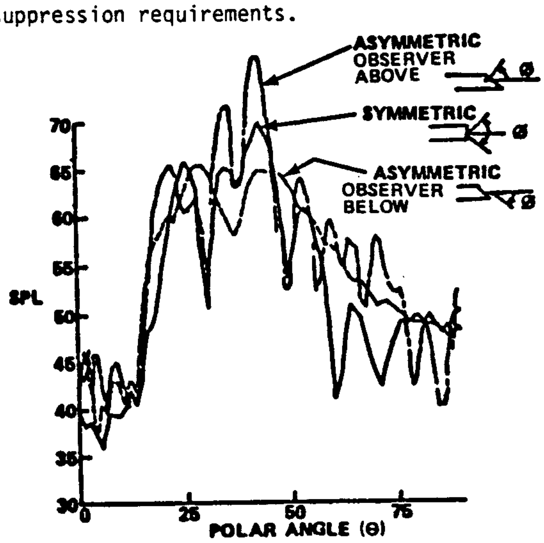

Fig. 11 EFFECT OF INLET SHAPE ON SOUND RADIATION WITH NO FLOW

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Core noise suppression progress will depend on better identification of the generation mechanisms, and may imply changes in combustor design. Low-frequency mufflers integrated into a plug design are a concept worthy of further development.

Airframe noise is the noise associated with parts of the airplane other than the propulsion system, and is not the target of suppression hardware on today's airplane designs. It is in the infant stages of understanding, with relatively very little known about the generation mechanisms. Associated with each postulated mechanism however, will be the potential for one or more hardware concepts aimed at suppression. Jet-flap interaction, which may possibly represent a barrier to achieving noise levels below Stage 3, will be strongly affected by engine placement and flap and cutout design.

In short, for each major component of noise, potentially-viable noise reduction hardware concepts can be conceived. However, in order that these concepts can be evaluated and those that merit further work economically integrated, considerable research aimed at identifying, modeling, and quantifying the noise generation and suppression mechanisms is necessary.

7.0 PREDICTION TECHNOLOGY

The importance of improved component noise prediction technology is evident for two reasons. First, the prediction technology is needed to properly screen and assess the effectiveness of various suppression concepts and is therefore an integral part of the airplane design process. Second, prediction capability is necessary to assess certification risk because of the vast financial commitment that a new airplane program requires prior to certification flight testing. The uncertainties in prediction result in the requirement for design margins. For example, a current technology subsonic transport airplane is felt to have an uncertainty in the eventual demonstrated certification levels which corresponds to a standard deviation of approximately 3 1/2 to 4 EPNdB. When translated into design requirements and confidence levels, Figure 12

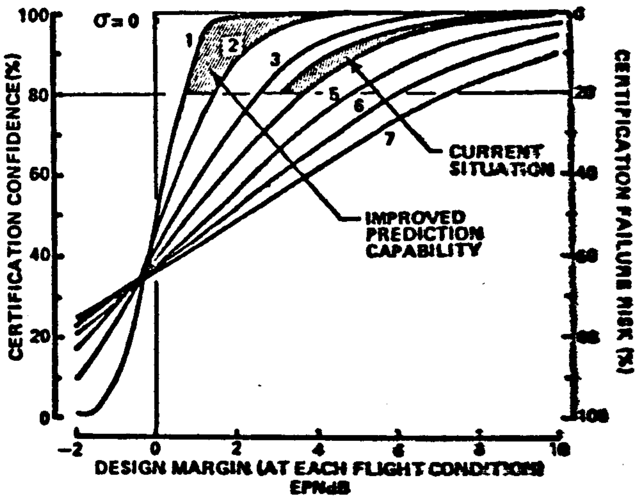

Fig. 12 CERTIFICATION CONFIDENCE AND RISK

shows that in order to reduce the risk of a certification failure to the zero to 20% range, the design margin must be a minimum of 3-4 EPNdB. By the same token, if the uncertainty in prediction could be reduced to the point where the total uncertainty is 1-2 dB, design margins could then be reduced by 1 1/2 to 3 EPNdB, with the resulting decrese in economic penalties to the airlines and flying public.

