MODEL AERONAUTICS MADE PAINLESS

BY

R. J. HOFFMAN

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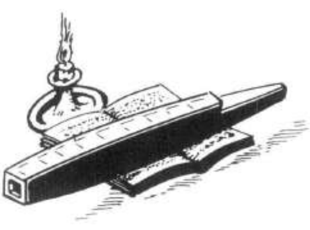

BY

R.J.HOFFMAN

1955

Raoul J. Hoffman

1884-1956

FOREWORD.

The purpose of this booklet is to appeal to a wider circle of readers to use the contents and take further steps in advancing model aeronautics to a higher level. The collection of references and investigations is the result of fifty years of experience in engineering, which includes designing of airplanes from 20 to 1200 h.p. since 1909 and scientific testing of models since 1912.

The mathematical aspect of model engineering is not very promising because of the great variations in basic designs. In spite of the lack of systematic tests with models an attempt has been made to use aerodynamics for large airplanes and low-velocity wind-tunnel reports in predicting the performances and the stability of models, many of them contradictory.

The relatively large forces of torque, power, gyroscopic moments of high speed propellers not present in large airplanes makes model aeronautics a new field of engineering. Attention is called to the problem of weight distribution, to the relative position of the zero-lift lines of airfoils and the distance of the thrustline to the center of gravity.

Thanks to all model builders who supplied flying characteristics of their models, that were incentive of many theoretical investigations appearing in this booklet.

If following, poorly arranged pages contribute only in a small share to greater interest in model aeronautics the labor involved was well rewarded.

The Author.

This Edition was prepared from
author's original paste-up copy.

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THE AIR- Moving an object through the air or any other fluid requires a force to overcome the resistance called the drag. The drag is influenced by the size, shape, velocity, surface texture of the body and the condition of the air. The drag is equal to the velocity squared, times half of the standard density, times the area and times the drag coefficient. The drag coefficient of a flat plate vertical to the air stream is 1.28. Other coefficients are given on page 6. The area is usually taken at the maximum cross sections of the bodies.

Inclining a flat plate in an air stream the air is deflected downward. The forming of a vacuum on the upper surface is out of question and air is transfered to the upper surface which may flow past the lower surface. This transfer is the upwash. The upwash can be analyzed as a combination of a straight flow and a circular flow about the airfoil. This circulation is accepted as the cause of lift. The upwash is not sufficient to meet the air flowing past the lower surface at the trailing edge and turbulent vortices will form in the air stream as it leaves the plate or the airfoil. The air leaving at an angle is the downwash. The turbulent wake behind the airfoil is the Karman vortex street.

The drag of a flat body can be reduced by streamlining, shaping it to a cross sectional outline similar to the one of a fish (trout). Such symmetrical sections are basic for designing sections for wings by bending, cambering the meanline with its maximum point at the 25 to 60 per cent of the chord and also varying the maximum thickness. The camber of the meanline and the thickness depend on the required performances.

Sections are tested in wind-tunnels at various angles. The resultants are inclined to the air stream, and for simplicity are resolved into two components, one parallel to the air stream,the drag and one vertical to the flow,the lift. For use they are reduced to CD, the drag coefficient and to CL, the lift coefficient.The angles are measured from any line selected by the laboratory. In addition to the coefficients points are given on the base line where the resultants intersect, which are the centers of pressure, c.p. With the angle of attack, the c.p. and the coefficients the resultants can be plotted. The resultant can be resolved into components again at any other point on the resultant. This shows that there is no center of lift or drag even for the reason of simplifying investigation of problems, except if the reference point is at a great distance.

The increase of the CL coefficient with increase of angle for regular airfoils is the same, except the zero-lift line will be at different angle of attack. The change per degree is the slope of lift coefficients, which however depends on the ratio of the span and the average chord, the aspect ratio. Tests on slopes are plotted on a diagram and an empirical equation is formulated which simplifies estimate that includes function of aspect ratio of areas.

The slope of lift coefficients is also the same for symmetrical sections, and call them non-lifting because their zero-lift lines coincide with the chord lines is a misnomer. The location of the zero-lift line for a cambered section is found by equations or by halving the thickness at the 40 per cent chord distance and connecting the point with the trailing edge. The angle to the zero-lift line is the absolute angle of attack, which is always used in stability calculations.

Increasing the angle of attack increases the lift until it reaches a maximum and its coefficient is the CLmax. The CLmax varies with the velocity of the air and with the velocity the angle is increased or decreased, as shown in a diagram. The speed obtained by the CLmax is the stalling or landing speed, which is only a few degrees higher than the angle for the minimum sinking speed and for the minimum power required for horizontal speed. This is the reason to use the CLmax in comparing sections for performances, and especially for durations. Maximum lift coefficients for rectangular and oval wings with various aspect ratios are plotted that shows that an oval wing with an aspect ratio of 1.28 (flying saucer) has the highest CLmax. Up to-date no equation is available for estimating CLmax coefficients.

Stalling does not start at the same angle along the span of a wing. The point, a stall starts depends on the plan view of the wing, if all ribs are of the same design and have the same angle. A few layouts give the places where a stall begins as the angle of attack is increased. An elliptical wing is the only design that stalls evenly. It is however desired that the stall should start at the center of the span in order to have lateral stability and to stop spiral diving. To overcome a probable stall at the tip the mean camber of the outer section of the wing is increased or stall strips are added to the center section. Some use washout of the tip section, which however reduces lift, but increases lateral and horizontal stability.

The radius of the leading edge (nose) has a certain influence on the CLmax, shown in a sketch. The larger the radius, the greater the movement of the stagnation point. This indicates that the effective mean line will increase with the increase of the angle of attack. Stunt models and stunt propellers should have a large nose radius. Speed wings and speed propellers have always pointted leading edges. Some manufacturer even give the size of the radius for their wings to insure performances predicted.

Many devices have been tried to increase the maximum lift of a section. Cambering, using flaps, sucking, blowing, extending the area for gliding or placing rotors in the leading edge. Attaching such devices to models may increase the weight but may not lower the sinking speed.

The minimum drag coefficient can be corrected for other speeds than the one given by the report by using the nomogram with the triple scale. Deduct the friction coefficient on the center scale for the R.N. of the report and add the friction coefficient for the product of the chord in inches and the velocity in ft.p.sec. The R.N. Reynolds number is explained in another section.

The combination of all drag forces excepting the one of the wing is the parasite drag and its coefficient the CDpar. The parasite drag is reduced to a flat plate area of equivalent resistance and the ratic of the equivalent area and the wing area times 1.28 is used in equations for performance calculations as a constant. This however will not give the correct answer if the investigation involves a great variation in wing areas.

The air flow reaching an airfoil will part at the leading edge, the stagnation point, its upper flow passing close to the surface in a laminar flow. Later it becomes turbulent at the point of transition until it breaks away at the point of separation, leaving eddies in its wake. Airfoils with laminar flow have a low drag at small angles and a low maximum lift; airfoils with turbulent flow have a high drag at low angles and a high maximum lift. Most models perform at large angles, especially when gliding and a turbulent flow would be the solution. Sandpaper located on various places on a thick section gave negative results. Turbulators are suggested to be carried close to the leading edge. Serrating the leading edge or the trailing edge reduced the drag of wings and propellers. Coincidently the spacing of the serrations are the same as the pins holding the turbulator. No fool proof theory has been advanced but it seems that a slight helical motion of the air flow delays separation.

Prof.Schmitz suggested for low R.N. up to 20,000 to use sections from 3.4 to 6 per cent thick with a camber of the mean line from 4.5 to 6 per cent and a leading edge radius up to 1 per cent of the chords. His design should automatically give a turbulent flow.

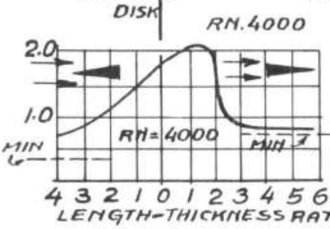
Many theoretical designs of airfoils are available. The best one is the Joukowsky transformation of a circle into an airfoil. The mechanism of the transformation is shown by a graphical construction and by an instrument using the same principle. This transformation does not consider the drag of the airfoil. Others are the Mises and the Trefftz constructions, and many designs developed by the N.A.C.A. Another sections were evolved by Davis from empirical equations of power and wing loadings of airplanes. The simplest and generally used section is the Clark Y with its flat lower surface, which was used to develop one of the N.A.C.A basic section. The thinning is usually done by reducing only the height of the section, instead in reference to the mean line, which would keep the quality of the Clark Y section.

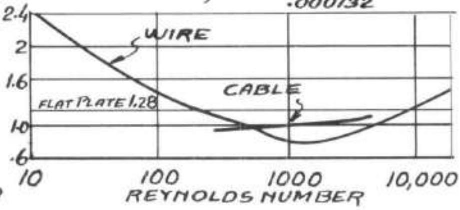
The interference of the fuselage on the coefficients of the wing at various locations is given by a diagram, which shows that the low wing is the worst location if no changes are made in the design. Sagging of the wing covering will not effect the efficiency of the airfoil if the waves are small, smooth and without ridges.

DRAG COEFFICIENTS ~ CD 6

STANDARD AIR DENSITY = .00238

CD = DRAG / .00119 × $V^2 A$
A SQ.FT, DRAG LBS, V FEET P. SEC.
IF A SQ.IN, DRAG OZ, THE CONSTANT IS .000132

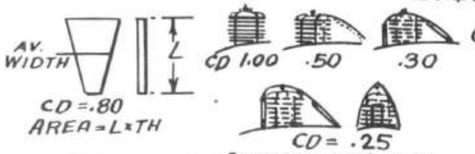




·80 1·55 2·05
CYLINDER

PLAIN .25
BALLOON .55
WITH SPOKES .80
AREA = DIA × WIDTH
WHEELS

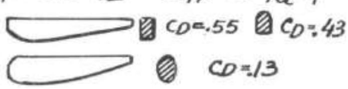
RN 1 5 100
CD 30 7 1.00
– SPHERE –



CD = .80
AREA = L × TH

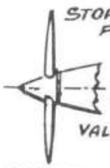
CD = .25
ENGINE AND COWLS

$C_{DP} = .45\, d/L$ $C_{Dfr} = .3\, L/d\, C_f$

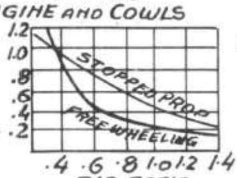


ENGINE EXPOSED AND LANDING GEAR CD = 1.30

STOPPED GAS PROPS



$C_D = 1.16 - .44\frac{P}{D}$
AREA = DIA × BLADE WIDTH
VALID UP TO $P/D = 1.00$



REG. PARACHUTE
$C_D = 1.40$
WITH 7.5% VENT ON TOP
$C_D = 1.70$

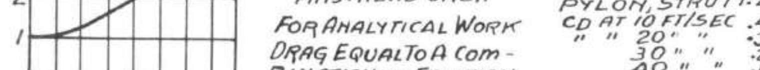
FLAT SHEET AT 26 F.P.S. CD = .10

MIN DRAG OF AIRFOILS

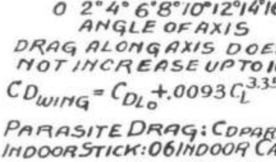
$C_{D\,MIN} = \frac{t}{20c} + 165/\sqrt{V \times CHORD_{IN}}$ $C_{D\,MIN} = C_{D_1\,MIN}(RN_2/RN_1)^{.11}$

FLAT DECK CD 50% HIGHER

FLOATS



CD .25 CD .50 CD = .40
FOR AREA USE CROSS SECTION

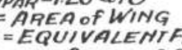


DRAG ALONG AXIS DOES NOT INCREASE UP TO 10°

ENGINE FRONTAL AREA

ROUND FUSELAGE + COWLED
A SQ.IN = 10.2 CU.IN $^{2/3}$
EXPOSED ENGINE .9 CU.IN. $^{3/4}$
FINS + HEAD ONLY.

COWL DESIGN



LOWEST DRAG AT LOW R.N.

PYLON, STRUT 1:2.5–5

CD AT	10 FT/SEC	.45
" "	20 " "	.30
	30 " "	.20
	40 " "	.18

FOR ANALYTICAL WORK DRAG EQUAL TO A COMBINATION OF FRICTION, PRESSURE, SEPARATION AND WAVE DRAG.

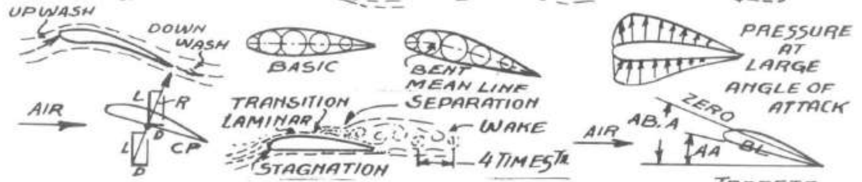
$C_{D_{WING}} = C_{D_{L_0}} + .0093\,C_L^{3.35} + \frac{C_L^2}{\pi AR}$

C_{D_0} = PROFILE DRAG COE.
C_{D_P} = PARASITE " "
C_{D_I} = INDUCED " "

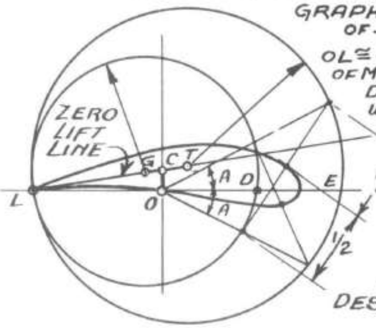
PARASITE DRAG COEFFI'T
$C_{DPAR} = 1.28\, a/S$
S = AREA of WING
a = EQUIVALENT FLAT PLATE AREA of ALL AREAS EXCEPT THE WING AREA. CDPAR MUST BE ADDED TO CD of THE AIRFOILS WHEN COMPARING THEM FOR PERFORMANCE.

PARASITE DRAG: CDPAR INDOOR GLIDER: .08
INDOOR STICK: .06 INDOOR CABIN .08 TOWLINE .05
OUTDOOR H.L. GLIDER: .08 RUBBER: STICK .06–.08
RUBBER. SCALE .12 GAS: REG.: .10 GAS. CABIN .14
GAS. FLY. BOAT.: .16 L/C. STUNT: .06 L/C. SPEED .04

C_{D_0} IS ADDED TO C_{D_P} IN ESTIMATING THE CLIMBING SPEED AT STEEP CLIMB.



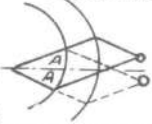
GRAPHIC CONSTRUCTION OF JOUKOWSKY SECTION



OL ≅ 1/2 CHORD, OC ≅ MAX. CAMBER OF MEAN LINE, CT ≅ 1/2 THICKNESS. DRAW CIRCLE THROUGH L WITH CENTER ON T, $OD = OL^2/OE$ FIND POINT G FOR CENTER OF CIRCLE PASSING THROUGH POINTS L AND D. TO FIND POINTS OF SECTION DRAW TWO LINES FROM O WITH EQUAL ANGLES A, A INTERSECTING BOTH CIRCLES CONNECT POINTS AS SHOWN AND DIVIDE. FINISH OUTLINE BY CHANGING ANGLES.

DESIGN WITHOUT PROFILE DRAG

TREFFTZ SYSTEM



ALSO CONSTRUCTED BY PARALLELS

$LIFT = AREA\, C_L q$

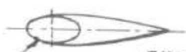
$q = V^2 \rho/2$

$\rho = .00238$

AT SEA LEVEL

FOR MODELS

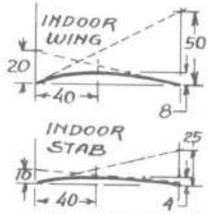
$V_{F.P.S} = 87\sqrt{\frac{W_{OZ.}}{A_{SQ.IN}\, C_L}}$



R.HOLLAND

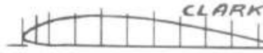
ELLIPSE

DESIGINING AIRFOILS



PER CENT of CHORD

CLARK Y



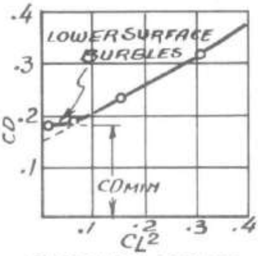
DIST.	0	5	10	20	30	40	50	60	70	80	90	100
UP'R	3.5	7.9	9.5	11.5	11.7	11.4	10.5	9.15	7.55	5.22	2.8	.12
LOW'	3.5	.95	.42	05	0	0	0	0	0	0	0	0

BASIC NACA

DIMESIONS FOR OUTLINE % OF t/c

DIST	0	5	10	20	30	40	50	60	70	80	90	100
UP'R	0	30	39	48	50	48	44	39	31	22	12	0

FOR LOWER SURFACE SAME DIM.



PLOTTING FOR CDMIN.

NACA 6412

$C_{L MAX} = 1.56$

$L/D = 2.75$

DIST	0	5	10	20	30	40	50	60	70	80	90	100
UPR	0	5.36	7.58	10.34	11.65	11.8	11.18	9.95	8.73	6.03	3.33	.12
LOW	0	-1.99	-1.99	-1.25	-.38	+.20	+.55	.78	.85	.73	.39	.12

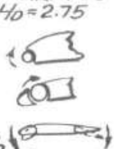
CLARK Y

ANGLE	-6	-4	-2	0	2	4	6	8	16
CD	.022	.018	.02	.024	.0314	.042	.055	.073	.16
CL	-.029	.111	.253	.395	.540	.683	.828	.97	1.25

ROUGHING LOWER SURFACE INCREASES CDMIN. BY 20% AND REDUCES CLMAX BY 5%.

ROUGHING WILL REDUCE CLMAX.

UPPER SURFACE BY 30%
BOTH SURFACES BY 43.5%
CENTER UPPER SURF. 2%
LEADING EDGE " " 30%
TRAILING " 0%



CHANGE OF CAMBER

AR 6 NACA 6412 RN-82,500

ANGLE	-6	-2	1.8	4.8	7	9.5	11.9	14	16.1
CD	.05	.03	.044	.058	.077	.10	.128	.145	.155
CL	0	.3	.6	.8	1.00	1.18	1.3	1.35	1.38

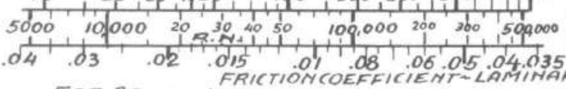
NACA AIRFOIL NUMBERS

FOUR-NUMBER SERIES

EXAMPLE: 2412

2 MAX. CAMBER % CHORD
4 = 40% CHORD LOCATION
12 = 12% MAX. THICKNESS

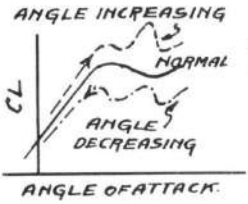
RN FACTOR: CHORD IN. TIMES VEL. FT. P. SEC.

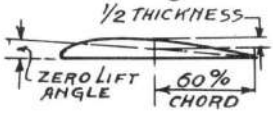


FRICTION COEFFICIENT - LAMINAR

FOR CD MIN WITH CHANGE OF R.N.

$CLARK Y\text{-}CD_{MIN} = .008 + \frac{.10}{\sqrt{C_{H\,IN} V_{F.P.S}}}$

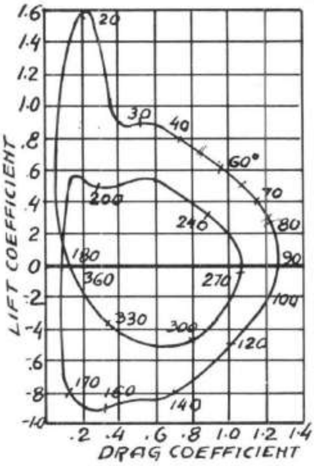


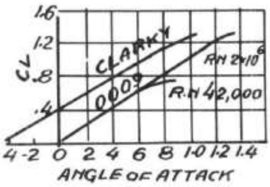


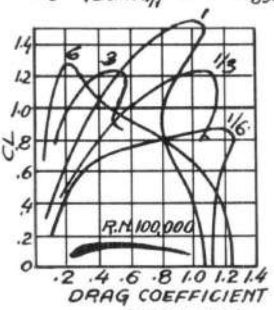
ZERO LIFT LINE

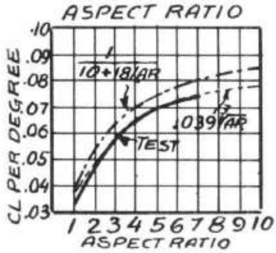
$\alpha_0 = -114.6\ TH._5$

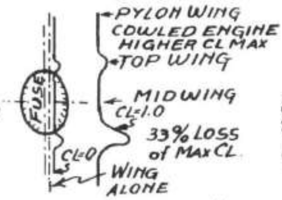
$\alpha_0 = -(32\ TH._{11} + 420\ TH._{83})$



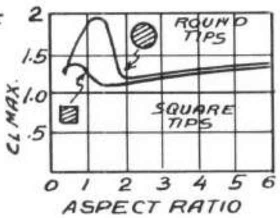


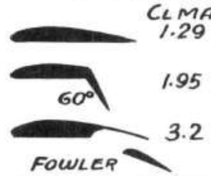






PROFILE DRAG of WING

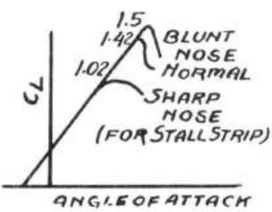




HIGH-LIFT DEVICES

SAGGING COVER.
IF AIRFOIL IS SMOOTH NO EFFECT.

LOCATION OF FIXED SLOT CANNOT BE DETERMINED FOR MODELS. THE HIGH DRAG INCREASES THE STABILITY AND THE SINKING SPEED.

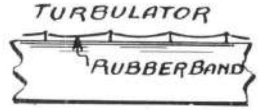


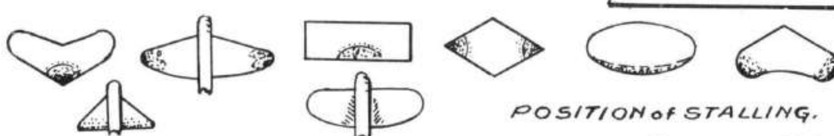
ANGLE OF ZERO LIFT LINE WITH BASE LINE

SYMMETRICAL 0°
6412 -5¼°, 6712 -4°
CLARK Y -5.5°, 2412 -1.5°
4609 -.2.5° 4412 -3°
RAF -7° 6409 -6°
EIFFEL -7°

SERRATED L. EDGE

TURBULATOR

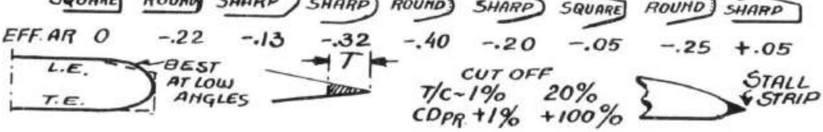




POSITION of STALLING.

CAMBERED AT THE TIP TO DELAY STALLING.

TIP DESIGN EFFECT ON ASPECT RATIO.



AIRFOIL SELECTION. The selection of an airfoil section for models is very involved by the various characteristics demanded by each category of models. The sections should be as thin as possible for the required performances with the necessary bending and torsional strength. The thickness will vary from 4 to 18 per cent of the chord and the camber of the mean line from zero to 6 per cent.

The control-line speed model requires a section that has a minimum drag at the lift coefficient for top speed. The section can be selected from N.A.C.A. reports because the Reynolds number may be the same as the one for the model. The camber should be 10 times the lift coefficient at top speed, if expressed in per cent of the chord. The wing is tapered, has a 10 per cent thickness at the fuselage and not less than 1/8 in. at the tip.

The control line stunt requires a symmetrical section with a large nose radius and extended flaps to increase the CLmax. The selection is based on the 1/2 power of the CLmax divided by the 1/3 power of the CDmin or the cube of the CLmax divided by the square of the CDmin. The CDmin must include the parasite drag coefficient.

The tow-line gliders require a section that has the lowest sinking speed. Plotting sections CL on a 3/2 power scale and the CD on a straight scale the tangent to the curve gives the angle for the lowest power or for the lowest sinking speed. The tangent drawn from the parasite coefficient indicates that the higher the parasite drag is the greater will be the camber.

