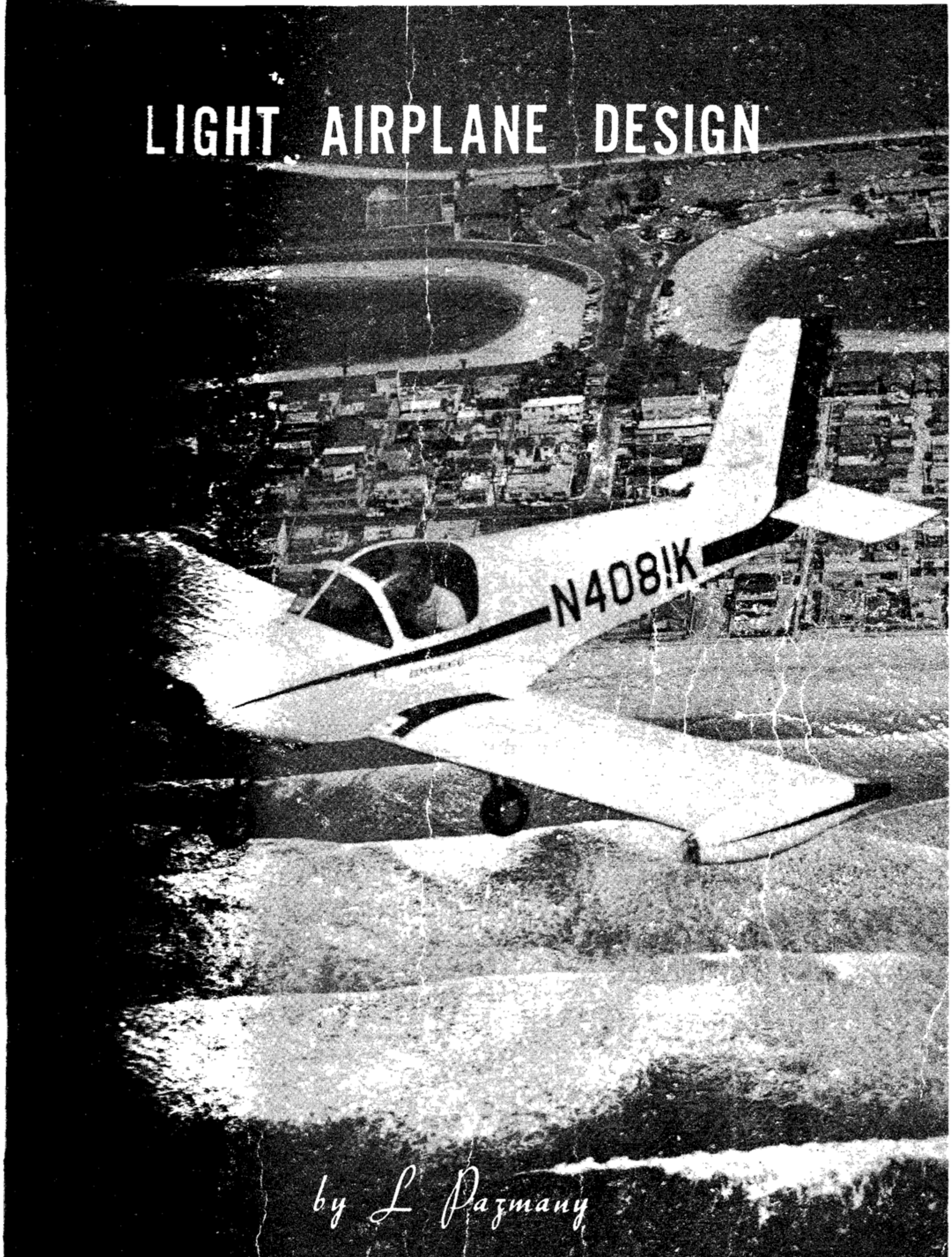
LIGHT AIRPLANE DESIGN



by L Pazmany

Cláudio Pinto de Barros
Agosto - 66

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L. Pazmany

(Illustrations by the Author)

Published by L. Pazmany
Printed in the United States of America
San Diego, California

THIRD EDITION

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PREFACE

This book describes the Preliminary Design of the Pazmany PL-1 "Laminar" airplane, which now has more than 450 hours of flying time. At a future date other volumes will be published covering: Performance, Stability and Control, Structural Design, Stress Analysis, Construction and Flight Tests.

It is not easy to predict all flying qualities of an airplane using calculations. This applies either to the simplest "home-built" or to a sophisticated supersonic fighter, mostly when they are of unconventional type, like the delta wing. There are too many variables in the game, especially when stability and control is the subject. The aerodynamicist needs the help of wind tunnel testing and simulators to rectify or ratify his calculations.

The PL-1 is a conventional airplane for present standards; nevertheless, modern aerodynamic and structural data were applied in every phase of the design.

At the present more than 40 PL-1 airplanes are in construction all over the United States and Canada. A few are in construction in Australia, India, New Zealand, Panama, British Solomon Islands and England.

The combination of the selected laminar airfoil with the untwisted rectangular wing planform resulted in a very efficient wing with extremely gentle stall and very good aileron control. The airplane has no "vices" and is easy to fly. The acrobatic capabilities of the PL-1 were demonstrated publicly in several air shows; all kind of acrobatic maneuvers were executed; barrel, snap rolls, immelmans, spins, loopings, stalls, etc. Also, the landing characteristics were evaluated. The comments can be summarized as "An excellent trainer," "a very good acrobatic airplane," "It is impossible to make a bad landing."

This book is dedicated to the great "amateur-builders" family hoping that it will encourage them to be a little more "amateur-designers." Particular thanks are due to Mr. Karl Sanders for assistance in proofreading and helpful criticism and suggestions.

San Diego, California   L. Pazmany
December 1964

Photo from AIR PROGRESS Magazine by Don Downie

INTRODUCTION

The most difficult problem in designing an airplane, when this job is done by an inexperienced person, is to find a guide that shows step by step the sequence in which the different problems have to be approached and solved.

The amateur designer can select between two extreme procedures: 1) Eyeball, 2) Engineering. The procedure described in the following pages is an intermediate way, and it must be stated here that this is only one of the many ways.

The necessary knowledge about different subjects, such as Aerodynamics, Stress, Structural Design, Air Regulations, is distributed in many sources of information. To find this knowledge it is necessary to invest a great amount of time searching and reading. Sorting out useful material takes up the most time because 90 per cent of the information is not related with the actual problem.

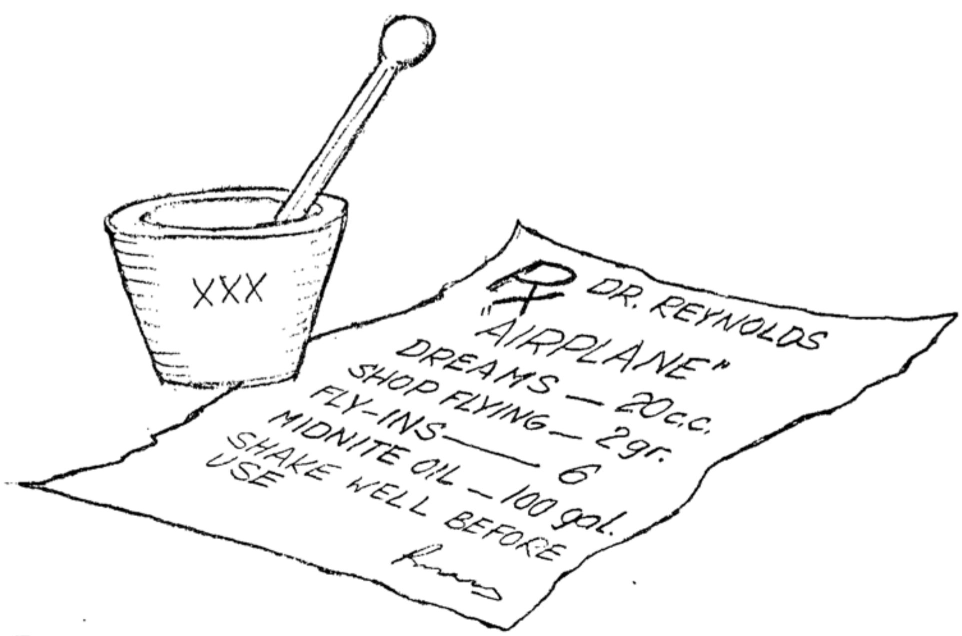
But it is not only enough to find the information in books and reports. The second problem is "How to put it to use" and for answering this question nothing is better than an example. This is the idea behind this publication--a guide for the amateur airplane designer.

No higher mathematics will be used, only the four basic algebra operations, along with many graphs and diagrams.

Sometimes more than one approach to a problem will be given, with the related comments about its usefulness. Of course, many solutions are influenced by a personal viewpoint.

The reader must keep in mind that there are no "prescriptions" to design airplanes. If some airplanes have large wing fuselage fairings or nicely rounded wing tips, that does not mean that "all" airplanes must have wing fillets or rounded tips.

A large wing fillet might reduce the interference drag, but its weights and production complications could be a good reason to leave it out and take the penalty in performance. The design of an airplane is not a simple task but a series of compromises.



1-PRELIMINARY DESIGN

The first step in the design of an airplane consists of defining the characteristics of the airplane and its use. In aeronautical engineering this is called "Mission Definition," and could be applied as well to an amateur-built airplane.

"What do you want to do with the flying machine?" is the first question. The design task will be very much simplified if a straight answer to this question could be spelled out. But when some advancement in the state-of-the-art is desired, it implies characteristics which are not always compatible.

If you will be happy with something to "fly around," you probably do not need to design an airplane in the first place. It will be much simpler to find some Piper Cub wings, Aeronca fuselage, Luscombe tail surfaces, and a Cessna gear, put it all together, and if you hit the C.G. at 25 per cent chord and have enough HP it will take you "around."

On the other hand, do not try to make a break-through in aerodynamics or structures. Private industry and government agencies are spending fantastic amounts of money in research, and the results are published in reports. Take advantage of this material which is generally free.

A few rules worthwhile keeping in mind are:

1- Make it big inside and small outside as the compact cars.

2- Make it strong enough to carry the loads. It is worthless to overstrengthen some non-critical parts while the main spar is weak.

3- Reduce weight even before you start your design. Assume optimistic weights for your components; they will go up anyway. You probably heard about the "weight spiral."

4- Do not penalize the design by using oversize or overweight components of "existent" airplanes.

5- Use a minimum choice of basic materials.

6- Use minimum number of parts.

7. Do not give up any reasonable chance to "clean up" your design.

8. Keep a continuous check of your weight and balance all through the design.

9. Build a full-size mock-up of the cockpit in the earliest stage of the design.

10. Do not hesitate to spend some thousand hours in the design of your "bird." It will be well rewarded during the construction, but mostly during flying.

1-1 GENERAL CHARACTERISTICS

Specific Use: Sport, Trainer, Acrobatic, Two Place.
Inherent Attributes: Safety.
Appearance: Functional.

Fuselage
Type of Construction: Semimonocoque.
Basic Structural Material: Aluminum Alloy.
Skin Material: Aluminum Alloy and/or Magnesium.

Wing
Type of Construction: Cantilever, Detachable, Carry-through Spar.
Shape: Rectangular.
Location: Low
Airfoil: To be selected.
Aspect Ratio: ~ 7
Area: ~ 100 sq.ft.
High Lift Devices: Flaps.
Wing Loading: ~ 10 lbs/sq.ft.
Spar Material: Aluminum Alloy.
Skin Material: Aluminum Alloy or Magnesium.

Empennage
Fin-Rudder Configuration: Conventional.
Stab-Elevator Configuration: Conventional or Slab Tail.

Power Plant
Type: Opposed, Air-cooled, 85-100 HP.
Fuel System: Gravity and electrical booster pump.
Tank Location: Wing tips. (Optional fuselage tank for extended range.)

Cockpit
Control Type: Stick.
Instruments: Nominal.
Canopy Type: Sliding, Bubble.
Visibility: Normal.

Landing Gear
Type: Tricycle, Fixed.

Weight
Empty: ~ 750 lbs.
Gross: ~ 1300 lbs.

Desired Performance
Stalling Speed: ~ 50 mph.
Cruising Speed: ~ 115 mph.
Max. Speed: ~ 135 mph.
Range: ~ 450 miles.
Service Ceiling: ~ 15,000 ft.

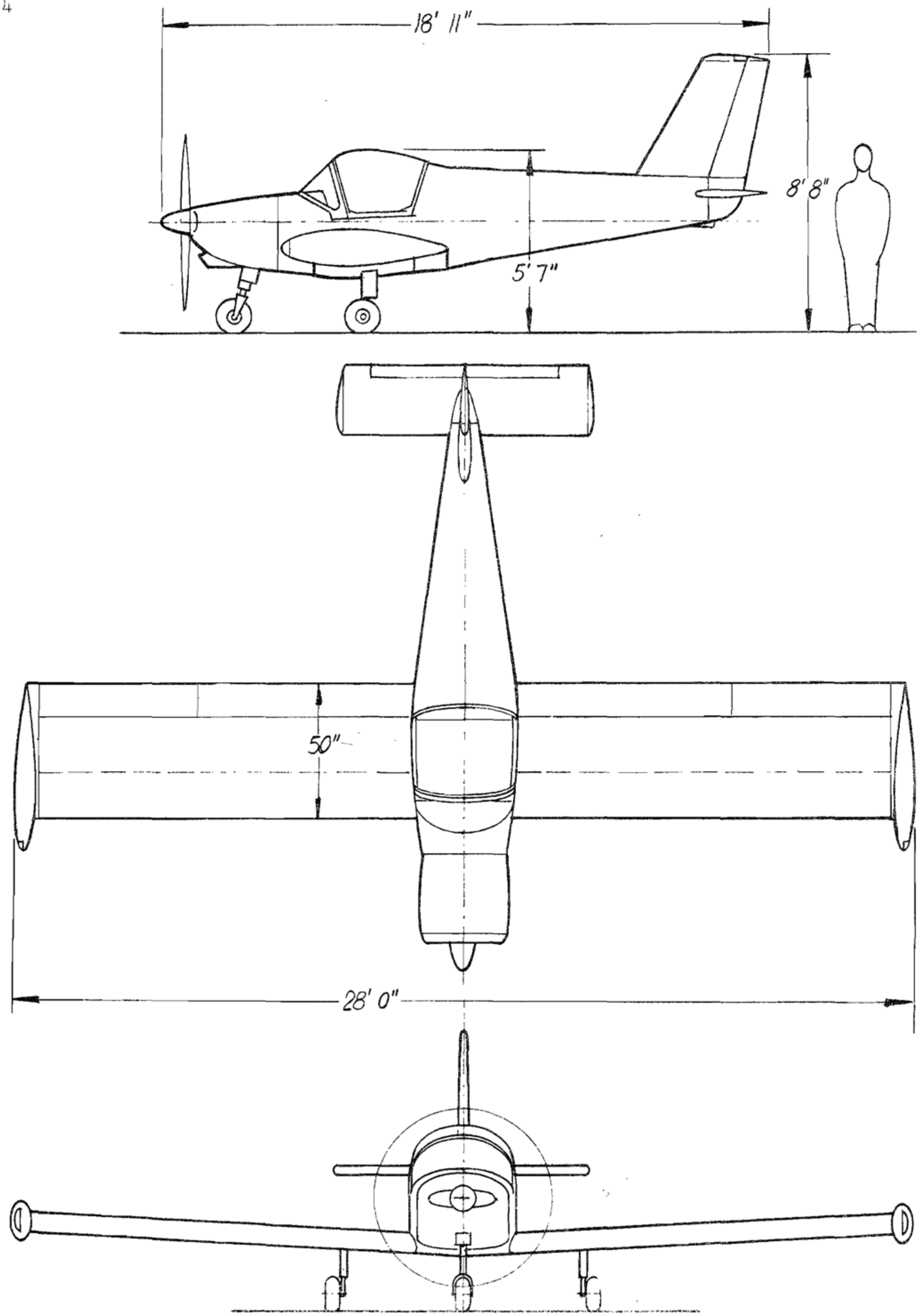


Fig. 1 - Pazmany PL-1 "Laminar" - General Arrangement SCALE 1:50

1-2 WHY AN ACROBATIC AIRPLANE?

CAR - Part 3 (Ref. 1) paragraph 3.186 establishes the following:

> "Maneuvering load factors (a). The positive limit maneuvering load factor shall not be less than the following values
>
> $$n = 2.1 + \frac{24,000}{W + 10,000} \qquad \text{Category Normal}$$
>
> except that n need not be greater than 3.8 and shall not be less than 2.5
>
> $n = 4.4$ Category Utility
>
> $n = 6.0$ Category Acrobatic

On paragraph 3.20 of the same regulation, Airplane categories are defined:

> "(1) Normal - Suffix N - Airplanes in this category are intended for non-acrobatic nonscheduled passenger, and nonscheduled cargo operation.
>
> "(2) Utility - Suffix U - Airplanes in this category are intended for normal operations and limited acrobatic maneuvers. These airplanes are not suited for use in snap or inverted maneuvers.
>
> "Limited Acrobatic Maneuvers' is interpreted to include steep turns, spins, stalls (except whip stalls), lazy eights, and chandelles.
>
> "(3) Acrobatic - Suffix A - Airplanes in this category will have no specific restrictions as to type of maneuver permitted unless the necessity therefor is disclosed by the required flight test."

Relatively few components are affected by the higher load factor due to the acrobatic maneuvers, while a great part of the structure is dimensioned by minimum practical gauges. Thus, with a small weight penalty, used to "beef-up" some critical parts such as the wing spar, the airplane can be designed to meet the requirements imposed by the "Acrobatic" category instead of the "Utility."

The additional strength also covers future possibilities of increasing the engine power. In such case the airplane could be reclassified in the "Utility" category with no change--or very minor changes, depending on the HP increase.

Another consideration which is becoming more and more important these days is the effect of wing tip vortices generated by fast flying highly loaded airplanes such as jet transports or bombers.

A CAB investigation report on the desintegration of a widely used personal airplanes relates that: "A light aircraft at 100 mph penetrating the vortices of a large jet aircraft at 90 degrees and one mile behind recorded an acceleration of plus 2.5 g and minus 3.5 g." Other aircraft at greater speeds have measured structural loads as high as 9 g in the wake of a large aircraft.

"When a large jet aircraft climbs at approximately 420 mph, the peak turbulence is 3-1/2 mi. in back and a relatively high degree of turbulence will exist at 7 mi. In relatively still air, the turbulence can persist for several minutes or long after the aircraft is out of sight. The study indicates that vortices can persist, theoretically, for as long as 30 minutes.

"Negative load factors higher than the minimum ultimate design requirements for Normal Category personal aircraft can reasonably be expected."

The previous report, only, should be enough to revise the existent CAR Part 3 to meet the problems of this "jet age."

1-3 WHAT CONSTITUTES A SAFE AND EASY TO FLY AIRPLANE?

"The flying qualities of an airplane may be defined as the stability and control characteristics that have an important bearing on the safety of flight and on the pilot's impressions of the ease of flying and maneuvering an airplane."

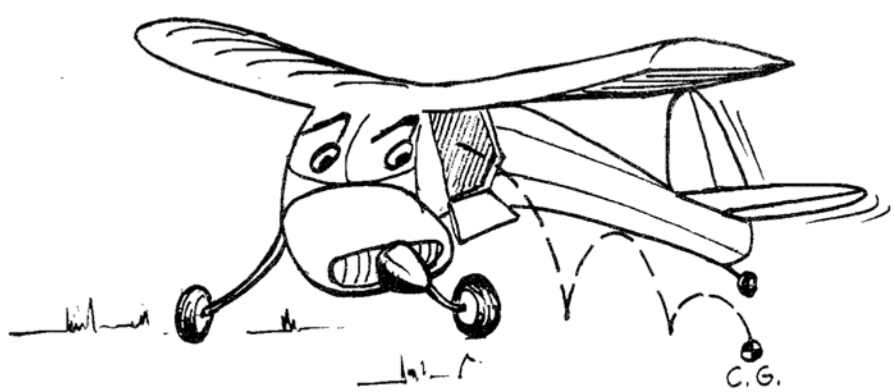
These words are reproduced from "NACA - Industry Conference on Personal Aircraft Research" - (Ref. 2). The reading of this publication is a must for the aircraft designer. The amount of experience and recommendations presented is so great that it is not possible to reproduce here. Following are a list of some of the papers included.

"History and Significance of Measured Flying Qualities"
"Flying Qualities Requirements for Personal Airplanes"
"Proportioning the Airplane for Lateral Stability"
"Design of Control Surfaces"
"A Flight Investigation to Increase the Safety of a Light Airplane"
"Factors Affecting Spinning of Light Airplanes"

Generally it can be said that a rough stability and control analysis is more important than a refined performance improvement.

It is often believed that the stability of an aircraft is only a function of the C.G. position. Many designers are satisfied when they balance the airplane at 25% of chord, but the stability is also controlled by other factors as will be outlined in the second part of this book.

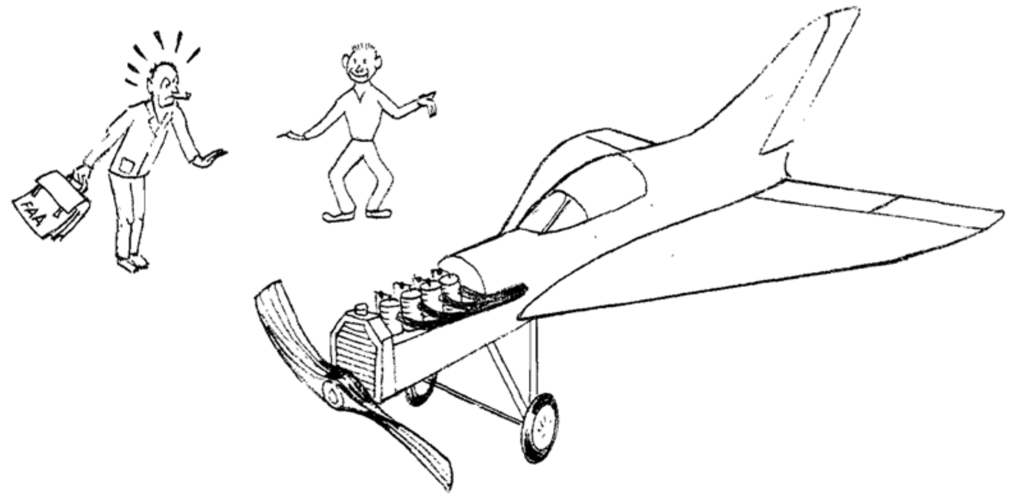
It is much safer to spend some time investigating at least some stability and control characteristics than it is to follow the usual method of "cut and try." Some designers are worried about the strength of a spar, yet they are completely careless about the location, size or travel of the elevator.



1-4 AESTHETIC OR FUNCTIONAL?

A functional airplane can be aesthetic, and by aesthetic it is meant that a certain harmony exists between the different components.

Of course that "certain" harmony depends upon the individual taste, but it is not difficult to recognize that an elliptical wing does not combine with a rectangular elevator; and obviously it would be useless to make a streamlined wheel fairing for an open cockpit airplane whose maximum speed is 60 miles per hour.



A fundamental idea in the design of an aircraft is to make it as small yet as functional as possible. In Figure 2 the side view of this airplane and a conventional two-place airplane are superimposed using the same scale.

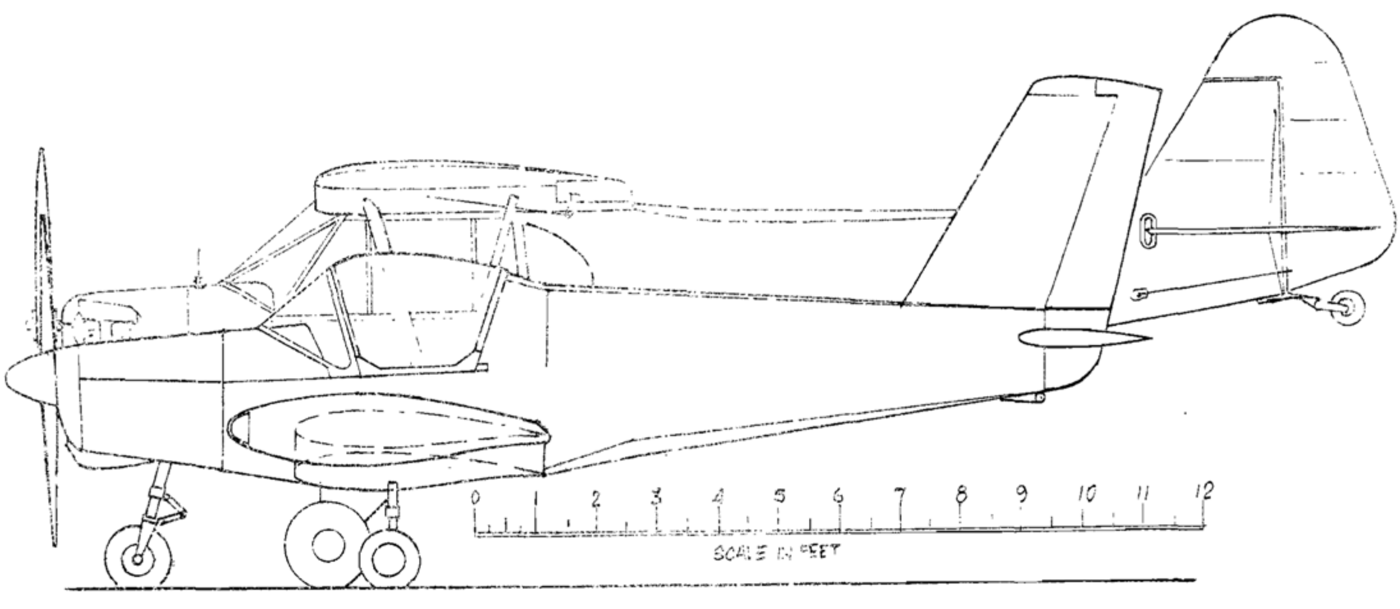
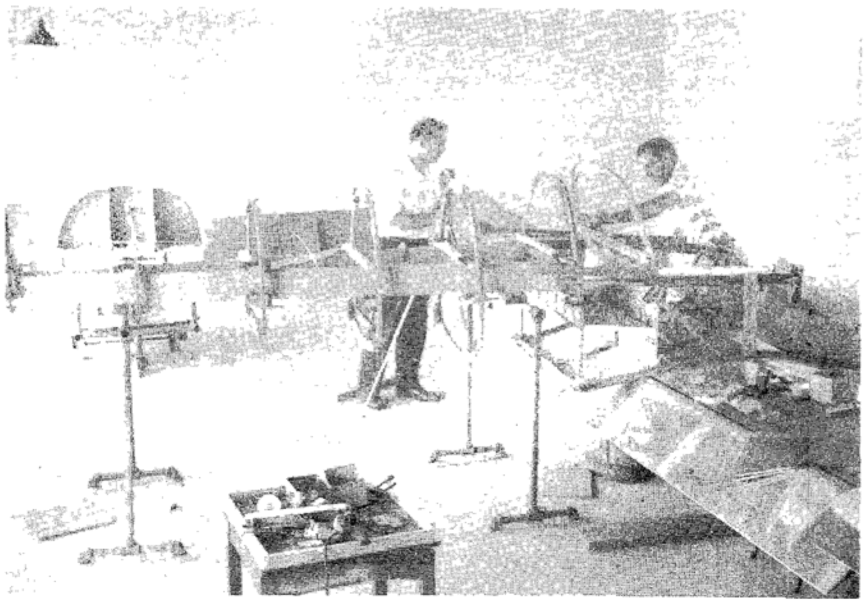
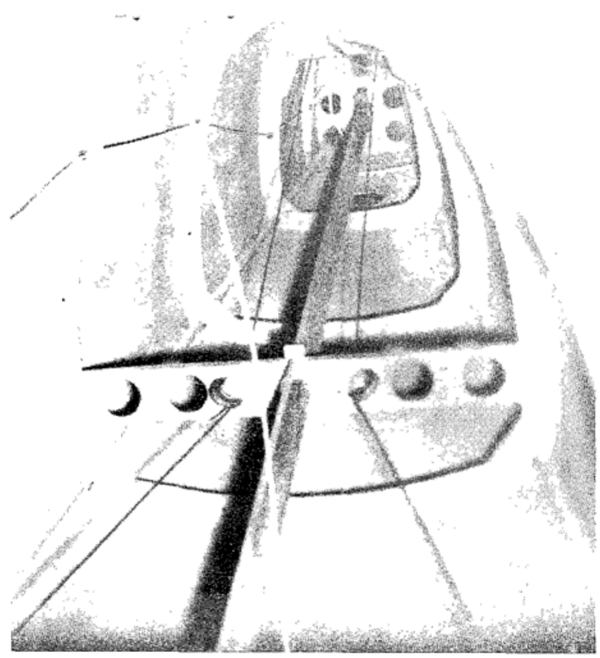


Fig. 2

1-5 WHY SEMI-MONOCOQUE CONSTRUCTION?

The Semi-monocoque construction is widely used for airplanes of this size and characteristics. The fuselage built around four longerons does not require complicated assembly jigs (see photo); and also, it is an efficient structure to transmit the loads. The stress analysis is simple; each side of the fuselage can be considered a beam, while the box formed by the four sides carries the torsional and shear loads.





1-6 CONSIDERATIONS ABOUT AN ALL-METAL AIRPLANE

a) Material

The Aluminum Alloy will be used for all the structural parts. The uniformity in quality is better than plywood or spruce. With a metallic skin, the expensive fabric finishing is eliminated, not only as an initial investment, but also at the periodical overhaul.

In the December 1945 issue of S.A.E. Journal (Transactions), a very interesting article was published. The title is "Wood vs. Metal Construction in Aircraft" by Herb Rawdon, Assistant Chief Engineer of Beech Aircraft Corp. In this article is related a comparison of wood and metal as material for aircraft construction based on the fortunate circumstance that the Beech Company was building an all-metal and a plywood covered airplane for the AAF at the same time.

After many interesting discussions and examples, one of the final considerations are: "The weight of the metal structure is less than wood, even in the smaller airplanes." At the end of the article there is a comparative table about weight of different materials used in the construction of equivalent Outer Wing Panels for the At-6 Aircraft.

TABLE 1

MATERIAL	WEIGHT	WEIGHT IN % OF AL.
Magnesium (riveted semi-monocoque)	158.6	87.4
Aluminum	181.5	100.0
Stainless steel	208.0	114.6
M 4610 steel	207.9	114.5
Magnesium (welded monocoque)	230.0	126.7
Plywood	296.0	163.0
Plastic plywood	293.0	161.0

b) Weight Comparison Between a Fabric and Magnesium Covered Wing

From the Volume 1 of the "Weight Handbook" of the S.A.W.E. (Society of Aeronautical Weight Engineers): (Ref. 3)

Page 3-11: Airplane Cotton Cloth-Mercerized: 0.0273 lbs./sq. ft.

Page 3-05: Flightex fabric (bare): 0.0281 lbs./sq. ft.
Fabric finish, regular 9 coat system: 0.0600 lbs./sq. ft.
This finish includes 4 coats Clear Nitrate Dope, 2 coats Pigmented Aluminum Dope, and 3 coats Colar Pigmented Dope.

Then:
Fabric...............0.0280 lbs./sq. ft.
Finish...............0.0600 lbs./sq. ft.
Tape and Stitching...0.0120 lbs./sq. ft.
Total .1000

Obtained from Page 33.03 of the same manual:

Fabric covering (including Tape, Stitching and Dope)
Unitary weight = 0.100 lbs./sq. ft.

This agrees with the previous value.

The estimated "wetted" wing area = 100 sq. ft. x 2 = 200 sq. ft.
Weight of fabric covering = 200 x .100 = 20 lbs.

Assuming that the leading edge of the wing up to the main spar will be covered with .040 inch magnesium "stressed" skin, while the remaining surface will be covered with .020 inch "non-stressed" magnesium skin:

Unitary weight of .040 Mg = .368 lbs./sq. ft.
Unitary weight of .020 Mg = .184 lbs./sq. ft.

.040 Mg surface = 35% of 200 sq. ft. = 70 sq. ft.
.020 Mg surface = 65% of 200 sq. ft. = 130 sq. ft.

Weight of .040 Mg sheet = .368 x 70 = 25.8 lbs.
Weight of .020 Mg sheet = .184 x 130 = 23.9 lbs.
49.7 lbs.

Additional weight due to protective coating $\cong$ 1 lb.

Total magnesium skin weight = 49.7 + 1 = 50.7 lbs.
Difference between fabric and Mg = 50.7 - 20 = 30.7 lbs.

This difference would be very near to the weight of the internal bracing necessary to carry the torsional and chord loads in a conventional two spar fabric covered wing.

The PL-1 airplane was originally designed with magnesium sheet covering all surfaces. In considering the use of this type material, the following objections arose:

(1) Special care is necessary to adequately protect magnesium against corrosion.

(2) Additional cost of magnesium as compared to aluminum.