In the prediction area, we believe that NASA's Aircraft Noise Prediction Program (ANOPP) is a national contribution but also needs to be improved. Noise engineers and noise regulators need to incorporate uncertainty analysis estimates into their predictions, and need to base analytical models on the acoustically important physics. Finally a strong interplay between experimental evidence and prediction methods is vital.

An example of some recent prediction technology which is based on the acoustically-important physics is the jet noise prediction method illustrated in Figure 13 and Reference 10, in which the jet noise generation and propagation is related to the velocity profile, temperature profile, and turbulence intensity profile. Reasonable agreement has been reached for an axisymmetric low-bypass jet in the transonic regime, as shown in Figure 14.

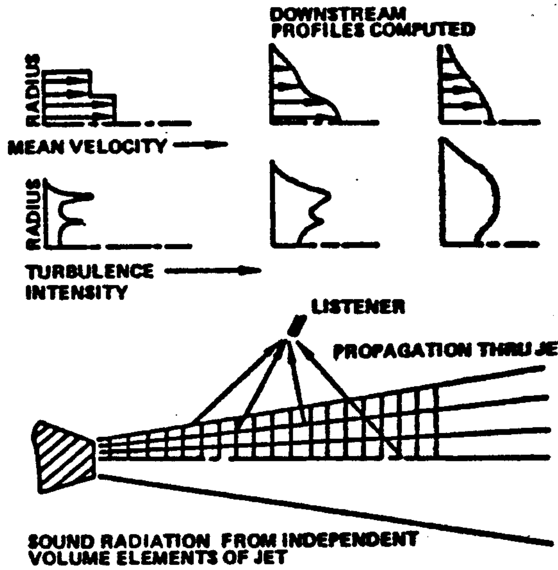

Fig. 13 ANALYTICAL JET NOISE PREDICTION

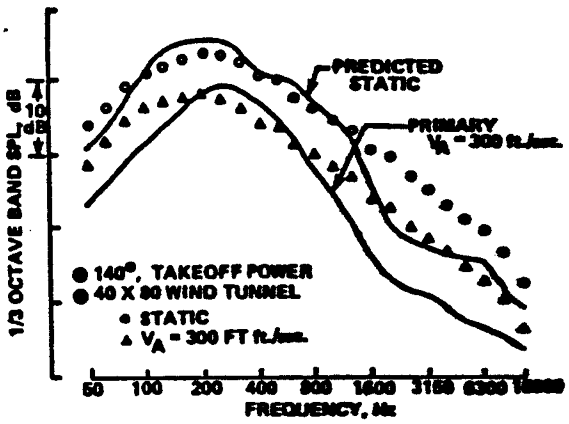

Fig. 14 JT8D STATIC AND FLIGHT COMPARISON OF PREDICTED AND MEASURED NOISE

8.0 TESTING TECHNOLOGY

One of the most critical areas in noise testing technology lies in the ability to make repeatable and reliable flight test measurements which can be analyzed in a manner such that various components of noise can be quantified. The typical certification testing process, although relatively repeatable in terms of EPNdB, does not lend itself to this type of detailed analysis, the results of which are essential to the development of quieter airplanes. Meaningful progress has been made in this area by Boeing in recent years. One technique that has been used involves flush mounted ground microphones with ensemble averaging of the various measurements. As an example, in Figure 15, we see the measurements

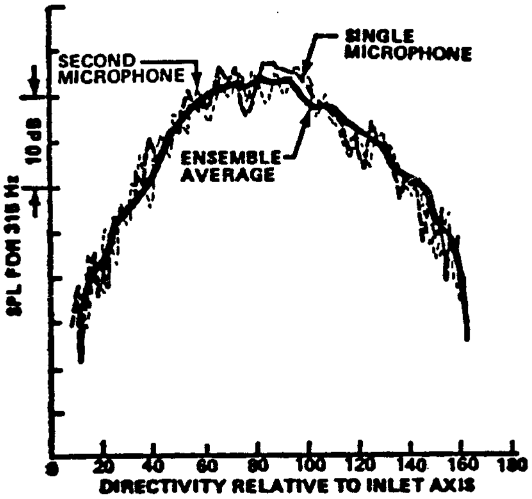

Fig. 15 ENSEMBLE AVERAGING OF FLIGHT DATA

from two of six microphones at a given frequency and as a function of polar angle; and we notice the poor repeatability and lack of smoothness. On the other hand, the ensemble average of six microphones shows a considerably smoothed directivity pattern. The corresponding spectral analysis (Reference 11), which amounts to a cross plot of a number of these plots, will tend to result in identifiable component spectrum shapes.