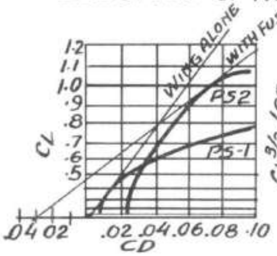
Indoor gliders require a section that has a low minimum drag when launched and a high camber when gliding downward. The gliders have usually wings with flexible rear portion to reduce the drag during the high speed launching. Disregarding the analysis on page 65, the selection may be simplified by dividing the slope of the tangent to the vectors by the CDmin. The low parasite drag of balsa gliders will result in a mean line camber of 2 to 2.5 per cent.

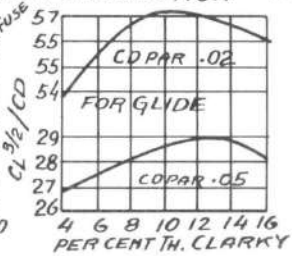
The section for free-flight models depends on the reserve power over the required minimum power. Models flying on minimum power have the same selection as the tow-line gliders. Lately many rubber driven models are designed to fly at minimum power. The section thus selected is tops for low sinking speed.

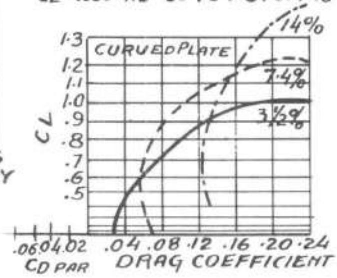
Gas jobs with their super power and short power run will follow the same selection as the hand launched gliders. The sections will be thicker and the camber larger because the parasite drag is also greater. The problem to be solved becomes more complicated if no limitation is set for the wing area. The equation should also include the weight variation with the change of area and aspect ratio.

SELECTION OF AIRFOIL SECTION 10

CL	.1	.2	.3	.4	.5	.6	
$CL^{3/2}$	.03	.09	.165	.252	.354	.465	
CL	.7	.8	.9	1.0	1.1	1.2	1.3
$CL^{3/2}$	.585	.715	.85	1.0	1.15	1.31	1.48





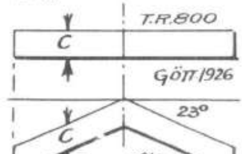


CLARKY	4%	6%	8%	10%	12%	15%
CLMAX	1.36	1.39	1.41	1.43	1.44	1.45
CDMIN	.00955	.0096	.0097	.0099	.0103	.011

STUNT: STREAMLINE AIRFOILS
FOR INVERTED FLYING
R.N 400,000 ~ 500.000

THICK. RATIO	C_{Lmax}	C_{Dmin}	$\frac{CL^{3/2}}{CD+.02}$	$\frac{CL^{3/2}}{CD+.20}$
9%	.85	.0065	29.5	3.70
12%	.90	.0110	27.4	4.05
15%	1.00	.008	35.8	4.75
18%	1.08	.013	34.0	5.25

SPEED
LINE CONTROL
MAKE CHORD TO
GET RN. 500,000
OR $CHORD_{IN} \times V_{MPH}$ =
ABOUT 650
160 MPH ~ CHORD = 4 IN.
FOR 0009 SECTION.
LOWEST CDMIN.
FOR SAME LAND'G SPEED
MAX. FOR C_{Lmax}^2/C_{Dmin}.
BEST FOR 0015 SECTION
A.R. NOT CONSIDERED.

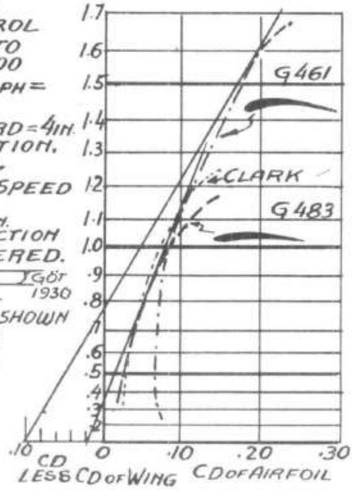


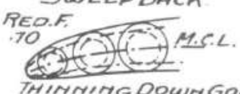
10° WASHOUT ON TIP
CL MAX 20% LOWER.
SWEEP BACK

BODY ADDED AS SHOWN
NO MEASURABLE
CHANGE OF C_D, C_L.

CD_0 LOW R.N
SYMMETRICAL SECTIONS

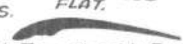
9%	.0185
12%	.0203
15%	.0227
18%	.0252
VERY THIN FLAT.	.03





THINNING DOWN GOOD
THICK AIRFOIL SECTIONS.

TANGENT LINE INDICATES
LOWEST SINKING SPEED
WITH ADDED DRAG COEFF.
OF EXPOSED STRUCTURE.



KEEP MEAN CAMBER LINE.
MULTIPLY DIA. OF CIRCLES BY THE RED. FACTOR.

NO TEST. BIRDS IN FLIGHT
HAVE NO SUCH WINGS.

SELECTION FOR GLIDE,
MIN. POWER.

~FOR CARRIER DECK EVENT ~ COMPARING SECTIONS
FOR SPEED DIFFERENCE.

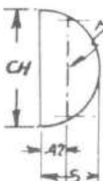
$$V - V_L = \frac{550\,BHP\,EFF}{W}\,\frac{CL}{CD} - \frac{W^{1/2}\,29}{A^{1/2}\,CLmax^{1/2}}$$ LB. FT. SEC SYSTEM.

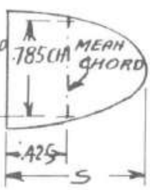
BEST: MAX. BHP, AREA, CLMAX., MIN. W, CD/CL

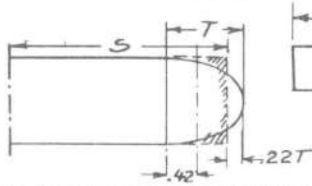
STOPPED PROP

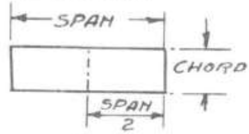
VERTICAL POS. CLMAX 1.32
HORIZONTAL POS'N " 1.19

AREA CALCULATION ~ AVERAGE CHORD = AREA/SPAN









HALF CIRCLE HALF ELLIPSE ERROR LESS THAN ½%

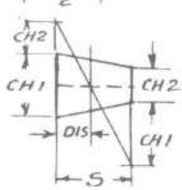
AREA: CH×S×.785 A=CH×S×.785

RECTANGULAR = CHORD × SPAN

TAPERED = (ROOT CHORD + TIP CHORD) $\frac{SPAN}{2}$

HALF CIRCLE = $CHORD^2$ .392

AVE. CHORD LOCATED
ON C.G OF AREA
USE ONE SIDE
OF WING FOR
ESTIMATE.



D = S(CH1 + 2CH2)/3(CH1 + CH2)

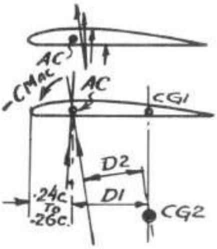
AERODYNAMIC CENTER. One point on the airfoil that simplifies stability calculations is the aerodynamic center. This is the point about which the moments of the air force resultants up to the stalling angles are the same, and the coefficient Cmac. The Cmac is zero for symmetrical sections and usually negative for cambered sections. The distance from the leading edge is 24 to 26 per cent of the chord. To find the moment of any resultant about a point usually about the c.g., add to the moment about the a.c. the moment of the parallel force passing through the a.c. The resultants of the airforces or their coefficients are plotted through the a.c. and the moments are obtained by the shortest distances from the reference point. Moment estimates are simplified for designs with the c.g. locations on the chord line by omission of the CD coefficients. This simplified method is used in some textbooks with the c.g. located well below the chord line and the moments obtaind are often twice of the actual moments. A graphic construction shows how the position of a.c. can be found.

SCALE EFFECT. The drag coefficient is influenced by the speed, the length and the air conditions. The factor denoting this relation is the Reynolds number (RN), which for standard air is equal to 6300 times length of the body, times the speed in ft.p.sec. The Reynolds number is applicable only to geometrical similar bodies or airfoils. The R.N. for model units is equal 780 times the chord in inches, times the velocity in m.p.h. In case the velocity is given in ft.p.sec. the constant is 532. The scale effect depends on the air moving over the bodies, which may be laminar or turbulent. The air flow will stay laminar over models due to the low R,N,. A few diagrams give the change of coefficients with variation of R,N,

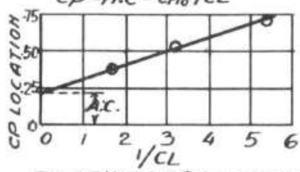
BOUNDARY LAYER. Air moving over a body will stick to the surface to a certain thickness and the rest of the air will move unrestrained. This stationary air is the boundary layer. The thickness of the layer depends on the Reynolds number and on the condition of the boundary layer, if laminar or turbulent. The flow is usually laminar at the start and later becomes turbulent. To keep the flow laminar over an airfoil at low angles the surfaces must be smooth and free from waves. Indoor gliders should be checked with french curves for irregularities on the wing surfaces.

To prevent transition or separation of the boundary layer many devices were tried. Slots give only limited effect and its exact location on model wings cannot be determined. They usually create high drag even at low angles. Sucking and blowing seems to give the most promising device. Suction may be applied to model wings if the engine intake can be connected to the slots located at the 60 per cent distance. The loss of power may be greater than the gain in drag.

AERODYNAMIC CENTER

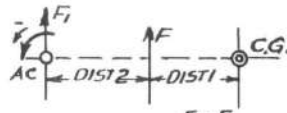


$C_P = A.C - C_{M_0}/C_L$



GRAPHIC A.C. CONSTRUCTION

A.C. LOCATION MOVES TOWARDS THE T.E WITH INCREASE OF R.N.
A.C. ON SYMMETRICAL SECTIONS ARE LOCATED ON THE UPPER SURFACE



$F = F_1$

$M = F \times DIST_1, M_{AC} = -F\,DIS_2$

$CG\text{-}M = M_{AC} + F(DIS_1 + DIST\,2) = F\,DIST_1$

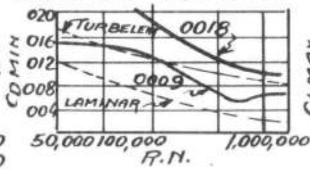
$C_{MA.C.} = M6 + .002$
CLARK Y −.07, 2409 = −.045
2412 −.043, 4409, 4412 −.088
4409 − 4412 = −.133, 6712 = −.021
ALL SYMMETRICAL = 0

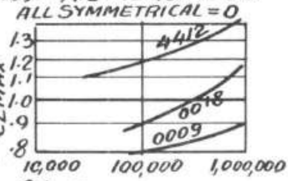
SCALE EFFECT (REYNOLDS NUMBER)

R.N = 6380 V×L − V·FPS, L = FT
RN = 525 V×CH − V = F.P.S. CH − IN.
FOR AIRFOILS:

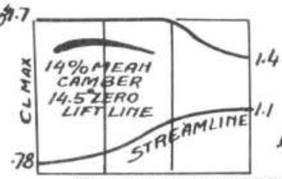
$C_{D\,FRIC.} = 2.66/\sqrt{R.N.}$

R.N INDOOR RUBBER 8000
OUTDOOR " 55,000
" GAS 150,000
SPEED LINE CONTROL 35×10^4
PROPELLER 300,000
$RN = 35\,R_{IN} \times CHORD_{IN}\,R.P.M.$

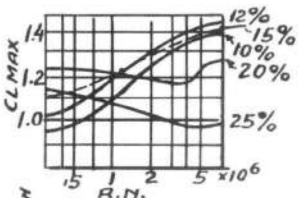


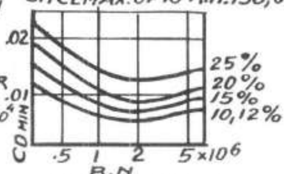


CL/A REDUCES WITH TURBULENCY.
CLMAX HOWEVER INCREASES. POLISHED OR ROUGH SURFACE NO INFLUENCE ON CLMAX. UP TO R.N. 150,000



REYNOLDS NUMBER
$C_F = .00242$ THIN SECT.
$C_F = .00328$ THICK " $RN\,28\times10^4$





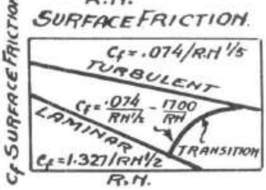
R.A.F. 32 CLMAX

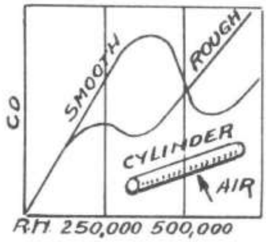
RN 150,000	1.3
75,000	1.25
25,000	1.08

$R.N = \frac{\rho}{\mu} V.L \quad \rho = .00238$
$\mu = 373\times10^{-9}$

$C_F = \frac{.472}{(\log R_X)^{2.58}}$ FOR TURBULENT FLOW

SURFACE FRICTION.





BOUNDARY LAYER

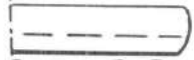
THICKNESS OF LAYER: LAMINAR
$\delta = 5.5C_X / R.N^{1/2}$
X − DISTANCE IN FT.

THICKNESS OF LAYER: TURBULENT
$\delta = .37X / R.N^{.2}$ T.M. 1167

BOUNDARY-LAYER CONTROL TO PREVENT SEPARATION.
BY SUCTION OR BLOWING
USUALLY AT 60% OF CHORD

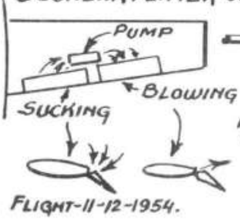
BLOWING

SUCTION



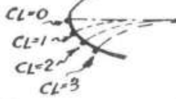
SUCTION BY GAS ENGINE INTAKE(?).
SLOTS .010 WIDE
SPACED ABOUT .60 CH.
SMALL ANGLES: LAMINARY
HIGH ANGLES: HIGH LIFT, TURBULENT FLOW.

BOUNDARY LAYER CONTROL



FLIGHT-11-12-1954.

"CUSTER CHANNEL WING"
$C_L = 2.87$



CHANGE OF STAGNATION POINT WITH CL.

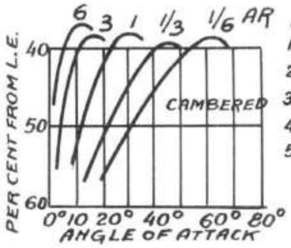
SUPER-CIRCULATION.

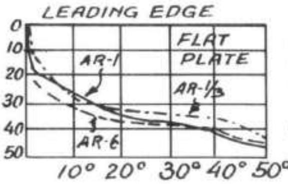
CENTER OF PRESSURE. The point on the base line through which the air force passes is the center of pressure (c.p.). The base line is an arbitrary line selected by the testing engineer, usually the tangent line to the lower surface of the airfoil. Now all base lines are from the leading edge to the trailing edge. In some reports one may find the base line far below the chord line. The best symmetrical sections have no c.p. movement. The c.p. for cambered sections moves toward the leading edge with increase of the angle of attack. Selecting the reference line below the chord line may reverse the movement of the c.p.. Bending the trailing edge downward moves the c.p. toward the trailing edge and also increases the maximum lift coefficient. This additional flexing of the airfoil is used in adjusting models for better atbility. Reversing the c.p. movement can be accomplished by bending the trailing edge upward. This is used in making a cambered airfoil stable without the use of a stabilizer. the angle of deflection is for regular sections about 6 to 7 degrees. The c.p. movement on fuselages are similar to the one on symmetrical airfoils.

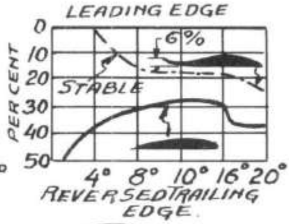
ASPECT RATIO. The ratio of the span and the average chord is the aspect ratio (AR). The higher the aspect ratio the smaller the angle of the wing for the same lift. Theoretically the air affected is within the circle drawn through the wing tips. The drag of an airfoil can be divided into the profile drag and the drag required to deflect the air downward. The profile drag is assumed to be constant. The force to deflect the air is the induced drag, its coefficient CDi, which is equal to the square of the lift coefficient divided by 3.14 times the AR of the wing. This indicates that by deducting the induced drag coefficient the profile drag coefficient is obtained, which is shown by its application to the CLARK Y section. Most tests on airfoils are made with an aspect ratio of 5 to 6 and to use the results for higher aspect ratio the angle of attack for the corresponding CL must be reduced by the difference of the induced angles estimated by the equation 18.24xCL/A.R. N.A.C.A. simplified this by giving the drag and the angle of attack for the infinite aspect ratio to which the induced drag and the induced angle is added.

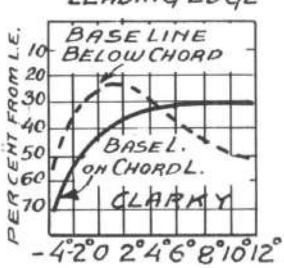
Elliptical wings give ideal load distribution for any aspect ratios. Similar load distribution may be obtained by a 50 per cent taper. Endplates increase the effective aspect ratio as given by an equation or determined by the graphical method shown. The proximity of the ground increases the effective aspect ratio and its value can be obtained by the empirical equation presented. Gliding tests with models should be madeat a certain distance from the ground, and the distance of a glide should never be measured to the point where it touches the ground.

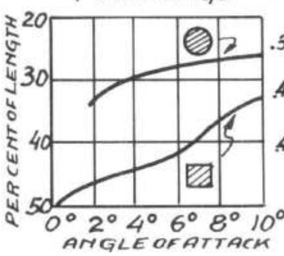
An interesting relation of profile drag and induced drag is developed, which shows that the higher the AR the lower the total drag. The increase of structural weight with increase of A.R. is not included in the investigation.

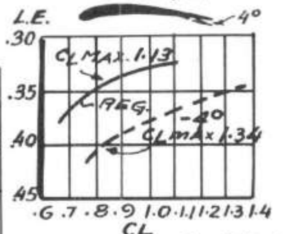






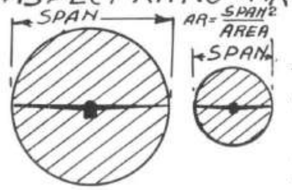


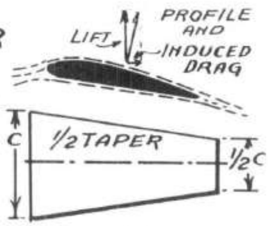




WING WITH REG T.E. + SAME WING WITH T.E DEFLECTED - 4° DOWNWARDS.

ASPECT RATIO ~ AR





INDUCED DRAG COEFF'T

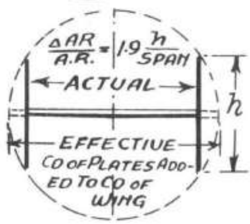
$C_{D\,IN.} = C_L^2/\pi AR$

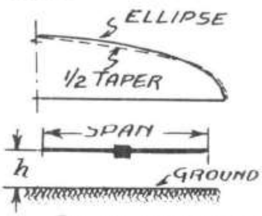
$C_D = C_{D\,PROF} + C_{D\,IND.}$

$IND.\ ANGLE = \frac{18.24\, C_L}{A.R.}$

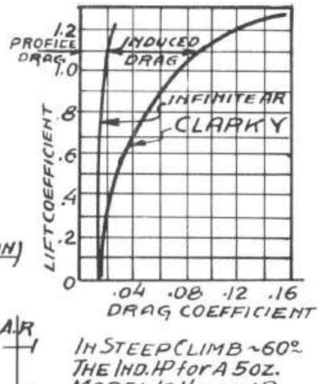
$IND.\ H.P. = \frac{274\ W/_{SPAN^2}^{2}}{V_{FT.P.SEC}}$

W = OZ., A - SQ.IN., SPAN - IN.





GROUND EFFECT



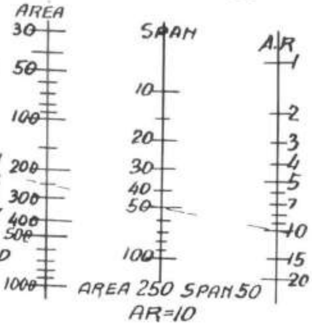
$C_D = C_{D\,PROF} + C_{D\,IND.}$ $\qquad AR_{EFF} = AR(1 + .07\,\frac{SPAN}{h})$

$C_D = \frac{2.36}{\sqrt{R.N}} + \frac{C_L^2}{\pi AR}$

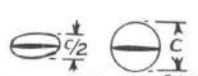
SUBSTITUT'G + DIFF'G AR FOR MIN. CD, $C_L = 1.0$
$AR = 23\, W^{1/6} \cdot A^{1/6}$
FOR OZ, SQ.IN.

$AR = 6.3\, W^{1/6} \cdot AREA^{1/6}$

FOR WAKEFIELD AR ~ 21
FOR INDOOR MICRO " 13
SMALL GAS " 19
FOR HIGH SPEED JOB X $C_L^{1.17}$
INCREASE of WEIGHT WITH AR NOT CONSIDERED



IN STEEP CLIMB ~ 60° THE IND. HP FOR A 5 OZ. MODEL IS 1/10,000 HP. NEGLECTED IN ESTIMATES.



ENDPLATE HEIGHT CHORD/2
CL INCREASES 5 1/2 %
ENPLATE HEIGHT - CHORD
CL INCREASES 11.5%
ADD CD OF PLATES

PROPELLERS. The least understood unit of a model is the propeller. With its gyroscopic forces, high torque, side thrust, slipstream and high static thrust makes design and adjustment a complicated task. The thrust necessary to move a model through the air is obtained by rotating one or more inclined blades that accelerate a mass of air through the disc area. Higher efficiency is obtained by imparting a low acceleration to a great mass of air. The power required to rotate the blades is supplied by gas engines or by twisted strands of rubber bands.

This acceleration is used in the momentum theory for estimating the thrust and the required power. This theory, however does not take into consideration the profile drag of the blades. There is no theory which would give a correct answer. The blade element theory, which uses the lift and drag of an airfoil agrees fairly well with tested propellers, if correction factors are used.

The angle of the lower flat surface of the blade is used to estimate the nominal or geometric pitch. This distance is equal to the distance a blade will travel in one revolution sliding on its flat surface. The diameter and the nominal pitch are usually marked on propellers. Mutiplying the distance from the axis to the blade section by 6.28 and marking it on a horizontal line with a vertical on one end and the blade angle on the other end, then the intersection of the line on the vertical is the nominal pitch. A sketch shows how to find the angle at any section along the blade if the pitch is given. The pitch measured to the zero-thrust line is the aerodynamic pitch.

The angle made by the flying speed, divided by the rev.p.sec. of the propeller and the circumference it traveled will be lower than the pitch angle and is denoted as the advance angle. The advance angle and the pitch angle have a certain relation to the efficiency of the propeller, which is the ratio of the power output and the power input.

In reports the advance angle is simplified by using the advance factor v/nD. The reports give the efficiencies and p/D ratios referred to the advance factor. In case the pitch varies along the blade, the nominal pitch is taken at the 3/4 distance from the axis. Another factor that simplifies selecting propellers is the speed-power coefficient, the Cs, which takes the speed, the power and the r.p.m. of the drive shaft for designing the propeller. For rubber driven propellers most of the equations for gas props are not directly applicable, because even with a given rubber the torque changes with the number of strands used. Any scientific approach is problematic.

Diagrams show forces on a blade, its efficiency and the maximum efficiency expected with the advance factor. One diagram gives the thrust and torque along the blade; another the relative axial and rotational flow, and two sketches show the direction of the air on the upper and on the lower surface. The spinner added to the propeller should not be greater than 25 per cent of the diameter.

Propellers with equal pitch distribution tested on a spindle lose 6 per cent of their efficiencies when mounted on a fuselage. Propellers with increasing pitch toward the tips lose only 2.5 per cent when tested on a fuselage.