(3) The possibility that this design will be released for the amateur builder may also confront him with the difficulty of procuring this material readily.

In view of these considerations, it was decided to use aluminum, resulting in a weight penalty.

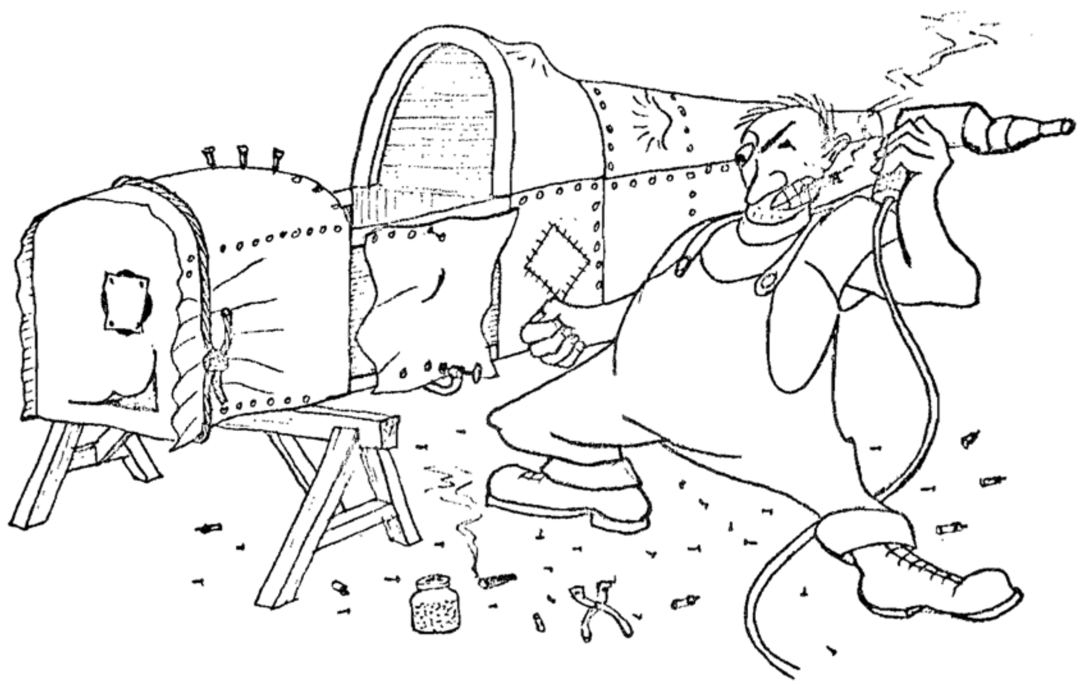
At the present time, there are many civil and military aircraft in operation incorporating the use of magnesium components. It appears that if appropriate measures are taken to prevent corrosion, no other problems arise. The best example is a Ryan Q-2C Firebee jet drone which drifted 13 months in the Pacific Ocean from the coast of California to near Hawaii. The Q-2C has several magnesium parts which remained in fairly good condition.

c) Aerodynamic Advantages

A relatively thick skin on the leading edge allows the use of countersunk rivets, and also reduces the wrinkles. Both conditions are very desirable to obtain some laminarity in the airflow, at least up to the maximum thickness of the airfoil. This is an ideal condition difficult to reach, but if obtained, will result in a general improvement in the performance.

d) Manufacturing Advantages

The construction of an all-metal small airplane does not require a big investment in tooling or machinery. Double curved parts much and can be avoided, but single curvature instead of flat panels is desired to reduce oil canning and improve stiffness. The inexperienced builder can learn the riveting process faster than the welding. A bad rivet can be detected, drilled out and replaced, while a burned welded joint requires more experience to be saved. Furthermore, a bad rivet in a row represents a small per cent strength reduction, while a bad welded joint can be a 100% strength loss.



1-7 WHY CANTILEVER WING?

With a 15% chord-thickness ratio airfoil, it is possible to build a cantilever wing with a weight comparable to a strut braced. It is not only the weight of the basic members that must be considered in the comparison, but also the extra fittings, bolts, turnbuckles, etc., along with the added loss in aerodynamic efficiency due to the additional parasite and interference drag. Finally there is the aesthetic consideration which was discussed on previous pages.

1-8 WHY DETACHABLE WING?

When the idea of folding wing is applied to a cantilever low wing airplane, the designer will find very serious problems such as heavy machined fittings, universal joints, control connections, and so on. So, as a compromise, "Detachable outer panels" were tried. The width of the center panel was fixed at 92" to comply with the highway regulations. A complete design was made, and final drawings of three different types of spar connections were developed. None of them looked satisfactory. They were too complicated and too heavy. So finally it was decided to redesign the whole wing, but this time with a different concept--"one-piece detachable wing."

This solution allows the possibility to tow the airplane to an airport or back home for maintenance or repair work once in a while.

The connection of the wing to the fuselage is made by means of two bolts at the main spar and about two dozen of platenuts and bolts along the wing drag angle.

The only control connection will be the elevator push-pull tube since the control column and the flap lever are integral parts of the wing. The seats are built integral with the wing also.

The "one piece" wing and the fuselage with the tail surfaces installed can be accomodated on a trailer and towed on a highway.

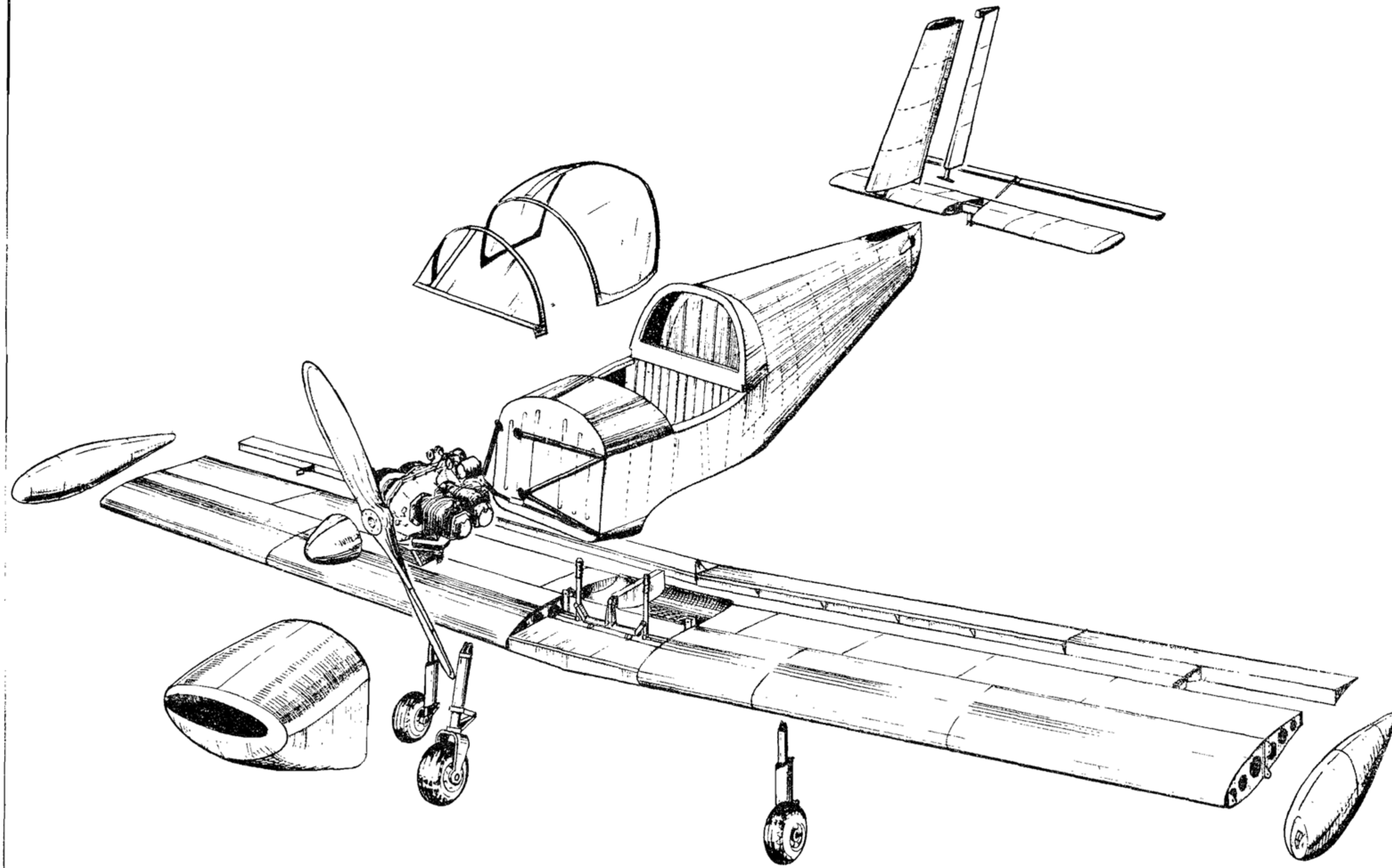


Fig. 3 - Pazmany PL-1 "Laminar" - Component Breakdown

1-9 WHY A RECTANGULAR WING?

From theoretical considerations and from pressure distribution tests, it can be demonstrated that the ideal wing form is the elliptical because it has the smallest induced drag. But using the same theory and tests, it was found that a rectangular wing of aspect ratio 6 has only about 5 per cent greater induced drag then that of an elliptical.

Between these two wing plan forms, there is the tapered, which has roughly one per cent more induced drag then the elliptical.

Both elliptical and tapered wings allow a lighter spar construction, but these advantages are of small importance when compared with the better stalling characteristics and simplified construction of a rectangular wing.

In the NACA Report 927 (Ref. 4), "Appreciation and Prediction of Flying Qualities," useful information can be found relating to the stalling characteristics as a function of wing plan form. Also, almost all the problems of the aerodynamic design are covered. If the designer does not have enough of a mathematical background, the formulas should be left out and the text read thoroughly and still provide many good ideas and basic knowledge.

Figure 4 shows the influence of the wing plan form on the stall properties. All wings are untwisted. It is evident that the rectangular planform has the best stall characteristics. The stall begins at the root of the wing progressing toward the tips, thus the ailerons remain effective while the center part of the wing is already stalled.

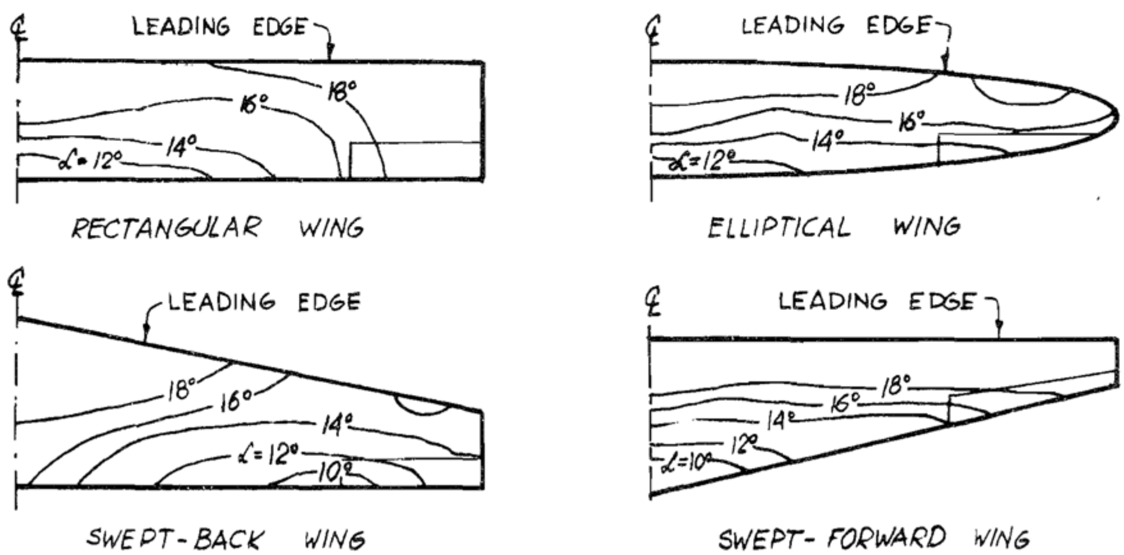
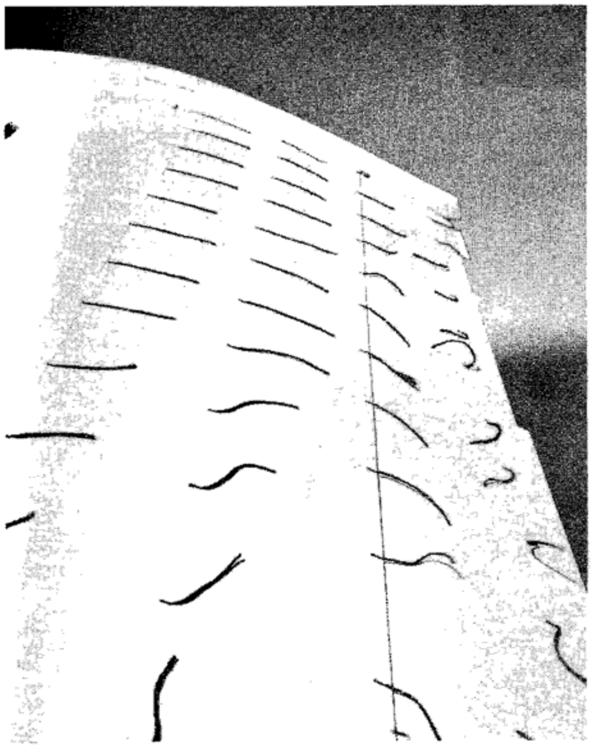


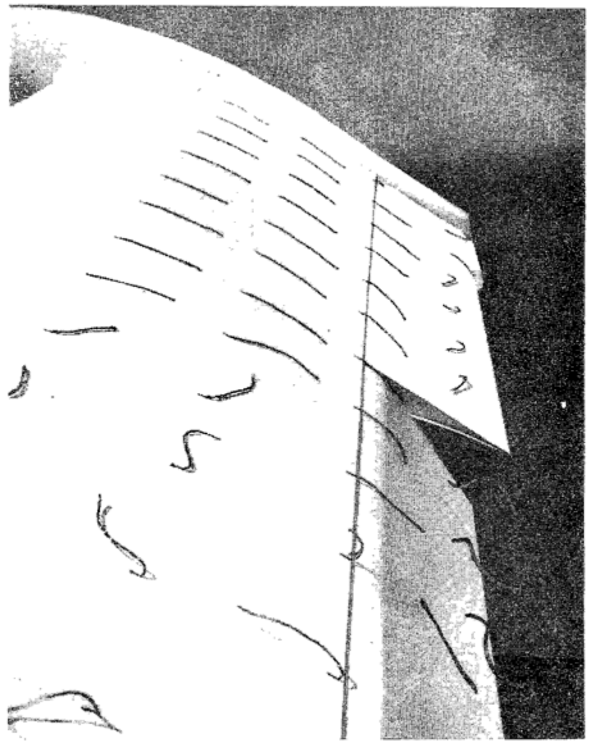
Fig 4- STALL PATTERN AS FUNCTION OF ANGLE OF ATTACK

Photos were made during stall investigations of the PL-L Laminar. The test conditions were 1400 rpm, 4000' altitude, and the stalls were approached very gradually so the angle of attack could be measured against the horizon with more or less accuracy.

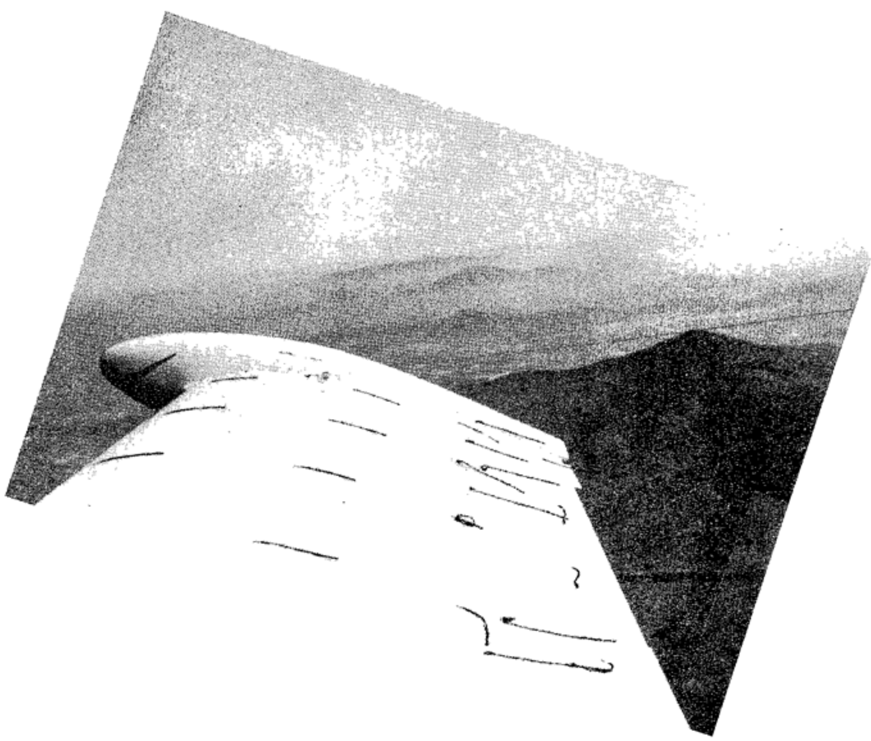
The first photo (see next page) shows the wing stalling with the flap retracted. The second photo shows the stall with maximum flap deflection.

In both photos, the tufts indicate that the outboard section of the wing is not stalled allowing very good aileron control all through.





The next photo was made during another test. This time, some tufts were attached to wire masts at 2.5" and 5" away from the wing surface. These tufts are out of the boundary layer and indicate attached flow while the tufts near the trailing edge and directly on the skin are oriented spanwise in the direction of the pressure gradient within the boundary layer.



1-10 WHY LOW WING?

The most dangerous parts of every flight probably are the take-off, landing, and flying the pattern. Visibility in a turn is greatly desired during these maneuvers. In a high wing aircraft the visibility during these critical moments is reduced mostly toward the inside of the turn. These considerations alone will decide the choice between high wing and low wing, but there are many others that can be enumerated.

Aircraft accident investigations and simple reasoning indicate that the more structure between the occupants and the ground, in case of crash, the higher are the possibilities of survival. A lot of energy can be dissipated in a low wing before starting with the passengers. In a high wing airplane crash, the energy will be dissipated by successfully collasping the landing gear, the fuselage nose, the occupants and finally the wing.

From aerodynamic viewpoints, the fuselage cross-sectional area of a low wing airplane could be made smaller than of a high wing; the occupants could be seated over the wing. In the PL-1, the seat is directly built-in between the main spar and the rear auxiliary spar. The seat sheet metal is also part of the carry-through torque box. In a high wing airplane, the occupants cannot sit directly on the floor because it will be very tiresome. Therefore, a seat has to be provided to take place of the previously mentioned torque box, but the high wing could not be lowered proportionally because the complete loss of visibility.

The interference drag of a high wing is generally smaller than of a low wing, but a good wing root fillet could reduce this disadvantage.

From structural considerations, the low wing has many advantages. The largest concentrated loads in a small airplane are the occupants; this load could be reacted directly by the low wing. This is not the case for a high wing where these loads should be transmitted from the seats to the fuselage up to the wing. No doubt that this represents a weight penalty.

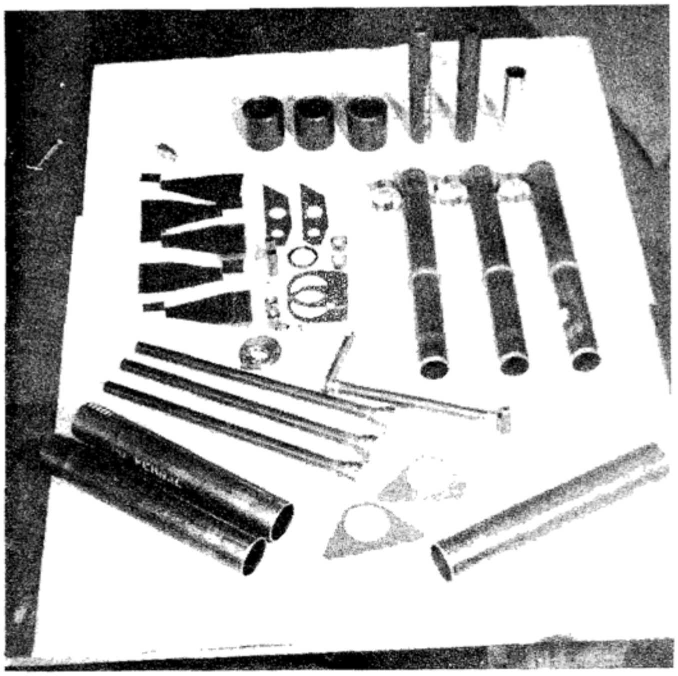
The door cut-out in the high wing airplane represents a weight penalty because the fuselage bending material has to be concentrated in very shallow beams, either under or over the door cut-out. This is mostly true in semi-monocoque type structures. In welded truss type fuselages, the door cut-out is generally designed into one of the truss modules.

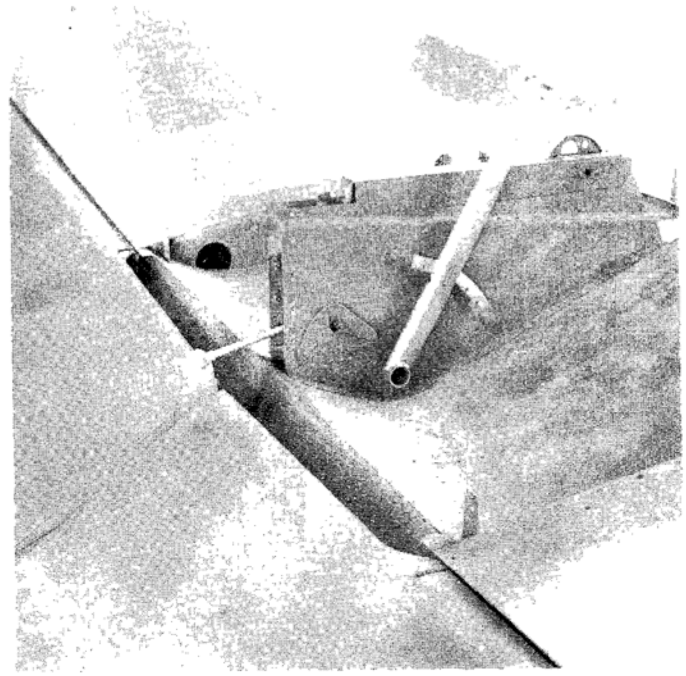
With semi-monocoque fuselages, sliding canopy and low wing arrangement, the fuselage bending material could be designed into two deep beams forming the fuselage sides, resulting in a considerable weight saving.

The main landing gear struts in a low wing airplane can be very short if they are attached to the wing spar. This represents a minimum weight and parasite drag. The wing spar does not need to be strengthened to take the landing gear loads because the air loads are critical. In the PL-1 airplane, the main gear and the nose gear shock absorbers are identical; therefore, a reduction in dissimilar parts has been achieved. (See photo next page)

The low wing configuration allows a running flap under the fuselage which provides a great increase in lift and drag when lowered. If we consider that the flap function is not only to provide high lift but to steepen the flight path, this is of appreciable value.

The flap and aileron control mechanism can be very simple in the low wing configuration. In the PL-1, the flap mechansim consists basically of a lever, a push-pull tube and a horn directly attached to the flap rib as shown in next photo. Obviously this simplicity cannot be achieved with a high wing arrangement.





1-11 CRASH WORTHINESS

Airplanes are not supposed to crash, but statistics have shown that a few of them do. High speed crashes in rough terrain are not survivable, but many crashes happen in such conditions that the chances of survival are great.

Every effort should be made to provide adequate protection to the occupants in case of a survivable crash. The wing tip tanks of the PL-1 are already a safety feature since the only fuel in the fuselage will be contained in the fuel lines.

The fuselage, safety belts, shoulder harness and the associated structure should be designed to take the ultimate accelerations specified by CAR 3.386, reproduced below.

TABLE 2

	Category	
Direction of Load	Normal & Utility	Acrobatic
Upward	3.0 g	4.5 g
Forward	9.0 g	9.0 g
Sideward	1.5 g	1.5 g

As a matter of comparison, the U. S. Navy requires 40 g's (ultimate) load factor for their airplanes. Airplane crash investigations have shown that even in very mild accidents, the occupants are subjected to accelerations well over 9 g's. On the other hand, investigations on the human tolerance to decelerations have shown that an adequately restrained body could tolerate up to 40 g's without injury.

Another design consideration is that there should be no heavy components or structure behind the occupants. A battery installed in the tail cone of the fuselage becomes a missile in a crash, and pusher engines should be directly forbidden by regulation.

CAR 3.386 establishes that penetrating or relatively solid objects should be avoided in the cockpit. To this can be added that the fuselage structure should be designed in such a way that it will bend or break outward, away from the occupants in case of an accident. The cockpit upper longerons in the PL-1 are curved outward so that under a compression load they will bend out. The instrument panel should be collapsible to avoid head injuries and the heavy instruments should be mounted on shear pins and as low as possible.

In other words, the two most important considerations in crash worthiness are:

1. If the cabin of the airplane did not collapse as a result of an accident, the occupants should survive.

2. Nearly 80 per cent of the aircraft accident fatalities are due to head injuries (Figure 5).

The next more dangerous detail in the cockpit is the control wheel. Again, statistics have shown that in many mild accidents where the cockpit remained survivable, the control wheels are slammed forward by the force of the crash, and if no shoulder harness is provided, the torso will be free to rotate and the head will hit the control wheel.

In other cases, the control wheel is pushed backward when the firewall collapses, producing chest injuries--just as in typical car accidents.

The PL-1 airplane has stick controls which, firstly, are short and very difficult to hit even with bent chest. Secondly, the elevator push-pull tube inertia, in case of a crash, will push the control sticks away from the occupants.

Fig 5

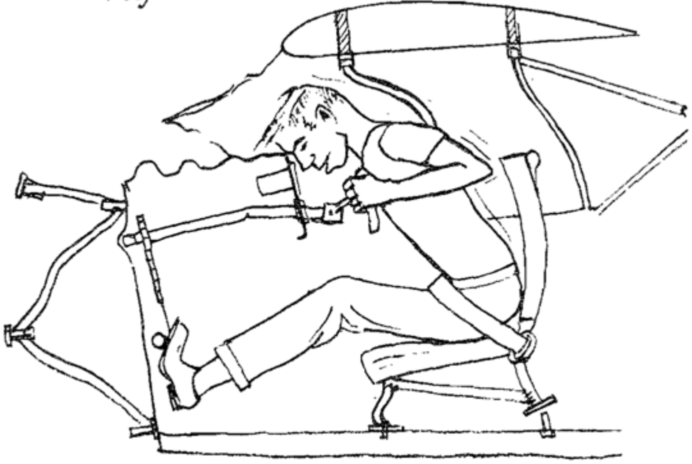
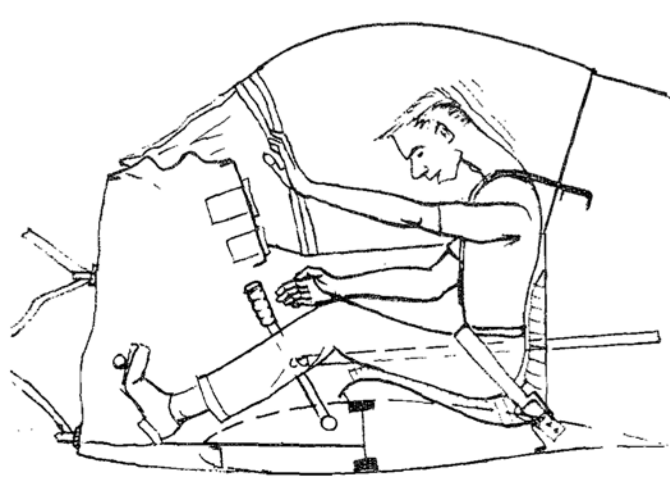


Fig 6



Another very common practice is the use of foam rubber seat pillows which have no energy absorption capacity. The sequence of what generally happens in a crash is as follows:

1. The airplane hits the ground and the structure starts collapsing. The occupant starts compressing the foam rubber cushion.

2. After the initial impact, the airplane structure rebounds but the passenger is still coming down with the initial speed because the foam rubber cannot provide any breaking reaction

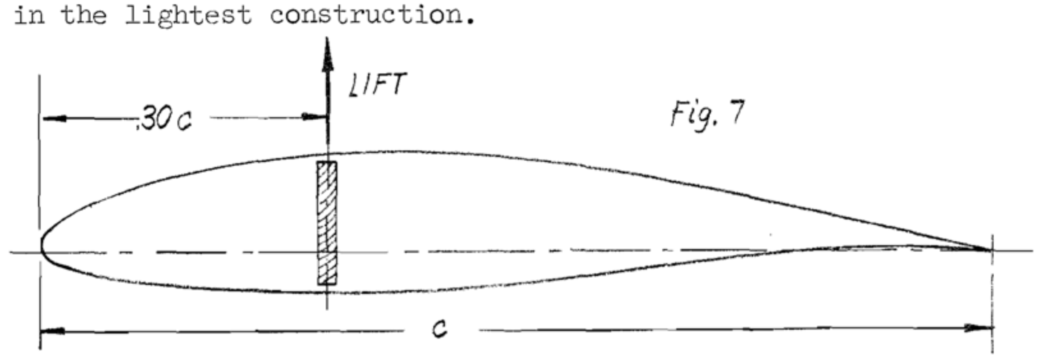
3. The cushion finally is completely depressed, and now the passenger hits the seat structure, still with the initial velocity; but the seat, due to the rebound, is already going up resulting in a "head on" collision. This type of load on a human body generally produces vertebral injuries. The solution is a crushable material such as foamed polystyrine or expanded polyvinyl chloride (commercially, ensolite). This last material can be cut, trimmed, cemented and heat formed.

There is a great amount of information about airplane crash worthiness. Institutions such as the Flight Safety Foundation, Inc.--2809 Sky Harbor Boulevard, Phoenix, Arizona--are devoted to this subject and will provide extensive data upon request.

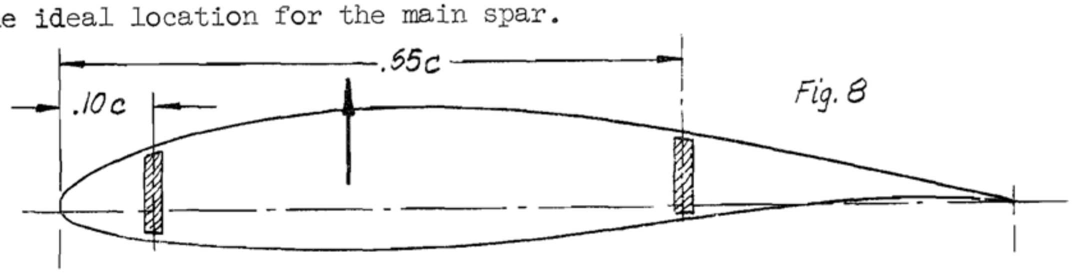
1-12 AIRFOIL SELECTION

a) Structural Considerations

The weight of a beam which has to carry a certain bending moment is inversely proportional to the square of the depth of the beam. Therefore the thickest airfoil will house the deepest beam, and this in turn will result in the lightest construction.



The lift force in an airfoil is approximately located at 30 per cent of the chord. If the maximum thickness is also at 30 per cent, obviously this will be the ideal location for the main spar.



In a two-spar wing, these considerations are not valid. The front spar is generally located at 10 per cent C, and the rear spar at 65 per cent C; both are points at shallower parts of the airfoil resulting in a heavier structure.

b) Aerodynamic Considerations

TABLE 3
Airplanes using "Laminar" Airfoils

Airplane	Country	Places	H.P.	Airfoil
Wassmer Super IV	France	4	180	63.618
Picchio F15	Italy	4	180	640 series
Aeromere Falco	Italy	2	150	640 series
Aviamilano Nibbio F14	Italy	4	180	640 series
Euklund	Finland	1	65	63_3-618
Heinonen	Finland	1	65	64_3A418 root 63_1A412 tip
Piper Cherokee	USA	4	150	65_2-415

TABLE 4
Gliders using "Laminar" Airfoils

Glider	Country	Weight(lbs)	Airfoil
Super Javelot (WA 22)	France	750	630 series
Standard (SF 26)	W. Germany	683	63_2-615
Rhönsegler (Ka 6CR)	W. Germany	661	63-618
D 34d	W. Germany	-	64_3-618
Eon 463	England	600	64_3-618/64_2-615
Standard (R-25)	Hungary	661	64_3-618
Mg 23	Austria	794	63-015
Strale (CVT-4)	Italy	661	64_2-515/64_2-512
Delfin 62	Yugoeslavia	701	63_3-618 M
Edelweiss (C-30)	France	838	700 series
Standard Austria	Austria	712	65_2-415
Foka (S2D-24)	Poland	688	63_3-618
Sagitta	Netherland	705	63_3-618
He 201	W. Germany	750	63_2-615
Vasama (PIK-16)	Finland)	617	63_2-615
Zefir 2	Poland	893	65_2-515
Assegai (BJ2)	South Africa	840	65_3-418
Movette (Br 901 S)	France	948	63 series
Skylark 4	England	830	63_3-620/6415
HKS 3	W. Germany	838	65_{215} 11^{16}
Favorit (Lom 61)	E. Germany	683	65_2-615.5
A-15	Russia	838	64_3-618/64_3-616
HP-10	USA	825	65-618 Mod.
Sisu-1	USA	711	65_3-418
Meteor	Yugoeslavia	1,113	63_2-616.5
EC 40	Italy	1,058	65-620/0009
Blanik (L-13)	Czechoslovakia	1,012	63_2A-615/63_2-A612
Capstan (T-49)	England	1,250	63_3-620/6412
Choucas (Br 906)	France	1,014	63-820/63-013
R-6	USA	1,226	63_2-615

Looking at Tables 3 and 4, evidently the newest "laminar type" airfoils were used in high performance airplanes and gliders all over the world. Unfortunately some designers seem to be reluctant to investigate the advantages of modern airfoils and recent designs are still using the prehistoric Clark Y or the obsolete 23012.