Another recent development which should have a significant impact on noise technology between now and the year 2000 is the ability to isolate noise sources in a ground test. Figure 16 shows schematically a method in which data is taken at multiple sidelines (Reference 12), and by examining the spectra projected back to the noise at a given frequency and given far-field angle, can be projected back to its apparent source location. In this way components which have the

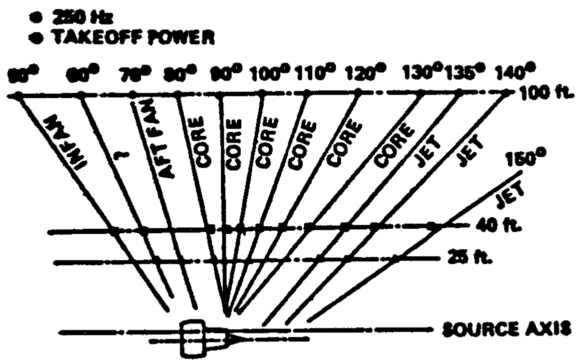

Fig. 16 NOISE SOURCE LOCATION FROM JT9D-7 STATIC DATA

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same frequency content and directivity can be distinguished from one another by the apparent location of the noise source.

Another aspect of static testing in which progress has been made is in control of the flow into an engine, particularly with regard to turbulence (References 13 and 14). Figure 17 shows a turbulence inflow control structure which has been developed to better simulate certain aspects of inlet fan noise. The ability to measure modes of acoustic propagation in ducts is also essential to the development of fan noise prediction and measurement technology.

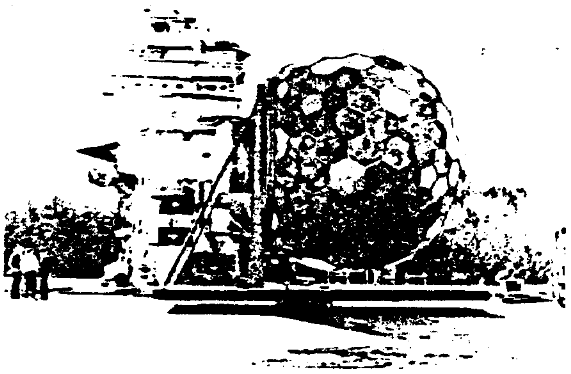

Fig. 17 BOEING JT9D INFLOW CONTROL STRUCTURE TULALIP WASHINGTON TEST SITE

The cost and complexity of flight testing, coupled with the requirement to realistically assess the in-flight noise performance of different configurations and suppression concepts, points to the need to test a single-engine full-scale propulsion system, or model-scale propulsion system components and/or airframe, in a ground-based flight environment. Although inflow control structures may suffice for fan noise, jet and airframe noise testing needs a large acoustic wind tunnel. At this point in this country we do not have a facility which will be capable of making such noise measurements at levels which are appreciably below that of a current high bypass engine operating at takeoff or cutback power. What is needed is a large facility with flight simulation capability, designed with acoustic testing as a primary use. Associated with this facility requirement is a need for microphone and noise-source location technology that will enable a thorough diagnostic job to be done.

9.0 CERTIFICATION

During the next twenty years one can reasonably expect changes in certification technology. In the flight testing area, there will be an emphasis on better repeatability in order that this contribution to the design margin requirement can be reduced. Furthermore, flight testing is an extremely expensive process, and it is expected that more sophisticated ground testing and analytical methods will be developed for certification of noise levels for nearly similar configurations.

10.0 CONCLUSIONS AND RECOMMENDATIONS

The continuing regulatory, airport, and community pressures for lower aircraft noise levels, coupled with the high economic leverage of noise technology establish a requirement for improved aircraft noise reduction technology. The complexity of the technology, and the potentially slow rate of progress, result in a need for an aggressive program aimed at design, prediction, and measurement technology of a number of noise components: jet noise, fan noise, core noise, and airframe noise.

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