Two nomograms and five diagrams give the diameter for rubber props and the diameter and the p/D ratio for gas propellers. Improving the efficiency of propellers were tried. End-plates to decrease tip losses and angles fastened to the flat surface to prevent radial flow did not improve efficiency. Ducted propellers give greater thrust at low speeds, but their maximum efficiencies fell off because the drag of the cowling was greater than the gain in thrust. Removing the boundary layer by the centrifugal improved the static thrust slightly. Serrating the leading edge and the trailing edge gives hopes for higher efficiency. The question is always- was the improvement due to the design or the thinning of the blades.

An equation and a chart are given for estimating the approximate static thrust of a propeller. For high p/D ratios the thrust will be lower than for lower p/D ratios, because the blade is above the stalling angle. A diagram is given for zero thrust which will help to layout the thrust curve.

The thrust of a propeller changes its direction in a skid (yaw) opposite to that of the skid, because the advancing blade has a larger angle of attack than the retreating blade. Its effect is assumed to be that of a plate with a span equal to the diameter and an aspect ratio of 8. This indicates that a tractor propeller requires an additional stab and fin areas to equalize the effect and that a pusher prop will aid stability or even fly a model without a tail if placed at a sufficient distance.

One condition is regrettable, that propellers are not manufactured with greater change in diameters. The change of a propeller from 8 to 9 inches in diameter increases the power by 60 per cent, and increasing the pitch from 6 to 8 inches increases the power by only 30 per cent. Blade width has an influence on the thrust and r.p.m. at the start, but if correctly designed the efficiency at maximum speed is nearly the same. A diagram shows the difference in performances of a narrow and a wide blade propeller. The advantage of a wide blade prop is that its selection is not very critical. The diameter and the pitch of a very narrow blade must be correct to give top performance.

A one-blader must have a counter-weight for balancing the centrifugal force and the one-sided thrust to prevent vibrations and friction losses. In case contra or tandem propellers are used, the diameter and the pitch is estimated for half the horse power or the diameter is estimated for the total power and then mutiplied by .84.

The outline of gas propeller blades is similar to the one shown in the sketch. Rubber driven propellers use wider blades in order to reduce the r.p.m. at the start. This may also be effected by designing the blade so it will flex to a higher pitch with the high initial torque. The airfoil section for gas props is a Clark Y with a larger nose radius and the section for a rubber prop has a camber on the under surface.

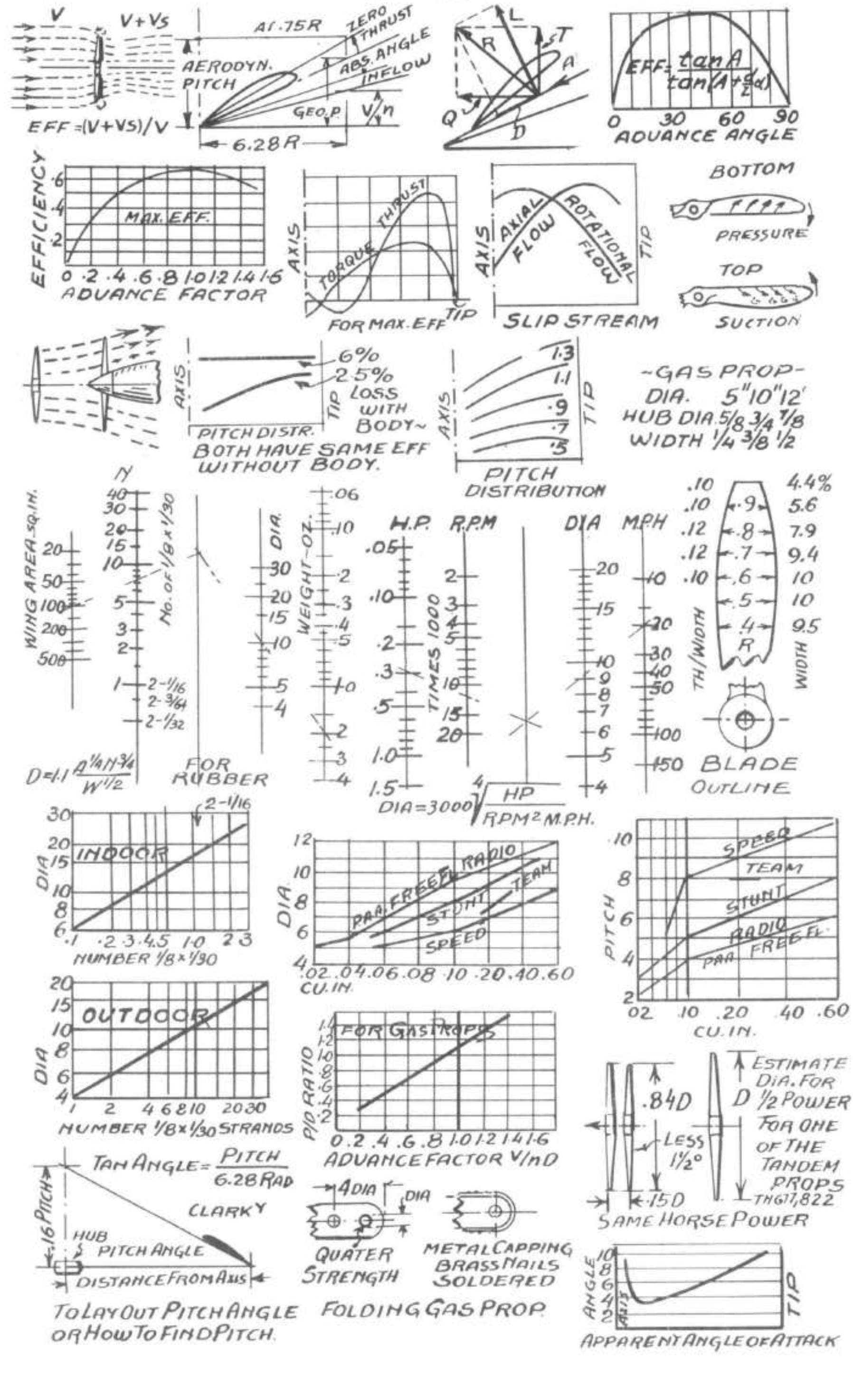
The advantages of folding props over stopped or wind-milling props are the lower drag and consequently a lower sinking speed. Folding props for gas engines must take into consideration the high centrifugal force and the bending moment of the thrust. The distance of the pin must be at least four times its diameter, because the shear strength is only 1/10 of the compression strength of the wood used. If the distance is lower metal capping must be provided. Many designs for rubber driven folding props are satisfactory. The problem is always the slope of the hinge axis. To fold a blade to the same level, the axis must be in a 45 degree plane half-way to the fuselage. The slope depends on the angle the blade should rotate. The slope of the hinge axis is given by a diagram and an equation. It may be approximated by taking 70 per cent of the blade angle. The direction of the hinge axis is with the pitch of the propeller. The rubber is kept under tension to preventbunching which may upset stability of the model and after the remaining turns cannot keep the model flying on an even level the shaft lug hits the stop and the prop will fold due to the air pressure.

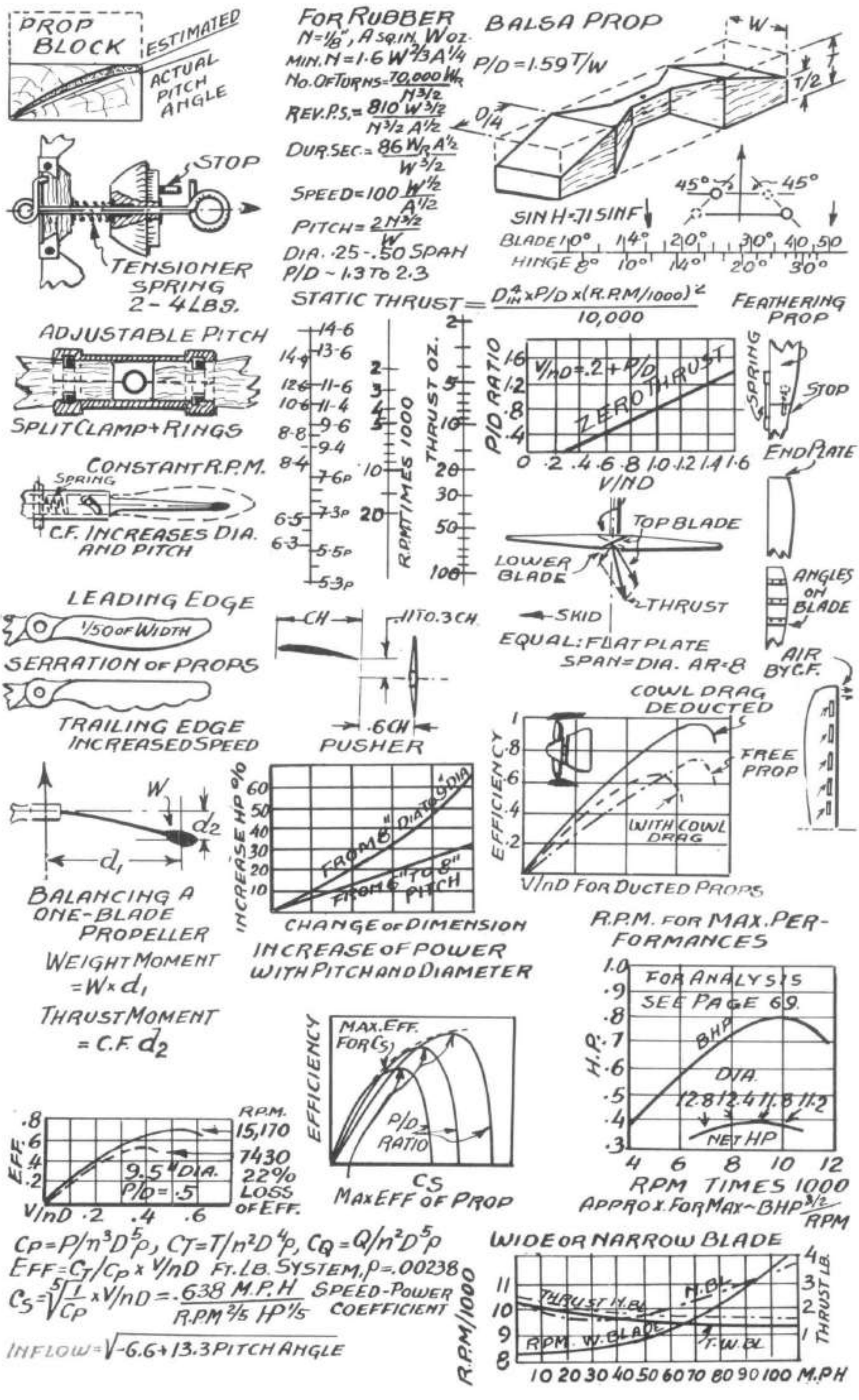
Many propeller designs were tried to reduce the starting burst of rubber motors due to their high initial torque. One was a prop with two pitch settings actuated by the air pressure. One used the centrifugal force to increase the diameter and the pitch by the action of a cam and the tension of a spring. The simplest and may be the best one is to design the blade with its flexural center behind the 30 per cent diagonal axis of the propeller. This will flex the propeller to a higher pitch at the start. The flexural center for microfilm propeller blades is controlled by the size of the leading and trailing edge.

Gas propellers with adjustable pitch did not become popular because of the initial cost, the lack of extra blades and the time required to set both blades to the desired pitch.

Two diagrams give the efficiencies of propellers with various p/D ratios. They indicate that the best propeller for a design will not operate at the highest efficiency of the propeller, except at very high p/D ratios. Another paradox that most "observing" modelers have found to be the fact that the maximum thrust of a propeller in flight is not obtained at the maximum power delivered at a high r.p.m.. The highest efficiency is reached at about 1500 to 2000 r.p.m. below the revolutions for maximum power. For proof see page 60.

The material used in manufacturing propellers limits the efficiency. Aluminum and dural with thin sections give the highest. Next is the compressed wood and then regular propellers. Metal increases the efficiency by 10 to 12 per cent; metal leading edges by 5 to 7 per cent. Thick blade shanks (airplane props) reduce the efficiency by 15 per cent. The gain however is from 3 to 10 per cent if a spinner covers the poor sections. If the propeller is located too close to the wing or to a relative large fuselage the efficiency may fall as much as 20 per cent. The extended crankcase has added to the efficiency of propellers. Tests also showed that the high r.p.m. of today's engines increased the efficiency by reducing the profile drag. The sharp leading edge of the blades,that give maximum efficiency is not in favor by modelers, because they may damage their fingers.

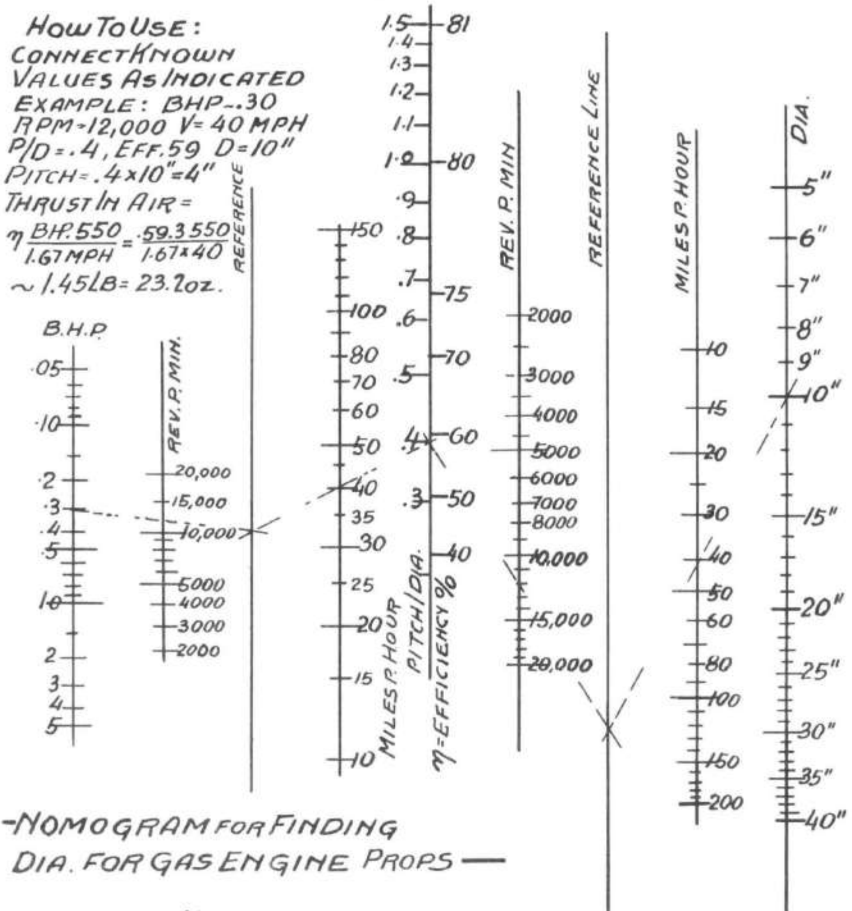

V
V+Vs
EFF =(V+VS)/V
AT .75R
ZERO THRUST
ABS. ANGLE
INFLOW
AERODYN. PITCH
GEO.P
V/n
6.28R
L
R
T
A
Q
D
EFF= tan A / tan(A+d/2 α)
0 30 60 90
ADVANCE ANGLE
EFFICIENCY
MAX. EFF.
0 .2 .4 .6 .8 1.0 1.2 1.4 1.6
ADVANCE FACTOR
AXIS
TORQUE
THRUST
TIP
FOR MAX. EFF
AXIS
AXIAL FLOW
ROTATIONAL FLOW
TIP
SLIP STREAM
BOTTOM
PRESSURE
TOP
SUCTION
AXIS
6%
2.5%
LOSS WITH BODY~
TIP
PITCH DISTR.
BOTH HAVE SAME EFF WITHOUT BODY.
AXIS
1.3
1.1
.9
.7
.5
TIP
PITCH DISTRIBUTION
-GAS PROP-
DIA. 5" 10" 12"
HUB DIA. 5/8 3/4 7/8
WIDTH 1/4 3/8 1/2
WING AREA SQ.IN.
N
No. OF 1/8 x 1/30
2-1/16
2-3/64
2-1/32
DIA.
WEIGHT-OZ.
D=1.1 A^1/4 N^3/4 / W^1/2
FOR RUBBER
H.P.
R.P.M
TIMES 1000
DIA
M.P.H
DIA=3000 ⁴√(HP / RPM² M.P.H.)
.10 .10 .12 .12 .10
TH/WIDTH
4.4% 5.6 7.9 9.4 10 10 9.5
WIDTH
R
BLADE OUTLINE
2-1/16
INDOOR
DIA
NUMBER 1/8 x 1/30
PAA. FREE FL.
RADIO
STUNT
TEAM
SPEED
DIA.
CU. IN.
SPEED
TEAM
STUNT
RADIO
PAA FREE FL.
PITCH
CU. IN.
OUTDOOR
DIA
NUMBER 1/8 x 1/30 STRANDS
FOR GAS PROPS
P/D RATIO
0.2 .4 .6 .8 1.0 1.2 1.4 1.6
ADVANCE FACTOR V/nD
.84D
LESS 1½°
.15D
SAME HORSE POWER
ESTIMATE DIA. FOR D ½ POWER FOR ONE OF THE TANDEM PROPS
THG 17,822
TAN ANGLE = PITCH / 6.28 RAD
.16 PITCH
CLARK Y
HUB
PITCH ANGLE
DISTANCE FROM AXIS
TO LAY OUT PITCH ANGLE OR HOW TO FIND PITCH.
4 DIA
DIA
QUATER STRENGTH
METAL CAPPING BRASS NAILS SOLDERED
FOLDING GAS PROP
ANGLE
AXIS
TIP
APPARENT ANGLE OF ATTACK


PROP BLOCK
ESTIMATED
ACTUAL PITCH ANGLE
STOP
TENSIONER SPRING 2-4 LBS.
FOR RUBBER
N=1/8", A SQ.IN. W OZ.
MIN. N=1.6 W2/3 A1/4
No. OF TURNS = 70,000 W_R / N3/2
REV.P.S. = 810 W3/2 / N3/2 A1/2
DUR.SEC. = 86 W_R A1/2 / W3/2
SPEED = 100 W1/2 / A1/2
PITCH = 2N3/2 / W
DIA. .25-.50 SPAN
P/D ~ 1.3 TO 2.3
BALSA PROP
P/D = 1.59 T/W
W
T
T/2
D/4
45°
45°
SIN H = .71 SIN F
BLADE 10° 14° 20° 30° 40 50
HINGE 8° 10° 14° 20° 30°
STATIC THRUST = D⁴IN x P/D x (R.P.M/1000)² / 10,000
FEATHERING PROP
SPRING
STOP
END PLATE
ADJUSTABLE PITCH
SPLIT CLAMP + RINGS
CONSTANT R.P.M.
SPRING
C.F. INCREASES DIA. AND PITCH
R.P.M. TIMES 1000
THRUST OZ.
P/D RATIO
V/nD = .2 + P/D
ZERO THRUST
V/ND
TOP BLADE
LOWER BLADE
THRUST
SKID
EQUAL: FLAT PLATE
SPAN = DIA. AR = 8
ANGLES ON BLADE
AIR BY C.F.
LEADING EDGE
1/50 OF WIDTH
SERRATION OF PROPS
TRAILING EDGE INCREASED SPEED
CH
.1 TO .3 CH.
.6 CH
PUSHER
COWL DRAG DEDUCTED
FREE PROP
EFFICIENCY
WITH COWL DRAG
V/nD FOR DUCTED PROPS
BALANCING A ONE-BLADE PROPELLER
W
d1
d2
WEIGHT MOMENT = W x d1
THRUST MOMENT = C.F. d2
INCREASE HP %
FROM 6" DIA TO 8" DIA
FROM 6" TO 8" PITCH
CHANGE OF DIMENSION
INCREASE OF POWER WITH PITCH AND DIAMETER
R.P.M. FOR MAX. PERFORMANCES
FOR ANALYSIS SEE PAGE 69.
BHP
DIA.
12.8 12.4 11.8 11.2
NET HP
H.P.
RPM TIMES 1000
APPROX. FOR MAX ~ BHP3/2 / RPM
EFF.
R.P.M. 15,170
7430
22% LOSS OF EFF.
9.5" DIA.
P/D = .5
V/nD
MAX. EFF. FOR Cs
P/D RATIO
EFFICIENCY
Cs
MAX EFF OF PROP
Cp = P/n³D⁵ρ, CT = T/n²D⁴ρ, CQ = Q/n²D⁵ρ
EFF = CT/Cp x V/nD FT.LB. SYSTEM, ρ = .00238
Cs = ⁵√(1/Cp) x V/nD = .638 M.P.H / R.P.M 2/5 HP 1/5 SPEED-POWER COEFFICIENT
INFLOW = √-6.6 + 13.3 PITCH ANGLE
WIDE OR NARROW BLADE
THRUST N.BL
N.BL
RPM W.BLADE
T.W.BL
R.P.M./1000
THRUST LB.
M.P.H

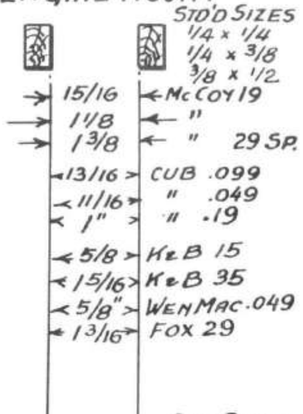
HOW TO USE:
CONNECT KNOWN
VALUES AS INDICATED
EXAMPLE: BHP-.30
RPM-12,000 V=40 MPH
P/D=.4, EFF.59 D=10"
PITCH=.4×10"=4"
THRUST IN AIR =

$\eta \frac{BHP.550}{1.67 MPH} = \frac{.59.3.550}{1.67 \times 40}$

$\sim$ 1.45 LB = 23.2 oz.



-NOMOGRAM FOR FINDING
DIA. FOR GAS ENGINE PROPS —



SEE PAGES
26, 59.

FREE-FLIGHT MODELS. Indoor models are for experts. They are not only microfilm covered, but also must use selected, light weight balsa for the frame work. The weight of each part msut be the same on both sides. A scale illustrated on page 77 can be built that may give weights within 1/10,000 of an ounce. A layout is shown with dimensions of a few structural details. Propellers are microfilm covered, which are 50 per cent lighter than the flimsiest balsa prop. Washers and bearings are amde small for minimum weights. The strength of indoor models is very low, they can hardly carry twice their own weight. The maximum camber for wing section is at 40%, windtunnel tests however show that a 50% distance will improve performances, The wing should be a high wing about 40% chord distance above the thrustline.to reduce the required stab area.

Balsa props are made from extra light balsa, cut and arranged so that the rings are vertical to both of the blade sections. The hook distance is 15 inches , the motor is a loop of 16 to 18 in. in length and about 50 per cent of the total weight.The durations of three wing-fuselage combinations as sketched gave very close .

Microfilm is made by adding a few drops of castoroil to nitrate dope and a few drops poured on water of a temperature of 65 deg. The success of a contest for indoor models depends on an auditorium with a 80 to 100 feet ceiling without drop lights and a minimum of super structure. Low ceilings will reduce the duration by damaging the microfilm prop.