The following lines are reproduced from NACA TN 1945 (Ref. 7) page 14: "In the smooth surface condition, the two NACA 230- series sections are seen to possess extremely undesirable stalling characteristics at nearly all the Reynolds Numbers," and other: "In the rough surface condition, nearly all of the plain airfoils have good stalling characteristics at most Reynolds Numbers. The NACA 230- series sections and, at the higher Reynolds Numbers, the NACA 0012 sections are notable exceptions for even in the rough condition the stalling characteristics of these airfoils are rather undesirable."

The meaning of these words can be seen in Fig. 9 where Section Lift Coefficient curves for four different airfoils are plotted.

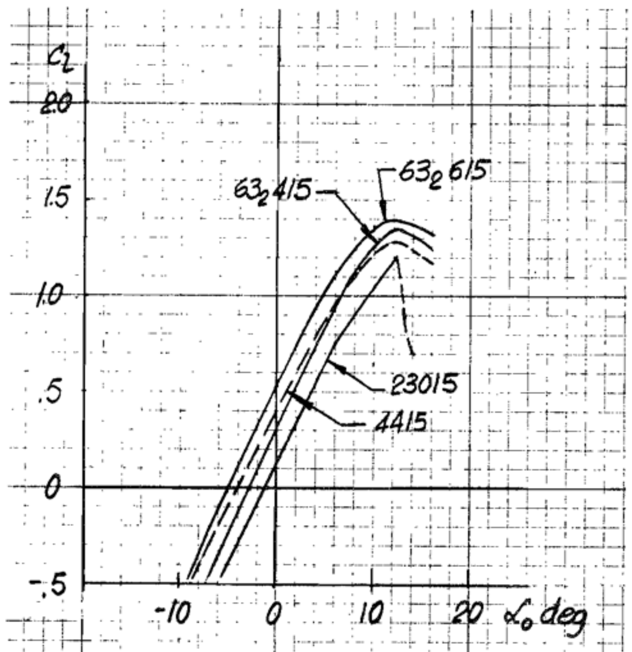


FIG. 9

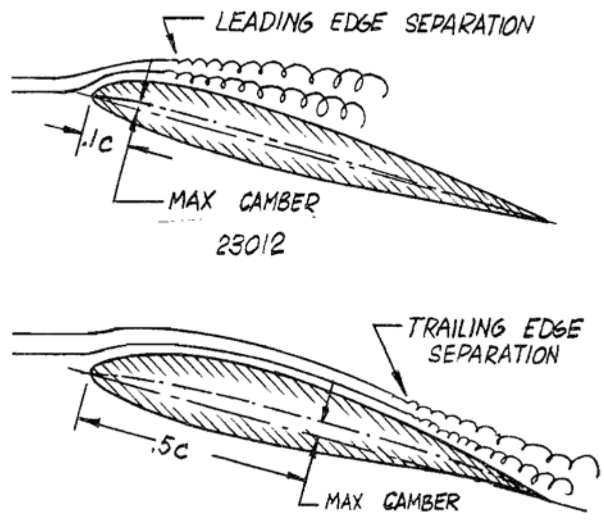
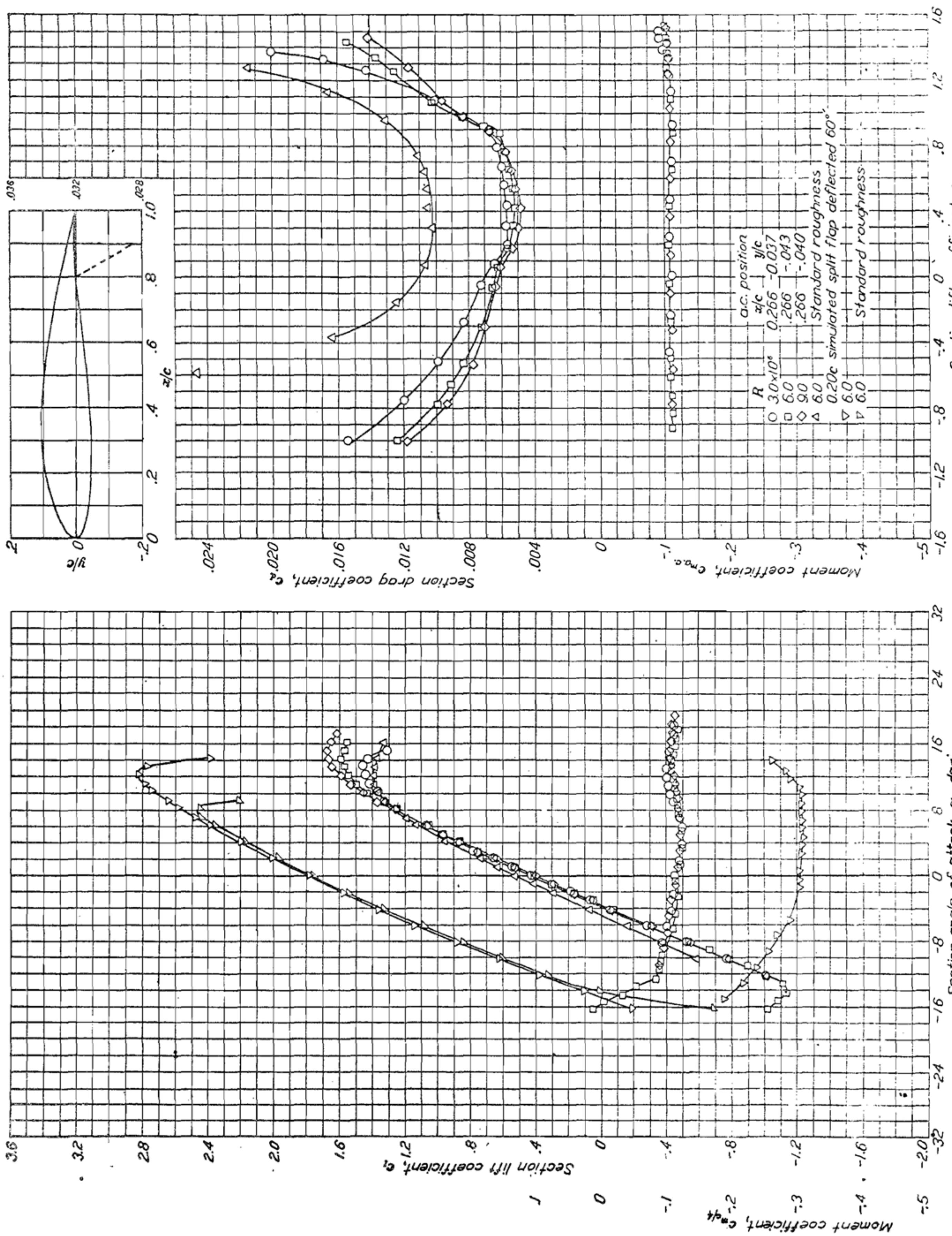


FIG 10 - FLOW SEPARATION

Looking at the 23015 curve it can be seen that it reaches a $C_{L_{max}} = 1.2$ and then drops sharply. This sudden loss of lift indicates a leading edge separation produced by far forward location of the maximum camber. The 4415, $63_2 415$ and the 63_2-615 airfoils show a gradual stall related to a more rearward location of the maximum camber as shown in Fig. 10.

In NACA "Industry Conference on Personal Aircraft Research" (Ref. 2) there is a paper titled "Development of Airfoils and High-Lift Devices" by L. H. Loftin Jr. from which the following paragraphs are reproduced:

"In any case, however, the characteristics of low drag airfoils are no worse than those of conventional airfoils and, if sufficient care is taken with the surface condition, definitive advantages are associated with their use."



Aerodynamic characteristics of the NACA 63_2-615 airfoil section, 24-inch chord.

On page 22 the aerodynamic characteristics of NACA 63_2-615 airfoil are reproduced from NACA Report 824. The curves in the right hand upper graph are the Section Drag Coefficients (c_d). The uppermost curve with "Δ" symbols represents the c_d at R.N. = 6,000,000 and "standard roughness." The lowest value is at c_l = .30 and the curve rises gradually at both sides of this point. The curves below with "O" "□" and "◇" symbols correspond to smooth airfoils.

These "smooth airfoil" curves are not representative of conventional light airplane wing surfaces, and therefore should not be used. On page 22 of NACA TR824, there is a chapter analyzing the "effect of surface irregularities on drag," which is very worthwhile reading. Some interesting thoughts for the amateur designer are reproduced next:

"It is important to maintain smooth surfaces even when extensive laminar flow cannot be expected, but the gains that may be expected from maintaining smooth surfaces are greater for NACA 6- or 7-series airfoils when extensive laminar flows are possible."

"It is known, at one extreme, that the surfaces do not have to be polished or optically smooth. Such polishing or waxing has shown no improvement in tests in the Langley two-dimensional low-turbulence tunnels when applied to satisfactorily sanded surfaces."

"Transition spreads from an individual disturbance with an included angle of about 15°. A few scattered specks, expecially near the leading edge, will cause the flow to be largely turbulent. Specks sufficiently large to cause premature transition on full-size wings can be felt by hand."

And on page 24, "All recent airfoil data obtained in the Langley two-dimensional low-turbulence pressure tunnel include results with roughened leading edge. This standard roughness is considerably more severe than that caused by the usual manufacturing irregularities or deterioration in service, but is considerably less severe than that likely to be encountered in service as a result of accumulation of ice or mud or damage in military combat."

In NACA TR824, the aerodynamic characteristics for many other airfoils can be found. For instance, on page 261, airfoil 66_4-221, the c_d curve for "standard roughness" rises rapidly at both sides of the Design Lift Coefficient, c_{l_i}, and for "smooth" airfoils, the rise is almost vertical; in fact, the c_d curve forms a bucket between c_l = -.3 and c_l = +.6. This is called the "Laminar Low Drag Bucket".

The characteristics of the NACA 6- and 7-series airfoils are coded in the airfoil numbering system. Assuming that the amateur is already familiar with the NACA 4 and 5 digit systems, a brief explanation of the meaning of the 6- and 7-series digits will be made:

63_2615

Series Designation

Position of Minimum Pressure or Approx. the Max. Thickness in % of chord

Maximum thickness, t/c

c_{l_i} in tenths

c_l range in tenths above and below of c_{l_i} in which low drag exists

The first digit (6) is the series designation. The second digit (3) denotes the chordwise position of minimum pressure in tenths of the chord behind the leading edge for the basic symmetrical section at zero lift. Figure 12 is reproduced from page 76 of TR824. The pressure distribution at $c_l = 0$ reaches a peak at 30% behind the trailing edge for the 63_2-015 airfoil.

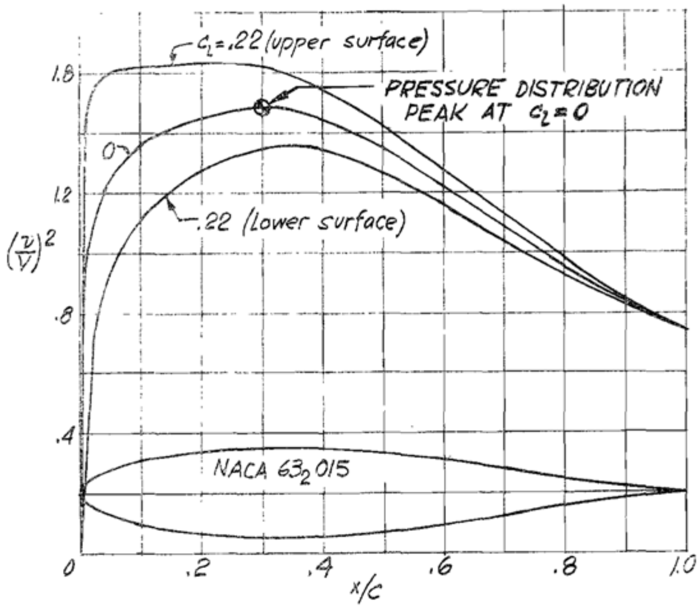


FIG. 12. PRESSURE DISTRIBUTION ON NACA 63_2 015

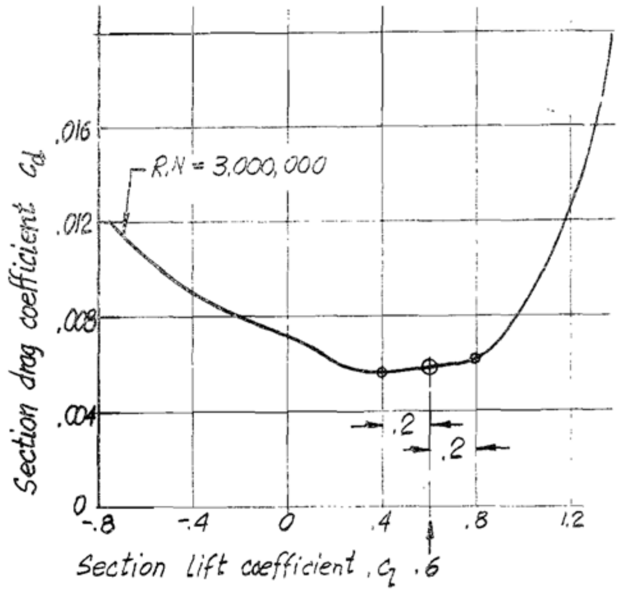


FIG. 13- DRAG BUCKET FOR NACA 63_2 615

The third digit and the fourth digit define the shape and location of the Laminar Low Drag Bucket illustrated in Figure 13.

The fourth digit (6) is the design lift coefficient, and the third digit (2) represents the low drag range at both sides of the design lift coefficient.

The determination of the Design Lift Coefficient (c_l) is described next. As most of the time the airplane is flying at cruise speed, this will be the "Design" condition to determine c_l:

Assume Cruise Speed = 110 mph

Then:

$$C_l = \frac{391}{V^2} \times \frac{W}{S} = \frac{391}{110^2} \times \frac{1250}{115}$$

$$= .347$$

$$C_l \cong .35$$

Where:

V = Speed in mph

W = Airplane Average Weight

$\cong$ 1250 lbs.

S = Estimated Wing Surface

= 115 sq. ft.

In Figure 14, the Section Drag Coefficients (c_d) for four different airfoils are shown.

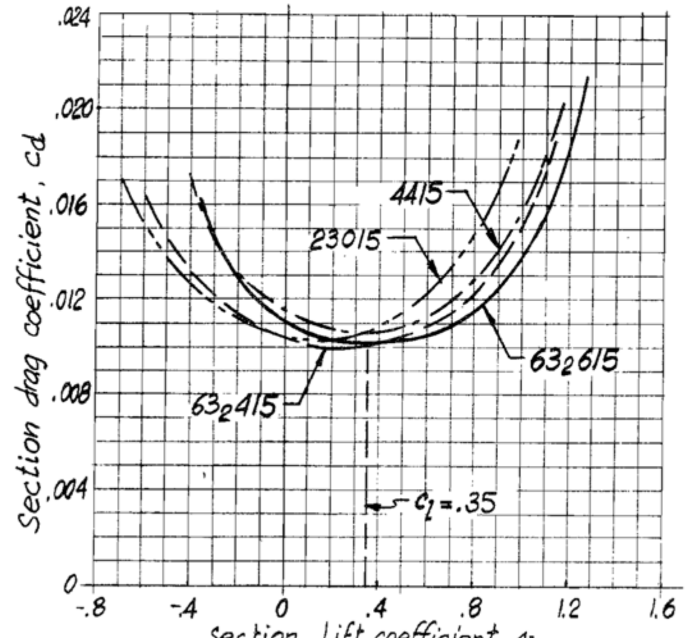


FIG. 14_ AIRFOIL DRAG COMPARISON

For the c_l = .35, the minimum c_d value is found on the 63_2415 airfoil curve. The number "4" in the airfoil designation indicates that the airfoil has a minimum c_d when c_l = .40.

The 63_2615 airfoil has almost the same c_d at c_l = .35, while the 23015 and the 4415 have higher values.

From the previous consideration, the 63_2415 will be the right choice, but in order to have a better ceiling, the 63_2615 was selected. When the airplane is flying at high altitude, the angle of attack is higher; this in turn means greater drag (c_d). At higher c_l the 63_2615 has less drag than the 63_2415.

Another reason in the selection is that, when the Section Lift Coefficient (c_l) curves are compared (see Figure 9), the 63_2615 has a $c_{l_{max}}$ = 1.39 while the 63_2415 has a $c_{l_{max}}$ = 1.32; therefore, landing speed without flap is slightly reduced.

The main disadvantage using the 63_2615 instead of 63_2415 is the greater moment coefficient $c_{m_{ac}}$, which for the 63_2615 is -.110 and for the 63_2415 is -.070, resulting in a proportionally larger trim drag. A greater negative elevator deflection for trim will be necessary to compensate for the larger nose down moment.

In the previous pages, the term "Reynolds Number" (abbreviated R.N.), was mentioned several times. Any good Aerodynamics text book will have a definition of the meaning of R.N. For practical purposes, the following equation can be used:

$$R.N. = v \times c \times 6{,}380$$

Where:

v = Speed in fps

c = Wing Mean Aerodynamic Chord

If we desire to calculate the R.N. of an airplane flying a 110 mph with a wing chord of 50 inches, we should proceed as follows:

Speed in fps = Speed in mph x 1.466

$$v = 110 \text{ mph} \times 1.466 = 161 \text{ fps}$$

Chord in feet = Chord in inches x $\frac{1}{12}$

$$c = 50 \times \frac{1}{12} = 4.17 \text{ ft.}$$

And substituting in the equation for R.N.:

$$R.N. = 161 \times 4.17 \times 6,380 = 4,280,000$$

In previous pages we have seen that most of the data presented by TR824 are for R.N. = 3,000,000, 6,000,000 and 9,000,000. It is evident that the Section Lift Coefficient (c_l) reaches higher values at the highest R.N.; on the other hand, the Section Drag Coefficient (c_d) is always smaller at highest R.N. If our airplane has a R.N. = 4,280,000 at a certain flying condition, it seems optimistic to use the values for R.N. = 6,000,000 (standard roughness). On the other hand, if the wing surface results in a very good quality, we may expect better values than the "standard roughness." Therefore, the "standard roughness", R.N. = 6,000,000 will be a good compromise for all practical purposes.

In Table 3 is listed the "Heinonen" airplane which uses 64_3 A418 airfoil at the tip. Also, in Table 4, the "Blanik L-13" glider uses a 63_2 A615 airfoil at the root and 63_2 A612 at the tip.

The meaning of the letter "A" in the code is that the basic airfoil has been modified to eliminate the trailing edge cusp as shown in Figure 15.

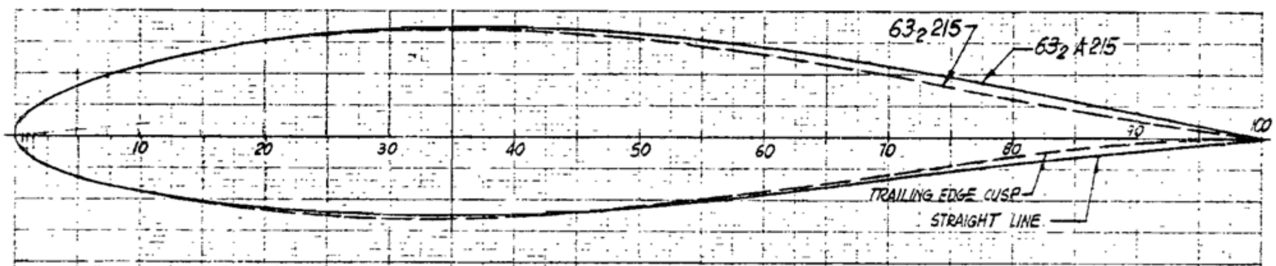


FIG. 15

The modified straight trailing edge airfoils are simpler to build and provide a deeper rib at the trailing edge which in turn will result in stiffer flaps or ailerons. The aerodynamic characteristics are practically the same as the original airfoils with the exception of the Moment Coefficient ($c_{m_{ac}}$) which is slightly more negative. The amateur builder could develop his own "A" modified airfoils using the information contained in TR824 and TR903 (Ref. 8).

As an example of this method, the ordinates for the 63_2A215 airfoil are calculated in Table 5.

1	2	3	4	5	6	7	8	9	10	11	12	13	14
	63_2A015 BASIC THICKNESS	FROM TR 903 Page 210 MEAN LINE a = 0.8 $c_{l_i} = 1.0$		MEAN LINE a = 0.8 $c_{l_i} = .2$		$x_U = x - y_t \cdot \sin\theta$ $x_L = x + y_t \cdot \sin\theta$				$y_U = y_c + y_t \cdot \cos\theta$ $y_L = y_c - y_t \cdot \cos\theta$			
x	y_t	y_c	$\tan\theta = dy_c/dx$	y_c'	$\tan\theta' = dy_c'/dx$	$\sin\theta$	$\cos\theta$	$y_t \cdot \sin\theta$	$y_t \cdot \cos\theta$	x_U	y_U	x_L	y_L
0.	0	0	—	0	—	—	—	0	0	0	0	0	0
.5	1.203	.281	.47539	.056	.0951	.09469	.99551	.1138	1.195	.3862	1.251	.6138	-1.139
.75	1.448	.396	.44004	.079	.0880	.08773	.99614	.1271	1.1440	.6229	1.519	.8771	-1.361
1.25	1.844	.603	.39531	.120	.0791	.07875	.99689	.1453	1.838	1.1047	1.958	1.3953	-1.718
2.5	2.579	1.055	.33404	.211	.0668	.06656	.99778	.1716	2.560	2.3284	2.771	2.6716	-2.349
5.0	3.618	1.803	.27149	.361	.0543	.05408	.99854	.1955	3.610	4.8045	3.971	5.1955	-3.249
7.5	4.382	2.432	.23376	.486	.04676	.04681	.99890	.2060	4.380	7.2940	4.866	7.7060	-3.894
10	4.997	2.981	.20618	.596	.04124	.04129	.99915	.2067	4.997	9.7933	5.593	10.2067	-4.401
15	5.942	3.903	.16546	.781	.03309	.03315	.99945	.1971	5.942	14.8029	6.723	15.1971	-5.161
20	6.619	4.651	.13452	.930	.02690	.02676	.99964	.1771	6.619	19.8230	7.549	20.1770	-5.689
25	7.091	5.257	.10873	1.051	.02175	.02181	.99976	.1546	7.091	24.8454	8.142	25.1546	-6.040
30	7.384	5.742	.08595	1.148	.01719	.01716	.99985	.1268	7.384	29.8732	8.532	30.1268	-6.236
35	7.496	6.120	.06498	1.224	.01299	.01309	.99991	.0982	7.496	34.9018	8.720	35.0982	-6.272
40	7.435	6.394	.04507	1.279	.00901	.00901	.99996	.0669	7.435	39.9331	8.714	40.0669	-6.156
45	7.215	6.571	.02559	1.314	.00512	.00512	.99999	.0369	7.215	44.9631	8.529	45.0369	-5.901
50	6.858	6.651	.00607	1.330	.00121	.00121	.99999	.0083	6.858	49.9917	8.188	50.0083	-5.528
55	6.387	6.631	-.01404	1.326	-.00281	-.00281	.99999	-.0179	6.387	55.0179	7.713	54.9821	-5.061
60	5.820	6.508	-.03537	1.302	-.00707	-.00707	.99999	-.0373	5.820	60.0373	7.122	59.9627	-4.518
65	5.173	6.274	-.05887	1.255	-.01177	-.01177	.99993	-.0608	5.173	65.0608	6.428	64.9392	-3.918
70	4.468	5.913	-.08610	1.183	-.01722	-.01722	.99985	-.0768	4.468	70.0768	5.651	69.9232	-3.285
75	3.731	5.401	-.12058	1.080	-.02411	-.02410	.99971	-.0899	3.731	75.0899	4.811	74.9101	-2.651
80	2.991	4.673	-.18034	.935	-.03607	-.03605	.99935	-.1077	2.991	80.1077	3.926	79.8923	-2.056
85	2.252	3.607	-.23430	.721	-.04686	-.04680	.99890	-.1054	2.252	85.1054	2.973	84.8946	-1.531
90	1.512	2.452	-.24521	.490	-.04904	-.04898	.99879	-.0740	1.512	90.0740	2.002	89.9260	-1.022
95	.772	1.226	-.24521	.245	-.04904	-.04898	.99879	-.0378	.772	95.0378	1.017	94.9622	-.527
100	.032	0	-.24521	0	-.04904	-.04898	.99879	-.0157	.032	100.0000	.032	100.000	-.032

EXPLANATION OF THE DIFFERENT STEPS IN TABLE 5

Column

1 and 2 - Ordinates for 63_2A015 Basic Thickness from page 206 of TR903

3 - Ordinates for Mean Line a = 0.8, $c_{l_i} = 1.0$ from page 210 of TR903

4 - Slopes for Mean Line a = 0.8, $c_{l_i} = 1.0$ from page 210 of TR903

5 and 6 - As the desired airfoil 63_2A215 has a $c_{l_i} = .2$, the values of columns 3 and 4 are multiplied by .2 to obtain column 5 and 6 respectively.

7 and 8 - Knowing "tan "Θ", the values of "sin "Θ" and "cos "Θ" are found in a trigonometric table.

9 - $y_t \sin\Theta$ = = 2 x 7

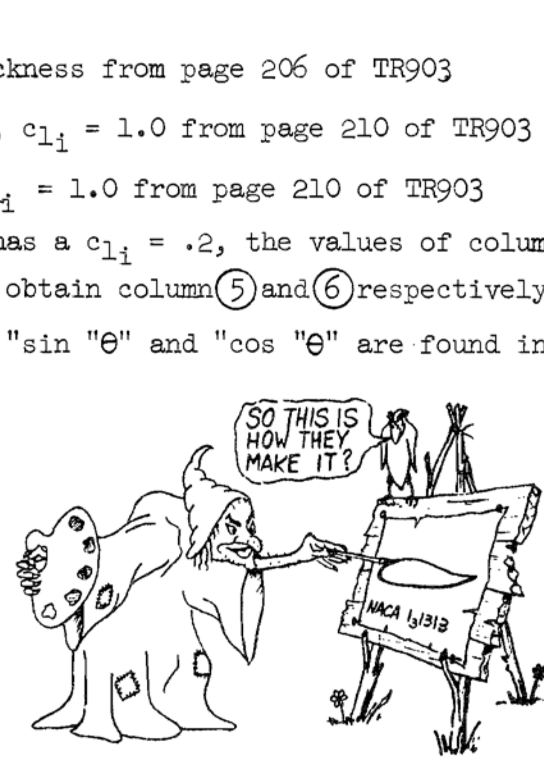
10 - $y_t \cos\Theta$ = = 2 x 8

11 - $x_u = x - y_t \sin\Theta$ = = 1 - 9

12 - $y_u = y_c' + y_t \cos\Theta$ = 5 + 10

13 - $x_l = x + y_t \sin\Theta$ = = 1 + 9

14 - $y_l = y_c' - y_t \cos\Theta$ = = 5 - 10



1-13 WING AREA DETERMINATION

The wing area is a function of the landing speed. In Table 6, the landing speeds or stalling speeds of several well known airplanes are listed. Unfortunately, some manufacturers list their landing speed as stalling speed. In "Airplane Performance Stability and Control" by Perkins & Hage, (Ref. 10), Page 199, the landing speed is estimated 15 per cent higher than the stalling speed.

TABLE 6 - Landing Speeds (V_L)

Piper PA-11	37 mph	Piper Colt	54 mph
Aeronca Champion	38 mph	Smith Miniplane	55 mph
Cessna 140	41 mph	Cessna 182	56 mph
Mooney Mite	43 mph	Stits Skycoupe	57 mph
Globe Swift	43 mph	Nesmith Cougar	59 mph
Luscombe Silvaire	45 mph	Beech Bonanza	60 mph
Fournier Ercoupe	48 mph	Meyers 200	62 mph
Cessna 150	50 mph	Whitman Tailwind	65 mph
Navion	53 mph	Heuberger Sizzler	68 mph

$$V_L = 1.15\ V_S$$

The desired stall speed for the Laminar is 50 mph; therefore, the landing speed will be:

$$V_L = 1.15 \times 50 = 57.5 \text{ mph}$$

This value seems fairly conservative compared with the values listed in Table 6.

The lift equation at sea level is:

$$L = W = \frac{S_W \times V^2 \times C_L}{390}$$

And solving for speed "V"

$$V = \sqrt{\frac{W \times 390}{S_W \times C_L}}$$

Where:

S_W = Wing area in square feet

V = Speed in mph

C_L = Wing lift coefficient

Looking at the last equation, we could do several things to reduce the landing speed.

First - Reduce Weight (W), which is always desirable, but rather difficult.

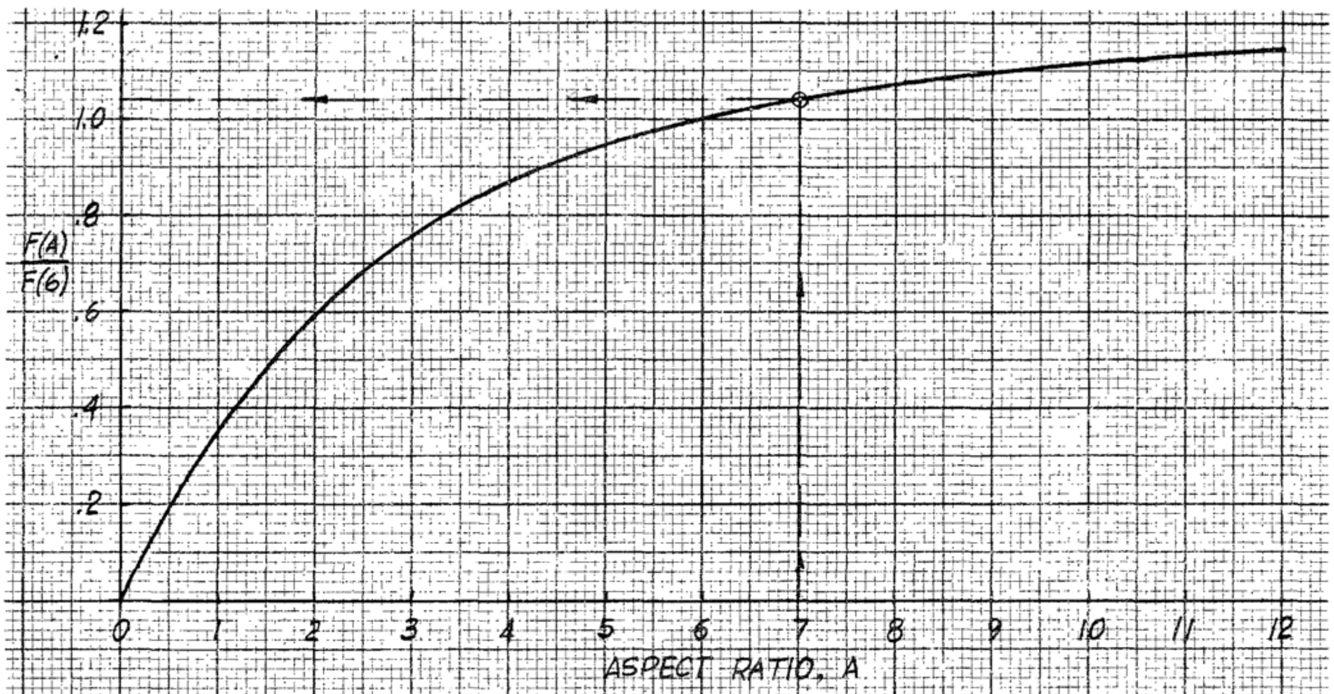
Second - Increase Wing Area (S_W). This is possible, but it will add weight and drag.

Third - Increase Lift Coefficient (C_l). This is probably the most appropriate term to work on. Airfoil selection and high lift devices are the ways to do it.