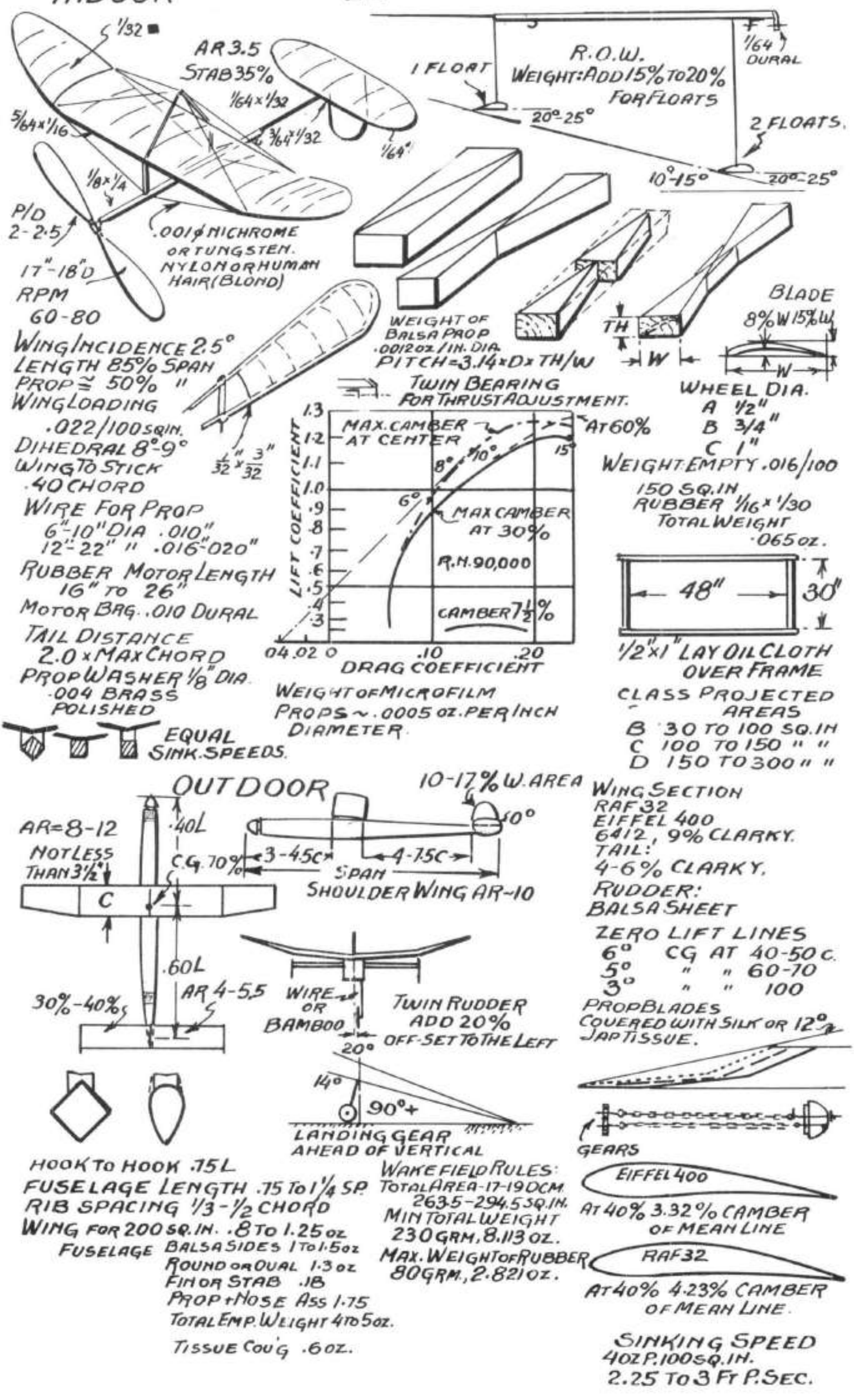
A general layout for outdoor models shows that the wing should be made adjustable for c.g. corrections. A cradle is used if possible to insure a fixed position without misalignment for succeeding flights. The area is usually limited and an aspect ratio of 10 to 12 may be used if the modeler can expertly cover a narrow wing with a cambered under surface. Endplates on the tips add only drag and reduce the duration. They will however increase stability. Airfoils for a slow climb should have high cambers and for a fast climb low cambers.

Fuselages may be square, octagon or round. The duration will not vary much if folding props are used. The adjustment with a round fuselage belongs to the expert. A low pylon will simplfy adjustment.

The tail may have endplates if the aspect ratio is low and the plates are large enough to replace the necessary fin area. The section is usually a thinned down Clark Y. The tail setting or the difference of the zero-lift lines depends on the weather conditions; for calm weather smaller setting and the c.g. towards the trailing edge and for turbulent weather a greater setting with the c.g. far forward. Any setting will be critical for the initial burst, because the rubber torque is often 10 times the one used in level flight.

A gear drive may be advantageous for unlimited rubber by saving weight and friction of a shorter fuselage, which will compensate for the weight of the gears and their frictional power.

R.O.G., rise-off-ground type models while at rest in a normal attitude should have no part other than the take-off gear touch the ground. This requires a three-point landing gear.



INDOOR
1/32 ■
AR 3.5
STAB 35%
1/64 x 1/32
5/64 x 1/16
3/64 x 1/32
1/64"
1/8 x 1/4
P/D
2-2.5
.001ϕ NICHROME
OR TUNGSTEN.
NYLON OR HUMAN
HAIR (BLOND)
17"-18" D
RPM
60-80
WING INCIDENCE 2.5°
LENGTH 85% SPAN
PROP ≅ 50% "
WING LOADING
.022/100 SQ.IN.
DIHEDRAL 8°-9°
WING TO STICK
.40 CHORD
1/32" x 3/32"
WIRE FOR PROP
6"-10" DIA .010"
12"-22" " .016"-020"
RUBBER MOTOR LENGTH
16" TO 26"
MOTOR BRG. .010 DURAL
TAIL DISTANCE
2.0 x MAX CHORD
PROP WASHER 1/8" DIA.
.004 BRASS
POLISHED
EQUAL
SINK. SPEEDS.
R.O.W.
WEIGHT: ADD 15% TO 20%
FOR FLOATS
1 FLOAT
20°-25°
1/64
DURAL
2 FLOATS.
10°-15°
20°-25°
WEIGHT OF
BALSA PROP
.0012 OZ./IN. DIA.
PITCH = 3.14 x D x TH/W
TH
W
BLADE
8% W 15% W
W
TWIN BEARING
FOR THRUST ADJUSTMENT.
WHEEL DIA.
A 1/2"
B 3/4"
C 1"
MAX. CAMBER
AT CENTER
AT 60%
MAX CAMBER
AT 30%
R.N. 90,000
CAMBER 7 1/2%
LIFT COEFFICIENT
DRAG COEFFICIENT
1.3
1.2
1.1
1.0
.9
.8
.7
.6
.5
.4
.3
04 .02 0
.10
.20
6°
8°
10°
15°
WEIGHT EMPTY .016/100
150 SQ.IN
RUBBER 1/16 x 1/30
TOTAL WEIGHT
.065 OZ.
48"
30"
1/2" x 1" LAY OIL CLOTH
OVER FRAME
WEIGHT OF MICROFILM
PROPS ~ .0005 OZ. PER INCH
DIAMETER.
CLASS PROJECTED
AREAS
B 30 TO 100 SQ.IN
C 100 TO 150 " "
D 150 TO 300 " "
OUTDOOR
10-17% W. AREA
40°
AR=8-12
.40L
NOT LESS
THAN 3 1/2"
C.G. 70%
3-4.5C
4-7.5C
SPAN
SHOULDER WING AR-10
C
.60L
AR 4-5.5
30%-40%
WIRE
OR
BAMBOO
20°
TWIN RUDDER
ADD 20%
OFF-SET TO THE LEFT
14°
90°+
LANDING GEAR
AHEAD OF VERTICAL
WING SECTION
RAF 32
EIFFEL 400
6412, 9% CLARK Y.
TAIL:
4-6% CLARK Y.
RUDDER:
BALSA SHEET
ZERO LIFT LINES
6° CG AT 40-50 C.
5° " " 60-70
3° " " 100
PROP BLADES
COVERED WITH SILK OR
JAP TISSUE.
12°
GEARS
HOOK TO HOOK .75L
FUSELAGE LENGTH .75 TO 1 1/4 SP.
RIB SPACING 1/3-1/2 CHORD
WING FOR 200 SQ.IN. .8 TO 1.25 OZ
FUSELAGE BALSA SIDES 1 TO 1.5 OZ
ROUND OR OVAL 1.3 OZ
FIN OR STAB .18
PROP + NOSE ASS 1.75
TOTAL EMP. WEIGHT 4 TO 5 OZ.
TISSUE COV'G .6 OZ.
WAKEFIELD RULES:
TOTAL AREA-17-19 DCM.
263.5-294.5 SQ.IN.
MIN TOTAL WEIGHT
230 GRM, 8.113 OZ.
MAX. WEIGHT OF RUBBER
80 GRM., 2.821 OZ.
EIFFEL 400
AT 40% 3.32% CAMBER
OF MEAN LINE
RAF 32
AT 40% 4.23% CAMBER
OF MEAN LINE.
SINKING SPEED
4 OZ P. 100 SQ.IN.
2.25 TO 3 FT P. SEC.

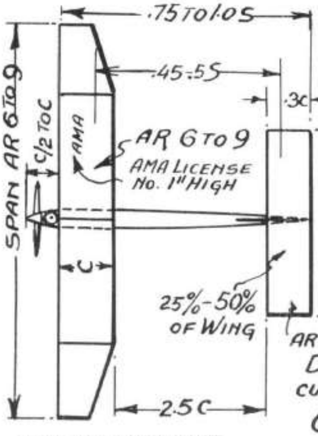
FREE FLIGHT. Models with commercially produced gas engine were flown since 1912. The gas engine advanced from the throublesome interim of spark plug ignition to a simpler glow plug and compression ignition. The total weight of a model reduced from 8 pounds to less than 5 ounces. A general design of a gas model is shown with the wing resting on a low pylon, the engine close to the c.g. and the tail from 25 to 50 per cent of the wing area. The aspect ratio is often as high as 11. If the wing area is restricted the airfoil is a Clark Y and if the area is unlimited the airfoil can be a thinned down Clark Y. The tail is usually on top of the fuselage and the airfoil a 6 to 7 per cent flat bottom section.

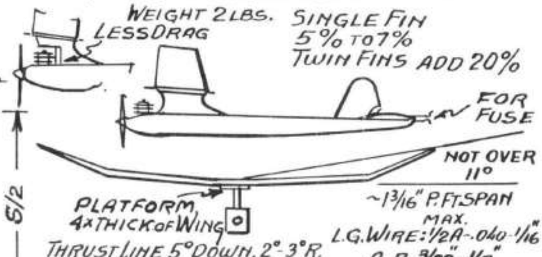
The modeler has always been puzzled by the better performances of a B gas over a 1/2A gas job, even if everything was in proportion to the displacement. A numerical estimate is given for both models. The duration of the B job will be 35% higher than the one for the 1/2A job. The 1/2A job has relatively larger prop, larger fuselage, however a lower slipstream. The worst offender is the stopped propeller on an 1/2A job. Improvement may be made on streamlining the engine, and folding the propeller when gliding.

TAIL FIRST (Canard)- Years ago the most popular models for contests were the tail-first twin-propeller pushers. They were flown for distance and their adjustment were easy because the torque and the gyroscopic precession were balanced by the counter rotating propellers. In addition pusher propellers are self-stabilizing. The fin area on tail-firsts is replaced by a high dihedral of the tail. The higher duration of tractor models soon replaced tail-first models and from then on contests were run for time instead for distance.

Wind tunnel tests on canards were made by the Eiffel Laboratory and are shown on page 43. The spacing of the vectors and their magnitude give an aspect of their restoring power on the longitudinal stability.

The canards have a higher sinking speed than the wing-first models and consequently the canards should give a lower duration than the standard models if the slipstream effect is not taken into consideration. The tail-first models should have at least a 4 degree difference of the zero-lift lines. Propeller driven tail-first models can be made stable by increasing the tail setting and the fin area. A rough graphic construction for locating the c.g. is illustrated.



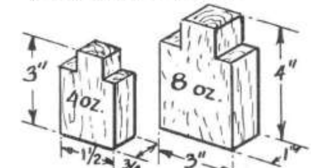


F.A.I. - GAS. 115.6 oz./cu.in. MIN.
.152 cu.in. (2.5 cm³) MAX - 15 SEC. RUN
WING LOADING 2.72 oz/100 sq.in. MIN.
SIX FLIGHTS - 5 BEST ONES.

DESIGN:
$CU.IN. = C \cdot W_{oz}^{3/2} / A_{SQ.IN}^{1/2}$ CU.IN = .0523 $W^{3/2}/A^{1/2}$ ~ LIMIT.

C = .04 TO .067 FAI = 5 FLIGHTS @ 2 ATTEMPTS

PAA FREE FLIGHT



1/2 A @ 4 oz, I.c.c. INT @ 8 oz
+1 oz. +2 oz. +8 oz.

CLIPPER CARGO:
20 SEC. ENGINE RUN
MIN. DURATION 40 SEC.

BEST 1/2A - ENGINE .05
FLY. WEIGHT 30 oz. W_E = 6 oz.

SMALL AND LARGE GAS MODELS CLIMB, AND GLIDE WITHOUT PROP AND WITH STOPPED PROP. SAME POWER AND WING LOADING.

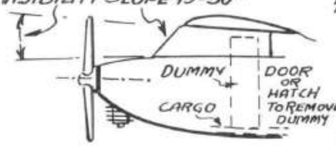
DESIGN - CARGO (BEST)
CU.IN = .006 $W^{3/2}/A^{1/2}$
FOR MAX. A = 3140 x CU.IN.$^{.57}$

GAS F.F. CLASSIFICATION
1/2 A = .000 - .050 CU.IN.
A = .051 - .200 " "
B = .201 - .300 " "
C = .301 - .650 " "
WEIGHT NOT OVER 7 LBS.
NOT LESS THAN 100 oz./CU.IN.

CU.IN DISPL.	.05	.30
PROP DIA	6"	9"
ADDED DRAG TO FUSE BY SLIPSTREAM	20%	40%
CLIMBING FT.P.SEC.	35	38
GLIDING SPEED	17.7	17.7
GLIDING ANGLE L/D		
NO PROP	9.5	12.4
STOPPED PROP	6.9	9.9
SINKING SPEED		
NO PROP	1.8	1.43
STOPPED PROP	2.56	1.86
TOTAL DURATION		
15 SEC. RUN	3M.40S.	
W. STOPPED PROP	4M57S.	

POWER PLANT NOT STREAMLINED.

VISIBILITY SLOPE 45°-50°

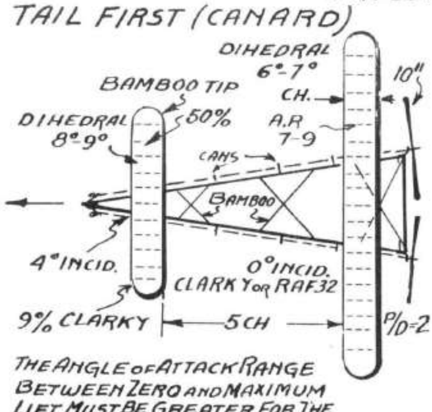


DUMMY IN THE UP RIGHT POSITION. EASY REMOVAL THROUGH SIDE OR TRAP DOORS.

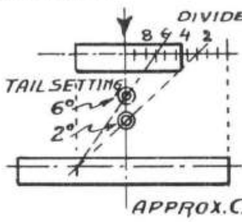
WING		FUSE	TAIL
PROFILE	INDUCED		

DRAG DISTRIBUTION
IN GLIDE THE DRAG OF STOPPED PROP ADDED

TAIL FIRST (CANARD)



THE ANGLE OF ATTACK RANGE BETWEEN ZERO AND MAXIMUM LIFT MUST BE GREATER FOR THE TAIL THAN FOR THE WING.
FOR SECTIONS 12° TO 31° TM 612



DIVIDE INTO 10 PARTS
FOR 50% TAIL
1° TO TAIL SETTING
FOR 100% TAIL
ADD 2° TO GET C.G. LOCATION.

WIND TUNNEL TESTS EIFFEL

APPROX. C.G LOCATION.

WEIGHT DISTR: FRAME 35% WING 43%
2 PROPS 14%, TAIL 8% ~
ADD: 72% FOR RUBBER

1912 - TAIL-FIRST SPRUCE + PINE
75 TO 100 SQ.IN. TOTAL AREA.
MAX. DISTANCE 700 FT.

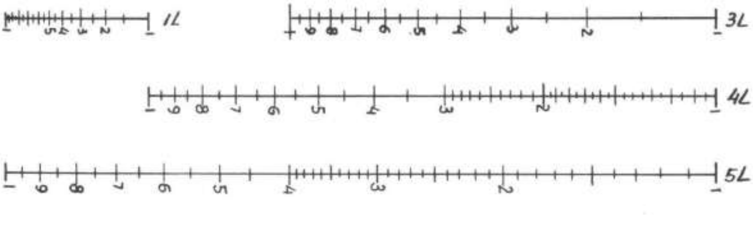
RADIO CONTROL, R/C. Radio controlled models should go into a turn without jerky movements when flying, gliding or diving and return to a straight flight after rudder is neutralized. It should have a 3 to 4 ft.per sec. climb in a straight flight, a slight descend in a circling flight and with full rudder it should dive. The gliding speed should be sufficient to overcome a slight breeze in order to be able to return to the base.

The average design follows a certain relation of the total weight, area and engine displacement, which is given by an equation and two nomograms. The total flying weight is estimated by taking twice the combined weights of the radio and power equipments. The resulting structural weight may give too rigid wings that may snap easily in a glide-landing. The wing section is usually a Clark Y and for the stabilizer a symmetrical 0009 to 0010 section. The c.g. location at the 35 to 40 per cent chord distance slightly above or on the thrustline. To reduce the mass moment of inertia the tail is short coupled and the tail setting has a 8 to 9 degree difference of the zero lift lines The tail and the wing should have a low aspect ratio. The thrustline is offset to the right by 2 degrees to neutralize the torque effect of the engine. In a bank the resultant airforce should pass close above the c.g., which demands a very deep fuselage with its masses evenly distributed about a line at an upward slope passing through the c.g.

This sloping line will be the rotating axis about which the model will rotate at the instant the rudder is operated. First it will have a slight skid, the dihedral will take effect banking the model with its thrustline pointing outward before it begins to turn. This outward position of the thrustline prevents the model from taking an initial dive which it will do in case the rotation is about the thrustline. The centroid of the vertical control surface should be below the c.g. with a dorsal fin to have stability at large angles of attack. The rudder should be of sufficient width to overcome the shift of the resultant, the outward yawing force of the thrust and especially the precession forces of the high speed propeller.

Manufacturers should give the exact position of the c.g., the total weight, the engine and propeller used for the final test. Models built from kits by old-timers dived right to the ground. Other models built from kits kept on flying straight ahead with full rudder. An addition of 1/2 inch to the rudder made it one of the best rudder-only kit.

REFERENCE: COMPUTATION BY NOMOGRAMS BY RAOUL J. HOFFMAN ~ POP. AVIA. 1938
FIVE LOG SCALES OF 1, 2, 3, 4, 5 POWER



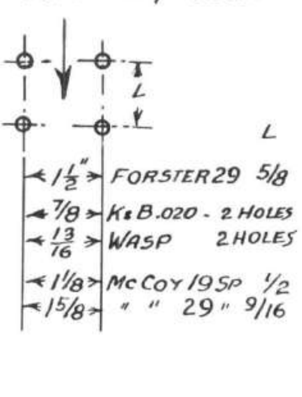
SNOWBANK THEORY. Very interesting observations may be made by watching nature reduce drag of an obstacle by minimum of work. A snow blizzard will streamline an antique car by creating an embankment of snow at the front end of the hood and a slight overhang at the top. In the empty space snow flakes will swirl. This work of nature can be dangerous to airplanes entering a region where a snow storm may be active and climatic conditions suitable. In a short time ice particles will settle on the nose of the wing in a way to reduce the camber of the wing, which will eliminate lift. The cross section will depend on the size of the nose and if the radius is large it will be on two points in the shape of shells with two counter rotating whirls on the inside. Another kind of streamlining may be seen on poles on a flat surface with space left around the pole in the shape of a doughnut. In time all this free space is filled with snow. This action of nature can be applied to model design to reduce drag of exposed engines or balloon tires. It was also used in a suggested design of airfoil sections. A few sketches shows the reduction or the creation of turbulency.

One racing pilot used nature's indication for fairing a LAIRD biplane in 1928 by flying in a downpour and watching the collection of water at the places to be streamlined. Other places were pointed out by darkened spots on the covering caused by swirling dust particles.

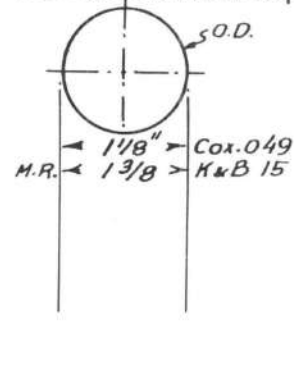
The profile of airfoil sections can be divided into the form drag and the friction drag. The reduction of the form drag may be accomplished by making the flow turbulent far enough forward of the body. It may be possible by roughening of the surface of the body. This increase may be greater than the decrease in the form drag if the form drag is only a small portion of the total drag. This condition takes effect for spheres at high R.N. as shown in two diagrams. It is doubtful that this will occur with model wings of low thickness ratios.

Tests with fluid passing through pipes indicated that if the R.N. is below a minimum turbulence will not start by itself and any turbulence started on the outside will convert to laminar flow again. The question of minimum drag for laminar or turbulent flow must be answered by systematic tests. Mathematical formulation of turbulent flow is unpromising except by averaging tests of similar designs.

MOUNTING HOLES



OUTSIDE DIAMETER

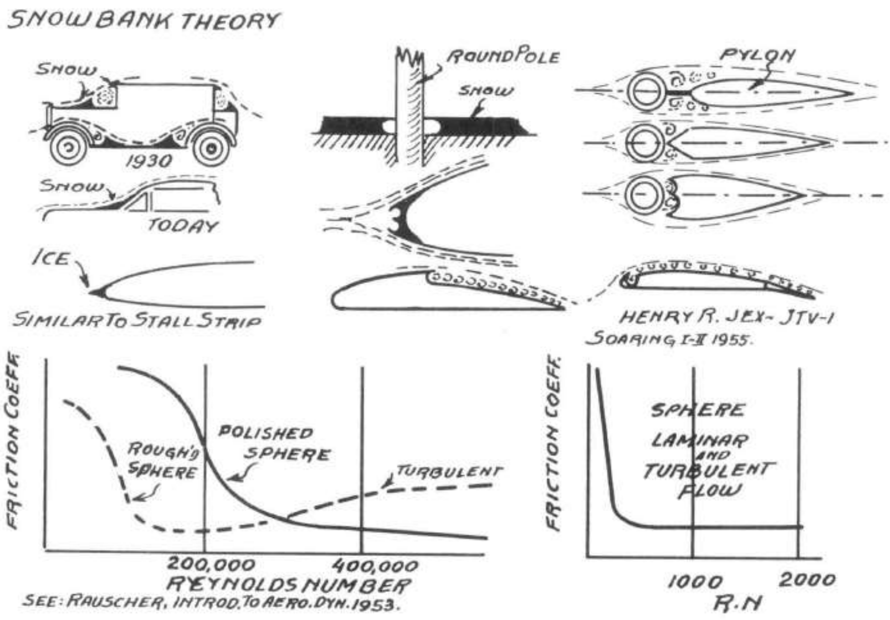


FLYING WINGS- All straight flying wings, planks or all-wings with a straight or cambered mean line will dive. Many arrangements can counteract the diving moments. One is to lower the c.g., which may work in calm weather, but the increase of the drag may be greater for the needed structure than for the tail to make it stable. Another one is to place a large fin or two tip fins of high aspect ratio above the c.g. This design will work only for symmetrical section or low cambered airfoils, because the drag moment of the fin or fins is not very great. This may work for jets and especially with pusher propellers Pusher props are stable when operating and may add to stability in a glide if correctly folded and located at a certain distance.

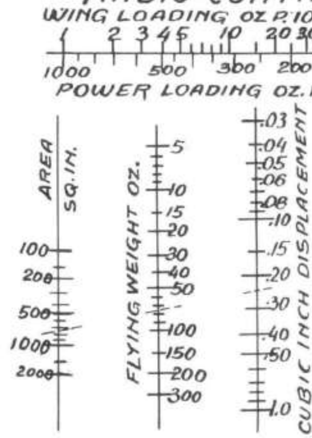
The best arrangement is a sweepback of the wings and a gradual washout toward the tip by about 10 degrees. This design cannot be classed as a true flying wing because part of the wing is used as a stabilizer. Even the CLmax is reduced. A partial negative dihedral may be the answer.

A rubber driven model made from a 100 h.p. low aspect ratio flying wing with zero dihedral needed a large dihedral and an increased fin with a large dorsal fin to make it stable. The stability was very good the duration however was low.

There are no information available on successful gas engine driven straight flying wings without washout.

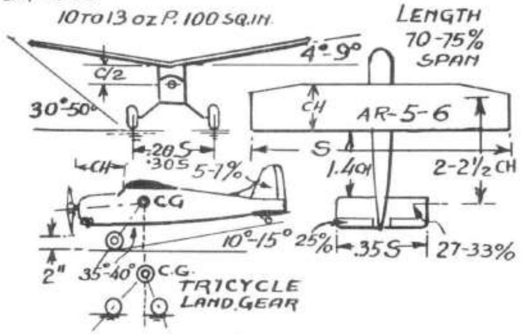


RADIO CONTROL



EQUATION FOR DESIGN

$CU.IN = .010\, W_{OZ}^{3/2} / A^{1/2}$



5-6 FT. P. SEC. CLIMB

WEIGHT ESTIMATE:
SUM OF ALL REMOVABLE WEIGHTS, NOT BELONGING TO THE STRUCTURE ~ RADIO, ENGINE, TANK PROP ~ MULTIPLY BY 1.75 ~ 2.00.

DISTRIBUTION OF WEIGHT NO EQUIPMENT OR ENGINE

WING 40%, FUSELAGE 40%, STAB 6%, LAND. GEAR 14%

WING SILK COVERED.

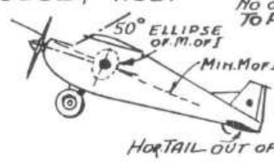
WEIGHT OF WING

$W_{\text{IN OZ.}} = A^{3/2}/1000$

AIRFOIL: WING: CLARK Y, STAB 0009-10
ASPECT RATIO: WING 6-7, STAB 3-4
LONGITUDINAL DIHEDRAL 8°-9° ZERO LIFT LINE
C.G SHOULD BE SLIGHTLY ABOVE THE THRUST LINE. RIGHT THRUST 2°-3°

PROP TO BE CHECKED FOR BALANCE.
RADIO EQUIPMENT TO BE SUSPENDED ON RUBBER BANDS OR PLACED BETWEEN FOAM RUBBER PADS.

NO OF RUBBER STRANDS TO FASTEN WING TO FUSE

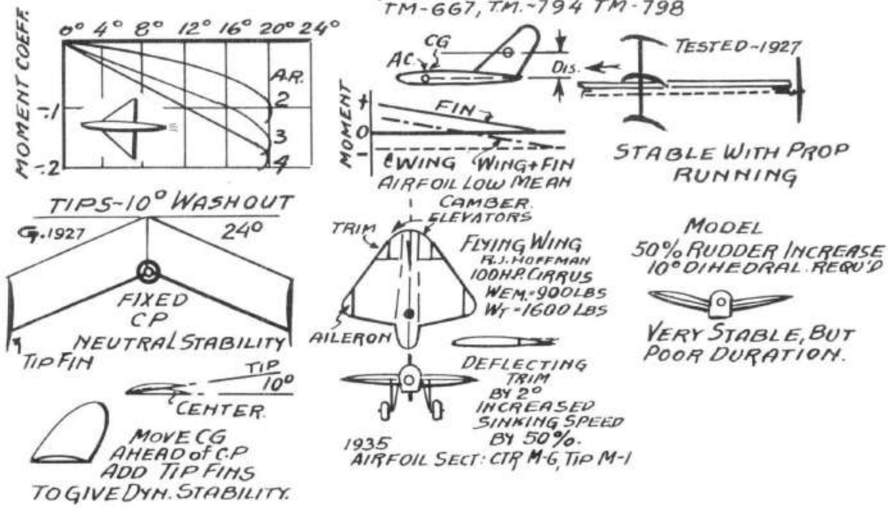


$No = K\, SPAN \times W_T / B$
$SPAN = IN.\ W_{TOT} = OZ$
B = WIDTH OF PLATFORM = IN.

K FOR 1/8 = .016

HOR.TAIL OUT OF DOWNWASH

FLYING WING (TAILLESS) MODELS



CONTROL-LINE MODELS- The performances of various line controlled models are judged for speed, high and low speeds, acrobatics, speed with a limited fuel supply, combat and some new ones in the experimental stages.

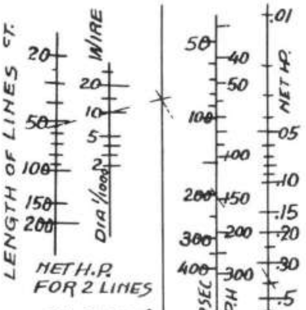
Modelers shy away from speed job, because souped-up engines are used. The preparation of such engines involves a great deal of machine work and ruining a few engines. To increase the speed, models are flown now on a single line. The design for speed is for a minimum frontal area, minimum inlet and outlet areas for the cooling air and a direct intake of cool air to the engine. The direction of the exhaust gases and the cooling air should be inline with the path of flight in order to aid the speed of the model. The wing area should be sufficient to give a fair landing speed, about 50 m.p.h. The tip of the wing should be not over 1/8 in. and 3/4 taper with a 15% thickness ratio at the center. The fuselage should be bent to follow the circle. The setting of the zero-lift lines should be about 4 degrees. The camber 10 times the CL, in per cent of the chord. In case the connections are made within the wings the speed will increase by about 5 m.p.h. The total weight will be about three times the weight of the engine, the c.g. at 20% of the chord. Use maple wood and metal skids. The drag of two lines is 70 per cent of the total drag; 10% for the wing and 20 per cent for the fuselage. For estimates the drag of the lines is equal to 1/4 of the lines moving with the speed of the model.

The team racers must have a minimum area of 125 sq.in., a fuel tank of one fluid ounce, 1.80 cu.in., maximum capacity and a 3.75 by 2 in. minimum fuselage cross section. The contest is for a minimum time of a number of laps on 60 ft. lines. The time for refueling is included. On engine shut-off operated by one of the lines must be installed. Engine displacement is from .15 to .40 cu.in.

The category of stunt models have the greatest number of followers because anyone who can hold a handle and follow the model in a circle will be able to enter a contest. The wing section is a 12 to 18 per cent symmetrical airfoil. Lateral balance is corrected by weights for the weight of the control lines. The c.g. location is at the 20 to 22 per cent of the chord. The flaps are usually external, tapered if for the full length of the span. To insure lateral stability flaps are only used for half of the span at the center section, because the flapped wing will stall 2 to 3 degrees ahead of the tips. The tail is at a very short distance and with a zero setting of the zero lift lines.

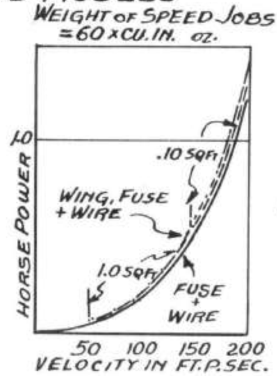
Combats are arranged to cut off paper strips attached to the model. The wing area is about 80 per cent of the regular stunt models or just plain flying wings. Some use plywood rudder and endplates sharpered to cut the streamer, if missed by the propeller. Also the center line of the span is moved about 10 per cent toward the inside of the circle.

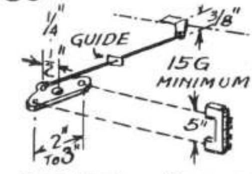
The CARRIER DECK EVENT MODELS. This contest gives the contestand three points for each mile difference between the high speed and the low speed flying 7 laps for each speed. In addition 100 points for a scale copy and 100 points for a hooked landing on a simulated carrier deck. The total weight of the model will be about 70 times the cu.in. displacement in ounces. The wing will weigh about 5.5 oz./100 sq.in. A .35 cu.in. will do the job.



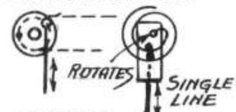
$HP = \frac{TH \times R \times F.P.S^3}{11 \times 10^6}$

TH-DIA. WIRE-INCH

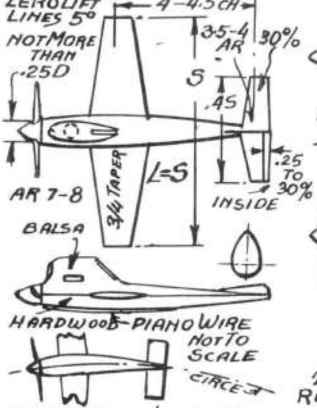




FOR SPEED MODELS THE DIMENSION ARE REDUCED IN ORDER TO SAVE SPACE AND MORE SENSITIVE.



SPEED



CLASS	DISPL'T	L-WIRE
1/2 A	000-.050 CU.IN	35 FT
A	.051-.200 " "	52 1/2 "
B	.201-.300	60 "
C	.301-.650	70

FOR MAX CU.IN IN CLASS

$L = 90\ CU.IN^{1/3}$

$\frac{AREA}{3}$ = 150 SQ.IN./CU.IN.

$$V = 100\sqrt{\frac{550\,HP \times EFF.}{8.3\,A\,C_D + .05\,L\,d}}$$

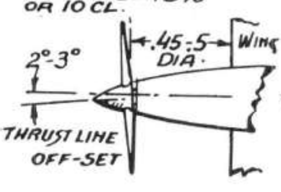
A = SQ.IN, L = FT, $d = \frac{1}{1000}$ IN

V = F.P.S. FOR 2 WIRES

IF LINE IS 35FT ~ 6 LAPS
52.5" ~ 4 "
70 " ~ 3 "

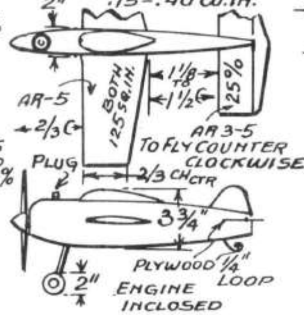
V IN M.P.H ≈ $\frac{900}{\text{TOTAL TIME IN SEC}}$

AIRFOIL 15% ℄ ~ 10% TIP
MEAN CAMBER 3% OR 10 CL.



TEAM RACING

.15-.40 CU.IN.



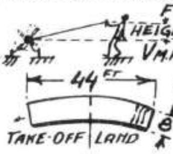
SPECIFICATIONS
125 SQ.IN MIN. AREA
20 G TEST. CIRCLE 65 FT RAD.
RACE: 21, 35, 70 LAPS
MAIN FEATURE 5 MILE 140 LAPS
TANK: 1 FL.OZ. = 1.8 CU.IN = 29.5 CC.
ST'D'D HEIGHT 15 FT. 25 FT.
APPROX SPEED WITH .29 ENG. 75-90 M.P.H.
MUST HAVE ENGINE SHUT-OFF OR ENGINE CONTROL.

WIRE .15-.200 1-.016 2-.010
" .201-.300 1-.018 2-.012
" .301-.400 1-.020 2-.014

FOR APPROXIMATION:
TIME 1 TANK = 22 TO 40/CU.IN.
LAPS 1 TANK = $9.5\text{-}16/CU.IN^{2/3}$
SPEED = $119\ CU.IN.^{1/3}$ M.P.H.
FROM CONTEST
TOTAL TIME IN SEC. = $LAPS\left(\frac{2.4}{CU.IN^{1/3}} + \frac{CU.IN^{1/3}}{9.5}T_{RS}\right)$

T_{RS} = TIME FOR RESTART IN SEC. 2 MIN. MAX.

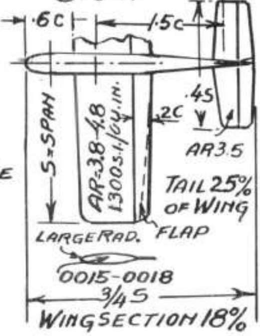
TAKE-OFF WITH CENTRIFUGAL FORCE ONLY



$V_{MPH} = \frac{3.88 R_{FT}}{\sqrt{HEIGHT}}$

AREA FOR DECK LANDING

STUNT



.29 TO .35 CU.IN.
400 TO 500 SQ.IN.
WEIGHT 25 OZ. TO 33 OZ.
.60 CU.IN. 625 SQ.IN
W_T = 50 OZ.

DESIGN:

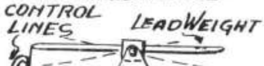
$CU.IN = C \times W_{OZ}^{3/2} / A_{S.IN.}^{1/2}$

$C = .033\text{-}.045$

C.G AT .20-.23 OF AVERAGE CHORD
ELEVATOR 40-50% OF STAB AREA.

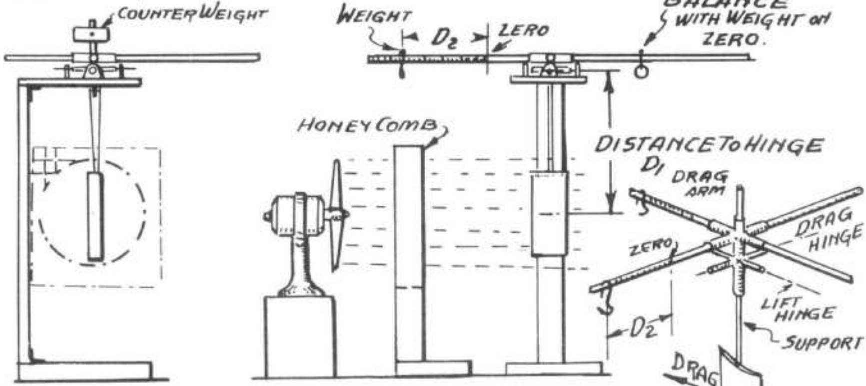
HALF FLAPS BETTER THEY STALL 2° TO 3° AHEAD OF THE WING TIPS

~LATERAL BALANCE~



COMBAT: 60 FT LINE
2-WIRES .014, 1 WIRE .020
0-.49 CU.IN 6-8 FT. LEVEL
DURATION 5 MIN.
PULL TEST
2 FT STRING, 8 FT PAPER
CUT-OFF 110-350 POINTS
AREA FOR COMBAT 1000 SQ.IN/CU.IN.

WIND TUNNEL



$$\text{DRAG OR LIFT} = \frac{\text{WEIGHT} \times D_2 \text{ LESS MOM. FOR SUPPORT}}{D_1}$$

$$CL = \frac{\text{LIFT}}{V^2 \times .0012 \times \text{AREA}}$$

$$CD = \frac{\text{DRAG}}{V^2 \times .0012 \times \text{AREA}}$$

TO FIND AIRSPEED USE DRAG OF A DISC IN EQUATION

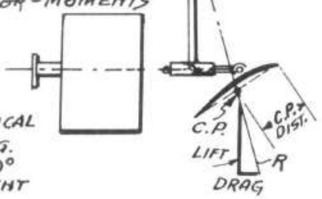
$$V = \sqrt{\frac{\text{DRAG}}{\text{DISC AREA} \times .0012 \times 1.1}} \quad \text{FT. P. SEC.}$$

$$R = \sqrt{\text{LIFT}^2 + \text{DRAG}^2}$$

$$D_3 = \text{MOMENT}_{90°} / R$$

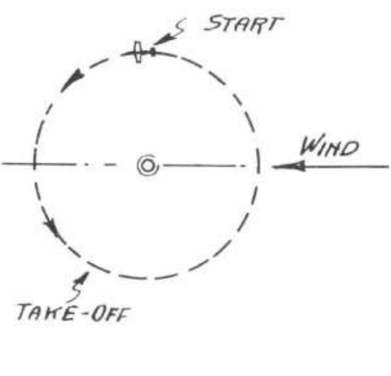
C.P. LOCATED BY GRAPHIC CONSTRUCTION.

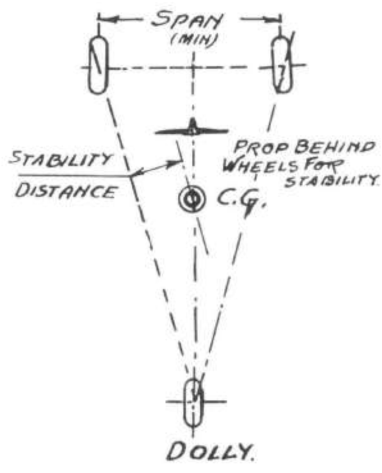
FOR CENTER OF PRESSURE WATCH FOR + OR - MOMENTS



TEST WING VERTICAL FOR LIFT AND DRAG. THEN ROTATE 90° AND FIND MOMENT

REF. POP. AVIAT. 1937 - R.J.H.





DOLLY.

SLIPSTREAM- The accelerated air behind the propeller is the slipstream, which increases the drag of parts in its wake. The slipstream is not evenly distributed over the propeller disc area and is not passing directly to the rear. The inclined propeller blades give the slipstream a slightly helical motion. The fin due to this spiral movement is often off set to prevent the model from turning to the left. In flight the angle, at a distance equals to the diameter of the propeller, is about 1:20. The beneficial effect of the slipstream is used for adjusting the longitudinal stability by increasing the upforce or the downforce of the stabilizer. The upforce is usually used to eliminate the looping tendencies of pylon gas models and a balance is obtained when the thrust times the distance to the c.g. is equal to the additional lift times the distance to the tail.

The velocity of the slipstream can be estimated for static or flying propellers with the given equations and nomograms. The slipstream of the flying propeller cannot be higher than the one found for the static propeller. The thrust for estimating the slipstream is obtained by a nomogram given in the propeller section. Usually only a portion of the stab is affected by the slipstream and the estimate of the effect becomes more complicated when the direction of the slipstream changes in a circling flight.

The slipstream has a pulsating flow, which increases with the number of blades. This however also decreases the efficiency of the propeller. This seems to be contradictory to the use of multibladers on airplanes, which has its answer that the diameter is limited by the Mach number (speed of sound) and the available power must be absorbed by the increase of the total blade area.

To eliminate the increase of the drag by the slipstream, pusher propellers are used. They aid stability and may increase the efficiency of the wing if in line with the downwash.

The slipstream will increase the lift of the wing if the propeller is placed ahead of the leading edge and will decrease the lift if placed slightly below the wing. Placing the propeller in front of the wing may delay the stall due to the reduction of the angle of attack by the resultant of the flying speed and the velocity of the slipstream, and consequently the tips may stall ahead of the center of the wing, reducing the lateral stability.

In a rare case a condition may arise in which the change of the thrustline to the right will turn the model to the left. This is possible if a large flat surface area lies ahead of the c.g.

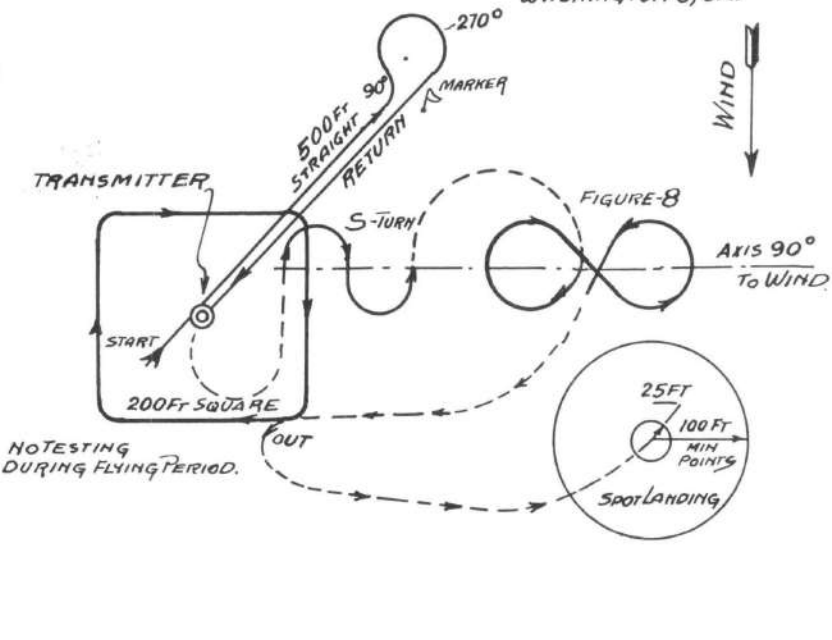
WIND TUNNELS. Small wind tunnels are very useful for comparing component parts and for eliminating poor design details. They are excellent for educational purposes. Open throat tunnels are better than venturi type ones. A 1/4 h.p. motor driving a 14 inch fan will give an airspeed over 30 ft.p.sec., which is sufficient for designing light airplanes. All small tunnels are turbulent.

LOCATION OF THE CENTER OF GRAVITY. The attraction of the earth is indicated by the gravitational forces acting on the masses of the model. If any mass is allowed to drop without the drag of the air it will accelerate by about 32.2 f.p.sec. The resultant of all gravitational forces will pass through the c.g. Balancing the model horizontally the line through the c.g. is easily located. For easy adjustment the c.g. location should be obtained on the sideview by balancing the model with the wing in a vertical position. This will give the distance of the c.g. to the thrustline and to the baseline of the average chord of the wing. This point should be given in all plans of manufactured kits.

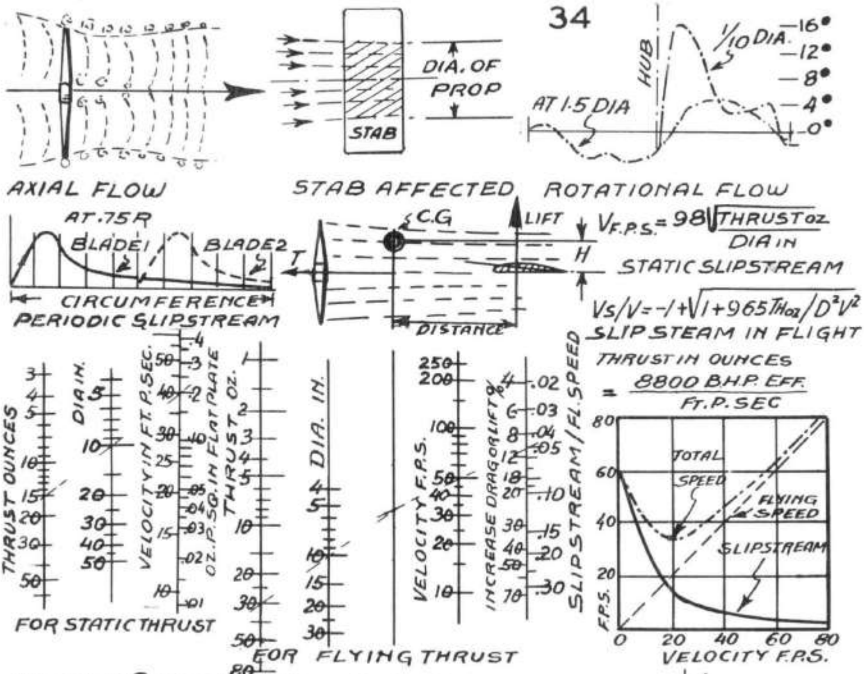
The c.g. of a plain glider is located on the air resultant of the wing and the tail that gives the lowest sinking speed. By adding the drag of the fuselage and the rudder the location of the c.g. will not change, the slope of the resultant however will be steeper. This well defined position of the c.g. does not happen on a pylon job because the resultant will change considerably with the addition of the drag forces of fuselage and landing gear. Therefore the location of the c.g. will depend on the design and on the magnitude of the additional drag of each unit. Years ago when adjustment was not so simple and competition was not so keen corrections were made by wrapping solder wire around landing gears or increase the size of the wheels to eliminate climbing moments.

The drag of the deflected rudder on a radio controlled model affects the longitudinal stability. In case the drag resultant passes below the c.g. the model will dive and if it passes above the c.g. the model will climb; the effect may be hardly noticeable.

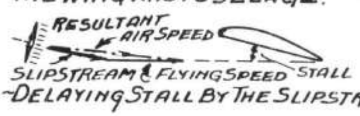
FLIGHT PATTERN FOR RADIO CONTOL MODELS
FOR OFFICIAL RULES WRITE TO ACADEMY OF MODEL AERONAUTICS
WASHINGTON 6, D.C.



SLIPSTREAM



PUSHER PROPS ELIMINATE SLIPSTREAM OVER THE MODELS. MAY RECOVER SOME ENERGY OF THE WAKE FROM THE WING AND FUSELAGE.