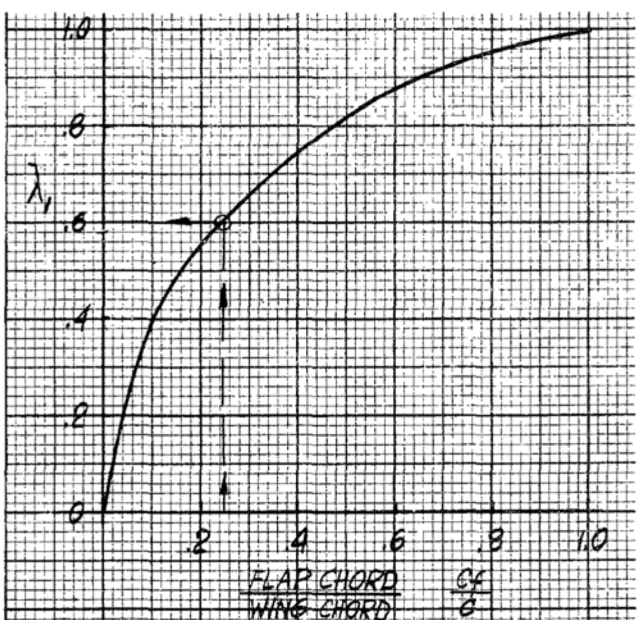
In the General Characteristics, Page 3, we listed "Flaps" in the High Lift Devices. Assuming that partial span plain flaps will be used, their effect on the $C_{L_{max}}$ will be calculated next.

A very good source of information on every kind of flaps is the British Aeronautical Research Council Report and Memorandum No. 2622, "The Aerodynamic Characteristics of Flaps" by A. D. Young (Ref. 9). In page 10 of this report, we found an equation to calculate the lift increment due to plain flaps.

$$\Delta C_L = \frac{F(A)}{F(6)} \cdot \lambda_1 \cdot \lambda_2$$



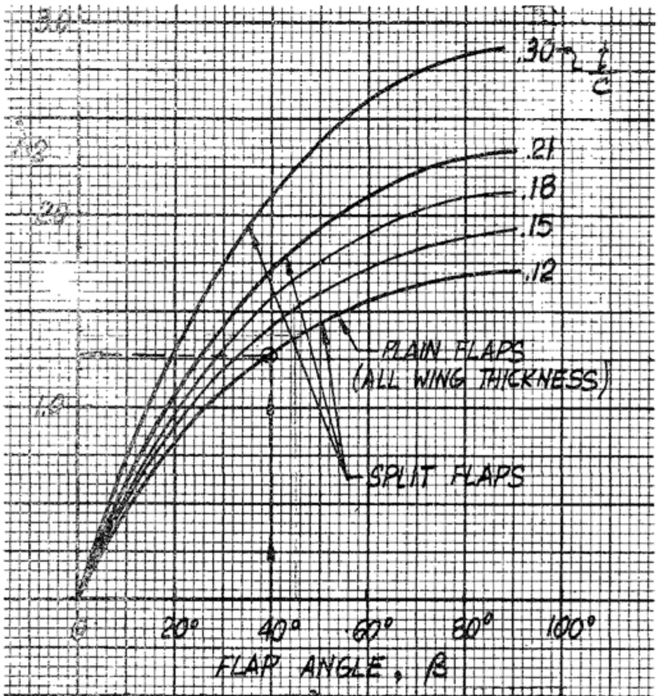
* FIG. 16 - VARIATION OF LIFT CURVE SLOPE WITH ASPECT RATIO



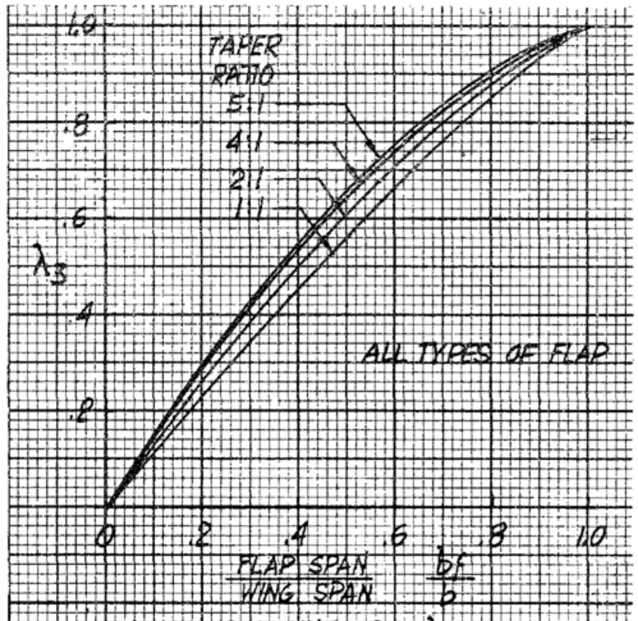
* FIG 17 - FUNCTION λ_1 - ALL FLAPS

$$\frac{c_f}{c} = \frac{\text{flap chord}}{\text{wing chord}} = .25 \xrightarrow{\text{Figure 17}} \lambda_1 = .6$$

* Note: Figures 16, 17, 18 and 19 are reproduced from R & M No. 2622 with the permission of the Controller of Her Britanic Majesty's Stationery Office.



* FIG. 18. FUNCTION λ2 - SPLIT AND PLAIN FLAPS



* FIG. 19. FUNCTION λ3

$\beta \cong 40°$ (flap angle) ⟶ Figure 18 $\lambda_2 = 1.27$

Replacing in the equation:

$$\Delta C_L = 1.04 \times .6 \times 1.27 = .792 \cong .79$$

The flap does not affect the whole wing; therefore, the lift increment just calculated must be reduced accordingly to the flap span. On page 11 of the same British report we found:

$$\Delta C_L' = C_L \times \lambda_3$$

$$\frac{b_f}{b} = \frac{\text{flap span}}{\text{wing span}} \cong .6 \longrightarrow \text{Figure 19} \rightarrow \lambda_3 = .67$$

$$\therefore \Delta C_L' = .79 \times .67 = .53$$

During the flare-out, the tail is producing a down load which should be subtracted from the wing lift, but the wing in proximity of the ground will develop a higher $C_{L_{max}}$. These two opposite effects are approximately of similar magnitude; therefore, we assume that they cancel each other.

The R.N. for stalling speed is calculated next.

$$R.N. = 6380 \times c \times V_S$$
$$= 6380 \times 4.17 \times 73.2 =$$
$$= 1,950,000$$

c = mean aerodynamic chord

= 50" = 4.17 ft.

V_S = 50 mph

= 50 x 1.466 = 73.2 fps

From Figure 11 on page 22, the $c_{l_{max}}$ for the plain 63_2615 airfoil at R.N. = 6,000,000 and standard roughness is

$$\text{Plain Airfoil} \quad c_{l_{max}} = 1.38$$

For the calculated R.N., this value will be reduced.

Also on Figure 11, we can read for the 63_2615 plain airfoil (smooth):

R.N.	$c_{l_{max}}$
9,000,000	1.66
6,000,000	1.58
3,000,000	1.46

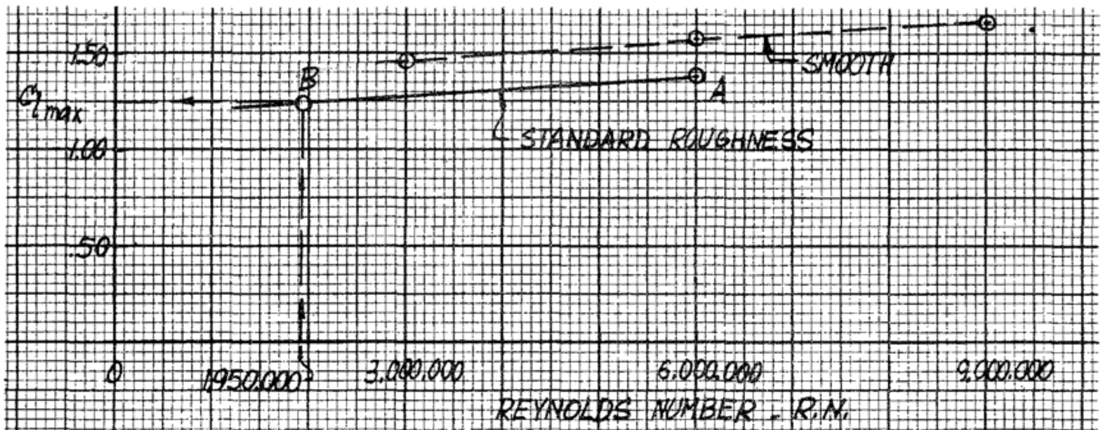


FIG. 20 - $C_{l\,max}$ CORRECTION FOR R.N.

In Figure 20, the three values of $c_{l_{max}}$ for the smooth airfoil are plotted. Point A is the single value for the rough airfoil ($c_{l_{max}}$ = 1.38). Assuming that the decrease of $c_{l_{max}}$ with reduction in R.N. is straight, a line is traced at point "A" parallel to the "smooth line", and by extrapolation point "B" is found at a R.N. = 1,950,000. At this point the $c_{l_{max}}$ will be 1.25.

To the corrected value of $c_{l_{max}}$ for the plain airfoil, we can now add the flap contribution calculated before.

$$\text{Flapped wing } C_{L_{max}} = \text{Plain Wing } C_{L_{max}} + \Delta C_L'{}_{(flap)}$$

$$C_{L_{max}} = 1.25 + .53 = 1.78$$

Now we have all the ingredients to calculate the wing area. The lift equation on page 28 can be solved for wing area:

$$S_W = \frac{390 \times W}{V^2 \times C_{L_{max_f}}} = \frac{390 \times 1300}{50^2 \times 1.78} = 114 \text{ sq. ft.}$$

A word of caution: The previous calculations are based mostly on wind tunnel data which are greatly affected by scale effects, tunnel turbulence and model finish. No adequate theories have been developed to correlate and explain the scatter in experimental data. Therefore, the calculated values could be off by a margin of plus or minus 10%, depending on the experience of the aerodynamicist.

1-14 ASPECT RATIO DETERMINATION

Aspect Ratio and Induced Drag are intimately related subjects. The significance of Aspect Ratio is well known, but the concept of Induced Drag might be rather obscure.

Induced Drag means "Drag Induced by the Lift," contrary to Parasite Drag, which could be produced by non-lifting bodies such as a landing gear.

A wing has both classes of drag, "Parasite" any time when moving, and "Induced" when lifting.

A car moving on a concrete road will have Parasite Drag, generated by the wind against the body. The engine has to produce a certain power to move the car at a certain speed. The same car riding on loose sand will sink continuously and will require a greater power to move at the same speed. This increase in power is due to Induced Drag. In an airplane, the wing is continuously supporting the weight on sinking air, and in fact has to climb out of the sinking air in order to maintain altitude. If the same car were equipped with wider tires, it would ride much easier over the sand. The wider tires do not sink so much. Another way to get out of trouble with standard tires is to go faster. The same thing happens with the airplane; a high Aspect Ratio wing, or flying faster will reduce the Induced Drag. Now that we are all convinced that a high Aspect Ratio is beneficial, the question is how much?

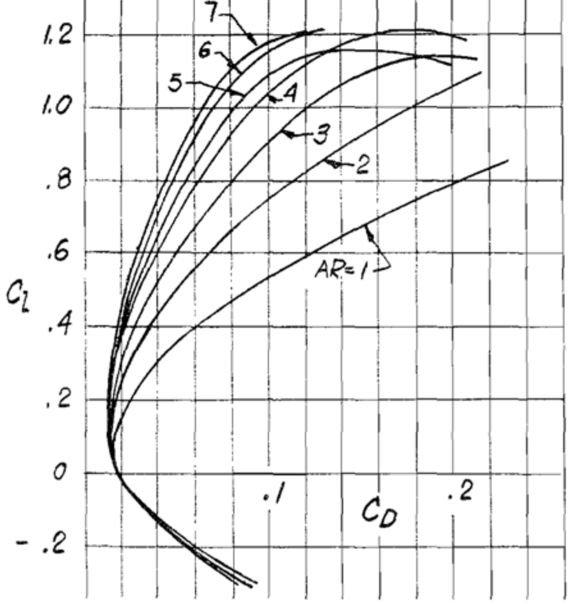


Figure 21 - Effect of Aspect Ratio on Wing Polar

In Figure 21, the polar diagrams for a series of wings with different aspect ratios are shown. It can be seen that there is not too much difference between AR = 7 and AR = 6 curves, but certainly there is a great difference between AR = 2 and AR = 1.

For light conventional airplanes a good compromise aspect ratio is about 7. A smaller aspect ratio will result in excessive Induced Drag, penalizing mostly the climb and ceiling. On the other hand, aspect ratios over 7 result in excessively reduced wing chords. When we calculated the Stalling Speed, the Reynolds Number effect on $C_{L_{max}}$ was analyzed and we remember that the smaller the R.N., more reduction in $C_{L_{max}}$. Remember that R.N. is a direct function of the wing chord.

From structural view point, the advantages of a small aspect ratio are double; first, because a larger chord will provide a proportionally larger depth for the wing spars, second, a shorter wing represents smaller bending moments, which in turn requires lighter spars.

We calculated the wing area as 115 sq. ft.

The equation for A.R. is:

$$AR = \frac{b^2}{S_W}$$

Solving for b (wing span):

$$b = \sqrt{AR \times S_W}$$

And substituting values:

$$b = \sqrt{7 \times 115} = \sqrt{805} = 28.3 \text{ ft.}$$

A round number will be easier to remember, so let us make the span 28 ft.

The average wing chord is calculated next:

$$c = \frac{S_W}{b} = \frac{115 \text{ ft.}^2}{28 \text{ ft.}} = 4.12 \text{ ft.} = 49.3''$$

Since we have no taper, c is constant along the span.

Again, a round number will be more convenient, so we make the chord $c = 50''$.

Based on these rounded figures, we recalculate the wing area.

$$S_W = b \times c = 28 \text{ ft.} \times \frac{50 \text{ in.}}{12} = 28 \times 4.17 = 116 \text{ sq. ft.}$$

And the final aspect ratio will be:

$$AR = \frac{b^2}{S_W} = \frac{28^2}{116} = 6.76$$

In the Laminar airplane it was decided to use wing tip fuel tanks, mostly based on safety reasons. By making the tank of elliptical shape (see Figure 22), the height is increased. A relation of 2 to 1 was selected for the major and minor axis, resulting in a maximum height of 14.8". The fuel tank is aerodynamically equivalent to an end plate.

End-plates increase the wing aspect ratio and this increase can be calculated with the following equation from "Fluid-Dynamic Drag" by S. F. Hoerner, (Ref. 11) Page 7-10, Equation 18.

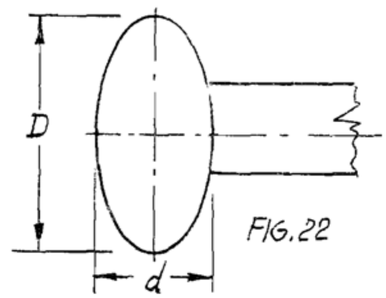


FIG. 22

$$\Delta A_i \cong A \times 1.9 \times \frac{h}{b}$$

Where h is the height of the plate and b is wing span,

$$h = 14.8'' = 1.23'$$

$$b = 28'$$

$$\Delta A_i = 6.76 \times 1.9 \times \frac{1.23}{28} = .57$$

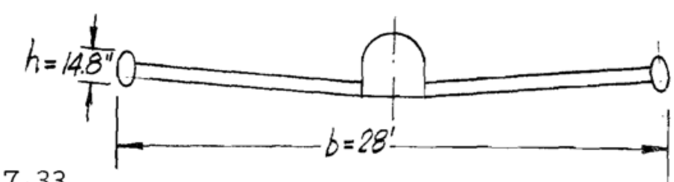


FIG. 23

The corrected aspect ratio will be:

$$A_i = A + \Delta A_i = 6.76 + .57 = 7.33$$

Other equations given by Hoerner are based on the area of the end plate:

$$\Delta A_i = A \times 1.1 \times \frac{S_{2E}}{S_W}$$

Where S_{2E} = Area of 2 end plates

S_W = Wing area

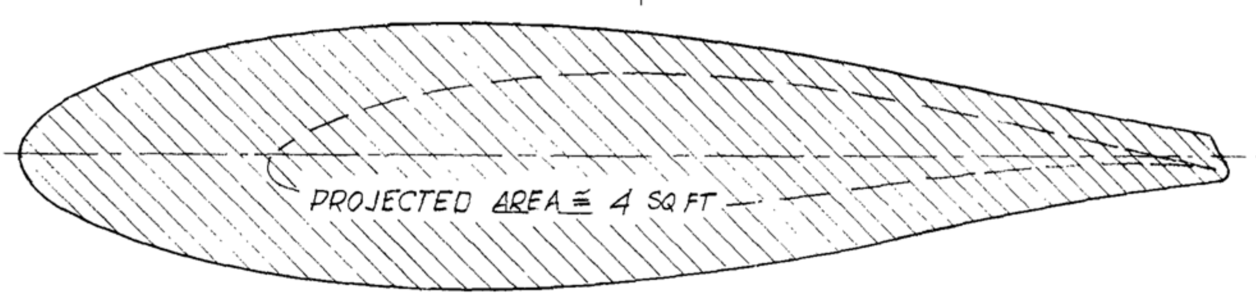


FIG. 24

$$S_{2E} = 4 \times 2 = 8 \text{ sq. ft.}$$

$$\Delta A_i = 6.76 \times 1.1 \times \frac{8}{116} = .51$$

The corrected aspect ratio will be:

$$A_i = A + \Delta A_i = 6.76 + .51 = 7.27$$

The two equations are in good agreement, we can use

$AR_i = 7.30$ for future calculations

1-15 WHY FLAPS?

From a constructional view point, it is easier, in this project, to make the wing with flaps than without them. The aileron and flap ribs will be exactly the same, otherwise a special form block would be necessary for continuous ribs. Following the same thinking, both aileron and flaps are piano-hinged at the bottom skin.

The proposed flap will be plain type, which is the simplest, but not the most effective. The increase of $C_{L_{max}}$ provided by this type of flap is relatively small ($\Delta C_{L_{max}} = .53$), but still will provide a landing speed reduction calculated as 10 mph.

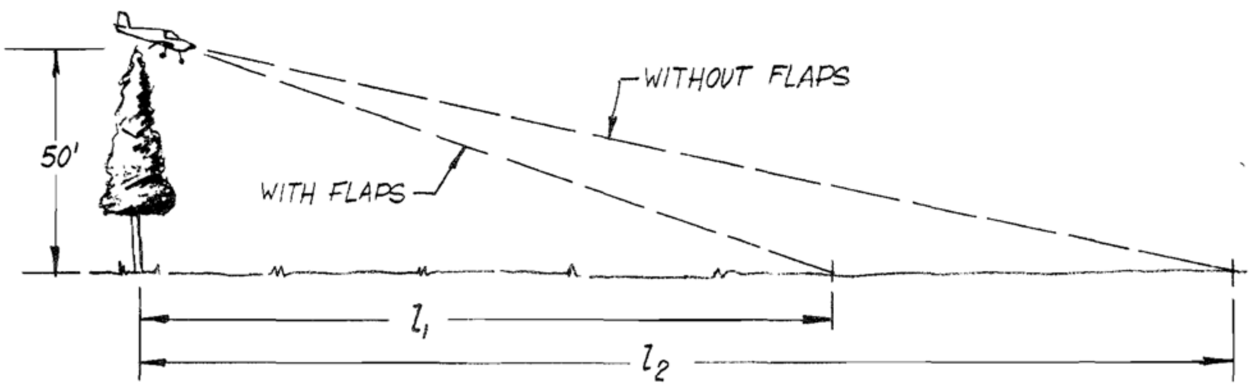


FIG 25. GLIDE PATHS

The second advantage of wing flaps is the increase in drag, resulting in a steeper glidepath, as shown in Figure 25. The landing distance over an obstacle of 50' can be calculated with a simplified equation given by AGARD Report No. 81 (Some Factors Affecting the Field Length of STOL Airplanes - Ref. 19).

$$S_L = 160 \sqrt{\frac{W/S}{c_{l_{max}}}} + \frac{510\ W/S}{a \cdot c_{l_{max}}} \quad \text{(ft)}$$

Where:

$W/S = \frac{1300}{116} = 11.2$ lbs/sq. ft.

$C_{L_{max}} = 1.25$ (Plain Wing)

1.78 (Flapped Wing)

$a = -7$ ft/sec^2 (Ground Run deceleration)

Then for the airplane without flaps:

$$S_L = 160 \sqrt{\frac{11.2}{1.25}} + \frac{510 \times 11.2}{7 \times 1.25} = 480 + 656 = 1136 \text{ ft.}$$

And for the airplane with flaps:

$$S_L' = 160 \sqrt{\frac{11.2}{1.78}} + \frac{510 \times 11.2}{7 \times 1.78} = 402 + 498 = 900 \text{ ft.}$$

1-16 WING AERODYNAMIC CHARACTERISTICS

The section aerodynamic characteristics of the 63_2615 airfoil were given in Figure 11. These curves represent an Infinite Aspect Ratio wing and cannot be used directly to determine the actual wing characteristics. Corrections based on Aspect Ratio should be made. For preliminary design, only the lift curve with flap down and flap up are necessary.

In Figure 26 the section lift curves for the 63_2615 (R.N. = 6,000,000 -- Standard Roughness) are reproduced. The slope of the lift curve can be determined from this plot. Within the straight portion of the curve, select a convenient angle of attack increment ($\Delta\alpha_0 = 10°$). Read-out the corresponding increment in the lift coefficient; in this case, $\Delta c_1 = 1.05$. The slope for the section (Infinite Aspect Ratio) will be:

$$a_o = \frac{\Delta c_1}{\Delta \alpha_0} = \frac{1.05}{10} = .105$$

The lift curve slope for a Finite Aspect Ratio can be calculated with the following equation from NACA TR665, "Calculation of the Aerodynamic Characteristics of Tapered Wings with Partial Span Flaps" (Ref. 14).

$$a = f \frac{a_o}{1 + \left(\frac{57.3\, a_o}{\pi A}\right)}$$

The slope lift curve factor "f" is plotted in Figure 27 also from NACA TR665.

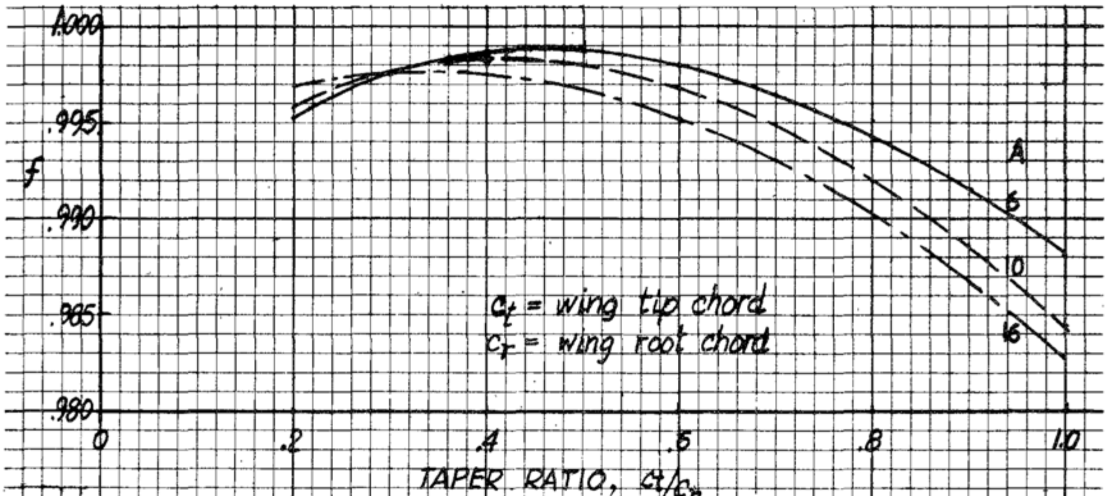


FIG. 27. SLOPE OF LIFT CURVE FACTOR, f.

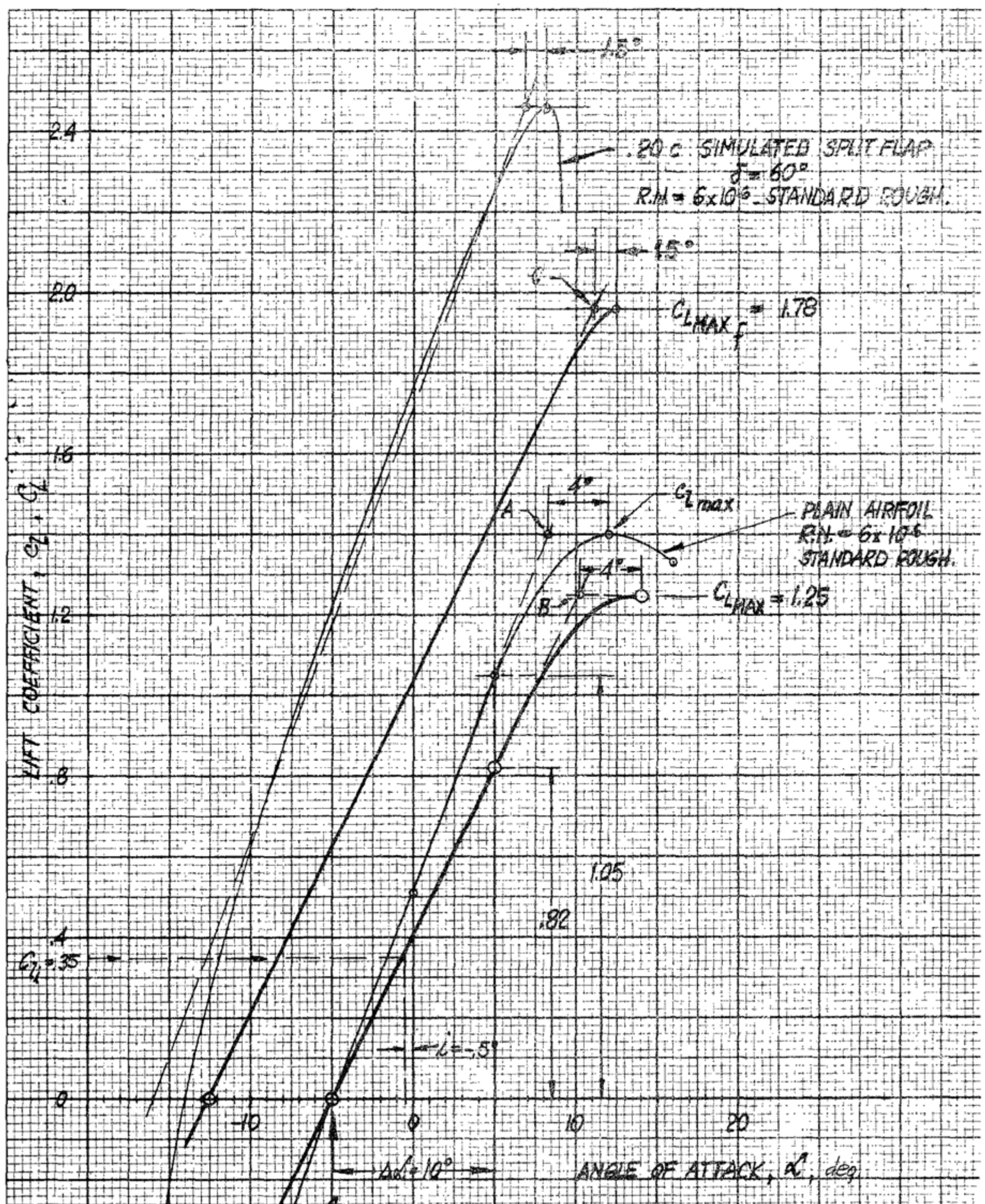


Figure 26 - AERODYNAMIC CHARACTERISTICS OF WING

For the Laminar PL-1 rectangular wing: (from Figure 27).

$$\left.\begin{array}{l} c_t/c_r = 1 \\ A = 7.3 \end{array}\right\} \quad f = .988$$

Then:

$$a = .988 \frac{.105}{1 + \left(\frac{57.3 \times 1.05}{3.14 \times 7.3}\right)} = .082$$

And from the lift curve slope equation, solving for Δc_L

$$\Delta c_l = a \times \Delta\alpha$$

And if we select again $\Delta\alpha = 10°$:

$$\Delta c_l = .082 \times 10 = .82$$

These values are plotted on Figure 26. Note that the angle for zero lift ($\alpha_{l_o} = -5°$) does not change with aspect ratio. Then, the straight portion of the lift curve could be traced. The curved upper part of the curve could be approximated with the following method.

Project the straight portion of the section lift curve up to the level of $c_{l_{max}} = 1.40$, (Point "A"). Measure the distance in degrees between Point "A" and $c_{l_{max}}$, which is 4° for this airfoil.
On page 31, we found that the $C_{L_{max}}$ for the plain wing was 1.25. Therefore, trace a line at this level. Project the straight portion of the lift curve up to intersect the $C_{L_{max}}$ line; obtain point "B". Measure 4° from point "B" to locate the point for $C_{L_{max}}$. Using the same curve as used for the section lift curve, complete the wing lift curve.

Somewhat similar procedure can be used to trace the flapped wing lift curve. Some simplifying assumptions could be made: (1) The flapped airfoil lift curve is parallel to the unflapped airfoil lift curve. This is substantiated in Figure 26, where the curve for the 63_2615 with a split flap deflected 60° is shown. The dashed line was traced parallel to the plain airfoil slope.

The zero lift angle for the flapped airfoil $\alpha_{l_{of}}$ should be calculated first.

In Figure 28, a series of curves are plotted, which gives the displacement of the zero lift angle ($\Delta\alpha_{l_o}$) as a function of flap chord and angle for plain flap.

$$\left.\begin{array}{l} \text{For } c_f/c = .25 \\ \text{and } \delta_f = 40° \end{array}\right\} \xrightarrow{\text{Fig. 28}} \Delta\alpha_{l_o} = -12.5°$$

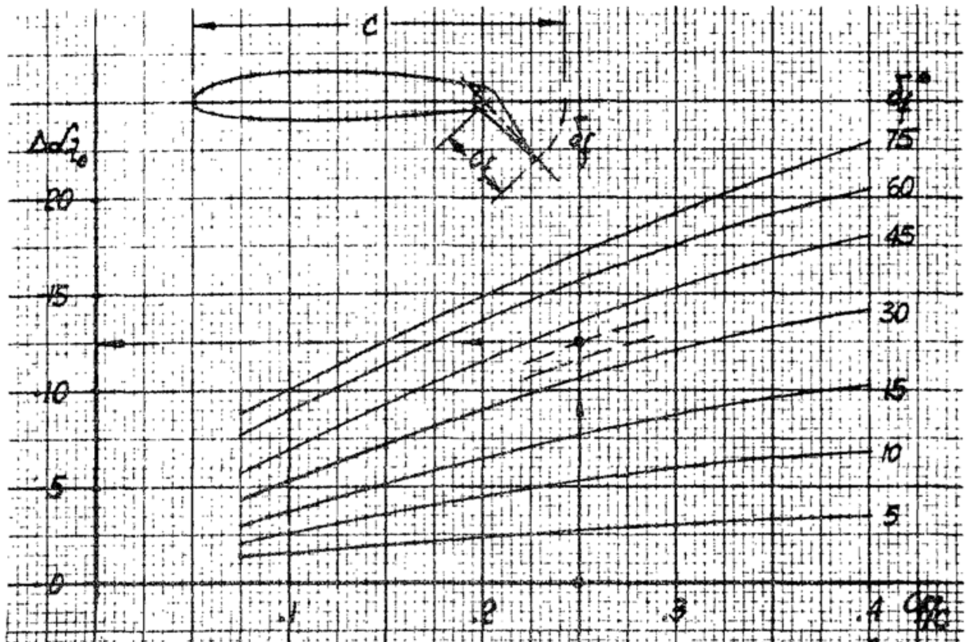


Fig. 28. DISPLACEMENT OF ZERO LIFT ANGLE ($\Delta\alpha_{l_0}$)

The calculated $\Delta\alpha_{l_0}$ is for a wing with full span flap. For partial span flap, the displacement will be proportional. Therefore:

$$\Delta\alpha'_{l_0} = \frac{b_f}{b} \times \Delta\alpha_{l_0} = .6 \times -12.5 = -7.5^\circ$$

Finally, the zero lift angle for the partially flapped wing will be:

$$\alpha_{l_{0_f}} = \alpha_{l_{0_f}} + \Delta\alpha_{l_{0_f}} = -5^\circ \ -7.5^\circ = -12.5^\circ$$

This point is plotted in Figure 26 and a straight line parallel to the plain wing lift curve is traced.

The maximum lift coefficient for the flapped wing was found on page 31.

$$C_{L_{max}} = 1.78$$

A line is traced at this level and point "C" is located. The shift of the $C_{L_{max}}$ point for the 63_2615 with split flap is 1.5°. This value could be used to locate the $C_{L_{max}}$ point for the partially flapped wing.

On page 24, the Design Lift Coefficient was calculated:

$$C_{L_{cruise}} = .35$$

The corresponding angle of attack from Figure 26 is:

$\alpha = -.5^\circ$ and is also the wing incidence (i) with respect to the fuselage horizontal reference line. A slightly smaller incidence was used in the PL-1, corresponding to a V_c = 120 mph.

The angle of attack for landing should be calculated next. It can be seen in Figure 26 that the angle of attack for $C_{l_{max}}$ with flaps-up ($\alpha = 14°$) is greater than the one with flaps-down ($\alpha_f = 12.5°$). Therefore, the landing gear position and tail clearance should be based in the flap-up attitude. The "Ground Effect" on the lift curve slope should be calculated first.

A wing flying at heights less than one semi-span above the ground will have less induced drag than at higher altitudes. The air is compressed between the wing and the ground, and the airplane virtually "floats." This effect is equivalent to an increase in Aspect Ratio, which in turn represents a change in the lift curve slope as we have seen before.

Figure 29 is **rep**roduced from NACA WR L-95 Report (Ref. 15).

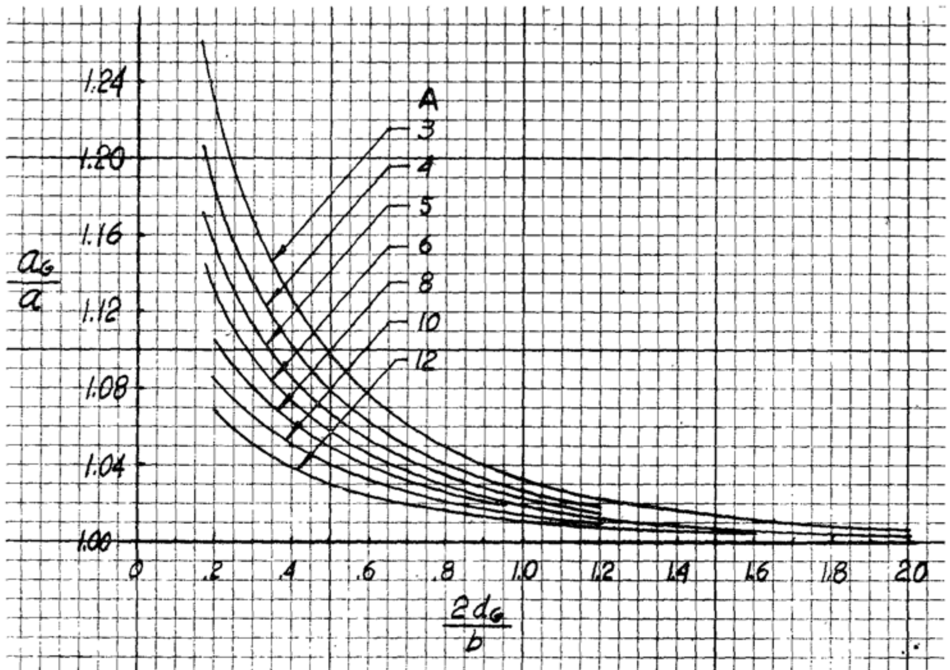


FIG. 29. GROUND EFFECT ON LIFT-CURVE SLOPE

Where:

a_G is the lift curve slope in ground effect

a is the lift curve slope at altitude

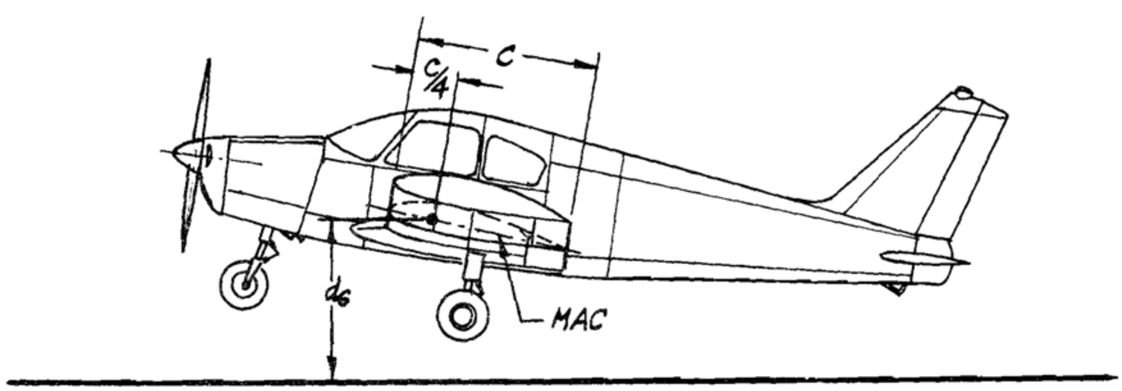


FIG. 30. GROUND DISTANCE d_G

Just before touch-down, the distance from the ground to the quarter chord point on the mean aerodynamic chord (MAC) may be assumed to be $d_G = 35''$.

Then:

$$\frac{2\ dG}{b} = \frac{2 \times 35''}{336''} = .21 \xrightarrow{\text{Figure 29}} \frac{a_g}{a} = 1.10$$

The change in angle of attack could be approximated by the following equation:

$$\Delta\alpha = \frac{C_L}{a} \left(\frac{1}{\frac{a_g}{a}} - 1 \right)$$

$$\Delta\alpha = \frac{1.25}{.082} \left(\frac{1}{1.10} - 1 \right) =$$

$$= 15.2\ (.91-1) = 15.2 \times -.09$$

$$= -1.36° \cong -1.4°$$

Where: $C_L = 1.25$

$a = .082$

$\frac{a_g}{a} = 1.10$

Then the angle of attack for landing with flaps-up in ground effect will be:

$$\alpha_g = \alpha + \Delta\alpha = 14.0° - 1.4° = 12.6°$$

1-17 EMPENNAGE DIMENSIONING

The tail surfaces of an airplane have to meet two basic requirements: stability and control. The design of tail surfaces, the determination of their size, position, angle of incidence is not an easy problem. The effects of factors such as slipstream, downwash, interference, C.G. position, Reynolds Number, and many others,complicate not only the problem, but also obscure the basic concepts for the amateur designer.

Even today, with the great amount of research data, the analytical approach for the tail surface design should be complimented with Wind Tunnel testing mostly when the design is unconventional.

The amateur designer willing to tackle the design of a Delta or Tail-less airplane will be foolish to risk his life without at least taking some college courses in Stability and Control, run some wind tunnel testing and have the data analyzed by experts.

For conventional configurations, the **problem** is well defined, but still complex. The reading of some textbooks such as Airplane Performance Stability and Control by Perkins & Hage is strongly recommended. For this preliminary design phase, we do not need to dip into differential equations; it will be enough to use the old Egyptian method of "follow the leader."

Let's have some statistics. But first, with the aid of Figure 31, some terminology will be defined.

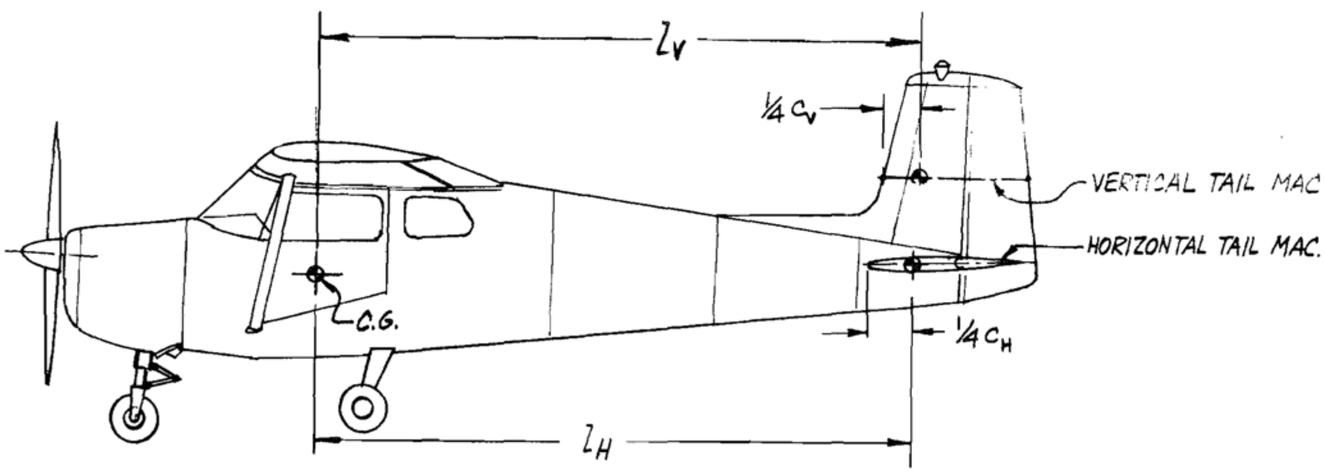


FIG. 31 - TAIL LENGHTS

If we multiply the Area of the Horizontal Tail in ft.^{2} by the distance between the C.G. and the quarter chord point of the tail (l_H) in ft., we obtain "ft.^{3}." And cubic feet are used to measure volumes. Therefore, the product ($S_H \times l_H$) is called Horizontal Tail Volume just for convenience.

Same with the Vertical Tail:

$$S_V \times l_V = \text{Vertical Tail Volume (ft.}^3)$$

The old question of "how much tail area" now should be changed to "how much tail volume." This is more convenient because it also considers the tail length. Comparisons based on tail area only, are misleading because 10 sq. ft. at the end of a short fuselage will not have the same effect than 10 sq. ft. at the end of a long one. But on the other hand, a 10 $ft.^2$ tail at 10 ft. from the C.G. will have nearly the same effect as 5 $ft.^2$ at 20 ft. from the C.G. Both have the same "tail volume."

$$10 \text{ ft.}^2 \times 10 \text{ ft.} = 100 \text{ ft.}^3$$

$$5 \text{ ft.}^2 \times 20 \text{ ft.} = 100 \text{ ft.}^3$$

The bigger the "Tail Volume" the greater will be the airplane stability, which is equivalent to allowable C.G. movement. An airplane with very small C.G. movement will need a relatively small tail volume. A conventional two place airplane, with the occupants seating over the C.G., and the fuel also near the C.G. will require less tail volume than a transport airplane, where a passenger could be seated far away from the C.G.

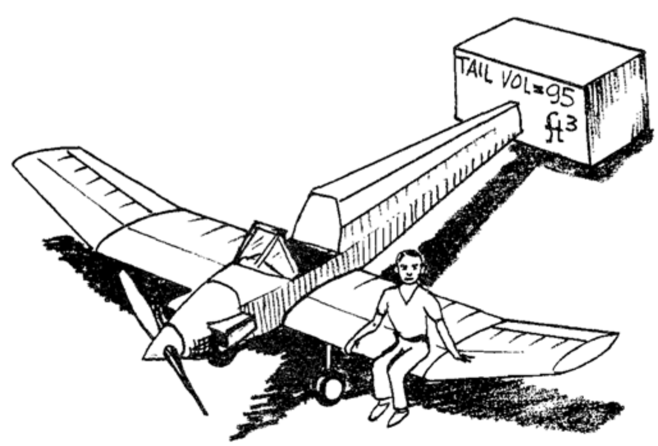
It is always convenient to use non-dimensional coefficients for comparative purposes. Therefore, if we divide the "Tail Volume" by another volume such as Wing Area ($ft.^2$) x Wing Chord (ft.) = $ft.^3$, we obtain a dimensionless expression called "Tail Volume Coefficient."

$$\frac{S_H \times l_H}{S_W \times c} = \overline{V}_H \quad \text{(Horizontal Tail Volume Coefficient)}$$

And for the vertical tail we could divide by the product of (Wing Area x Wing Span) and obtain another non-dimensional expression:

$$\frac{S_V \times l_V}{S_W \times b} = \overline{V}_V \quad \text{(Vertical Tail Volume Coefficient)}$$

In this manner, both tail surfaces are related to the wing area, but the horizontal tail is also related to the wing chord which has a great importance on the airplane longitudinal stability and control, while the vertical tail is related to the wing span which has a great significance on directional stability and control.



In Table 7 are calculated Horizontal Tail Volume Coefficients for some well-known airplanes. The Piper J3 has the lower value ($\overline{V}_H$ = .340) while the Navion has the higher ($\overline{V}_H$ = .692). These extremes represent a great spread.

TABLE 7

HORIZONTAL TAIL STATISTICS

Airplane	$S_W (ft^2)$	$\overline{c}$(ft)	$S_H (ft^2)$	S_H/S_W(%)	l_H(ft)	l_H/c	$\overline{V}_H$
Piper J3	178.5	5.33	24.5	13.7	13.2	2.47	.340
Piper Cherokee	160.0	5.08	23.0	14.4	13.1	2.57	.371
Cessna 140	159.6	4.90	23.3	14.6	12.5	2.55	.374
Cessna 150	160.0	4.95	23.7	14.8	13.0	2.63	.390
Shin 2150-A	144.0	4.80	20.8	14.5	12.7	2.64	.392
Thorp T-18	86.0	4.17	14.2	16.5	10.4	2.50	.412
Luscombe Silvaire	140.0	4.17	21.7	15.5	11.8	2.84	.442
Nesmith Cougar	82.5	4.00	14.0	17.0	11.3	2.83	.480
Emeraude CP 301	118.0	4.33	24.0	20.3	12.1	2.79	.568
Cessna 170	175.0	4.92	34.2	19.5	14.6	2.97	.580
Navion	184.0	5.22	42.8	23.3	15.5	2.96	.692

S_W = Wing Area

$\overline{c}$ = Wing M.A.C.

S_H = Tail Area

l_H = Tail Length

$\overline{V}_H$ = Tail Volume Coefficient

Assuming that the C.G. is located at 25% M.A.C., the following generalized criteria for V_H selection could be used:

TABLE 8

TAIL VOLUMES

Typical Applications	Tail Volume $\overline{V}_H$	Stability Margin	Elevator Area (% of S_H)	Control Effectiveness
Split flaps Small $C_{l_{max}}$ Small C_{mac}	.300	Small	30	Very poor
			50	Poor
			* 100	Fair
Plain Flaps Moderate $C_{l_{max}}$ Moderate C_{mac}	.450	Average	30	Poor
			50	Fair
			* 100	Good
Slotted flaps High $C_{l_{max}}$ High C_{mac}	.700	Large	30	Fair
			50	Good
			* 100	Very good

* The 100% Elevator Area represents the "All Movable Tail."

Stability Margin (S.M.) is the distance between the actual C.G. and the Neutral Point which by definition is that C.G. location for which there would be no

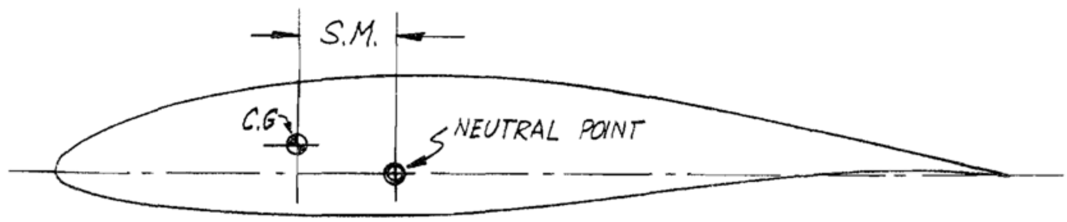


FIG. 32 - STABILITY MARGIN (S.M.)

stability. Consequently, a Large Stability Margin will cause the airplane to return quickly to a trim speed after some aerodynamic disturbance such as a gust, even stick free (see Figure 33a).

With small Stability Margin, the opposite is true and perhaps the airplane will not return to trim speed with the stick free (Figure 33b).

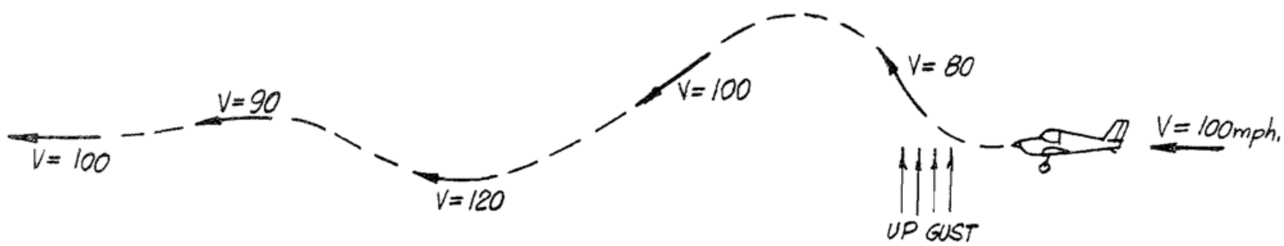


FIG. 33 a

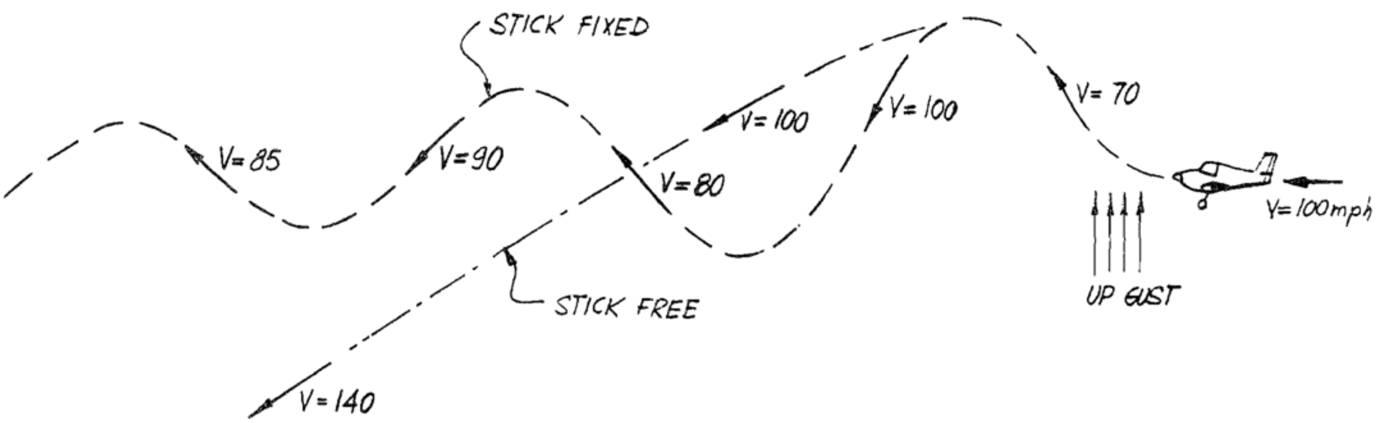


FIG. 33 b

For the Laminar PL-1 we assume $\overline{V}_H$ = .430 and the tail length $l_{H/c}$ = 2.75.

Solving the tail volume equation for S_H:

$$S_H = \overline{V}_H \times \frac{S_W}{l_{H/c}} = .430 \times \frac{116}{2.75} = 18.2 \text{ sq. ft.} \cong 18.0 \text{ sq. ft.}$$

From "Airplane Design" by K. D. Wood (Fourth Edition, Ref. 12), on page 7:1 - Table 7:1, the A.R. for the horizontal tail varies between 3.5 to 4.5. The lower value seems to be adequate for this airplane; the high values will result in heavier construction.

$$b_H = \sqrt{S_H \times AR} = \sqrt{18.0 \times 3.5} = \sqrt{63} = 7.94 \text{ ft.}$$

And as a round number will be easier to remember, we fix the tail span in 8 ft. The tail chord will be:

$$c_H = \frac{S_H}{b_H} = \frac{18.0}{8} = 2.25 \text{ ft.} = 27''$$

The next question is: "How much elevator?" It all depends what we want to do with the airplane. TABLE 8 provides a general idea of the control effectiveness related to elevator area. For an aerobatic airplane, a large elevator is desired. For a spin-proof airplane, a small elevator or restricted travel elevator will be used so the airplane could not even be stalled. These considerations are of general nature and a more accurate determination of the elevator area requires a careful and lengthy calculation described in detail in a future volume. But we can anticipate here that after investigating eight different tail combinations for the PL-1, it was found that an all-movable tail mounted on the top of the fuselage provides the best control and stability with the minimum area.

The concept of "Tail Volume" is mostly related to Stability, while the elevator angle (δe) and area are related to Trim and Control. Obviously, the larger movable surface will provide the greatest control or trim force. The maximum is reached when the whole tail moves, resulting in an "all-movable" tail. The elevator angle also has a limit. In general, no more than 25° should be used; any deflection beyond this value will not add appreciable value to the tail force. It is a good practice to dimension the elevator assuming $\delta e_{max} = \pm 20°$.

One last word on this subject: The elevator deflection has nothing to do with the airplane stability. Deflecting the elevator will change the trim speed but not the stability.

TABLE 9

VERTICAL TAIL STATISTICS

Airplane	$S_W(ft^2)$	$b(ft)$	$S_V(ft^2)$	$\frac{S_V}{S_W}(\%)$	$l_V(ft)$	$\overline{V}_v$
Piper J3	178.5	35.2	10.2	5.7	13.4	.022
Shin 2150-A	144.0	30.0	9.5	6.1	11.0	.024
Bebe Jodel D-9	97.3	22.9	4.9	5.0	11.4	.025
Luscombe Silvaire	140.0	35.0	10.6	7.6	11.9	.026
Cessna 140	159.6	33.3	11.5	7.2	12.8	.028
Cessna 150	160.0	33.3	11.7	7.3	13.1	.029
Piper Cherokee	160.0	30.0	10.8	6.7	12.9	.029
Mooney Mark 20	167.0	35.0	12.9	7.7	13.2	.029
Bellanca 260	161.5	34.2	16.4	10.2	12.6	.037
Beechcraft D-50	277.0	45.3	27.0	9.7	18.0	.039

TABLE 9 Cont'd

Airplane	$S_W(ft^2)$	$b(ft)$	$S_V(ft^2)$	$\frac{S_V}{S_W}(\%)$	$l_V(ft)$	$\bar{V}_V$
Ryan Navion	184.0	33.4	14.6	7.9	16.9	.040
Taylorcraft Model 20	178.5	34.7	18.7	10.5	13.8	.042
Beechcraft T-34	177.6	32.8	16.9	9.5	14.5	.042
Midget Mustang	69.3	18.6	6.7	9.6	8.2	.042
Cessna T-37 A	184.0	33.8	18.7	10.2	14.5	.043
Cessna TL-19D	174.0	36.0	18.4	10.6	15.4	.045

S_W = Wing Area

b = Wing Span

S_V = Vertical Tail Area

l_V = Vertical Tail Arm

$\bar{V}_V$ = Vertical Tail Volume Coefficient

The general sequence described for the horizontal tail also could be applied for the vertical tail. An average value from Table 9 is selected: Assume $\bar{V}_V$ = .033. The equation for Vertical Tail Volume can be solved for tail area S_V:

$$S_V = \bar{V}_V \times \frac{S_W \times b}{l_V} = .033 \times \frac{116 \times 28}{10.6} = 10.2 \text{ sq. ft.}$$

The Vertical Tail arm (l_V) was selected based on several cut-and-try layouts. The idea was to combine all the known values such as horizontal tail area and position, ground clearance, aesthetics, structural arrangement, control mechanisms, etc., and also to obtain a clean vertical tail for spin recovery as illustrated in Figure 34 and discussed in detail in a future volume. The horizontal tail blanks the vertical tail within the cross-hatched zone. To avoid this loss in effectiveness, the simplest resource is to move the vertical ahead of the horizontal. Also sweeping forward the vertical will help, like in the Mooney airplanes, but aesthetics and "jet-age" styling push in the opposite direction. Again it is a matter of compromises. Another solution is the "T" tail, (horizontal on top of the vertical). For spin recovery this is ideal, but from structural and control mechanism viewpoints, this is heavy and complicated.

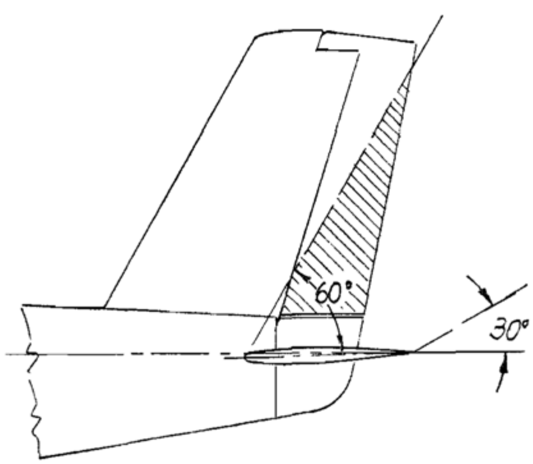


FIG 34

The amount of Rudder area was based in Directional Control calculations and resulted in 30% of the total vertical tail area. But flight tests results indicated that this was not enough. A modification of the PL-1 incorporates a larger rudder area giving very satisfactory results.

During the early stage of design of the PL-1 airplane, the possibility of using a Vee-tail was considered. The Vee-tail has some advantages and disadvantages compared with a conventional tail.

From NACA Report 823 (Ref. 13), "Experimental verification of a simplified Vee-tail theory and analysis of available data on complete models with Vee-tails," we reproduce some of the conclusions:

"The Vee-tail could have the following advantages over the conventional tail assembly:

(1) Less drag interference because the Vee-tail has fewer fuselage-tail junctures.
(2) Less tendency toward rudder lock.
(3) Higher location of tail surfaces, which tends to reduce elevator deflection required for take-off and landing, to keep the tail out of spray in flying-boat take-off, and to reduce possibilities of tail buffeting from the wing and canopy wakes in high-speed flight.
(4) Fewer tail surfaces to manufacture."

On the other hand, the analysis indicate the following disadvantages that a Vee-tail might have when compared with conventional tails:

(1) Possible interaction of elevator and rudder control forces.
(2) Possible interaction of elevator and rudder trimming when tabs are at fairly large deflections.
(3) More complicated operating mechanism.
(4) Greater loads on tail and fuselage, which would tend to increase the weight."

The relative merits of the Vee-tail and conventional tails for spin recovery have not been established, but it appears that the Vee-tail should be at least as good as the conventional tail assembly in this respect, except possibly in cases in which simultaneous full deflection of both rudder and elevator is required for recovery from the spin."

On page 12 of the same report appears the following equation:

$$S_h + S_v = S_{vee}$$

Where:

S_h = Surface of conventional horizontal tail

S_v = Surface of conventional vertical tail

S_{vee} = Surface of Vee-tail

This means that the surface of a Vee-Tail is equal to the sum of the vertical and horizontal surfaces of a conventional tail, and not smaller as apparently it seems to be.

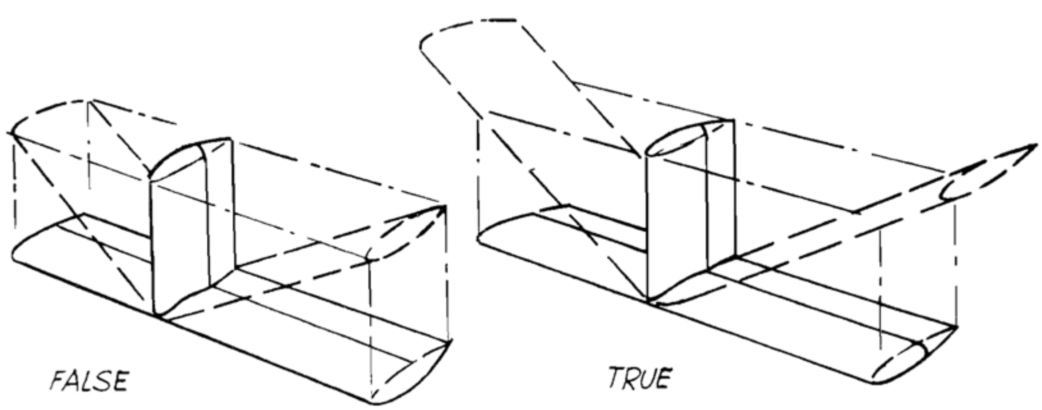


FIG. 35 - COMPARATIVE SIZE OF CONVENTIONAL AND "V" TAILS

If there is no reduction in the total surface, the structural weight is roughly the same, but the controls are more complicated and a "mixer" mechanism is necessary in addition to the standard controls. All these considerations were enough to decide on a conventional tail.

Some modern high-performance gliders are using Vee-tails mostly because this arrangement provides the greatly needed ground clearance.

1-18 POWER PLANT SELECTION

For the American Amateur Designer the choice is well defined: An air-cooled, four cylinder opposed engine. Figure 36 is a three-view (1/10 scale) of the Continental C90-12F, but the same drawing can be used for the C-75, C-85, and the O-200-A.

CONTINENTAL C90-12F
SCALE 1/10

Max. Cont. Rating:	90 HP @ 2475 RPM
Take-off Rating:	95 HP @ 2625 RPM

Basic Engine Weight	168.60	lbs
Starter	15.50	"
Generator	9.70	"
Voltage Regulator	1.68	"
Prop. Attachment	1.70	"
Rubber Mountings	.40	"
Carburetor (Marvel)	3.00	"
Magnetos (B.S.)	11.68	"
Spark Plugs	1.36	"
Ignition Cable	1.82	"
Standard Engine Dry	215.44	"

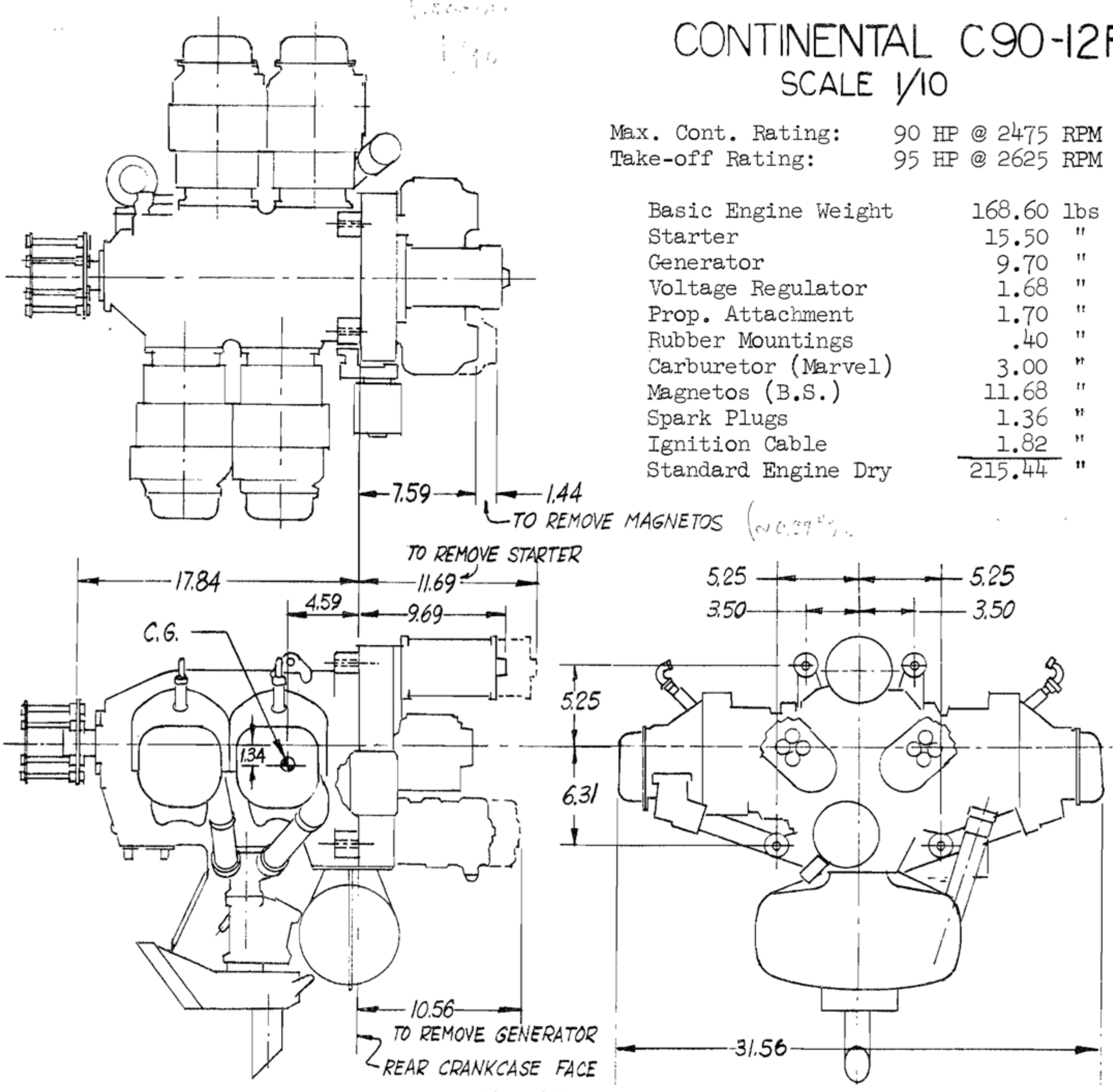
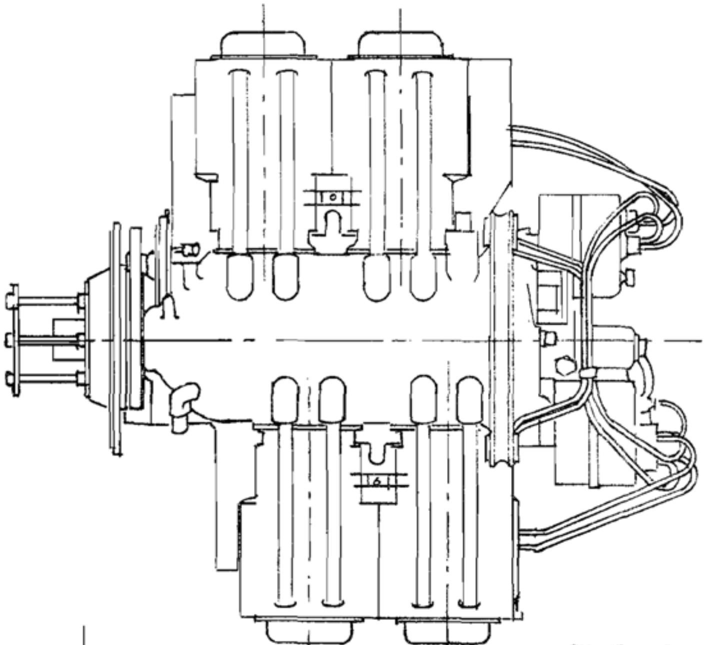


FIG. 36

The O-290-GA engine made by Lycoming for use in Air Corps Model C-21 and C-22 generator units which supplied power for starting jet aircraft is basically the same as the O-290-D2B aircraft engine shown in Figure 37. The O-290-G could be purchased for $100 to $160 making it an extremely attractive power plant for the home-builder.