~DELAYING STALL BY THE SLIPSTREAM~

STATIC SLIPSTREAM EFFECT ON WINGS.

STATIC EFFECT T.R. 569

THRUST PULLS MODEL TO THE RIGHT SLIPSTREAM TO THE LEFT.

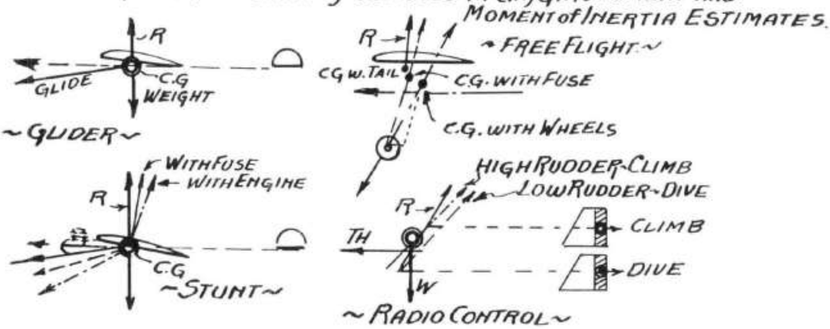
THE SLIPSTREAM IS USED TO GIVE AIRPLANES A V.T.O.

LOCATION OF CENTER OF GRAVITY.

GRAVITATIONAL FORCE = ACCELERATION 32.2 F.P.S. ≅ 32 F.P.S. = $a = g$

DISTANCE = $\frac{1}{2} g\, SEC^2$ SPEED = $V = g\, SEC$ SPEED FREE DROP - $V = \sqrt{2gh}$

ACCELERATION CONSTANT g - 32.2 USED IN C.F., GYRO MOMENT AND MOMENT OF INERTIA ESTIMATES.

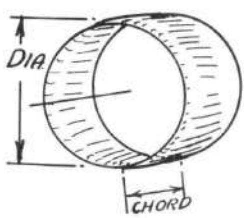


BIPLANES. The advantages of a biplane is its concentration of areas and masses, which makes it one of the best stunt control line model and also the easiest to adjust. Change of the stagger and the decalage give very versatile wing combinations. Stagger improves stability and if a 4 degree decalage is given to the upper wing the biplane becomes stable by itself, as shown in a vector diagram. In case this arrangement is changed by moving the lower wing farther to the rear it converts into the basic combination of a light plane known years ago as the "Flying Flea", with the lower wing used for an elevator control, often with disastrous results. Increasing the distance of the lower wing the biplane ends up as a tandem with reduced efficiency due to the downwash from the main wing.

TORQUE AND COUNTER TORQUE- The counter-torque is the reaction of the engine or the motor which drives a propeller or any other unit of propulsion. The counter torque will roll the model to the left if the propeller is a right-hand propeller used as a tractor. The counter torque is usually denoted as the torque of the drive. To overcome the rolling moment an equal moment of opposite sign must be created. One method is to offset the wing to the left and set the rudder to the right. Another arrangement is to bend the boom carrying the tail to the left. The lift of the tail and the new location of the c.g. will give the model the required right-hand moments. These adjustments are for indoor models with a single bearing. For outdoor models the correction is made by off-setting the thrustline to the right with the rudder set to the left for left-hand turning glide. For left-hand glide the stab is tilted with the right side down or with the wing down on the left side. Tilting the wing is very sensitive due to the great moment arm. The adjustment of the thrustline is about 2 to 3 degrees. An equation is given for estimating the off-set angle.

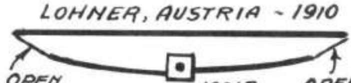
In case the pylon is low the torque and the gyroscopic moments are fairly balanced the rudder is set for a left-hand turn. The model climbing in a slow spiral does not need a thrust adjustment. Auto flaps are effective at the take-off and in a glide, but in a steep climb they do not affect the spiral. The flap is loosely hinged made of thin aluminum, attached to the left panel.

The counter-torque of helicopter models were balanced by the slipstream acting on three vanes having 1/7 camber ratios. They may not work for very high pitch settings.



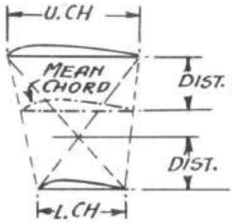
BARREL-WING
AREA = DIA. x CHORD
CHORD/DIA. = .60
CL MAX 1.1 - 19°
CHORD/DIA. = .40
CL MAX = 1.4
CD TWICE OF THAT OF A REGULAR WING
BEST GLIDING ANGLE AT R.N. 850,000 L/D = 10

LOHNER, AUSTRIA - 1910

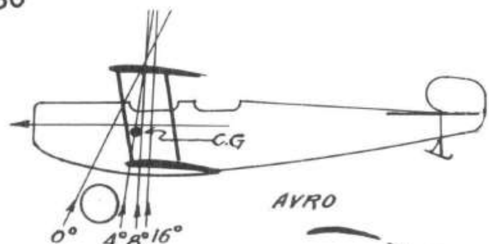


BIPLANE WINGS CONNECTED AT TIPS. PLANE ROCKED 90° BOTH WAYS UNTIL TIP SECTIONS WERE OPENED.

BIPLANE

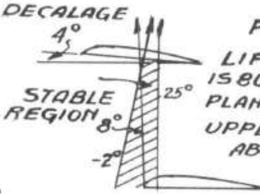


CONSTRUCTION OF MEAN AERODYNAMIC CHORD. APPLICABLE ONLY TO ORTHOGONAL BIPLANES

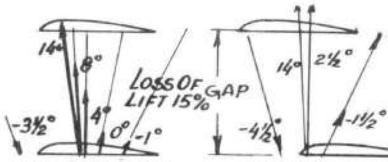


DECALAGE

FLYING FLEA



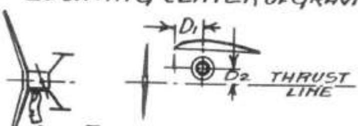
LIFT OF BIPLANE IS 80% OF A MONOPLANE OF SAME AREA. UPPER WING CARRIES ABOUT 56% OF TOTAL



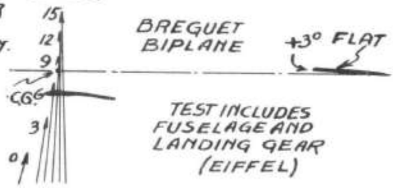
ORTHOGONAL | POSITIVE STAGGER

POS. STAGGER + DECALAGE

LOCATING CENTER OF GRAVITY.



FINGER OR PENCIL



BREGUET BIPLANE

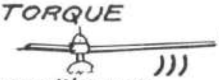
TEST INCLUDES FUSELAGE AND LANDING GEAR (EIFFEL)

TORQUE AND COUNTER TORQUE

$$TORQUE_{OZ.IN.} = \frac{B.H.P. \times 1{,}000{,}000}{R.P.M.}$$

ASSUMING BHP = 1.4 CU.IN.

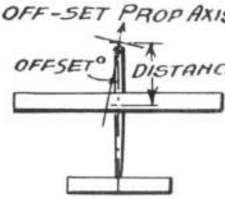
$$TORQUE_{OZ.IN.} = \frac{CU.IN.\ 1{,}400{,}000}{R.P.M.}$$

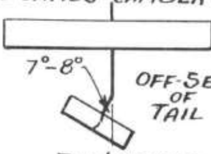


DOWNWASH OF HELICOPTER BLADES. ~TESTED~ 3 VANES-CAMBER 1/7

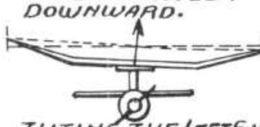
TILTING THE RIGHT SIDE OF THE STABILIZER DOWNWARD.

OFF-SET PROP AXIS





FOR INDOOR MODELS RIGHT HAND GLIDE



TILTING THE LEFT SIDE OF THE WING DOWNWARD.

FOR LEFT HAND GLIDE

$$OFF\text{-}SET^\circ = \frac{1.42\ \alpha_a^\circ\ TORQUE_{OZ.IN}}{AR^{2/3}\ THRUST_{OZ}\ DIS_{IN}}$$

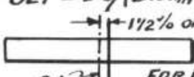
IN.OZ, OZ, IN SYSTEM.

AFTER SUBSTITUTIONS:

$OFF\text{-}SET^\circ = 90/(DIST_{IN}\ AR^{2/3})$

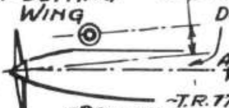
AUTO FLAPS FOR STEEP CLIMBING GAS MODELS: IN CLIMB FLAP IN LINE WITH CHORD, IN GLIDE FLAP ADDS DRAG FOR LEFT HAND TURN.

FOR LEFT HAND GLIDE



OFF-SETTING WING

FOR INDOOR DIST. TO C.G.



5° DOWNTHRUST FOR LARGE PLANES

NO POWER TURNS BY OFF-SETTING RUDDER

TO OBTAIN SIMILAR PERFORMANCES TOTAL WEIGHT, ENGINE, PROPELLER DIA AND D, DISTANCE OF C.G. TO THRUST LINE AND DISTANCE OF C.G TO L.E. OF WING.

MANUFACTURERS PLEASE NOTICE.

STABILITY. A flying model should keep on its intended path. Any deviation should be corrected by a force that will return it to its original position. The model will rotate about its c.g. at the instant it is disturbed. This restoring by an airforce is stability. There is no PENDULUM stability.

To simplify investigation of stability only three main axes about which the model will rotate are to be considered. The movement about the lateral axis for longitudinal stability, about the vertical axis for directional stability and about the longitudinal axis for lateral stability.

The restoring airforce must locate in the direction of the change. Three sketches indicate their locations. The restoring velocity depends on the magnitude of the force times its distance from the c.g., its weight distribution and on the general design of the model. If the restoring moment is too large or too small the model is unstable. With too large a rolling moment the model will flip over, too large yawing moment the model goes into a spiral dive and with a too great pitching moment the model will loop or dive. With the correct amount of restoring moments the model will have small oscillations about their axes, which is an indication of dynamic stability. Moments are positive if they increase the angle of attack and negative if they decrease the angles. Climbing moments are positive and diving moments negative.

Plotting the moments or their coefficients for various angles of attack will give a line which has a certain slope. The magnitude of the slope is indicative of the dynamic stability of each specific design. The point where the sloping line intersects the zero moment line is the angle of trim, the angle at which the model will fly on an even keel if conditions are fair.

LONGITUDINAL STABILITY.- An airfoil with a cambered mean line is unstable, its resultants move forward with increase of angle. Lowering the c.g. may make the model stable in calm weather. If the camber is small, a large, high aspeet ratio fin slanting backwards may give stability by the positive moments created by the drag of the fin.

The simplest means to make a wing stable is to place a tail in front or in the rear. The tail or stabilizer placed in back of the wing is more efficient. The incidence angle of the stab is always smaller than the one for the wing. This setting increases the moments faster than the ones created by the wing, a moment balance called longitudinal stability.

Many methods are possible for obtaining numerical values for the moments. All are only approximations and especially the ones calculated for dynamic stability. The graphic method used in the early stages of airplane designs can be applied to model design in which the c.g. is well below the wing. Nomograms, alignment charts are very useful for stability calculations of models with the wing, fuselage, tail and thrustline on the same line. Moment calculations are based on the distance of the c.g. to the air resultants, on the magnitude of the air forces, on the area and on the dynamic pressure, q. Usually in comparing moments the dynamic presuure is omitted.

The propeller moments are estimated by an equation or a nomogram that give the value for one degree of yaw. To obtain the fuselage moments the maximum cross sectional area and the length are used. In summation the moments, the plus and the minus signs should be considered.

A graphic method uses a plan or a scaled-down drawing on which the CL and the CD coefficients are plotted for various angles of the wing, with an interval of two, three or even four degrees. Similar layouts are made for the fuselage and the stabilizer. The relative angles between the units must be coordinated, especially the effect of the downwash must be considered when plotting the vectors of the tail. Adding all moments for a certain angle of attack, dividing it by the wing area and by the wing chord, the pitching moment coefficient is obtained. To find the actual moments in oz.in. the sum of the moments is mutiplied by the dynamic pressure and divided by 9.

In case a drawing is not available the diagrams on page 40 should be used and the positions of the c.g. are located on the diagrams. The diagram for the wing is based on an A.R. of 6. Two sets of vectors are plotted; one set gives the resultants of the lift and induced drag coefficients; the other gives only the direction of the profile drag coefficient, because the coefficient is assumed to be constant. Two diagrams are given for estimating the effect of the fuselage. For any powered model use the diagram for the rectangular fusealge. The diagram for the wheels is based on a drag coefficient of 1.00.

In all estimates the shortest distance from the c.g. to the vector is used. The moment or the moment coefficient is positive if it tends to increase the angle and it is negative if it causes the decrease of the angle. The pitching moment coefficient is obtained by measuring the distance and multiplying it by the value noted on the vector. One such moment is found for the lift vector and one for the profile drag coefficient. Adding to these two moment coefficients the Cmac and mutiplying it by the chord and the area of the wing the factor for one angle is determined. The same procedure is followed for the other angles of the wing.

The factor for the fuselage is obtained by measuring the distance and multiplying it by the drag coefficient, by the value on the vector, by the length and by the cross-sectional area of the fuselage. A similar method is used for estimating the factor for the wheel or wheels by taking the distance and multiplying it by the diameter and by the cross-sectional area. The prop factor is obtained from the nomogram on page 40. The tail factor is found by the nomogram on page 41. Summing the factors for each angle and dividing them by the wing chord and the wing area, the pitching moment coefficients are determined. Plotting the coefficients against the angles of attack and connecting the points, a line is obtained whose slope is indicative of the dynamic stability for each specific design. To estimate the actual moments in oz.in. the sum of all factors is multiplied by the dynamic pressure and divided bv 9.

In case the thrustline is at a distance from the c.g. the factor can be calculated with the equations if the speed of the model is assumed. In the above estimates the slipstream effect is neglected.

Another method uses a greater amount of mathematical operations. The wing and the tail momentfactors are estimated by equations and nomograms given on page **41**. The propeller factor is estimated by a nomogram on the page **40**. The fuselage moment factor is omitted.

Dynamic stability can be estimated by finding first the neutral point that is a location about which the static moments of the wing and the tail are the same for all angles of attack. The distance to the neutral point is from the a.c. Subtracting a certain distance, called the static margin, should give the location of the c.g. for dynamic stability. The static margin is about 15 to 25 per cent of the chord. An equation is given for close estimate of the neutral point. For approximation a nomogram may be used, which is plotted by assuming that a few variables are constant. The static margin should be obtained for each specific design. It is obvious that static margin decreases as the distance of the wing from the fuselage increases.

Vector diagrams, illustrated on a following page were taken from the reports of the Eiffel Laboratory 1915, give the airforces of regular and tandem combinations, and the loads carried on the tail.

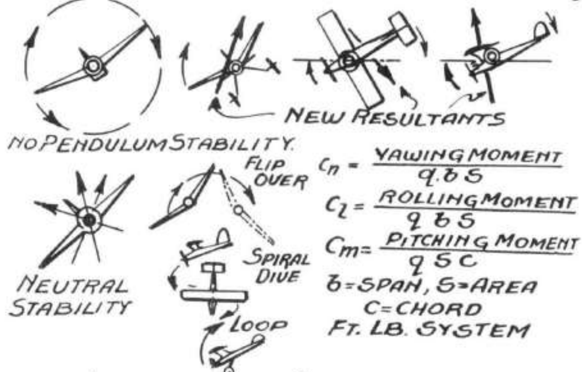
Nomograms on page 42 are for finding the neutral point of a model withe c.g. inline with the propeller axis, the wing and the fuselage. The exposed engine may change the location of the c.g., because it will give a certain climbing moment, which is balanced by moving the c.g. slightly ahead.

To obtain the c.g. location of a control-line flying wing with the aid of nomograms, it is assumed that the area and the aspect ratio of the wing and the tail are the same, and that the distance is taken from the 1/4 point to the elevator. The chord of the elevator should be at least 15 per cent of the wing chord. The location of the c.g. on a flying saucer can be assumed to be at 22 to 24 per cent of the maximum chord.

RUBBER: INDOOR CLASS B 30-100SQ.IN., C-100-150 SQ.IN, D-150-300
STAB 50% MAX. HAND-L'D STICK ~ FUSE $L^2/150$ MAX.
FLYING WING INCL. CLASS B, C AND D
R.O.G. CABIN: CLASS B, C. WHEEL DIA. B-3/4", C-1" FUSE
$L^2/100$ MIN. 90% COVERED, OUTRIGGERS + BOOMS
R.O.W RISE-OFF-WATER CABIN ~ B 5 SEC. FLOTATION.
ALL 6 FLIGHTS ~ 3 BEST-TOTAL.
OUTDOOR: H.L. LIMITED, 200 SQ.IN. COMB., 5 oz. MIN. 6 FLIGHTS, 3 BEST.
R.O.W. LIMITED. 30 SEC. FLOTATION
WAKEFIELD, R.O.G. 263.5-294.5 SQ.IN. 8.113 oz MIN, RUBBER 2.821 oz. MAX.
P.A.A-LOAD

GAS FREE FLIGHT 1/2A -.050 CU.IN, A-.051-.200, B .201-.300, C .301-.650. 60% FOR
4-STROKE CYCLE. R.O.G. 100 oz. P. CU.IN. 15 SEC OR 20 SEC.'
R.O.W ~ 6 FLIGHTS - 3 BEST.
P.A.A.-LOAD 1/2A, A, B.-1/2 A = 1 4 oz. DUMMY + 1 oz. 1 cc. 1 4 oz. DUM. + 2 OZ.
P.A.A. CLIPPER CARGO P.A.A. INT. 8 oz. DUM. + 8 oz.
F.A.I. 115.6 oz/cu.IN MIN, .1525 cu.IN MAX. 3.93 oz/sq.FT MIN. 15 SEC.
R.O.G. 20 SEC. OFF. 2 TIMERS. 5 FL. 3 MINUTES.-6th.

STABILITY

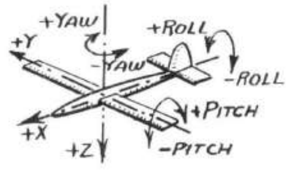


$$C_n = \frac{\text{YAWING MOMENT}}{q\,b\,S}$$

$$C_l = \frac{\text{ROLLING MOMENT}}{q\,b\,S}$$

$$C_m = \frac{\text{PITCHING MOMENT}}{q\,S\,C}$$

b = SPAN, S = AREA
C = CHORD
FT. LB. SYSTEM

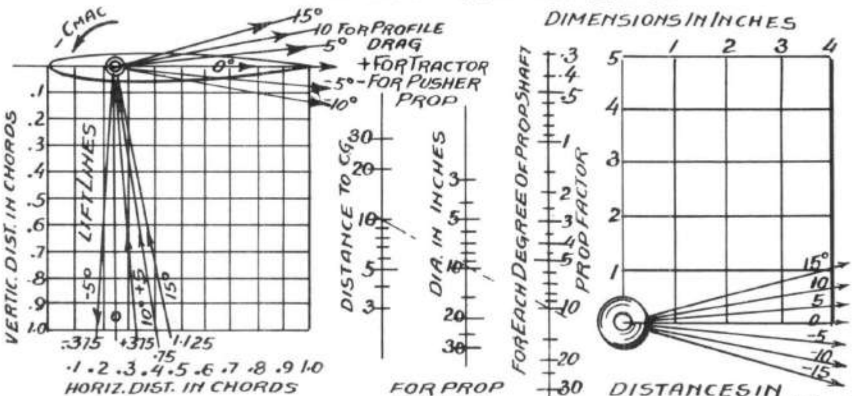


AUTOROTATION ANGLE AT HALF SPAN

$$= \frac{\rho' \text{ SPAN}}{2 \times \text{VELOCITY}}$$

ρ' = ANGULAR VELOCITY
ANGLES IN RADIANS

LONGITUDINAL STABILITY ~ GRAPHIC METHOD

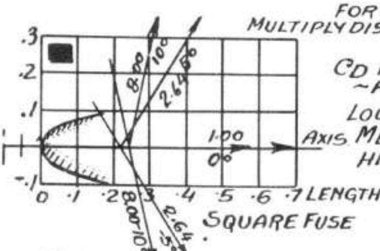


FOR WING FACTOR =
DIST. RATIO × CHORD × WING AREA

FOR PROP FACTOR EQUATION
$F_p = .01\,DIA^2\,DIST$

DISTANCES IN DIAMETERS OF WHEEL
FOR MOM'T MULTIPLY DIST. BY WHEEL DIA, CROSS-SECT.,

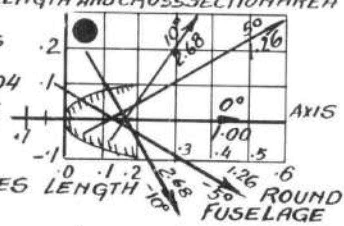
FOR FUSELAGE FACTOR =
MULTIPLY DIST. RATIO BY LENGTH AND CROSS SECTION AREA



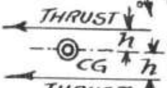
SQUARE FUSE

C_D FOR WINGS
~PROFILE~
LOW SPEED .04
MED. " .025
HIGH " .010

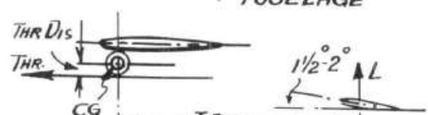
C_D FOR FUSELAGES



ROUND FUSELAGE



$$M_{TH} = \frac{TH_{oz}\,h_{IN} \times 9}{.00119\,V^2}$$



RADIO STUNT ELEVATOR CONTROL

YAW'G MOM. OF WING = $A_W\,q\,\text{SPAN}\,\dfrac{(a^2_W + \Delta a^2/4)}{4170\,AR^{1/3}_W}$

ROLL'G MOM OF WING = $A_W\,q\,\text{SPAN}\,\Delta a\,.039\,AR^{1/3}_W/2$

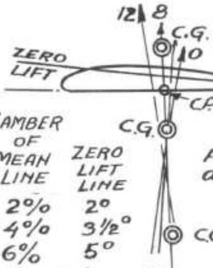
YAW'G MOM'T OF FIN = $A_{FIN}\,q\,B\,.039\,AR^{1/3}_F$ DIST.

a_W = ABSOLUTE ANGLE OF ATTACK
Δa - CHANGE OF ANGLE = $B\,G/57.3$
B = SKID ANGLE, G = DIHEDRAL

BALANCE OF FORCES: $L \times T_{DIS} = THR \times THR_{DIS}$.

TO OBTAIN C_m COEFFICIENT
ADD WING, FUSE, TAIL AND PROP FACTORS
DIVIDE BY WING AREA AN CHORD
THEN ADD $C_{m\,a.c.}$

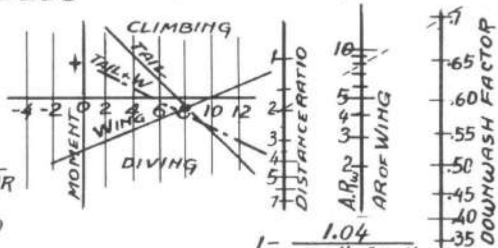
STABILITY OF MODELS LONGITUDINAL 41



CAMBER OF MEAN LINE	ZERO LIFT LINE
2%	2°
4%	3½°
6%	5°

HEAVY LAND. GEAR LOWERS C.G. IMPROVES STABILITY.

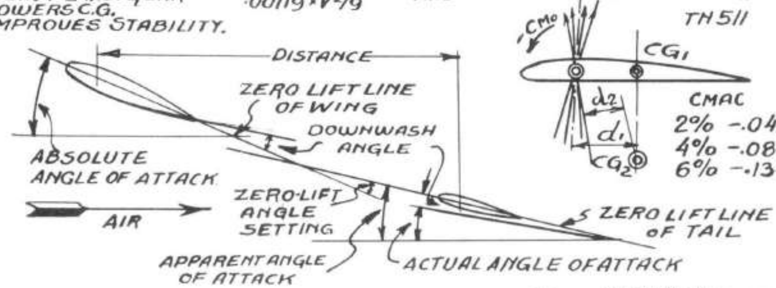
ASSUMING: $dC_L/d\alpha = .039\sqrt[3]{AR}$
VALID FOR AR 2 TO 10



$$1-\frac{1.04}{D.RAT^{1/4}\,AR_W^{1/2}}$$

$$\frac{\text{WING MOMENT}}{.00119 \times V^2/g} = C_{MAC} \times CH \times A_W + A_W \times DIST.AC \times .039\sqrt[3]{AR_W}\ ANG_W$$

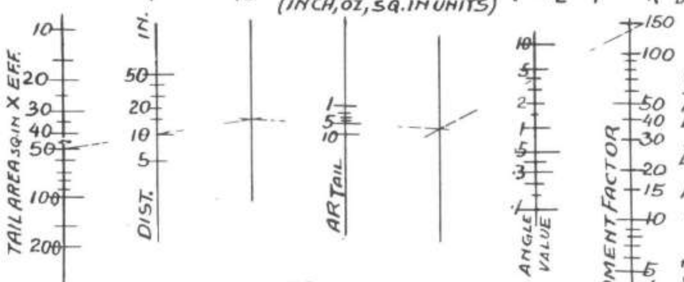
TN 511



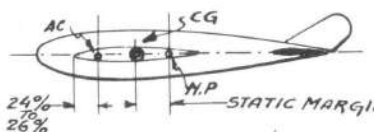
CMAC	
2%	-.042
4%	-.084
6%	-.130

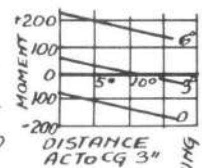
$$\text{TAIL MOMENT}/.00119\,V^2/g = -\text{TAIL AREA} \times \text{DIST} \times \text{EFF} \times .039\sqrt[3]{AR_T}\left[\underbrace{\text{ANG. WING}\left(1-\frac{1.04}{D.RAT^{1/4} \times AR_W^{1/2}}\right)-\text{SETT.}}_{\text{ANGLE VALUE}}\right]$$

(INCH, OZ, SQ.IN UNITS)

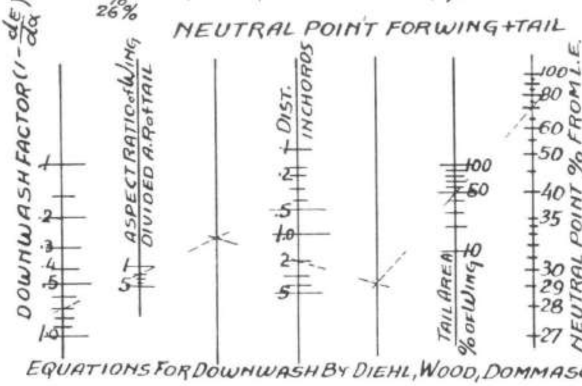


EXAMPLE
$C_{MAC} = .084$
CHORD = 5" A = 200 S.I.
SPAN 40" $AR_W = 8$
A_T 67 S.I. $AR_T = 4$
EFF .80 DIST RAT. 3
DIST TO TAIL 15"
DOWNW. FACTOR .73
C.G TO A.C. 3"
$M_W = -84 + 15\ DIS.\alpha$
$M_{TAIL} = -36.5\alpha + 50\ SET$





NEUTRAL POINT FOR WING + TAIL



NEUTRAL POINT

$$N.P. = \left(1-\frac{d\epsilon}{d\alpha}\right)\frac{dC_{LT}/d\alpha}{dC_{LW}/d\alpha} \times \frac{q_t}{q}\frac{S_T}{S_W}\frac{\ell}{C}$$

DOWNWASH ANGLE

$$E = \frac{52\,C_L}{AR}(X+1)^{-.38}(Y+1)^{-.23}$$

X = DIST. BEHIND TR. ED.
Y = " BELOW TR. ED.
IN CHORDS

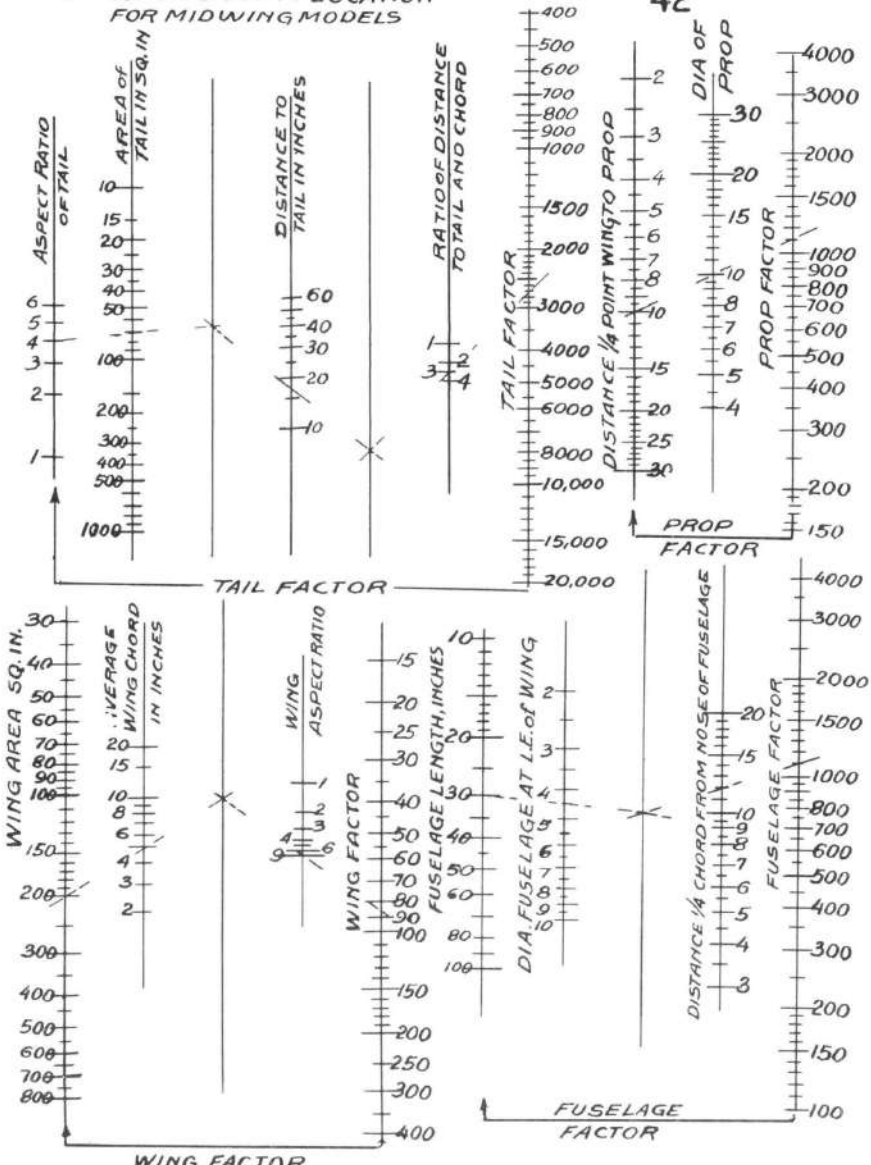
$$E = 20\,C_L\frac{\lambda^{.3}}{AR^{.725}}\left(\frac{3c}{l}\right)^{.25} \quad \lambda = \text{TAPER}$$

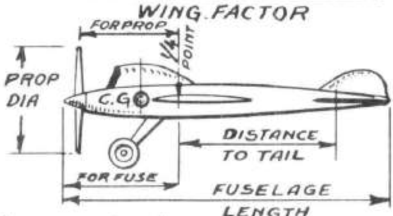
$$E = 36.5\,C_L/AR$$

$$\frac{E}{\alpha} = \frac{4 - L_T/3C}{AR+2}$$

EQUATIONS FOR DOWNWASH BY DIEHL, WOOD, DOMMASCH

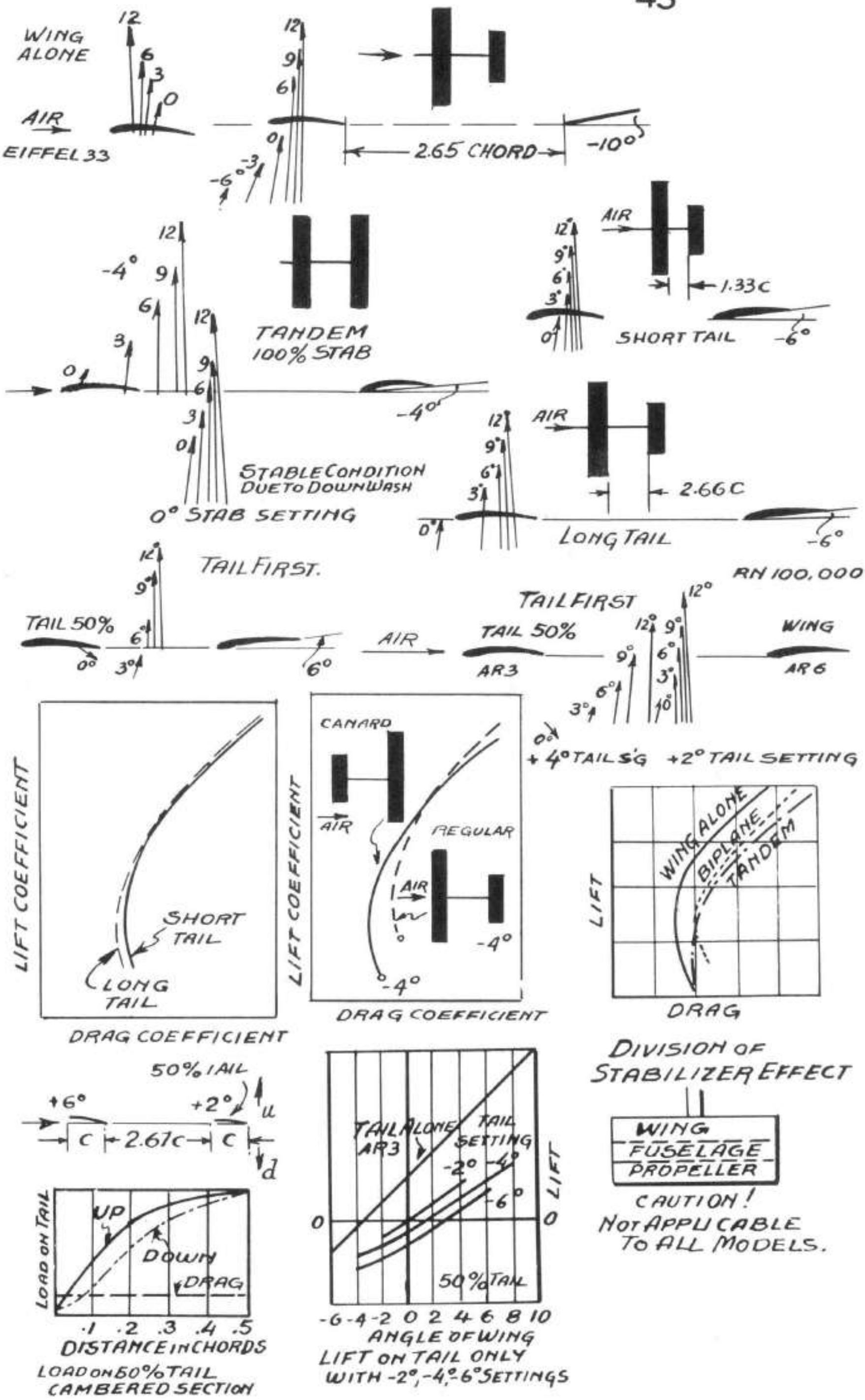
CENTER OF GRAVITY LOCATION FOR MIDWING MODELS





ANSWER FOR INDICATED EXAMPLE TO BE 23%C

HOW TO USE NOMOGRAMS:
SUBTRACT PROP AND FUSELAGE FACTORS FROM TAIL FACTOR. DIVIDE BY WING FACTOR. THEN ADD 25. TO GET THE CG LOCATION IN PERCENT OF THE CHORD FROM LEADING EDGE of WING SUBTRACT 15 FOR GLIDERS, 20 FOR FREE FLIGHT MODELS AND 15 FOR CONTROL LINE MODELS.
THE SUBTRACTIONS-15, 20, 15,-ARE STATIC MARGINS


WING ALONE
AIR
EIFFEL 33
2.65 CHORD
-10°
TANDEM 100% STAB
STABLE CONDITION DUE TO DOWNWASH
0° STAB SETTING
SHORT TAIL
1.33C
LONG TAIL
2.66C
TAIL FIRST.
TAIL 50%
RN 100,000
TAIL FIRST
TAIL 50%
AR3
WING
AR6
+4° TAIL S'G
+2° TAIL SETTING
LIFT COEFFICIENT
SHORT TAIL
LONG TAIL
DRAG COEFFICIENT
CANARD
REGULAR
LIFT COEFFICIENT
DRAG COEFFICIENT
LIFT
WING ALONE
BIPLANE
TANDEM
DRAG
DIVISION OF STABILIZER EFFECT
WING
FUSELAGE
PROPELLER
CAUTION!
NOT APPLICABLE TO ALL MODELS.
50% TAIL
2.67c
LOAD ON TAIL
UP
DOWN
DRAG
DISTANCE IN CHORDS
LOAD ON 50% TAIL CAMBERED SECTION
TAIL ALONE AR3
TAIL SETTING
50% TAIL
ANGLE OF WING
LIFT ON TAIL ONLY WITH -2°, -4°, -6° SETTINGS

DIRECTIONAL STABILITY. Directional stability is not only the restoring effect of vertical surfaces that keeps a model from yawing but also to keep it intentionally in a circling position. The resultant sideforce of the balancing moment should pass just above the c.g. This sideforce could be created by large vertical surfaces placed above the c.g. but the same effect could be obtained by a slight lateral (dihedral) angle of the wing. This interchange of wing dihedral for the necessary vertical surface effect brings the fin and the dihedral in a direct relation.

The size of the fin depends on the relative location of the wing, the fuselage, the propeller, the gyroscopic forces and many other disturbing forces. Fuselage and fin tested separately did not give the same result with the fin attached to the fuselage. This relation makes estimate of the fin area an approximation not only for the model designer but also for the large airplane designer.

The area of the fin depends on its aspect ratio, The higher the aspect ratio the greater the effect, as illustrated in a diagram. It indicates that an aspect ratio higher than 4 will only add weight, especially to models with low wing loading. The dynamic stability will depend on the weight distribution, the moment of inertia of the masses. The dorsal fin is not included in the estimate of the fin because it action starts only after the yawing angle passes 20 degrees as given in a diagram. For approximation for each specific design a nomogram gives the fin area in per cent of the wing area if the span and the distance to the fin is known. Two diagrams shows the interaction of fin area and dihedral angle. If the fin area is too large the model will go into a spiral dive if its too small it will have a dutch roll, weaving from one side to the other. A slight dutch roll is preferred.

The fin area required to balance all yawing moments will give only neutral stability, to which 5 to 10 per cent area is added for dynamic stability. The yawing moments of an oval fuselage are small and therefore the estimate of the required fin area must be accurate. Designing and adjusting models with oval or round fuselages are only for experts.

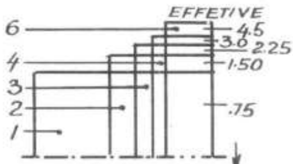
Twin rudders should only be used if the height is at least half of the stab chord and the aspect ratio of the stabilizer is low. In any case it should be 20 per cent higher than a single fin. To reduce the interference drag of a single fin a Vee-tail or butterfly tail is used which gives the combined effect of the horizontal stab and the vertical fin. An equation is given for the estimate of the dihedral angle of the tail. Care should be taken not to make the angle higher than 15 degrees.

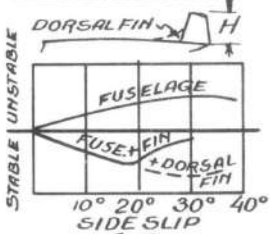
Radio controlled models use a portion of the fin for a rudder for performing most of the flying patterns a model can do with the addition of elevator control. The rudder should be aerodynamically and statically weight balanced. The rudder should be an extension of a symmetrical section. Many rudders have failed to overcome the gyro forces of an over-running propeller in a straight or spiral dive.

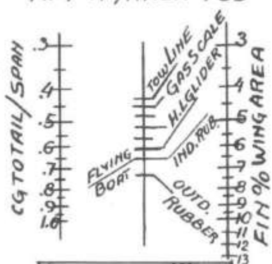
FIN AND RUDDER.

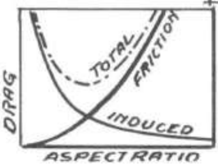
AREAS OF EQUIVALENT DIRECTIONAL EFFICIENCY

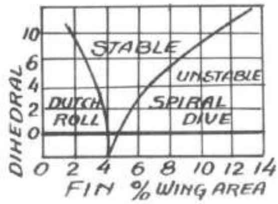
$A.R = H^2/AREA \times 1.55$

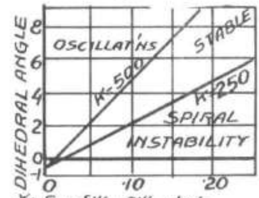




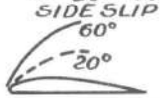








$\gamma + .5 = K^{SV}/_{S} \times {}^{a\gamma}/_{a} \times {}^{Lt}/_{b}$



BLANKETING AR 4

FUSELAGE EFFECT

$$FIN\ SQ.IN. = \frac{.60\,FA \times LENGTH_{IN}}{DIST._{IN}\ TO\ FIN\ \ AR^{1/3}_{FIN}}$$

C.G.–1/3L

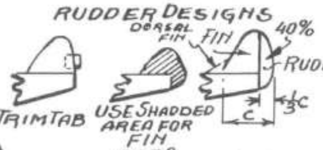
FA = CROSS-SECTION IN FUSE IN SQUARE INCHES

THE EFFECT OF AN OFF-SET FIN IS ALSO OBTAINED BY CAMBERED SECTION.

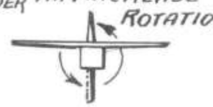
PROPELLER EFFECT ON FIN AREA

$$AREA = \frac{.20 D^2_{IN} CG\ TO\ PROP}{AR^{1/3}\ CG\ TO\ FIN}$$

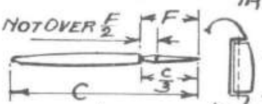
RUDDER DESIGNS



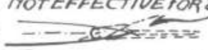
ADDING SUB FIN MAY INCREASE ROTATION.



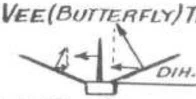
REDUCING HINGE MOMENT

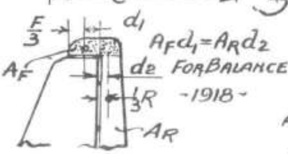


NOT EFFECTIVE FOR 2°-3°



VEE (BUTTERFLY) TAIL

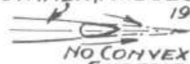




$A_F d_1 = A_R d_2$ FOR BALANCE

-1918-

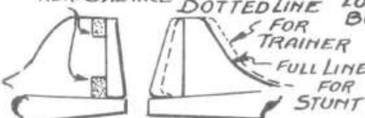
SYMMETRICAL SECTION 1928



NO CONVEX SURFACE ON MOVABLE CONTROLS

VEE TAIL AVANTAGES WEIGHT SAVING, LESS CORNER INTERFERENCE LOWER BLANKETING BUT MORE PROPWASH.

AER. BALANCE



STATIC WEIGHT BALANCE

R/C MODELS

BUTTERFLY TAIL TO GIVE EFFECT OF FIN AND STAB.

$$DIH.^\circ = 30\left(\frac{AREA_F}{AREA_T}\right)^{1/2}\left(\frac{A.R_F}{AR_T}\right)^{1/6}$$

F = FIN, T = STAB

DIH° NOT OVER 15°

WEIGHT TO BALANCE RUDDER WEIGHT

RUDDER AND TAIL WHEEL CONNECTED TO PREVENT GROUND LOOP ON TAKE-OFF

T.R. 803

THINNING REDUCES HINGE MOMENT.

FIN AREA FOR YAWED WING (NO FUSE, NO PROP)

YAWING MOMENT OF FIN

$= A_F\, q\, \beta \frac{dC_L}{d\beta} \times DIST$

$= A_F\, q\, \beta\, .039\, AR_F^{1/3}\, DIST.$

COMBINING WITH YAWING MOMENT OF WING

FIN EFF. TESTED. T.R. 437

EFFECT OF A.R. AND AREA AT 0° PITCH. BOX, ROUND AND CABIN IS STANDARD, AT 25° PITCH LOW A.R. FIN ON CABIN BLANKETED.

EFFECT OF WHEELS

$$A_{FIN} = \frac{DIA^2\ CG.\ To\ WH}{AR_F^{1/3}\ CG.\ To\ FIN.}$$

EFFECT OF EXPOSED ENGINE

$$A_{FIN} = \frac{A_{ENG} \times DISTANCE}{AR^{1/3}_{FIN}\ \ DIST._{FIN}}$$

SEE PAGE

$$\frac{A_{FIN}}{A_{WING}} = \frac{SPAN\,[a^2_w + \Delta a^2/4]}{DIST.\ 160\,B\,AR_F^{1/3}\,AR_W^{1/3}}$$

SIMPLIFIED

$$\frac{A_{FIN}}{A_{WING}} = .60\,\frac{SPAN}{DIST \times AR_F^{1/3}\,AR_W^{1/3}}$$

REF: TM 278, TM 685, TR 360

CENTRIFUGAL FORCE. A model in a curved flight path is subject to a centrifugal force (c.f.), which is considered to be acting through the c.g. In a steady circular flight the c.f. is usually balanced by aerodynamic forces or by the tension of the lines held by the modeler. Equating the weight with the c.f. the speed is eliminated and the radius of a loop is found to be a function of wing loading and the CL. Therefore the smallest diameter of a loop will be performed if the model flies with the angle of CLmax. The power however must be sufficient to generate a loop. The centrifugal force is reduced on top of the loop by the weight of the model and increased at the bottom.

Theoretically the model should follow a descending oval. Many times however free flight models made ascending loops due to the increase of the propeller thrust in the upward flight. The propeller thrust may finish a loop without the aid of the c.f.

Equations and a nomogram for estimating the c.f. are given. One is to give approximate estimate for the diameter of the steel wires used in control line flights. The stress is assumed to be 150,000 pounds p.sq.in. if the weight, the length of the line and the velocity of the model are known. Some of the wires, the genuine piano wire may go as high as 230,000 p.s.i.

GYROSCOPIC FORCES- The force created by the angular movement of the axis of rotating bodies, such as propellers, engines and the model itself, is the gyroscopic moment, which acts 90 degrees to its forced displacement in the direction of the rotation. If the displacing force is steady the axis of the masses will follow the surface of a cone, its velocity is the precession and the rate it may straighten out is the nutation.

To visualize the direction the rotating axis will deflect, a finger is pressed against a rotating pencil, which moves into the direction of the gyroscopic moment. Thus the gyro force of a right-handed propeller makes the model climb in a left turn and dive in a right-hand turn. The model turns to the left if its nose is turned downward and turns to the right is its nose is turned upward. The gyro force is one of the MAIN force affecting the flying characteristices of propeller driven models in a climb.

Two general equations are given for estimating gyro moments. A nomogram will aid in finding the gyro force of an average weight propeller and of a 1/32 in. thick steel plate. For a heavier plate the gyro force is in direct proportion to its thickness. For dural discs use only 1/3 of the forces estimated.

In order to use the equations the moment of inertia-I-or the weight of the body with its radius of gyration about its axis of rotation must be known. The radius of gyration can be found by the bifilar suspension method. The time indicated in the equation is for one full free axial, back-and-forth swing in seconds. The length of the lines should be at least 10 times their spacing. The moment of inertia should be carefully investigated for radio control models. The I may be also called the weight distribution.

Testing a 5 ft. propeller-driven helicopter, the gyroscopic moment was mistaken for the torque of the electric motor and the power estimated to be 2 h.p., which was impossible because it was only a 1/8 h.p. motor. Relocating the motor the gyro forces were eliminated from the torque meter and the right answer was obtained.