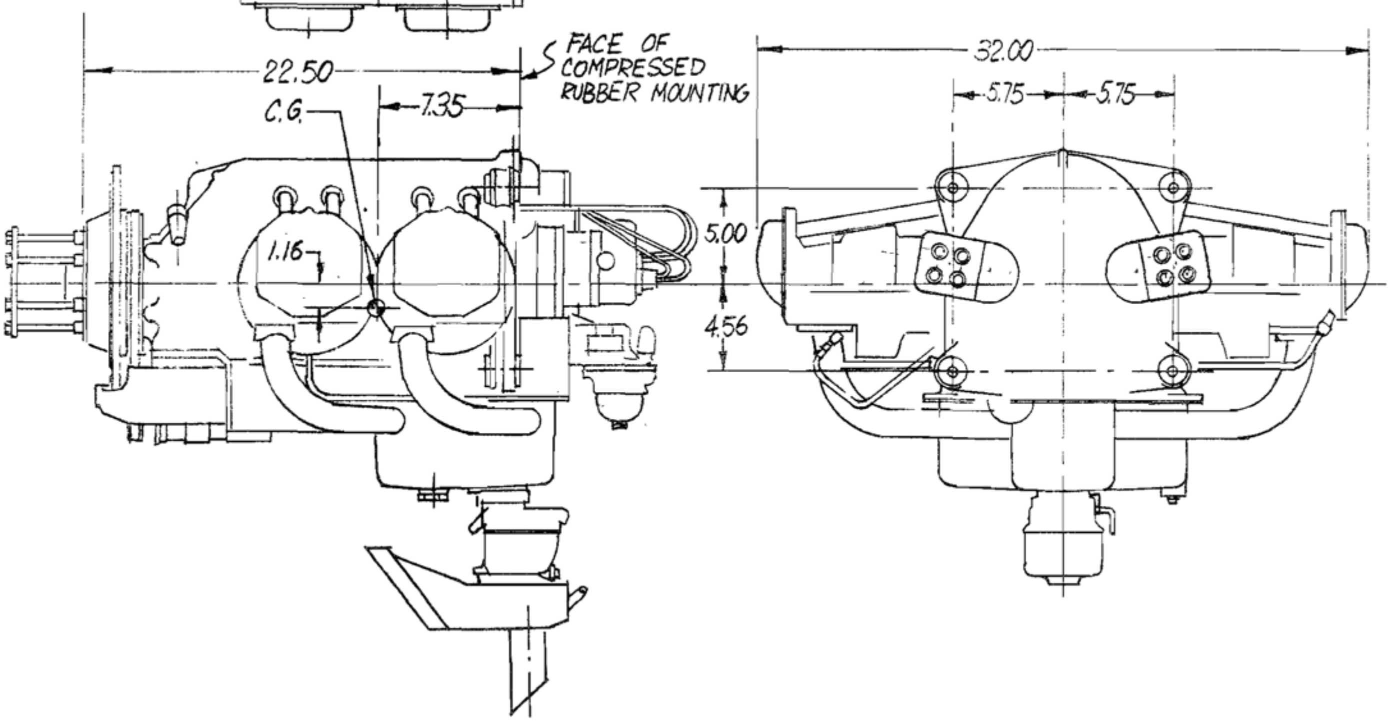


LYCOMING O-290-D2B

SCALE 1/10

Max. Cont. Rating:	135 HP @ 2600 RPM
Take-off Rating:	140 HP @ 2800 RPM

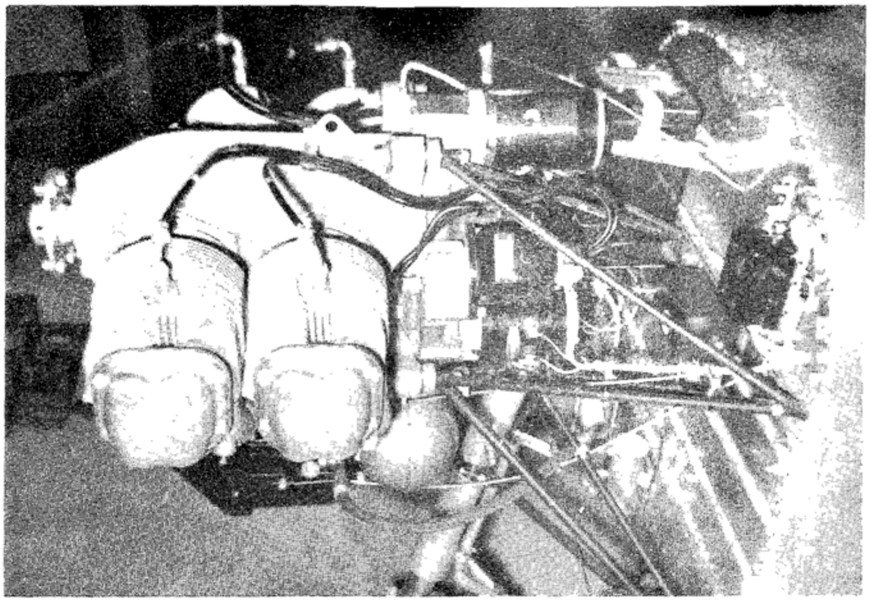
Basic Engine Weight	215.00	lbs.
Carburetor	3.00	"
Magnetos	10.25	"
Spark Plugs	1.36	"
Ignition Cables	1.00	"
Starter & Generator Drive	5.49	"
Starter	16.98	"
Generator	10.64	"
Standard Engine Dry	264.00	"



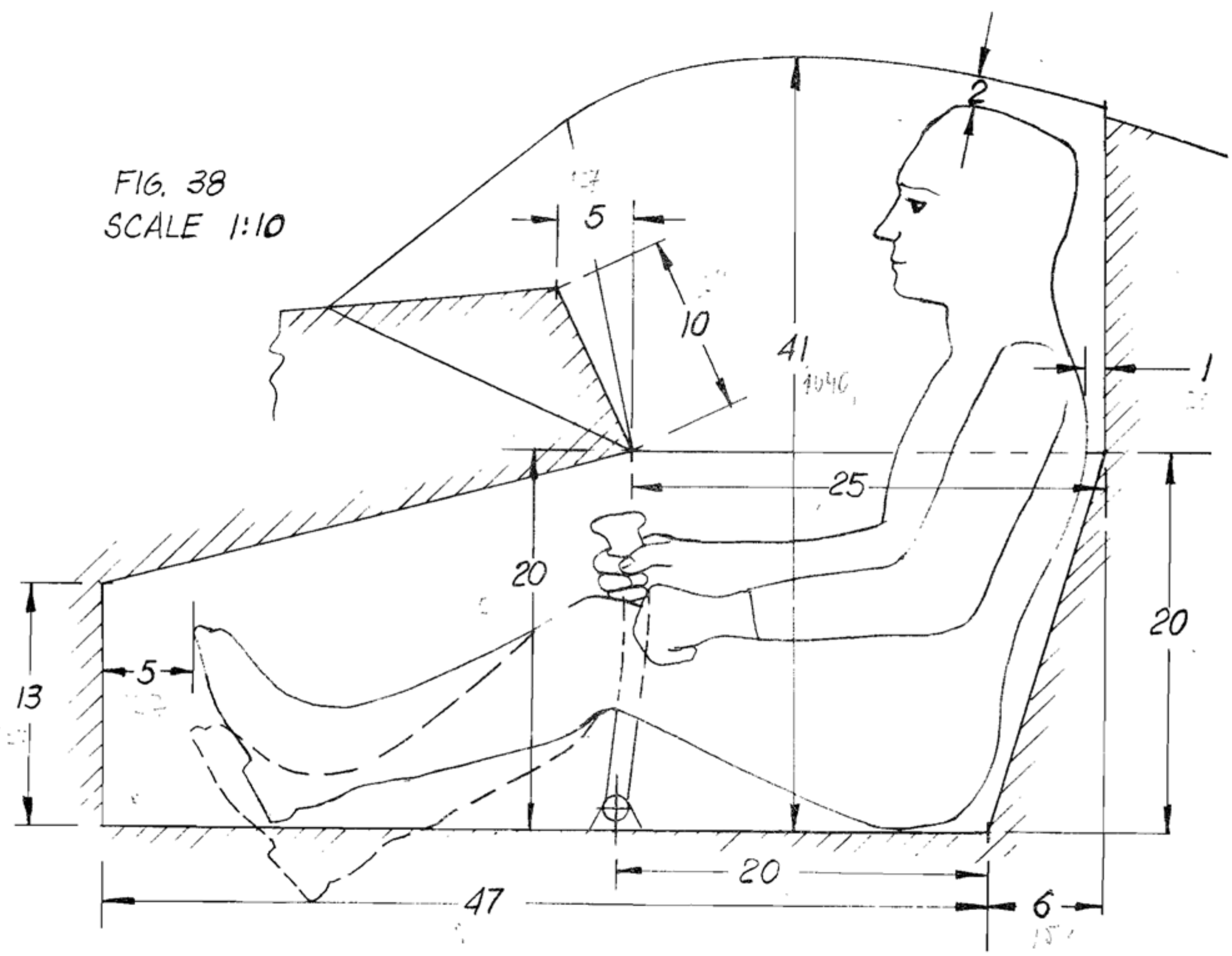
The PL-1 was designed for the Continental O-200-A engine which has 100 hp maximum continuous rating. The prototype airplane N4081K was equipped with a C90-12F requiring only four spacers to make up the difference in the engine mounting. (See Photo). The C90-12 uses the rubber cones, while the O-200-A has the Lord Mountings; otherwise both engines are physically the same. A redesign for the Lycoming O-290-D2B or the converted O-290-G is in the works. This power plant requires a different engine mounting and a modified cowling.

The wing tip tanks have 12.5 gallon capacity each. At the recommended cruising for the C90-12F, which is 2350 RPM and 24.5 in Hg of manifold pressure, the approximate fuel consumption is 5.9 gal/hr. The endurance is then:

$$\frac{25 \text{ gallons}}{5.9} = 4.25 \text{ hours}$$



1-19 COCKPIT DESIGN



The first step in the cockpit design is to outline on aluminum or celluloid sheets the silhouette of a standard size person in a suitable scale. For preliminary design work, the 1/10 scale is considered adequate. In Figure 38, the minimum dimensions for a cockpit are shown. It is advisable to provide a difference of level between the seat bottom and the floor to avoid leg tiring, as shown by dotted line.

The components of a standard size man are shown in Figure 39. These components can be traced directly on aluminum or celluloid sheets, then contour sawed and assembled using small screws at the indicated articulation points.

The cockpit is the best place to start the layout, and "design the airplane around the occupants." The cockpit minimum width is 40" for "side by side" configuration and 22" for single place or tandem configuration.

In Figure 40, the recommended control movements and locations are shown. The stick at neutral position is the reference point. This point is located at 20" forward of the seat back (Fig. 38).

The dual stick control is simple and light. Some details of the PL-1 cockpit are shown in the next photos. The first one shows the flap control lever at the "flap-up" position, the elevator trimtab wheel, and position indicator, the dual control sticks, and the seat pan which is also torque box for the wing.

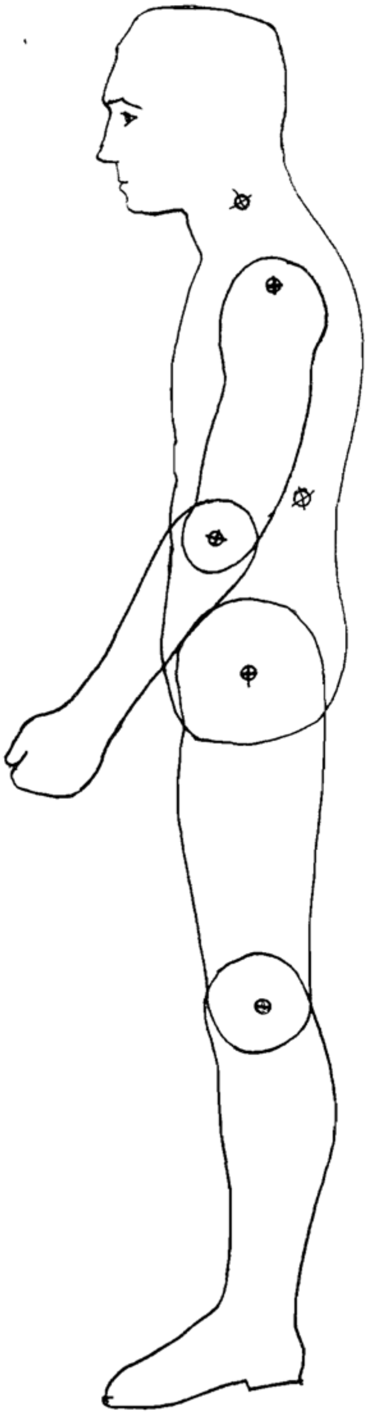


Figure 39
Standard Man
Scale 1:10

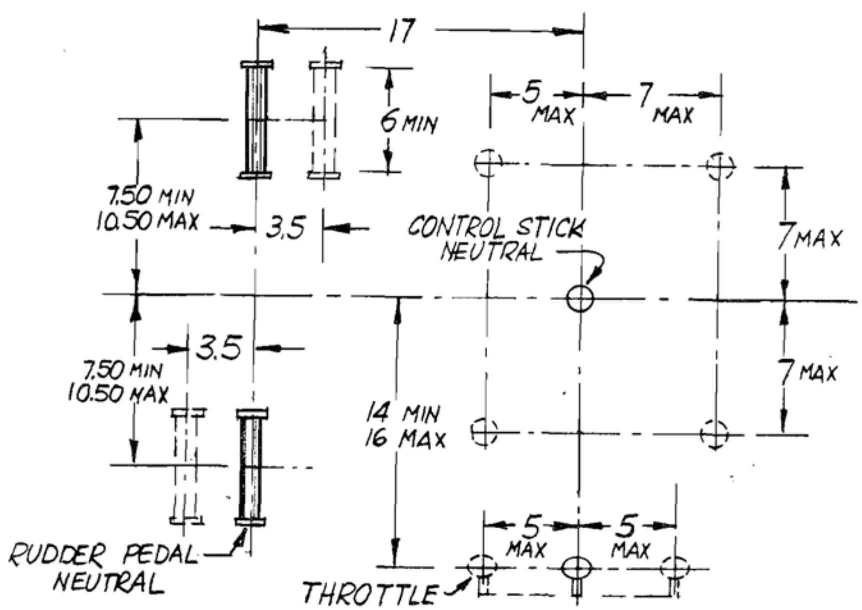
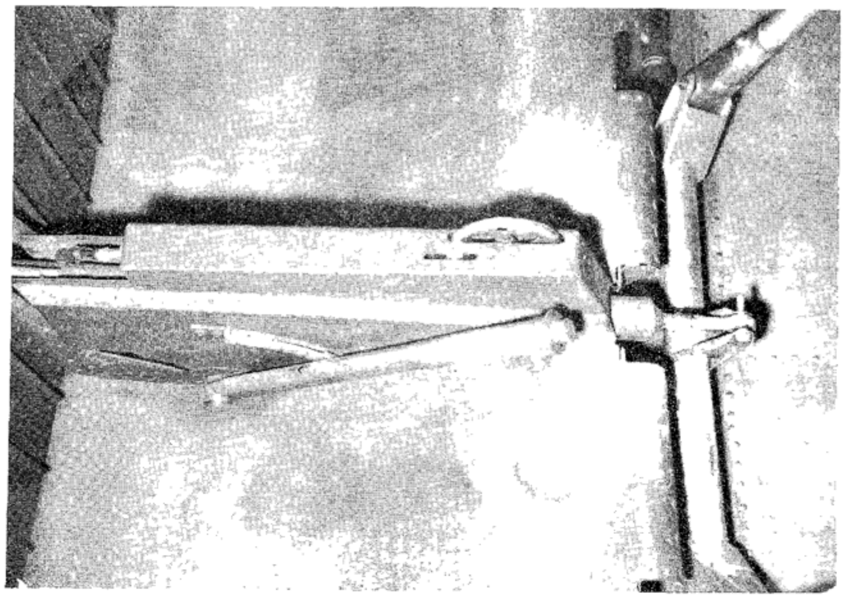
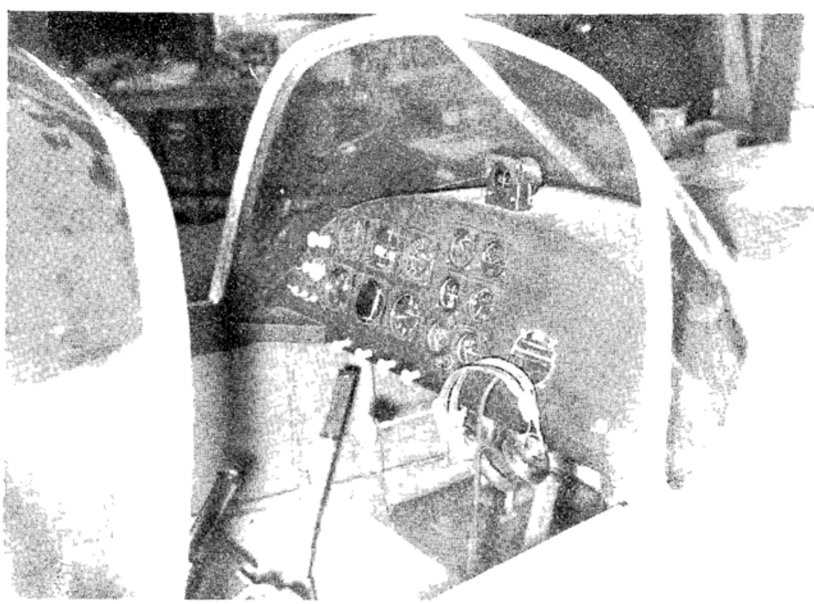


FIG. 40 - CONTROL MOVEMENTS
PLAN VIEW - SCALE 1:10

When the wing is removed from the fuselage, the bolt at the forward end of the elevator push-pull tube and the universal coupling on the trim-tab torque tube, (visible at the open rear end of the box), are disconnected. These two are the only control connections which have to be disconnected.

The second photo shows the instrument panel, the windshield and the forward end of the bubble canopy slides.





The choice of a tricycle landing gear is justified by the following reasons:

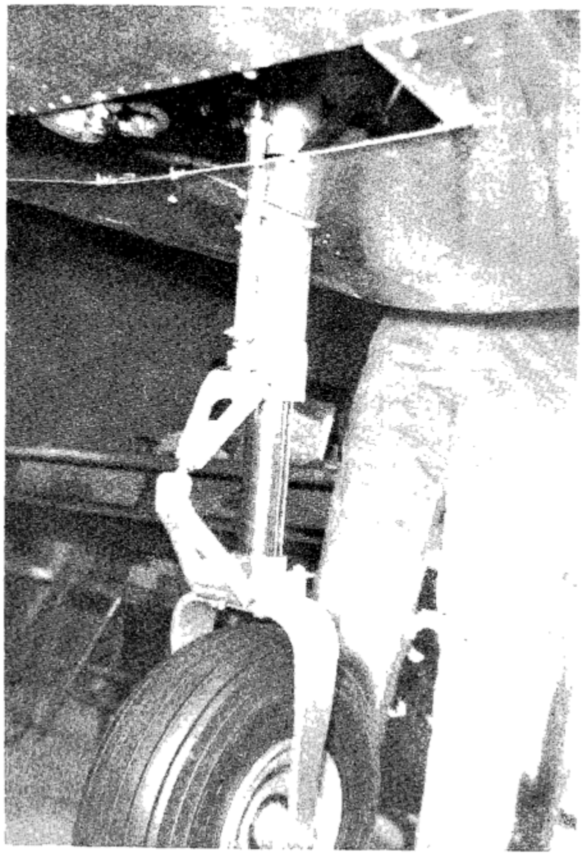
1. A leveled position is more comfortable when entering or leaving the cockpit.

2. There is an improved forward vision from the cabin during ground runs.

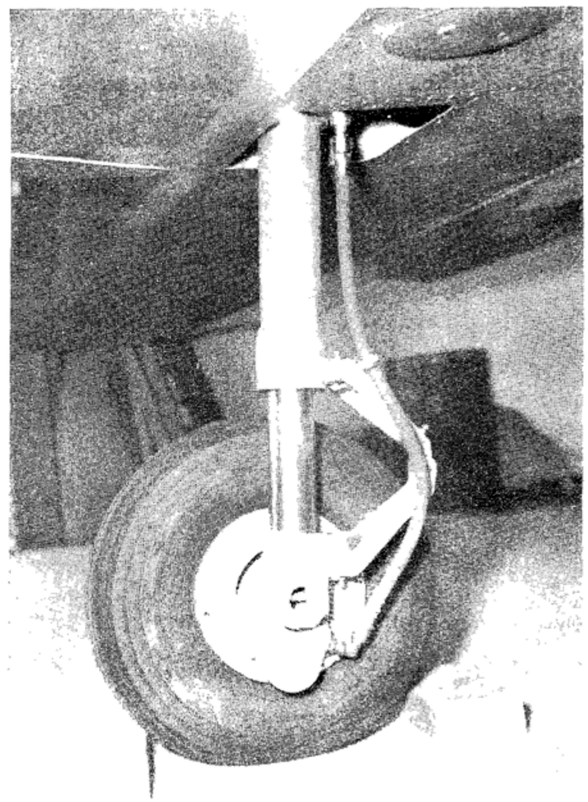
3. The tricycle landing gear eliminates the ground loop; it gives better ground stability and permits full braking which in turn reduces the landing distance.

4. The small wing incidence permits a faster acceleration, thus a reduction in take-off distance.

5. With a leveled taxiing position the chances of damaging the tail with stones blown up by the propeller are reduced.



NOSE GEAR



MAIN GEAR

The ground clearance requirements specified in CAR 3.422 (Ref 1) are reproduced next:

"(1) Seven inches (for airplanes equipped with nose wheel type landing gears) or 9 inches (for airplanes equipped with tail wheel type landing gears) with the landing gear statically deflected and the airplane in the level, normal take-off, or taxiing attitude, whichever is most critical."

"(2) In addition to subparagraph (1) of this paragraph, there shall be positive clearance between the propeller and the ground when, with the airplane in the level take-off attitude, the critical tire is completely deflated and the corresponding landing gear strut is completely bottomed." (See Figure 41)

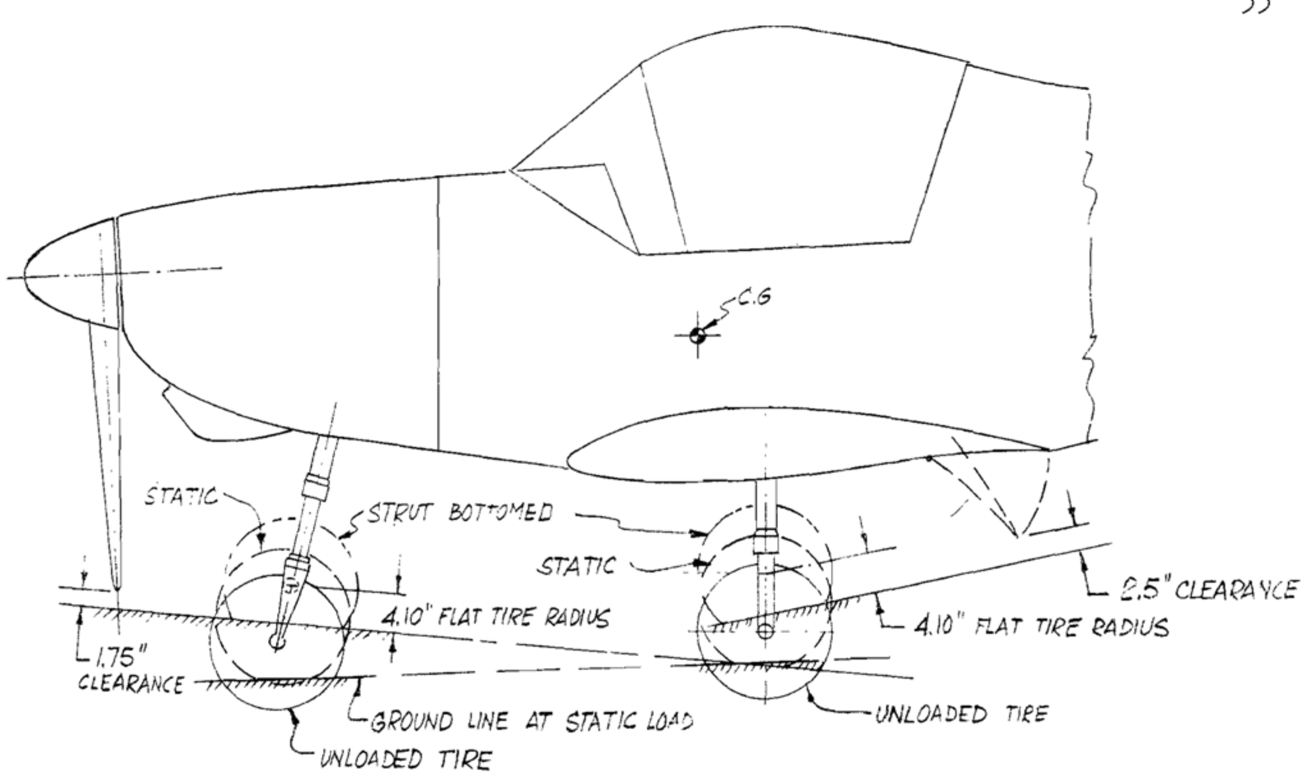


FIG. 41. LANDING GEAR CLEARANCES. PL-1 LAMINAR

Some General Considerations on Landing Gear Design

(a) Tail Wheel Type (Figure 42)

The C.G. for the Design Gross Weight should fall inside the cross-hatched area enclosed between 15° and 25° from the vertical. The wheel motion due to shock absorber deflection should fall inside the cross-hatched area enclosed between the vertical and 5°.

The tail wheel knuckle spindle axis should be inclined forward 5° from the normal to the ground line in the taxiing position. The spindle axis should intercept the ground line ahead of the wheel contact point at a distance equal at least 1/10 of the wheel diameter. The tail wheel shock absorber deflection must be within the cross-hatched area between the normal to the ground and 45° from the normal, preferably at 30° from the normal.

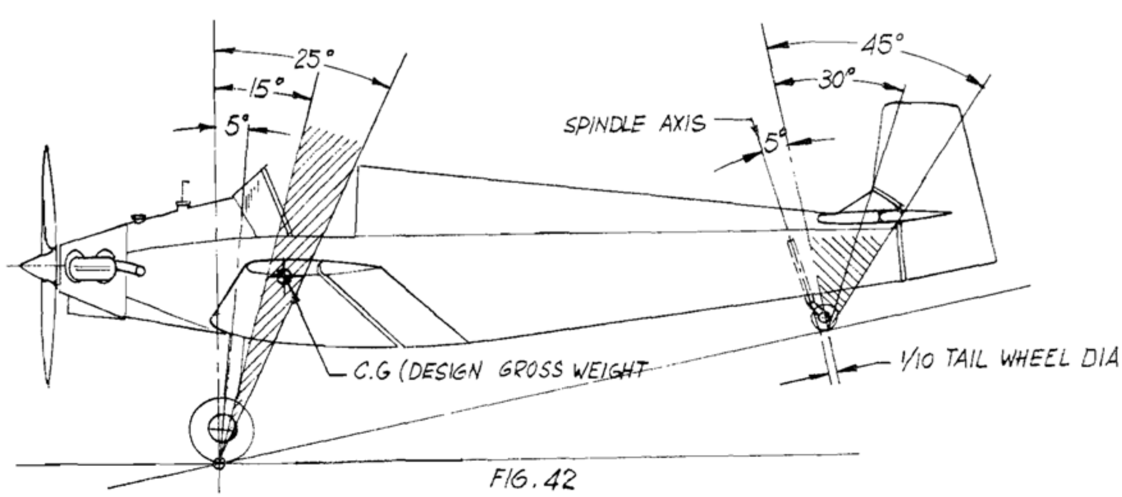


FIG. 42

The C.G. should be located as shown in the front view (Figure 43).

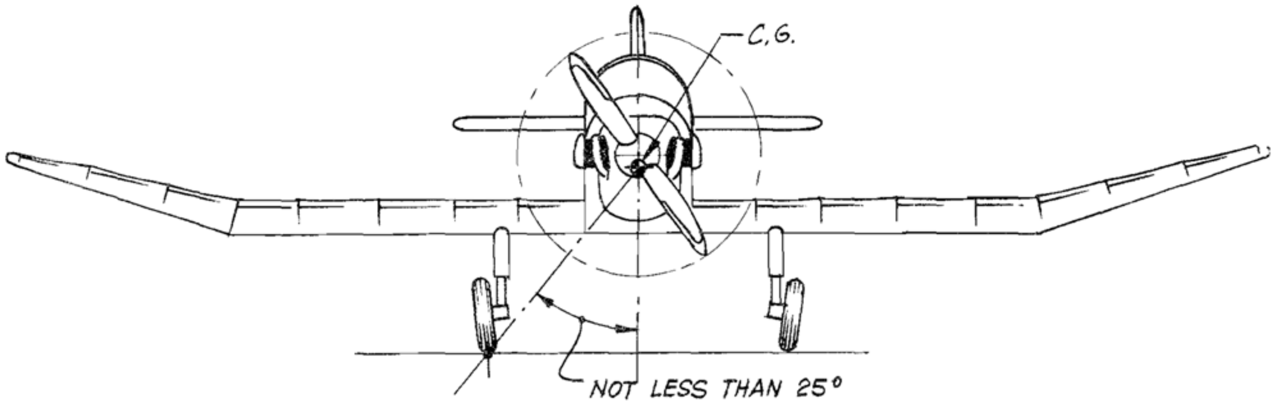


FIG. 43

(b) Nose Wheel Type (Figure 44)

The main wheels position with respect to the C.G. is determined as follows:

1. Calculate the angle of attack (α) at $C_{L_{max}}$ with flaps-up (See page 40
2. Locate the maximum aft C.G.
3. Draw a 1/10 scale side view of the airplane with the wing at the angle of attack α at $C_{L_{max}}$.
4. From the C.G. draw a vertical line, and from the tail skid a horizontal line.
5. At the intersection point "A" locate the center point of the tire contact area.
6. Draw the landing gear with the tire and shock absorber completely deflected.
7. After the shock absorber deflection is calculated, the extended (unloaded) gear can be drawn.
8. See Figure 41 for clearance requirements.

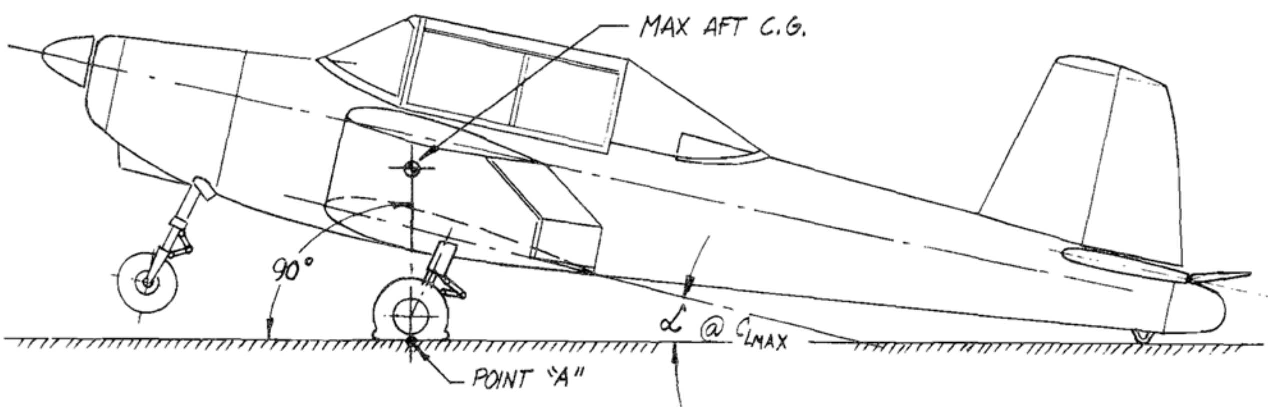


FIG. 44

The track and wheel base should be determined next. The relationship between the track and wheel base is dictated by the Turnover Angle which is determined as follows:

1. Draw a top view showing the desired nose wheel and tail wheel positions. Also show the C.G. location.

2. Draw a side view showing the landing gear with shock absorbers and tires statically deflected and the C.G. position.

3. Establish line A-B. Extend the line to a point "C".

4. Through point "C" draw a perpendicular to line A-B.

5. Through the C.G. (in the plan view) draw a line parallel to A-B and obtain point "D".

6. From point "D" measure the height of the C.G. (h) obtained from the side view and obtain point "E".

7. Trace line E-C and measure angle "β". This is the turnover angle and should be less than 60°.

If the turnover angle is more than 60° increase the track or the wheel base and try again.

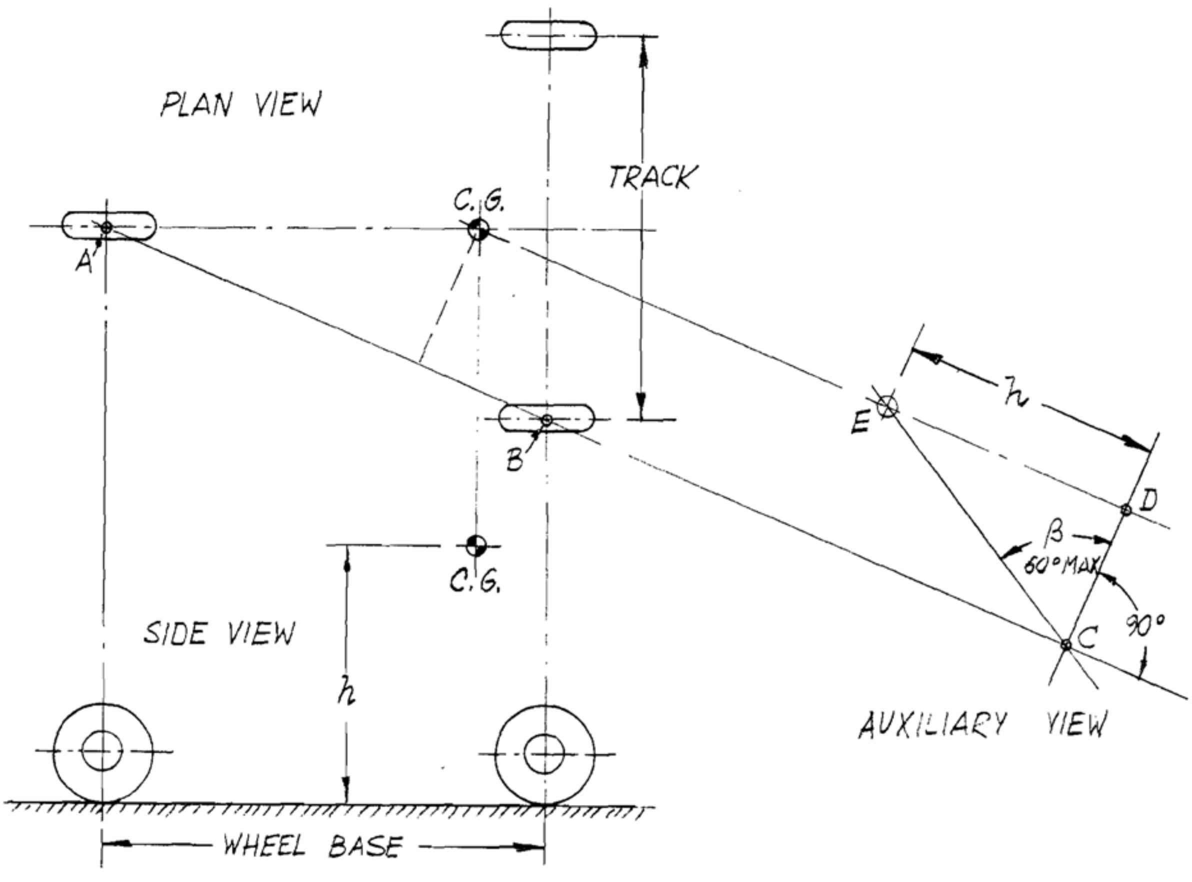


FIG. 45 - TURNOVER ANGLE

For a tailwheel type airplane, the checking of the turnover angle should be made using the same procedure. The angle β should not exceed 60°.

The steerable nose wheel should have an angular movement θ such as the turning point falls inside the wing tip as shown in Figure 46. Some airplanes have a large steer angle on the nose wheel which enables it to turn around on one wheel. To check the position of the turning point, simply project the main wheel axis and the nose wheel axis at the maximum steer angle until they intersect, as shown in Figure 46.

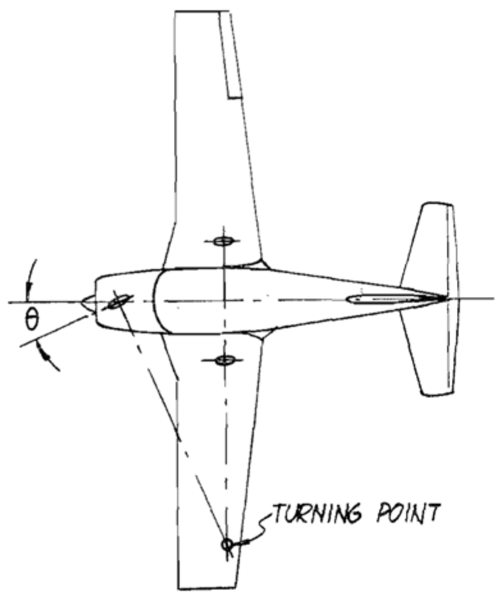


FIG. 46 _ TURNING POINT

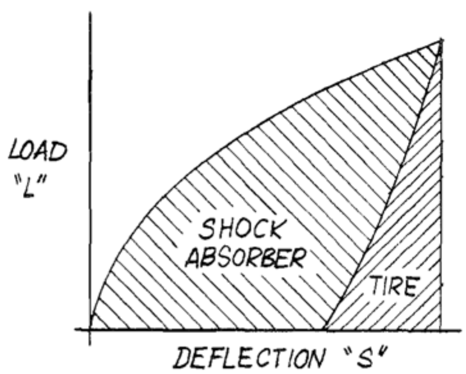


FIG. 47

Shock Absorber Travel

The shock absorber travel could be estimated with an approximate method described next:

The energy stored in the landing gear is represented by the cross-hatched areas in Figure 47 and expressed by:

$$S.E. = \eta . L . S$$

Where:

η = Efficiency
L = Max. Vertical Load
S = Total deflection (Tire + Shock Abs.)

The efficiency of various types of shock absorbers are given in Table 10.

TABLE 10

Type of Shock Absorber	η
Tires	.47
Steel Springs	.50
Rubber Rings	.60
Oleo-Pneumatic	.75

The total vertical energy of the airplane is given by the following equation:

$$K.E. = \frac{Wv^2}{2g}$$

Where:

W = Airplane Gross Weight
v = Maximum Descent Velocity
g = Gravity acceleration = 32.17 ft/sec.^{2}

The airplane vertical energy will be absorbed by shock absorbers. Therefore:

$$K.E. = S.E. \qquad \text{Then:} \quad \frac{Wv^2}{2g} = \eta .L.S.$$

Solving for η.S:

$$\eta .S = \frac{v^2}{\frac{L}{W} .2.g}$$

The maximum descent velocity "v" need not exceed 10 ft/sec. according to CAR 3.243. The relation $\frac{L}{W} = n$ is the landing gear limit load factor. The minimum value of n is 2 (CAR 3.243); normally 3 is used for standard aircrafts. A very high value will be rather uncomfortable. Introducing all these values in the previous equation:

$$\eta \cdot S = \frac{10^2}{3 \times 2 \times 32.17} = .527 \text{ ft.} = 6.32 \text{ in.}$$

The term $\eta \cdot S$ represents the whole shock absorber, which can be separated in tire + strut. Then:

$$\underbrace{\eta \cdot S}_{\text{Total}} = \underbrace{\eta_t \cdot S_t}_{\text{Tire}} + \underbrace{\eta_s \cdot S_s}_{\text{Strut}}$$

In Table 11, the most commonly used light airplane tires are listed. The tire maximum deflection can be calculated by subtracting the flat tire radius from the nominal radius.

TABLE 11

Static Load	Tire Size	Ply Rating	Tire O. Diam.	Flat Tire Radius	Max. Tire Deflection
700	5.00-4	4	13.25	3.6	3.02
800	5.00-5	4	14.20	4.1	3.00
1100	6.00-6	4	17.50	4.5	4.25

The static load is calculated as follows:

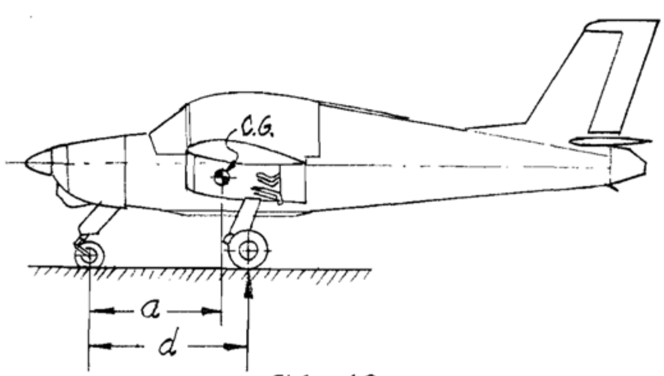


FIG. 48

$$\text{Nose Wheel or Tail Wheel Load} = \text{G.W.} \times \frac{(d-a)}{d}$$

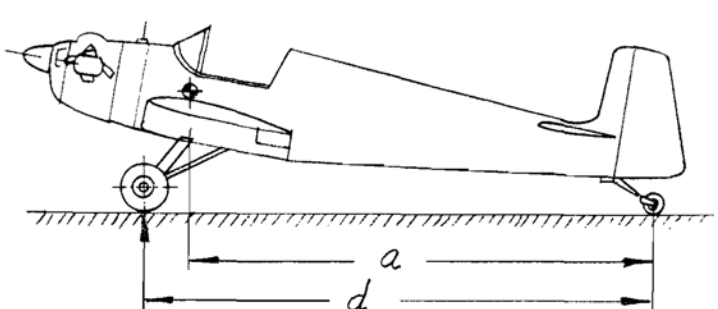


FIG. 49

For the Laminar PL-L we selected 5.00-5 tires, and oleo-pneumatic shock absorbers. The shock absorber strut travel is calculated next.

$$S_s = \frac{\eta_t \cdot S - \eta_t \cdot S_t}{\eta_s} = \frac{6.32'' - (.47 \times 3.00'')}{.75}$$

$$= 6.5 \text{ in.}$$

1-21 WEIGHT ESTIMATION

Some statistics on existent airplanes will give us an idea of "what to shoot for". Table 12 presents data on single seat light airplanes, and Table 13, data on two seaters. In both tables, the airplanes are listed by increasing Gross Weight. The relation between Useful Load and Gross Weight varies very much depending on the airplane. It is difficult to establish a trend because many factors are involved such as Load factor, Engine Weight, Aspect Ratio, Type of Construction, etc. Therefore, instead of calculating an average value based on the whole table, it will be advisable to select two or three airplanes with similar characteristics to the proposed design and calculate the average value based on these few samples.

For the PL-1:

Piel Emeraude 310A................	.437
M.S. 880-B Rallye.................	.412
PZL-102...........................	.360
Thorp Sky-Scooter.................	.425
	1.634

$$\frac{1.634}{4} = .408$$

TABLE 12 - SINGLE PLACE AIRPLANES

Designation	Gross Weight	Empty Weight	Useful Load	Useful L. / Gross W.	Max. HP	Wing Area	Lbs./ Sq. ft.
Driggs Dart DJ-1	511	330	181	.335	25	74.8	6.82
HM-200 Flying Flea	530	309	221	.417	30	101.0	5.25
Tipsy S-2	550	286	264	.480	25	100.0	5.50
Lincoln Sport	600	370	230	.383	35	108.0	5.55
Druine Turbulent	606	341	265	.437	30	80.7	7.52
Luton Minor	620	330	290	.468	37	125.0	4.96
Tipsy Nipper TG6	660	360	300	.455	30	80.7	7.75
Euklund	695	420	275	.396	65	50.0	13.90
Jodel D-9	700	440	260	.371	31	96.8	7.20
Turbulent Stark	727	463	264	.364	45	91.5	9.95
Fournier RF 01	727	462	265	.364	35	118.5	6.20
Mooney Mite 18 LA	780	520	260	.333	65	95.0	8.20
Andreasson BA6	800	530	270	.338	65	93.5	8.57
Piel Pinochio CP-20	800	390	410	.512	25	97.0	8.25
Heuberger Doodle Bug	830	620	210	.253	85	68.0	12.20
Loving-WR-1 Love	839	631	208	.248	85	66.0	12.70
Honey Bee	860	609	251	.291	65	96.0	8.95
Midget Mustang	875	575	300	.343	85	69.0	12.70
Heinonen HK-1	880	550	330	.375	65	75.0	11.70
Corben Baby Ace	950	575	375	.395	65	112.3	8.45
Smith Miniplane	1000	616	384	.384	100	100.0	10.00
Salvay Stark Skyhopper	1000	650	350	.350	65	100.0	10.00
Stolp Adams Starduster	1080	700	380	.352	125	110.0	9.80
Scweizer 1-30	1100	700	400	.363	65	160.0	6.88

TABLE 13 - TWO PLACE AIRPLANES

Designation	Gross Weight	Empty Weight	Useful Load	Useful L. / Gross W.	Max HP	Wing Area	Lbs/ Sq. ft.
Druine Turbi D-5	1090	610	480	.440	45	139.0	7.85
Jodel D-111	1144	616	528	.462	75	136.6	8.38
Taylorcraft	1200	750	450	.375	65	185.0	6.50
Wittman Tailwind	1250	700	550	.440	115	83.5	15.00
Thorp Sky Scooter	1250	720	530	.425	90	104.0	12.00
Böelkow Junior	1270	750	520	.410	100	94.0	13.50
Nesmith Cougar	1316	624	692	.526	85	82.5	16.00
Piel Emeraude 301 A	1345	758	587	.437	90	116.7	11.50
Aeronca Super Chief	1350	820	530	.393	85	180.0	7.50
Job 5	1350	944	376	.278	90	129.0	10.50
PZL-102 Kos	1390	890	500	.360	90	119.0	11.70
Silvaire 8-F	1400	870	530	.378	90	140.0	10.00
Aircoupe Forney	1400	890	510	.364	90	142.6	9.83
Champion Traveler 7 EC	1450	929	521	.359	95	170.2	8.55
Cessna 150	1500	946	554	.370	100	160.0	9.40
Cessna 140-A	1500	907	593	.396	85	154.0	9.75
Victa Air Tourer	1500	750	750	.500	90	120.0	12.50
Piper PA-18 "95"	1500	800	700	.467	90	178.5	8.40
Kiebitz LF2	1505	990	515	.343	90	158.9	9.50
Stits Sky Coupe	1525	1000	525	.345	100	125.0	12.20
Putzer Elster B	1540	1012	528	.343	90	188.0	8.20
Piper Colt	1650	940	710	.430	108	147.0	11.20
M.S. 880-B - Rallye	1698	1000	698	.412	100	132.0	12.88
Temco Swift	1710	1185	525	.307	125	132.0	13.00
Shinn 2150-A	1817	1125	692	.381	150	144.0	12.12
Zlin Z 326	1984	1404	580	.298	210	166.0	12.00

Determination of Useful Load

Pilot and Passenger....................170 lbs. each
Gasoline...............................6 lbs. per U.S. gallon
Lubricating Oil........................7.5 lbs. per U.S. gallon

These values are fixed by CAR 3.1.

For the PL-1 we assume the following values:

Pilot and Passenger........	(2 x 170)	=	340.0 lbs.
Fuel (25 gal)..............	(25 x 6)	=	150.0 lbs.
Oil (1 gal)................	(1 x 7.5)	=	7.5 lbs.
Baggage or Parachutes..............			40.0 lbs.
			537.5 lbs.

Estimated Gross Weight $= \frac{537.5}{.408} = 1316$

Estimated Empty Weight = 1316 - 538 = 778

Structural Weight Estimation

The structural weight is equal to the empty weight, less the engine weight. The engine dry weight for the C90-12F is listed on page 49 (215.44 lbs.).

Then: Structural Weight = 778 - 216 = 562 lbs.

The weight of the structure major assemblies could be estimated with the following graphs and formulas derived by K. L. Sanders.

Wing

First calculate factor B as shown next:

$$B = \frac{GW(lb) \times n_{ult} \times S_W(ft^2) \times \left[(1.9\ AR) - 4\right]}{1 + (.11\ t/c_r(\%))}$$

Then:

$$B = \frac{1316 \times 9 \times 116 \times \left[(1.9 \times 6.76) - 4\right]}{1 + (.11 \times 15)}$$

$$= 4{,}570{,}000 \xrightarrow{\text{Figure 50}} W_W = 180 \text{ lbs.}$$

Where:

G.W. = Airplane Gross Weight = 1316 lbs.

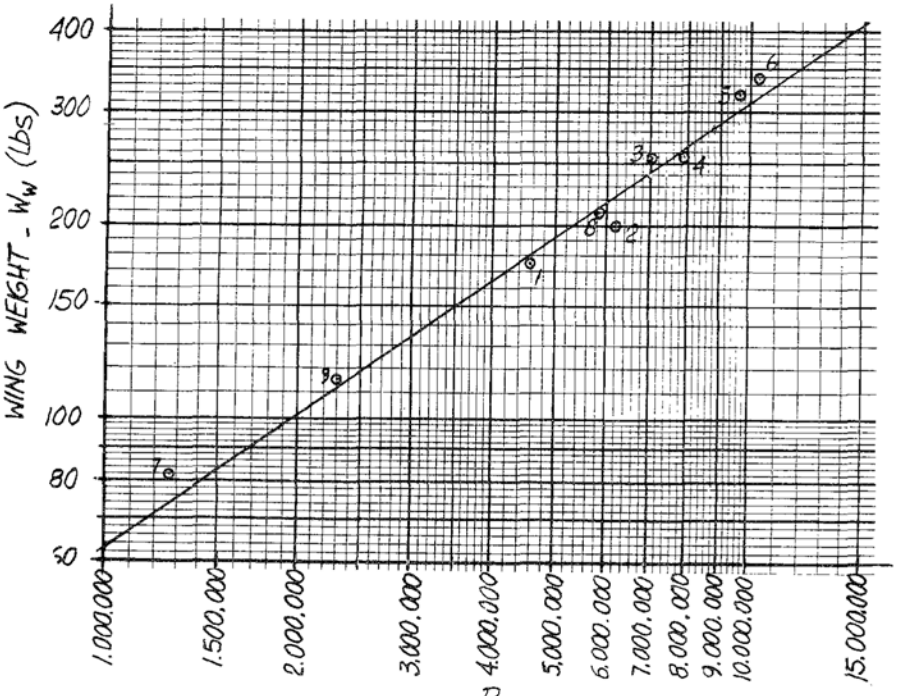
n_{ult} = Ultimate Load Factor = 6 x 1.5 = 9

S_W = Wing Area = 116 sq. ft.

A.R. = Geometric Aspect Ratio = 6.76

t/c_r = Root Airfoil Thickness = 15%

The actual wing weight of the PL-1 resulted in 175 lbs. This remarkable agreement can be seen also for some other wing weights checked with this graph.



$$B = \frac{GW(lb) \times n_{ULT} \times S_W(ft^2) \times \left[(1.9AR) - 4\right]}{1 + \left[.11\ t/c_R(\%)\right]}$$

AR : 5 TO [illegible]

$(t/c)_{ROOT}$: 12 TO 16 %

TAPER RATIO : .7 TO 1.0

Λ : 0 TO 10° (SWEEPBACK)

W_W : WING WEIGHT (INCL. FLAPS AND AILERONS).

1 : PL-1
2 : BEECH MUSKETEER
3 : SCHWEIZER 1-21
4 : SAAB SAFIR
5 : KLEMM 363
6 : BEECH MENTOR
7 : DRIGGS DART I
8 : PIPER J3C
9 : THORP T-18

FIG. 50_ FACTOR "B" FOR WING WEIGHT ESTIMATION

Fuselage

In Figure 51, two curves are shown based on statistical data. The lower curve represents "Optimized Designs," such as light sheet metal structure. The upper curve is representative of a more conservative type of construction, such as welded steel tube and fabric, also wood structures or flat sided heavy sheet metal.

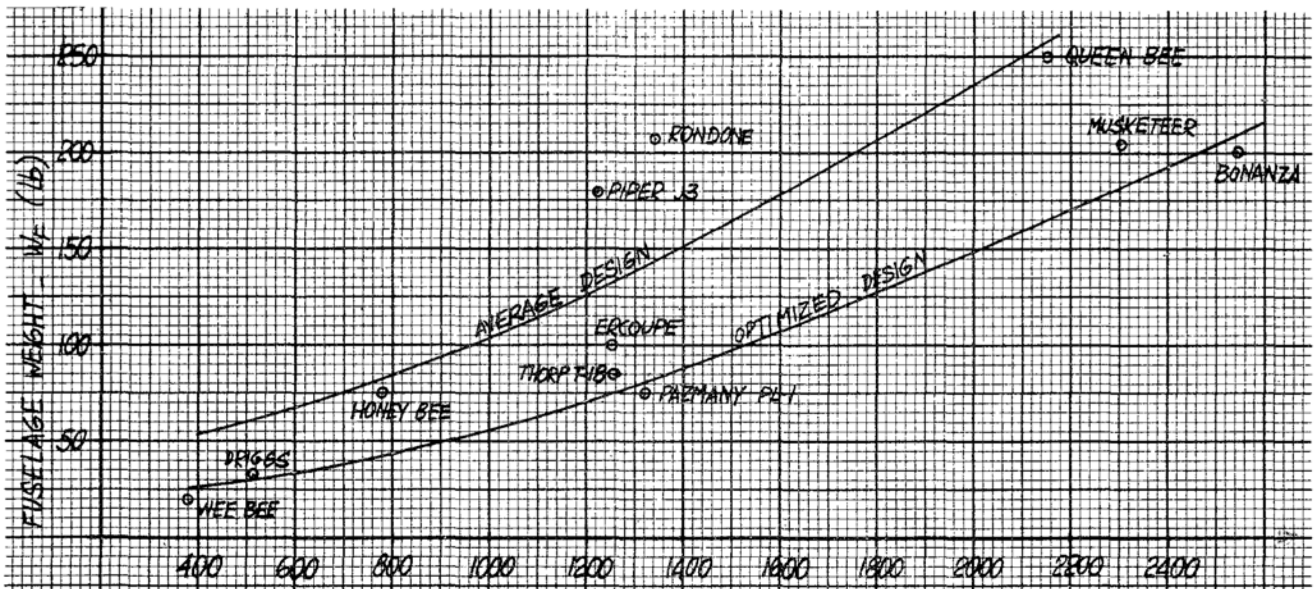


FIG. 51 FUSELAGE WEIGHTS

For the PL-1:

Gross Weight = 1316 lbs. ⟶ Figure 51 ⟶ W_F = 80 lbs.

Horizontal Tail

The weight of horizontal tail is estimated from Figure 52 based on factor A. The calculation of factor A for the PL-1 is shown next:

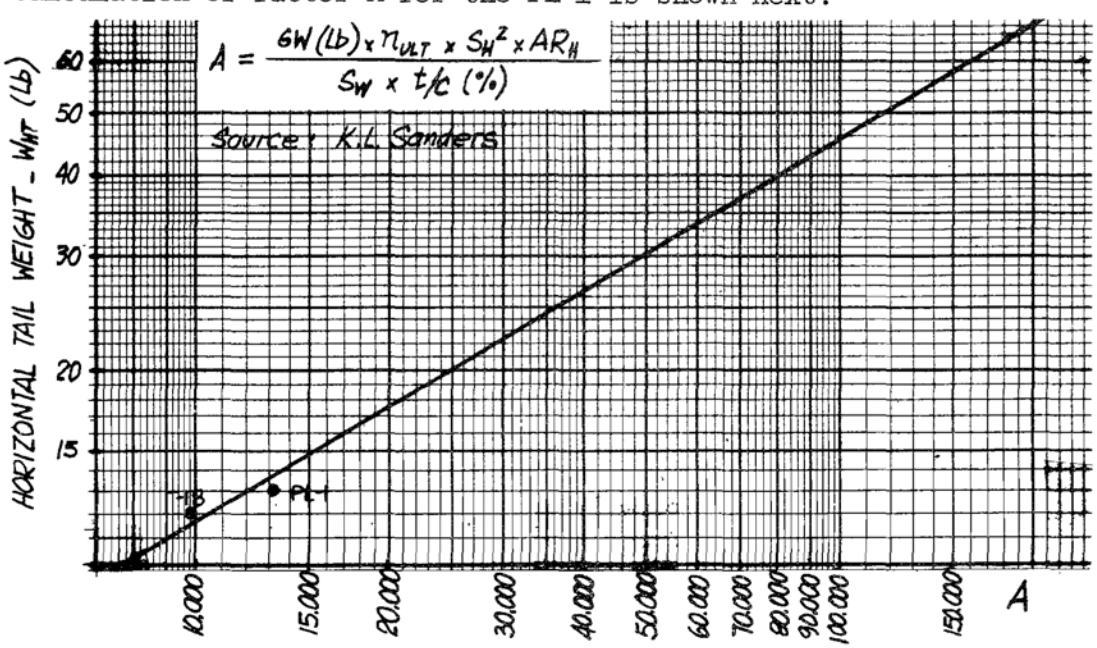


FIG. 52. FACTOR "A" FOR HORIZONTAL TAIL WEIGHT ESTIMATION

$$A = \frac{G.W.\ (lbs.) \times n_{ult} \times S_H^2 \times AR_H}{S_W \times t/c_r\ (\%)}$$

$$A = \frac{1316 \times 9 \times 18^2 \times 3.5}{116 \times 9} = 12.900$$

at A = 12.900 → Figure 52 → W_{HT} = 17.5 lbs.

The actual weight of the PL-1 Horizontal Tail resulted in 18 lbs.

Where:

G.W. = 1316 lbs.

n_{ult} = 9

S_H = 18.0 sq. ft.

AR_H = 3.5

S_W = 116 sq. ft.

t/c_r = Horizontal Tail Airfoil Thickness = 9%

Vertical Tail

The weight of the vertical tail is also estimated based on the horizontal tail weight previously found as follows:

Horizontal Tail Weight = 17.5 lbs.
Horizontal Tail Area = 18.0 sq. ft.
Unitary Weight = $\frac{17.5}{18} \cong$ 1 lb/sq. ft.
Vertical Tail Area = 10.2 sq. ft.
Vertical Tail Weight = 10.2 sq. ft. x 1 lb. sq. ft. = 10.2 lbs.

Landing Gear

The weight of Landing Gears could be estimated in 4.5% of the Gross Weight for tail wheel types and 5.5% for tricycle gears. For the PL-1:

$$1316 \times \frac{5.5}{100} = 72 \text{ lbs.}$$

Assume 70% of this weight for the main gear and 30% for the nose gear. Then:

$$\text{Main Gear} = \frac{70}{100} \times 72 = 50 \text{ lbs.}$$

$$\text{Nose Gear} = \frac{30}{100} \times 72 = 22 \text{ lbs.}$$

Controls

For light aircraft the surface controls weight could be estimated in 2.5% of the Gross Weight. For the PL-1:

$$1316 \times \frac{2.5}{100} = 33 \text{ lbs.}$$

Weight of Major Assemblies

Wing	180.0
Fuselage	80.0
Horizontal Tail	17.5
Vertical Tail	10.2
Landing Gear	72.0
Controls	33.0
	392.7

The weight of major assemblies calculated before should be checked as soon as preliminary layouts became available. The volume of each part is calculated and then multiplied by the specific gravity of the material. The weight of raw material and hardware is listed in books such as S.A.W.E. Weight Handbook (Ref. 3). Also, "Airplane Design Manual" by F. K. Teichmann (Ref. 16) is a very good source of information, either for hardware or airplane components.

"Practical Light Plane Design and Construction" by W. J. Fike (Ref. 17) has some information on light airplane components weight.

Another very valuable source of information is the "Air Associates" catalog (Ref. 18) which provides dimensional and weight data on thousands of standard hardware items.

When there is no weight data on some components, the solution is to prepare simple drawings (1/10 scale is adequate) and determine the weight analytically.

Good results can be obtained by using some simplifying assumptions. For instance, in all sheet metal parts, rivet and bolt holes are not deducted and rivet weights are not added to the sheet metal weights. Instead, 5% is added to the calculated weight to take care of rivets and anti-corrosive paint. Next, some examples for the PL-1:

Spinner

Cone Surface: $\dfrac{\pi \times d \times h}{2} = \dfrac{3.14 \times 10 \times 10}{2} = 157$ sq. in.

Backing Plate: $\dfrac{\pi \times d^2}{4} = \dfrac{3.14 \times 10^2}{4} = 79$ sq. in.

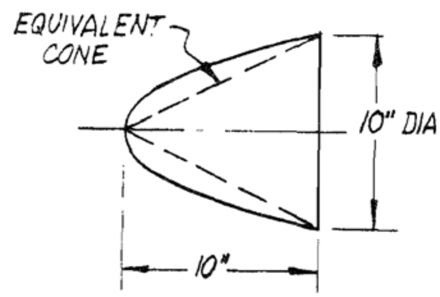


FIG. 53

Total Surface = 157 + 79 = 236 sq. in.

Dural Sheet .064" thick, Vol = 236 x .064 = 15.1 in.^{3}

Dural Spec. Weight = .100 lbs./in.^{3}

Weight = 15.1 in.^{3} x .100 lbs/in.^{3} = 1.51 lbs.

On the backing plate, there are some lightening holes which are not deducted because they are compensated by the weight of the fasteners, lips, reinforcements, etc.

Engine Cowling

The weight will be calculated based in projected areas. The nose piece is made of fiberglass .050" thick. (Weight of fiberglass Laminate ≅ .07 lb/in^3).

Frontal Projection	= 17.0 x 33.0	= 560 in.^{2}	
Top and Bottom	= 33.0 x 6.0 x 2	= 396 in.^{2}	
Sides	= 22.0 x 6.0 x 2	= 264 in.^{2}	
	Total	=1220 in.^{2}	1220 in.^{2}

Prop. Hole $= \dfrac{3.14 \times \overline{13.0}^2}{4}$ = 133 in.^{2}

2 Cooling Air Holes = 7.5 x 5.5 x 2 = $\underline{82 \text{ in.}^2}$
215 in.^2 $\quad$ $\underline{215 \text{ in.}^2}$
1005 in.^2

Volume = 1005 x .050 = 50 in.^{3}
Weight = 50 x .07 = 3.5 lb.

The carburetor scoop is made of fiberglass .030" thick.

Frontal projection = (5.0 x 6.6) - (3.0 x 4.6) = 19 in.^{2}
Bottom = 17.0 x 6.6 = 112 in.^{2}
Sides = $\frac{17.0 \times 3.0}{2}$ x 2 = $\underline{51 \text{ in.}^2}$
182 in.^{2}

Volume = 182 x .030 = 5.5 in.^{3}
Weight = 5.5 x .07 = .4 lb.

The cowling is made of .025" aluminum sheet metal. (Weight of aluminum = .100 lb/in.^{3}.

Side projection = $\left(\frac{22.0 + 28.0}{2}\right)$ x 28.0 x 2 = 1400 in.^{2}
Top and Bottom = 33.0 x 28.0 x 2 = $\underline{1848 \text{ in.}^2}$
3248 in.^{2}

Volume = 3248 x .025 = 81.2 in.^{3}
Weight = 81.2 x .10 = 8.12 lb.

Resume:

Front Piece.............3.50 lb.
Carburetor Scoop........ .40 lb.
Cowling.................$\underline{8.12 \text{ lb.}}$
12.02 lb.

The equivalent projected areas used in this estimation are larger than the real developed areas, but the increase can be considered representative of unaccounted reinforcements, baffles and fasteners.

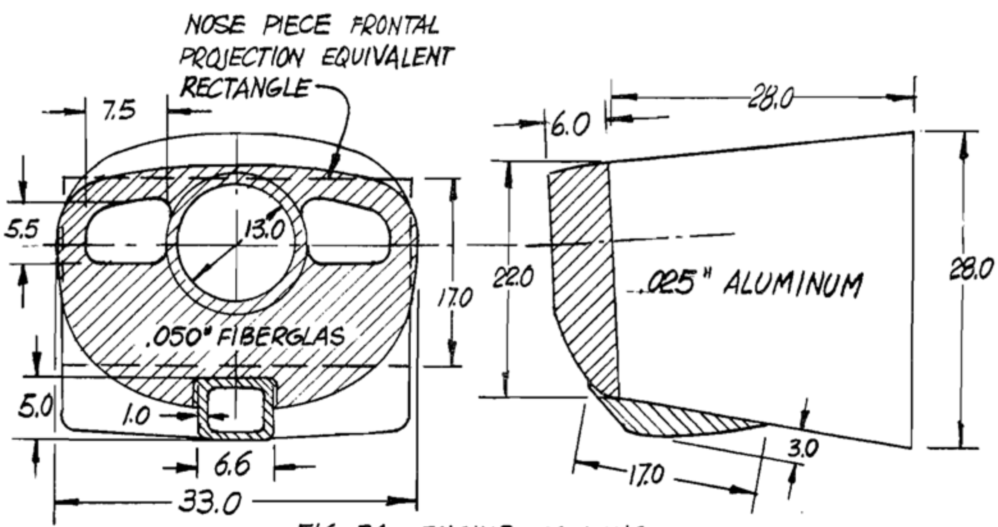


FIG. 54 - ENGINE COWLING

Engine Mounting

Figure 55 shows the geometry of the engine mounting tubes, which are assumed to be 3/4" dia. x .035". The lengths of the tubes are calculated analytically, based in the three ortogonal projections:

Length Calculations

Tube a: $l_a = \sqrt{\overline{14.5}^2 + \overline{6.0}^2 + \overline{11.5}^2} =$

$= \sqrt{210 + 36 + 132} = \sqrt{378} = 19.4"$

Tube b: $l_b = \sqrt{\overline{14.5}^2 + \overline{5.4}^2 + \overline{9.7}^2} =$

$= \sqrt{210 + 29 + 94} = \sqrt{333} = 17.7"$

Tube c: $l_c = \sqrt{\overline{14.5}^2 + \overline{10.6}^2 + \overline{9.7}^2} =$

$= \sqrt{210 + 123 + 94} = \sqrt{427} = 20.7"$

Tube d: $l_d = \sqrt{\overline{14.5}^2 + \overline{10.6}^2 + \overline{5.3}^2} =$

$= \sqrt{210 + 123 + 23} = \sqrt{361} = 19.0"$

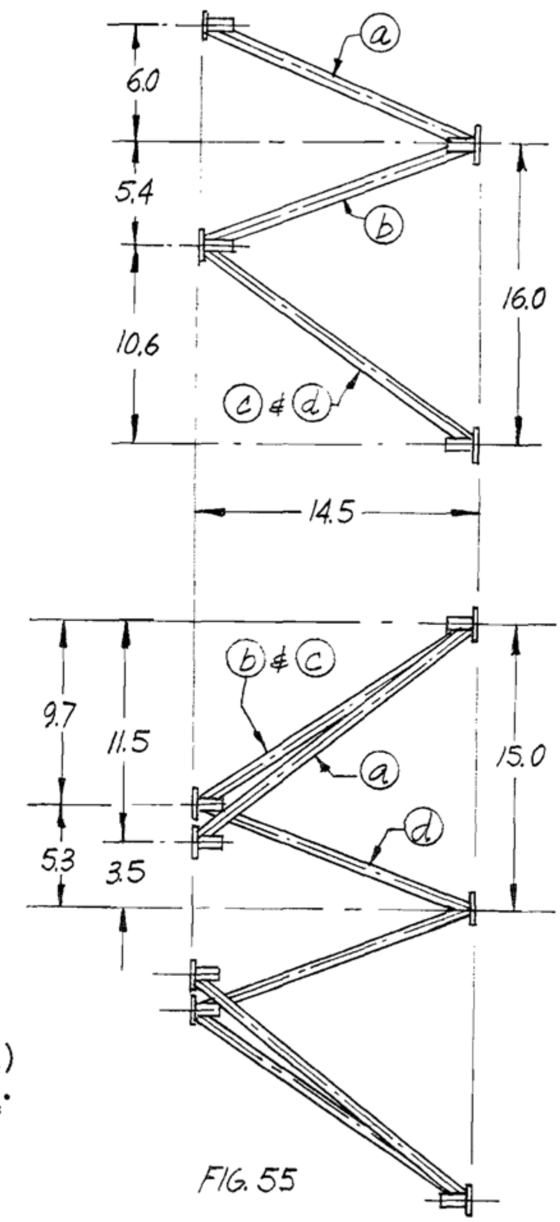


FIG. 55

Resume:

2 Tubes "a" = 19.4" x 2 = 38.8"
2 Tubes "b" = 17.7" x 2 = 35.4"
2 Tubes "c" = 20.7" x 2 = 41.4"
2 Tubes "d" = 19.0" x 2 = 38.0"
153.6"

Tube 3/4" x .035"; section = .0786 sq. in.
Volume = .0786 x 153.6 = 12.1 in.^{3}
Steel Specific Weight = .283 lbs/in.^{3}
Weight of Tubes = .283 x 12.1 = 3.43 lbs.
Weight of Bushings, Bolts, etc. = 1.00 lb. (estimated)
Engine Mounting Total Weight = 3.43 + 1.00 = 4.43 lbs.

Fuel Tanks

The weight of aluminum fuel tanks including filler neck and cap could be estimated in .75 lbs per gallon. Terne plate tanks weighs approximately 1.00 lb. per gallon.

The PL-1 has wing tip fiberglass tanks. Their weight will be calculated next:

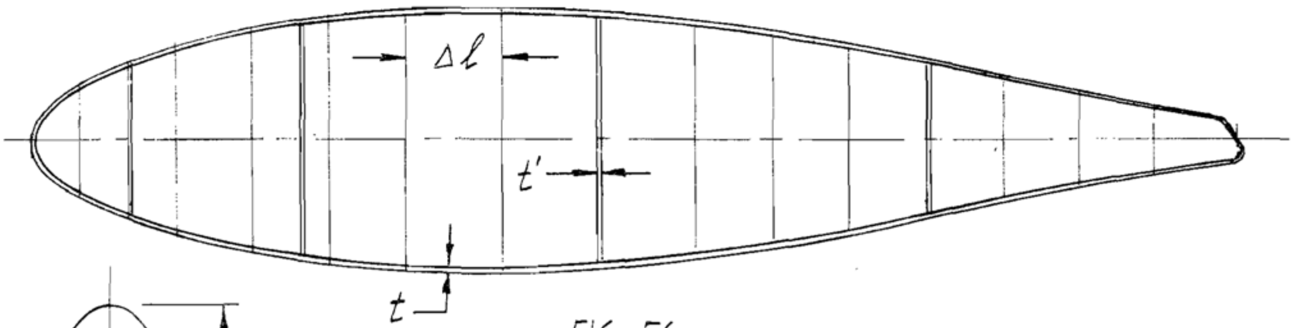
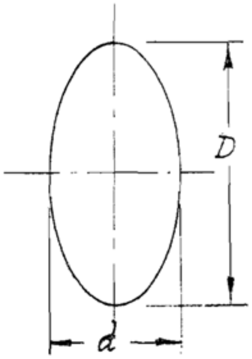


FIG. 56



$$D = 2d \begin{cases} \text{Elipse perimeter} = \frac{\pi}{2} \times (d + D) = 1.57 \times 3d = 4.72 \times d \\ \text{Elipse area} = \frac{\pi}{4} \times D \times d = .79 \times 2d \times d = 1.58\ d^2 \end{cases}$$

Weight of fiberglass laminate $\cong$ .07 lbs/in.^{3}

Divide the tank in sectors as shown, and calculate the volume of each shell sector (ΔV_{shell}).

ΔV_{shell} = Perimeter x Δl x thickness = 4.72 x d x Δl x t

The total shell volume will be the summation of all sectors:

$$V_{shell} = \Delta V_1 + \Delta V_2 + \text{------} + \Delta V_{10} = 77 \text{ in.}^3$$

The volume of baffles is:

$$V_{baffle} = \text{Elipse area} \times t' = 1.58\ d \ \times t'$$

and for the 4 baffles results: 17 in.^{3}

The weight of the fiberglass will be:

$$(77 \text{ in.}^3 + 17 \text{ in.}^3) \times .07 \text{ lbs/in.}^3 = 6.58 \text{ lbs.}$$

The weight of reinforcements, filler necks, fuel strainer, latches, pipes and fittings is estimated in 2.0 lbs. Therefore, the weight of each tank will be:

Weight of complete tank = 6.58 + 2.00 = 8.58 lbs.
Weight of two tanks = 8.58 x 2.00 = 17.16 lbs.

<u>Fuel Lines</u>

3/8" x .035 tubes Aluminum - Length $\cong$ 380"

Tube cross section = .0374 in.^{3}
Tube Volume = .0374 x 380 = 14.2 in.^{3}
Tube Weight = 14.2 x .100 = 1.42 lbs.

Tube Fittings $\cong$ .50 lbs.

Total weight of fuel lines = 1.42 + .50 = 1.92 $\cong$ 2.0 lbs.

Equipment

These weights are determined either by weighing or from catalogs.

Propeller - Metal - 66" dia.		19.50 lbs.
Engine Baffles		1.00 lbs.
Exhaust Pipes Cabin and Carburetor Heater		9.00 lbs.
Battery and Case - 23 + 1.25		24.25 lbs.
Auxiliary Fuel Pump		1.00 lbs.
Instruments		
Airspeed Indicator	1.00 lbs.	
Altimeter	1.00 lbs.	
Magnetic Compass	1.00 lbs.	
Oil Pressure Gauge & Tubes	1.00 lbs.	
Oil Temperature Gauge	.50 lbs.	
Tachometer & Cable	1.00 lbs.	
Turn & Bank Indicator	1.50 lbs.	
Climb Indicator	1.00 lbs.	
Manifold Pressure Indicator	.80 lbs.	
Clock	.70 lbs.	
Ammeter	.40 lbs.	
	9.90 lbs.	9.90 lbs.
Seat Belts & Shoulder Harness		2.00 lbs.
Cushions		2.00 lbs.
Cockpit Lights (2)		.30 lbs.
Landing Light		1.00 lbs.
Position Lights (wing tip: 3 oz each - tail: 5 oz.)		.70 lbs.
Rotary Beacon		1.00 lbs.
2 Brake Cylinders (Scott 4408)		1.00 lbs.
Radio & Power Supply (VHF)		6.00 lbs.
Sound Proofing		2.00 lbs.
		80.65 lbs.

Windshield and Canopy

The weight was calculated based on drawings and resulted:

Windshield	6.0 lbs.
Canopy	14.0 lbs.
	20.0 lbs.

Engine Controls

The weight is estimated in 3 lbs.

Resume of Structural Weight

Major Assemblies	392.7 lbs.
Spinner	1.5 lbs.
Engine Cowling	12.0 lbs.
Engine Mounting	4.5 lbs.
Fuel Tanks	17.2 lbs.
Fuel Lines	2.0 lbs.
Equipment	80.7 lbs.
Windshield and Canopy	20.0 lbs.
Engine Controls	3.0 lbs.
	533.6 lbs.

This result is very close to the Estimated Structural Weight (562 lbs.) on page 62 . The difference, 562 - 533 = 29 lbs., probably will vanish during the construction of the prototype. After the first airplane is built, it is always possible to refine the design. Simplifications could be made; sometimes one part could be redesigned to make the work of two. All this will eventually reduce the Structural Weight, but for preliminary desing, performance and loads calculation, the high value should be used.

1-22 AIRPLANE BALANCE

Now we are in condition to proceed with one of the most important steps in the aircraft design. This is the location of the C.G. The aircraft designer should permanently keep track of the weight and balance of the airplane. This is so important that every aircraft factory has a "Weights Group" in its Engineering Department. "Weight Engineers" continuously check the weight of each component during the design. Each drawing should be signed by them before release. Sometimes the weight of parts or assemblies result higher than estimated, then a decision should be made to either redesign the part and try to make it lighter even if it results more complicated, or to leave the design as it is and take the weight penalty. Parts located away from the C.G. are more critical than parts close to the C.G.

The C.G. position is calculated simply by calculating the moments of each component with respect to reference lines. The following procedure is recommended: (See Table 14)

1. Draw a side view of the airplane at a convenient scale (1/10 is adequate). Indicate the C.G. of each component by a small circle. It requires some practice to estimate by "eye ball" the position of the C.G. of some components. As a general guide, the C.G. of wings lies at 40% of the Mean Aerodynamic Chord. The C.G. of Vertical and Horizontal Tails can be located at 50% of the respective mean chord. The C.G. of fuselage could be estimated at 40% of the fuselage length measured between the firewall and the tail cone.

2. Enter the weight of each component in column 3 of Table 14.

3. Draw a vertical reference line at the spinner vertice and a horizontal reference line at ground level. (See Figure 57)

4. Measure the horizontal and vertical distance of each component C.G. from the reference lines. Enter these values in columns 4 and 6 of Table 14.

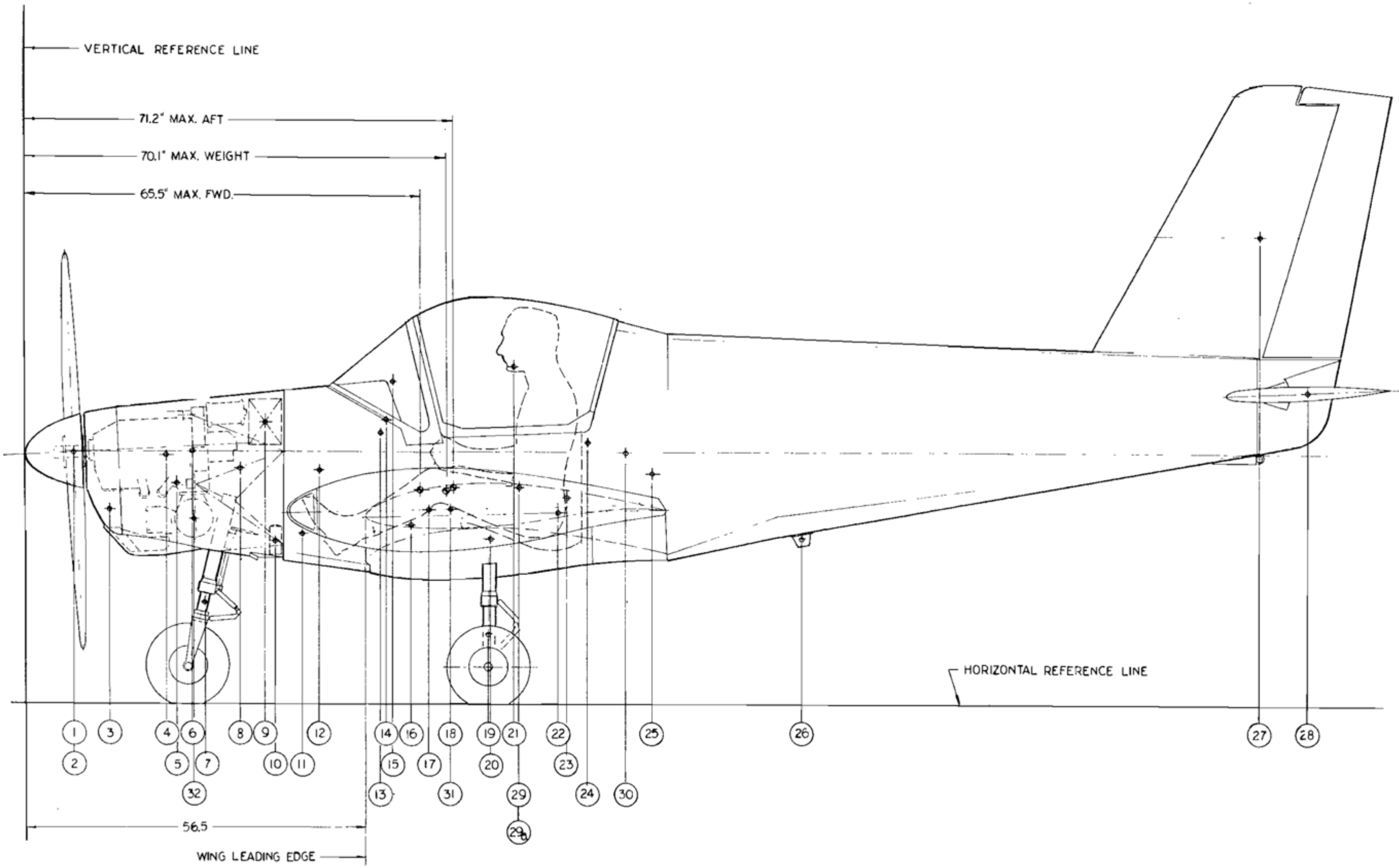
5. Multiply the weight of each component by its horizontal distance (column 3 x column 4) and enter the result in column 5.

6. Multiply the weight of each component by its vertical distance (column 3 x column 6) and enter the result in column 7.

7. Add column 3 to obtain the sum of weights. Add column 5 to obtain the sum of horizontal moments. Add column 7 to obtain the sum of vertical moments.

8. Divide the sum of horizontal moments by the sum of weights to obtain the horizontal location of the C.G.

9. Divide the sum of vertical moments by the sum of weights to obtain the vertical location of the C.G.


VERTICAL REFERENCE LINE
71.2" MAX. AFT
70.1" MAX. WEIGHT
65.5" MAX. FWD.
HORIZONTAL REFERENCE LINE
56.5
WING LEADING EDGE
1
2
3
4
5
6
7
32
8
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11
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Figure 57

TABLE 14- BASIC BALANCE

①	②	③	④	⑤	⑥	⑦
Item	Designation	Weight	Horz. Arm.	Horz. Mom.	Vert. Arm.	Vert. Mom.
1	Spinner	1.51	8.0	12	42.0	63
2	Propeller	19.50	8.0	156	42.0	820
3	Landing Light	1.00	14.0	14	32.0	32
4	Engine and Baffles	216.44	23.5	5,080	41.5	8,980
5	Exhaust Pipes and Heater	9.00	25.0	225	37.0	333
6	Engine Cowling	12.02	28.0	336	42.0	504
7	Nose Gear	22.00	30.0	660	17.0	374
8	Engine Mounting	4.43	36.0	159	39.0	173
9	Battery and Case	24.25	40.0	970	47.0	1,140
10	Auxiliary Fuel Pump	1.00	41.5	41	27.0	27
11	Brake Cylinders	1.00	46.0	46	28.0	28
12	Engine Controls	3.00	49.0	147	39.0	117
13	Radio and Power Supply	6.00	59.0	354	45.0	270
14	Instruments	9.90	60.0	594	47.0	465
15	Windshield	6.00	61.0	366	53.5	321
16	Sound Proofing	2.00	64.0	128	30.0	60
17	Fuel Lines	2.00	67.0	134	32.0	64
18	Fuel Tanks	17.16	70.5	1,210	32.0	548
19	Main Gear	50.00	77.0	3,850	12.0	600
20	Wing	180.00	77.5	13,950	27.5	4,950
21	Canopy	14.00	81.0	1,134	56.0	785
22	Belts and Cushions	4.00	88.5	354	32.0	128
23	Controls	33.00	90.0	2,970	34.5	1,138
24	Cockpit and Position Lights	1.00	93.5	94	43.5	43
25	Fuselage	80.00	104.0	8,320	38.0	3,040
26	Rotary Beacon	1.00	129.0	129	27.5	28
27	Vertical Tail	10.20	205.0	2,090	78.0	795
28	Horizontal Tail	17.50	213.5	3,740	52.0	910
		Σ = 748.91		Σ = 47,263		Σ = 26,736

$$\text{Horizontal Position of C.G.} = \frac{47{,}263}{748.91} = 63.2''$$

$$\text{Vertical Position of C.G.} = \frac{26{,}736}{748.91} = 35.7''$$

The maximum Aft C.G. Position is the most critical for stability; therefore, this will be calculated first. The most rearward C.G. position will occur under the following assumption:

No oil in the engine tank
Baggage overload (assume 60 lbs.)
Two heavy passengers
Airplane in climb, assume 1/2 fuel in tanks piled up in the rear half of the tanks

TABLE 15 - MAXIMUM AFT C.G. POSITION

Item	Description	Weight	Horz. Arm.	Horz. ~~Arm.~~ Mom.	Vert. Arm.	Vert. ~~Arm.~~ Mom.
	Basic Airplane	748.91	63.2	47,263	35.7	26,736
29	Pilot	170.00	82.0	13,940	36.0	6,120
29a	Passenger	170.00	82.0	13,940	36.0	6,120
30	Baggage	60.00	100.0	6,000	42.0	2,520
31	12 Gallon Fuel in Rear of Tanks	72.00	80.0	5,760	32.0	2,302
		Σ = 1220.91		Σ = 86,903		Σ = 43,798

$$\text{Horizontal Position of C.G.} = \frac{86{,}903}{1220.91} = 71.2''$$

$$\text{Vertical Position of C.G.} = \frac{43{,}798}{1220.91} = 36.0''$$

The leading edge of the wing is at 56.5" from the reference line, therefore the horizontal distance between the wing leading edge and the C.G. will be:

$$d = 71.2 - 56.5 = 14.7 \text{ in.}$$

And in % of wing chord:

$$d(\%) = \frac{14.7 \text{ in.}}{50 \text{ in.}} \times 100 = 28.4\%$$

This value looks good. In general, it is desirable to keep the C.G. at any condition ahead of the 30% of the Mean Aerodynamic Chord.

The most forward C.G. position should be calculated next. This condition is critical for elevator dimensioning, as will be seen in the next volume.

The most forward C.G. occurs under the following assumptions:

No baggage - No passenger - No fuel
Very light pilot
Maximum oil in the engine tank

TABLE 16 - MAXIMUM FORWARD C.G. POSITION

Item	Description	Weight	Horz. Arm.	Horz. Mom.	Vert. Arm.	Vert. Mom.
	Basic Airplane	748.91	63.2	47,263	35.7	26,736
29	Pilot	120.00	82.0	9,840	36.0	4,320
32	Oil (1 gallon)	7.50	28.0	210	31.0	233
		Σ = 876.41		Σ = 57,313		Σ = 31,289

Horizontal Position of C.G. = $\frac{57,313}{376.41}$ = 65.5"

Vertical Position of C.G. = $\frac{31,289}{876.41}$ = 35.7"

And in % of wing chord:

65.5 - 56.5 = 9.0 in.

$\frac{9.0}{50}$ x 100 = 18%

This value also looks good. For preliminary design purposes, the Maximum Forward C.G. should be kept behind 15% of the Mean Aerodynamic Wing Chord.

And finally the C.G. for airplane Gross Weight is calculated. Obviously, it must fall between the two extremes calculated before:

TABLE 17 - MAXIMUM WEIGHT C.G. POSITION

Item	Designation	Weight	Horz. Arm.	Horz. Mom.	Vert. Arm.	Vert. Mom.
	Basic Airplane	748.91	63.2	47,263	35.7	26,736
29	Pilot	170.00	82.0	13,940	36.0	6,120
29a	Passenger	170.00	82.0	13,940	36.0	6,120
30	Baggage	40.00	100.0	4,000	42.0	1,680
31	Fuel (25 gal)	150.00	70.5	10,775	32.0	4,800
32	Oil (1 gal)	7.50	28.0	210	31.0	233
		Σ = 1286.41		Σ = 90,128		Σ = 45,689

Horizontal Position of C.G. = $\frac{90,128}{1286.41}$ = 70.1"

Vertical Position of C.G. = $\frac{45,689}{1286.41}$ = 35.5"

And in % of wing chord:

d = 70.1 - 56.5 = 13.6 in.

$\frac{13.6}{50}$ = 27.2%

If the C.G. does not fall between the desired extremes, something must be shifted. Fuel and passengers should be located as close to the C.G. as possible so the changes in trim will be minimized. Other items could be moved around within their own limitations.

The most common remedy in case of trouble is to move the engine. In extreme cases use ballast, but this certainly is a poor solution. The best way is to start all over leaving everything in its place except the wing which is moved to the desired position. This "trial and error" process should be repeated as many times as required. Good Luck!

3) The M.A.C. of the entire wing will be:

$$\text{M.A.C.} = \frac{c_1 A_1 + c_2 A_2}{A_1 + A_2}$$

4) The distance from the Airplane Center Line to the M.A.C.:

$$y = \frac{y_1 A_1 + y_2 A_2}{A_1 + A_2} \qquad (y_1 = a_{1/2})$$

5) The distance from a spanwise reference line to the M.A.C.:

$$x = \frac{x_1 A_1 + x_2 A_2}{A_1 + A_2}$$

6) The distance from a Reference Plane to the M.A.C.: (See Figure 60 - Front View)

$$z = \frac{z_1 A_1 + z_2 A_2}{A_1 + A_2}$$

d) Elliptical Wing

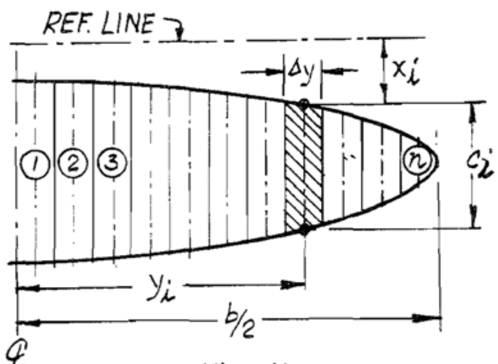

Fig. 61

1) Divide the wing in "n" strips of equal width (Δy).

2) Measure the mean chord of each strip (c_i) and the distance from the airplane center line to the strip mean chord (y_i).

3) Prepare Table 18.

4) Add Columns 2, 3, 5 and 7.

5) The M.A.C. of the wing will be:

$$\text{M.A.C.} = \frac{\Sigma\,③}{\Sigma\,②}$$

TABLE 18

1	2	3	4	5	6	7
Strip Number	Strip Chord	2 x 2	Distance to ℄	2 x 4	Distance to Ref. Line	2 x 6
1	c_1	c_1^2	y_1	$c_1 y_1$	x_1	$c_1 x_1$
2	c_2	c_2^2	y_2	$c_2 y_2$	x_2	$c_2 x_2$
3	c_3	c_3^3	y_3	$c_3 y_3$	x_3	$c_3 x_3$
-	-	-	-	-	-	-
-	-	-	-	-	-	-
-	-	-	-	-	-	-
n	c_n	c_n^2	y_n	$c_n y_n$	x_n	$c_n y_n$
Σ	-------	-------		-------		-----

6) The distance from the Airplane Center Line to the M.A.C.

$$y = \frac{\sum ⑤}{\sum ②}$$

7) The distance from a spanwise reference line to the M.A.C.

$$x = \frac{\sum ⑦}{\sum ②}$$

APPENDIX "B"

Some Useful Conversion Factors

Multiply	By	To Obtain	Multiply	By	To Obtain
Atmosphs.	14.69	Lbs/sq. in.	kilogs	2.204	pounds
Atmosphs.	2,116.	Lbs/sq. ft.	kilogs	35.27	ounces
Centimets.	.3937	inches	kilog/sq.m	.2048	lbs/sq. ft.
Centimets.	.0328	feet	kilomets	.6213	miles
cm./sec.	.0328	feet/sec	knots	1.688	feet/sec.
Cubic cm.	.0610	cubic in.	knots	1.151	miles/hr
Cubic ft.	1,728.	cubic in.	liter	.2641	gallons
Cubic ft.	7.481	gallons	meters	39.37	inches
Cubic ft.	.0283	cub. meters	meters	3.280	feet
Cubic in.	1/231	gallons	miles	5,280	feet
Feet	12	inches	miles	1.609	kilomets.
Feet	1/3	yards	miles/hr.	1.466	feet/sec.
Feet	30.48	centimet.	miles/hr.	.8683	knots
feet/min	.0113	miles/hr.	Naut. mi.	6,080	feet
feet/sec	.6818	miles/hr.	Naut. mi.	1.151	miles
feet/sec	.5920	knots	ounces	1/16	pound
gallons	231	cubic in.	ounces	28.35	grams
gallons	.1336	cubic feet	pounds	453.6	grams
gallons	3.785	liters	pounds	16	ounces
gallons	.8326	imp. gallons	lbs/sq. ft	4.882	kilog./sq. m
Horsepower	33,000	ft. lb/min.	quarts(liq	57.75	cub. in.

Standard gravity (g) = 32.174 ft/sec.^{2}

Atmospheric Standards at Sea Level:
Pressure: 29.92 in hg. = 2116 lb/ft^2
Temp. NACA: 59°F.
Density: = 0.002378 lb sec^2/ft^4

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Ref.	Publication Title	Author	Comments	Editor & Address
1	Civil Air Regulations Part 3	Civil Aeronautics Board	Government publication which establishes standard regulations for aircraft certification	U.S. Government Printing Office Washington 25, D.C.
2	NACA Industry Conference on Personal Aircraft	NACA - Staff Members	This publication is a very valuable source of information for small aircraft designers	NASA Office of Scientific and Technical Information Washington 25, D.C.
3	Weight Handbook	S.A.W.E.	A reference book which contains the weight of thousands of standard parts, raw materials, liquids, etc. An abbreviated pocket book could be obtained for $1.	Robert G. Farr 8428 Lurline Canoga Park, California Executive Secretary Society of Aeronautical Weight Engineers
4	NACA Technical Report 927 - Appreciation and Prediction of Flying Qualities	W. H. Phillips	Summary of data to provide satisfactory stability and control characteristics	U.S. Government Printing Office Washington 25, D.C.
5	NACA Technical Report 824 - Summary of Airfoil Data	Abbott, Doenhoff, & Strivers Jr.	Ordinates and characteristic curves for a great number of airfoils	U.S. Government Printing Office Washington 25, D.C. Out of Print - See Ref. 6
6	Theory of Wing Sections (Book)	Abbott & Doenhoff	Contains all information of Ref. 5 plus more material. Extremely valuable book for $2.95. (Paperback)	Dover Publications, Inc. 180 Varick Street New York 14, New York
7	NACA Technical Note 1945 - Aerodynamic characteristics	L. K. Loftin & H. A. Smith, NACA	This report shows the influence of the R.N. on the airfoil characteristics	NASA, Office of Scientific and Technical Information Washington 25, D.C.

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15	NACA WR L-95 A method for predicting the Elevator Deflection required for Land	R. F. Goranson	This report is mostly useful for elevator dimensioning	NASA - Office of Scientific and Technical Information Washington 25, D.C.
16	Airplane Design Manual	F. K. Teichmann	A great source of information, easy reading, covers all aspects of design in great detail	Pitman Publishing Company New York
17	Practical Lightplane Design and Construction	W. J. Fike	A useful source of information with much practical data	W. J. Fike Anchorage, Alaska
18	Air Associates Catalog	Air Associates	All standard parts needed in an airplane are listed here: dimensions, weights, specs.	Air Associates Inc. Teterboro Air Terminal Teterboro, New Jersey
19	AGARD Report 811 Factors Affecting the Field Length of STOL Aircraft	G. W. Johnston	Performance implication as reflected by the requirement of operating aircrafts from 500 foot fields	NASA - Office of Scientific and Technical Information Washington 25, D.C.

Ref.	Publication Title	Author	Comments	Editor & Address
8	Technical Report 903 - Theoretical and Experimental data for a number of NACA 6A-series airfoil sections	L. K. Loftin	This report presents aerodynamic and geometric data on NACA 6A-series airfoils, designed to eliminate the trailing edge cusp characteristic of the NACA 6-series sections	U.S. Government Printing Office Washington 25, D.C.
9	The Aerodynamic Characteristics of Flaps Aeronautical Research Council, Technical Report No. 2622	A. D. Young	A very complete summary on aerodynamic characteristics of every type of flaps	Her Majesty's Stationery Office - London, England
10	Airplane Performance Stability and Control	C. D. Perkins and R. E. Hage	A college text book, and a classical reference book for stability and control work	John Wiley & Sons Inc. New York
11	Fluid-Dynamic Drag	S. F. Hoerner	The most complete book on drag. Probably the only technical book ever to be found beside a bed in the morning. Fascinating!	S. F. Hoerner 148 Busteed Drive Midland Park, New Jersey
12	Airplane Design	K. D. Wood	A text book of airplane design with special emphasis on costs.	University Bookstore Boulder, Colorado
13	NACA Technical Rep. 823 Experimental verification of a simplified Vee-tail theory and analysis of available data on complete models with Vee-tails.	P. E. Purser and J. P. Campbell	Vee-tails do not provide reduction in area compared with conventional tail. Methods for designing Vee-tails.	U. S. Government Printing Office Washington 25, D.C.
14	NACA Technical Report 665 Calculation of the Aerodynamic Characteristics of tapered wings with partial span flaps	H. A. Pearson and R. F. Anderson	This report also represents wind tunnel tests of flapped 23012 and 23015 airfoils and complete wings with partial span flaps.	U. S. Government Printing Office Washington 25, D.C.