DIHEDRAL. A banked straight wing will sideslip due to the resultant of lift and weight of the model. The resultant will, within limits, not move and the wing will keep on diving. Vertical surfaces placed on top of the wing will create small righting moments not sufficient for safe flying.

Greater moments are obtained by giving the wing a lateral angle, called a dihedral. In a sideslip a dihedraled wing increases its angle of attack on the downwing and decreases it on the upwing. Two equations and a nomogram give the increase. The dihedral may be 100% or less of the semispan. A numerical example shows the advantages of a tip dihedral and the poor showing of a vertical surface equal to the projected wing. The comparison is for one wing only. The height of a vertical surface must be at least 40% of the span to give the same moment as a wing with a 10 degree dihedral. Square tips give greater moments than round tips. Equal moments are obtained by a 100% semi-span with a 10 degree dihedral, a 50% semi-span with a 15 degree dihedral and a 25% semi-span witha 25 degree dihedral. The dihedral of a 25% semispan may be increased to 45 degree for greater moments. To estimate the moments the distance must be taken to the c.g. of the model and subsequently the higher the wing the greater the righting moments. Normal dihedral will not increase the gliding angle but it will slightly decrease the lift.

Dihedral setting of the wing in cooperation with the fin gives lateral stability against gusts, torque and gyroscopic moments. Equations and a nomogram give dihedral angles for gas, rubber and gliders. No safe dihedral can be given for rubber models to counteract the maximum torque of a rubber motor weighing over 50% of the total weight of the model. A wing with a zero dihedral has actually a 2 degree effective slope. At very large angles of attack the zero dihedral wing will give greater moments than the wing with dihedral. A negative dihedral, cahedral may produce the so elusive lateral stability of flying wings.

Tests on a radio control model gave perfect smooth flying with a 4 degree dihedral. Increasing to 6 degrees, stunting with rudder-only became easier. The amount of dihedral, fin area and c.g. position will be specific on any design category.

REGULATIONS FOR CONTEST RULES NOT OFFICIAL, WRITE FOR OFFICIAL MODEL AIRCRAFT REGULATIONS TO A.M.A.

GLIDERS: INDOOR HAND LAUNCHED-H.L. MAX AREA 100 SQ.IN. TAIL AREA MAX. 50% OF WING AREA. NINE FLIGHTS, ALL OFFICIAL. HIGHEST SCORES 3 GLIDERS. 6 FT MAX. LAUNCHING HEIGHT.

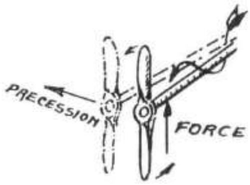
OUTDOOR HANDLAUNCHED 30 TO 130 SQ.IN. STAB 50% MAX. FLYING WING 2/3 OF AREA FOR CLASSIFICATION. 9 FLIGHTS TOTAL OF BEST 3.

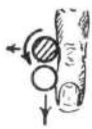
TOWLINE: LIMITED COMBINED AREA 350 SQ.IN. MAX., 10 OZ. MIN. 1-GLIDER - 40 SEC. OFF., 6 FLIGHTS, TOTAL OF BEST 3. 6 MIN. LIMIT. 328 FT. LINE 39" ELASTIC MATERIAL. 12" DIST. STREAMER

NORDIC, TOTAL SURFACE 495.9 TO 526.9 SQ.IN. MIN. TOTAL WEIGHT 14.46 OZ. LINE 164 FT.

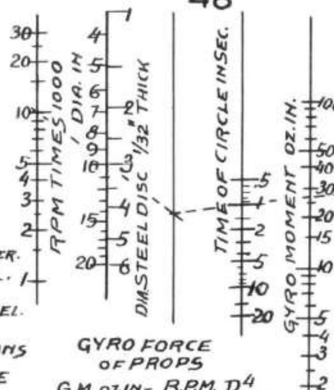
FOR OFFICIAL A-M.A MODEL AIRCRAFT REGULATIONS WRITE TO ACADEMY of MODEL AERONAUTICS - 1025 CONNECTICUT AVE. N.W WASHINGTON 6, D.C.

GYROSCOPIC FORCE

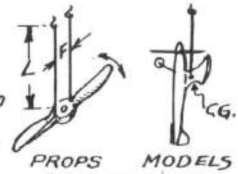




GYRO FORCE
$M = I \times \omega_1 \times \omega_2$
I MASS MOM. of INER.
ω_1 ANGULAR VEL. OF PROP
ω_2 = ANGULAR VEL. of SHAFT
ω_1, ω_2 IN RADIANS



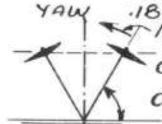
GYRO FORCE OF PROPS
$G.M\ OZ\cdot IN = \frac{R.P.M.\ D^4}{T\ 3{,}500{,}000}$
D - INCH, T TIME SEC.



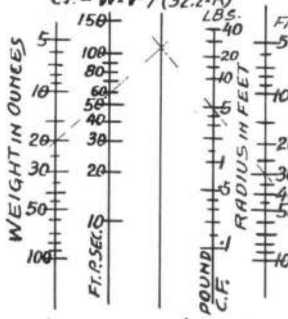
PROPS MODELS
RAD. OF GYRATION
$K = 1.58\, F t/\sqrt{L}$
F, L, INCHES
t - TIME FULL SWING, SEC.
PROP
$WEIGHT = D_{IN}^2 \times .004$ OZ.
K = .17 - .20 D

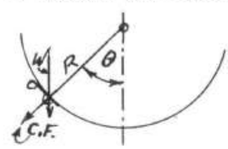
MODELS K	PYLON GAS	RUBBER
PITCH	.29	.20 L.
ROLL	.11	.26 SP.
YAW	.17	.32 SP.

AIRPLANES K	TN 780
PITCH	.20 - .22 L
ROLL	.125 - .150 SP.
YAW	.18 SPAN.



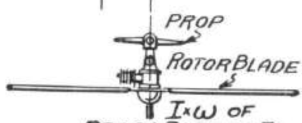
CENTRIFUGAL FORCE
$CF = W \times V^2/(32.2 \times R)$





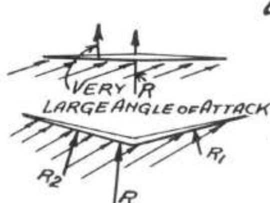
COMBINE EQU'NS
C.F. + LIFT. LOOP DIA.
$DIA = \frac{470\ Woz}{A_{SQ.IN.} \times CL}$ IN FT
DIA OF WIRE (2 WIRES)
$d(\frac{1}{1000})'' = .406 V_{M.P.H} \sqrt{\frac{W}{R_{FT}}}$
W IN POUNDS

$LIFT = C.F + W \cos\theta$
C.F. P. POUND = $V^2/15R$, V - M.P.H.



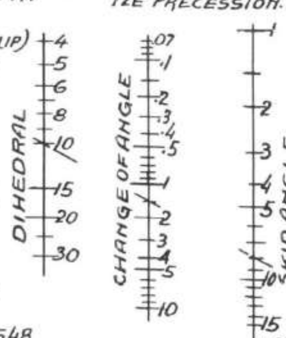
$I \times \omega$ OF PROP + BLADES TO BE EQUAL TO NEUTRALIZE PRECESSION.

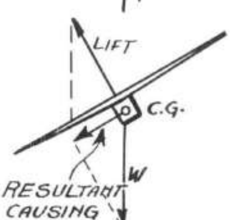
DIHEDRAL - Γ GAMMA



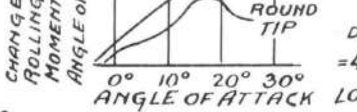
TAN (CHANGE) = SIN (DIH) SIN (SLIP)
$CHANGE° = \frac{DIHED. \times SIDESLIP}{57.3}$

DIHEDRAL	10	30	90
LENGTH	46"	16"	8"
CHANGE ANGLE	1.71	4.95	10
DISTANCE CP-CG	23	35	4
TOTAL MOMENT	1809	2772	320





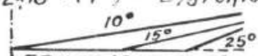
RESULTANT CAUSING SIDESLIP IN A BANK



DIHEDRAL ANGLE
$= 450 \sqrt{\frac{AR_E}{AR_W}} \cdot \frac{TAIL\ DIS}{SPAN} \cdot \frac{AREA_F}{AREA_W}$
LOSS OF EFFECTIVE AREA
$-2 \times 10^{-4} (\Gamma°) - 2\%$ FOR 10°

15° SIDESLIP 10% LOSS CL
25% INCREASE CD
30° SIDESLIP
28% LOSS OF LIFT
100% INCREASE OF DRAG

APPROX DIHEDRAL ANGLE FOR GAS MODELS
$DIHED° = \frac{80{,}000 \times CU.IN.}{W_{OZ.} \cdot SPAN_{IN.}}$
FOR RUBBER MODELS
$DIHED° = \frac{10 \times N_{1/8}^{3/2}}{W_{OZ.} \cdot SPAN_{IN.}}$



EQUAL MOMENTS
CHANGE OF DIHEDRAL

T.M. 502 TM-730 TR 112

TAKE-OFF DISTANCE. Some models will take off without any rolling on the ground, others need a long starting run. The distance. depends on the design,wing loading power loading, thrust at the take-off and the resulting velocity of the slipstream.

Propellers of high powered free-flight models usually have a thrust greater than the weight of the model, which may lift the model off the ground, if the angle of the wing and the angle of the thrustline are over 15 degrees. This direct take-off will depend on the streamlined fuselage and on the cowled engine, because the drag of the parts affected by the slipstream reduces the resulting thrust, which is not sufficient for a non-run take-off. The take-off distance from a smooth surface is approximated by an equaticn, which includes the static thrust of the propeller.

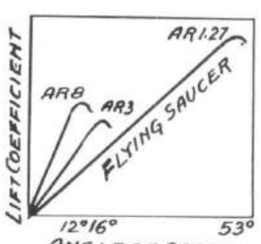
CLIMB PATTERN A well designed and adjusted model will climb to the left, straight or to the right, and after the power is cut will glide to the left. The reason for the left glide is to take advantage of the ever present thermals. The highest altitudes are reached with models climbing with a slow spiral.

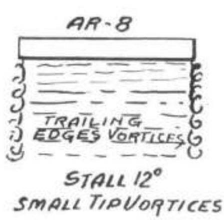
The climb pattern depends mainly on the distance of the c.g. to the thrustline and also on the distance of the c.g. to the wing chord. In a steep climb the aerodynamic lift is reduced and two forces will decide on the pattern, the torque moment and the gyroscopic moment of the fast running propeller. The final pattern is fixed by the air forces of the rotating model about a vertical axis. The rotating model creates additional gyro forces which realy complicates the problem.

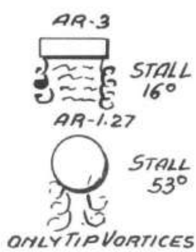
All models are set for a minimum sinking speed in a glide. This setting has a large difference of the zero-lift lines, which gives the tail a negative angle of attack in a steep climb. This unwanted drag can be reduced by climbing in a slow spiral, which may even place the tail into a zero angle of attack and give the model a maximum climbing velocity.

A model with a shoulder or top wing has its thrustline close to the c.g. If the model is not adjusted, it will, after take-off, due to the torque moment, bank to the left.. The generated gyro forces will press the nose upard, circling to the left. The c.f. may flip over the model to the left side and dive in a tight spiral with the aid of the torque and gyro forces.

To adjust a model for a climb to the left in order to take advantage of the nose-up tendencyof the gyro forces a wash-in is given to the left wing and to meutralize the torque a right turn is given to the fin. A slight left direction to the thrustline forces the model into a left climb. Left turn adjustment of pylon jobs is a chalenge to many old-timers.







A model with a high or pylon wing has the thrustline at a distance from the c.g. and from the chord of the wing. If the model takes off in a straight line it will turn under the influence of the torque to the left. This turning produces precession that forces the model into a circling to the right, and the same time adds gyro forces to the counter rotating model. This rotation of the model is retarded by the drag of the wing.

The adjustment is usually a right-hand thrust and a left-hand fin. Down thrust may increase spiraling. Some designers suggest a slight wash-in of the left wing.

In a spiral of a high-thrust pylon job the propeller axis describes a cone, with the c.g. and the wing on the inside. This conical rotation may increase to such an extend that the revolutions of the prop are reduced and the model stops climbing. The wing rotating on the inside of the cone gives a lower drag than rotating it on the outside. Giving the prop axis of a stick model sufficient downthrust to pass over the c.g., the model will climb with a slow spiral, with the c.g. on the inside of the cone, and the wing and the prop axis on the outside.

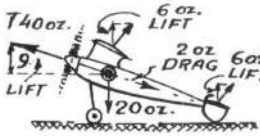
Sketches show the climb of models with pylon and high wings, Also sketches give the positions of a wing rotating on the inside and on the outside. Two equations give the expected revolutions of a model with the wing on the inside.

The upward spiraling is also affected by the relative position of the propeller axis and the minimum inertia axis. The lower the angle difference, the higher the number of counter revolutions. The aerodynamic qualities of a pylon job has small effect on the climbing velocity. The pylon model flies with the thrust and the gyro forces of the prop.

It is apparent that the distance of the thrustline to the c.g, decides on the flight pattern. The torque predominates on models with a small distance to the c.g., and with a great distance the gyro forces take over. To take advantage of the helpful gyro-Forces and balance them partly by the torque, a model can be designed by placing the wing on a low pylon and the rudder below the fuselage in order to have the axis of minimum moment of inertia is at angle with the thrustline. This model may not need thrust adjustment, but must have a left-hand fin setting. The effect of the gyro forces of high speed propellers is the reason for reducing the r.p.m. on radio controlled models.

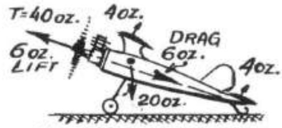
UP-GUSTS. Up-gusts will influence the stability of models , because they do not reach the complete model at the same time. They reach the wing ahead of the stabilizer and if the c.g. is behind the air resultant the model will balloon, and if the c.g. is ahead of the resultant the gusts will decrease the angle of attack and in some designs the model will fly on an even level with a slight oscillation, the model is dynamically stable.

TAKE-OFF, NO RUN



STREAMLINED COWLED ENGINE

TAKE-OFF, WITH RUN

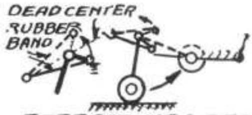


NO STREAMLINING EXPOSED ENGINE

MAX. WING + POWER LOADING RELATION

$$CU.IN. \cong .35 \frac{A^{1/2}}{W^{3/2}}$$

TO BE ABLE TO R.O.G FROM SMOOTH SURFACE

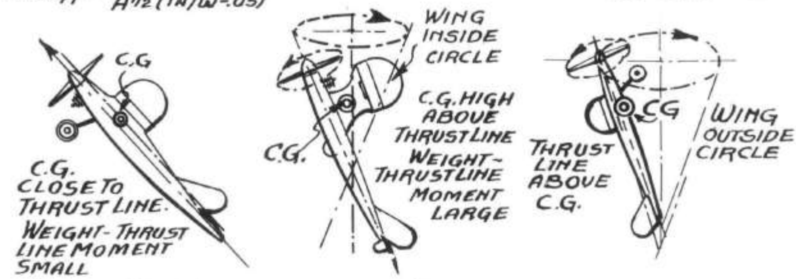


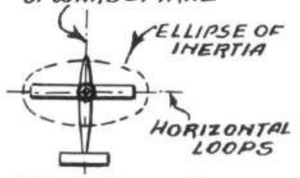
RETRACTING L. GEAR BY TOGGLE DESIGN

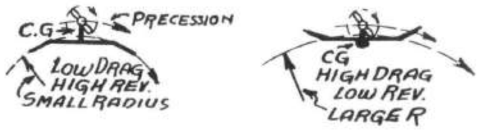
TAKE-OFF DISTANCE OF HEAVILY LOADED MODELS SMOOTH HARD SURFACE

$$DIST._{FT} = \frac{55\,W^{1/2}}{A^{1/2}\,(TH/W - .05)}$$

775 SQ.IN. 82 OZ. .23 CU.IN. DISTANCE 120~150 FT.
504 SQ.IN. 30 OZ. .05 CU.IN. DISTANCE 30-35 FT.







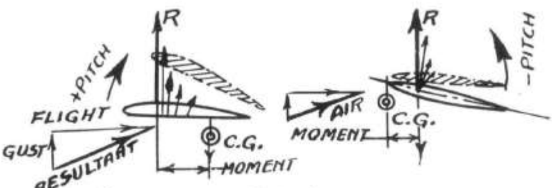
R.P.S. OF MODEL FROM COUNTER TORQUE OF PROP

$$R.P.S. = \frac{18{,}500 \times CU.IN^{1/2}}{SPAN^{3/2} \times AREA^{1/2}}$$

$C_D = 1.0$, SPAN-IN. AREA SQ.IN

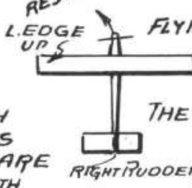
$$R.P.S. = \frac{200\,N^{3/4}}{SPAN^{3/2}\,AREA^{1/2}}$$

N = EQUIVALENT STRANDS OF 1/8 x 1/30 RUBBER



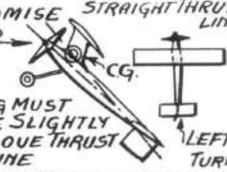
FLYING TO THE LEFT

D.K. FOOTE.



WARP LEFT SIDE OF WING W. LEADING EDGE UP (WASH-IN) LEFT THRUST, RIGHT RUDDER

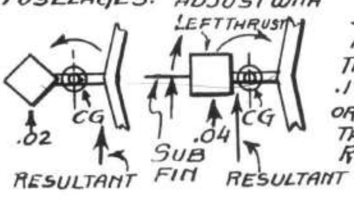
THE COMPROMISE



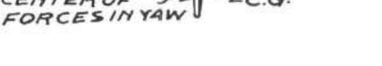
CG MUST BE SLIGHTLY ABOVE THRUST LINE

GYRO MOMENT EQUAL TO COUNTER FORCE OF SUB-RUDDER. SLOW TURNING, CLIMBING TO THE RIGHT.

ADJUSTING MODELS WITH STREAMLINED FUSELAGES NOT AS EASY AS WITH SQUARE FUSELAGES. ADJUST WITH



SOME GAS JOBS FLY WITH A .15 TO THE RIGHT, WITH A .19 TO THE LEFT OR WITH A 9-6 TO THE LEFT, 9-4 TO THE RIGHT OR UP.



TAIL SETTING SUFFICIENT FOR INDOOR FLIGHTS MANY GLIDERS WILL SPIRAL DOWN FLYING OUTDOOR IN LIGHT THERMALS.

SPIRAL STABILITY. Models may circle on the end of a line or in a free-flight attitude. Control-line models require only horizontal stability, which is often replaced by the dexterity of the operator. Spiral stability covers a large section of the general stability problem.

Free-flight models will rotate about an axis through the c.g. which has the lowest mass-moment x distance value. The first force acting on a model starting a circling flight is the c.f., which causes the model to skid and consequently increase the lift on the outer wing. This places the model into a bank, followed by downward spiraling, which would become tighter if a rolling moment started on the inside wing is not adequate to keep the model on an even keel.

In a steady circling flight the outside wing has a smaller angle and a higher velocity than that of the inner wing. In a balanced condition the resultant air forces should meet at the line of symmetry passing through the c.g. Also the resultant of the c.f. and the gravitational forces should balance the air forces. This is an ideal balance of forces in which the fin moment may be equal to the propeller and fuselage moments. The torque and gyro moments need additional corrective measure to balance their effect. In an ideal circling flight, known to pilots of light planes, the yawing moment of the fin is equal to the combined moments of the propeller, the fuselage and the wing.

A model stalls in a straight glide and flies in a large circle to the left and with power on, however it stalls flying to the right. If moving the thrustline closer to the c.g. (downthrust) and the action of the slipstream cannot be made to correct the stall, the model should be adjusted to fly in a smaller circle. A diagram shows the moments of a model with the thrustline below the c.g. causing a positive (stalling) moment of 12 oz.in. The effective tail setting is minus 6 degrees when flying in a straight line. Adjusting the model to fly in a 60 ft. circle the stabilizer will add a 12 oz.in. negative moment, which was obtained by the change of the tailangle from minus 6 degrees to minus 4 degrees. In case smaller circles are required for stability the reserve power and the vertical component of the lift forces must be sufficient for a banked flight.

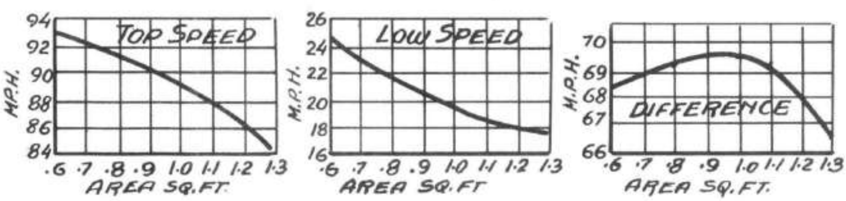
Another case of spiral stability occurs when a model flies into a rising air column. This rising air (thermal) moves upward to the right in a slow spiral. A model reaching this column of circling air will reduce its tail setting by increasing the lift on the tail, which makes the model dip immediately. This diving is the result of the movement of the vectors toward the trailing edge due to the reduction of the tail setting. This places the c.g. in balance with a resultant of a smaller angle of attack of the tail, which increases the gliding and the sinking speeds. Two diagrams show the two position of balance. After the model reaches the thermal and the diameter of the rising air is suitable for a balance it may stay with the riser. The high peripheral velocity of the riser will give the model a large banking angle.

Too high directional, weathercock stability will result in a spiral instability. In crease of the fin area should follow with an increase of the dihedral angle. The increase of the dihedral is of less influence on directional stability than the increase of the fin area. Spiral instability is easily observed by longer spirals in the direction of the disturbance. Due to the everpresent thermals some models are stable to the left and others flying to the right.

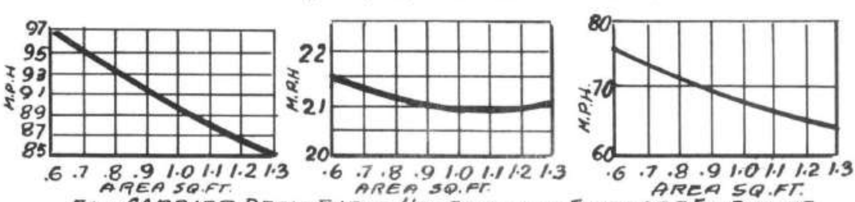
In circling the ratio of rolling moment and yawing moment is different for each kind of design. The rolling moment is proportional to the lift coefficient and the yawing moment to the change of induced drag. The greater the ratio the faster the recovery. The ratic shows that the recovery increases with the increase of the aspect ratio of the wing. This however does not give the real condition of dynamic recovery, which must use the function of mass distribution, the moment of inertia of the wing in the equation. The moment of inertia is in direct relation to the aspect ratio, and the rolling moment is affected in a greater amount than the yawing moment is. therefore dividing the ratio by the aspect ratio an equation is obtained that favors lower aspect ratio for recovery, a characteristic observed by pilots of racing planes. The equation also indicates that the recovery is slower with increase of the angle of attack.

A fast diving, spiraling model is the outcome of an oversized fin, a low dihedral or a low tail setting. This kind of diving is observed when flying indoor gliders in the outdoor. This spiraling does not signify spinning, which is the descend of a model at an angle past the stall angle on the inside wing. Diagrams shows the inclination to spinning of a rectangular wing and a wing with rounded tips. Another diagram give the range of the angles of attack that tend to induce spinning of a monoplane and a biplane.

CONTROLLINE ~ 3LINES, 60 FT. 1 LINE FOR ENGINE.
SAME FUSELAGE AND .35 CU.IN. ENGINE, AREA VARIED.

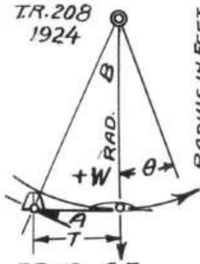


SAME .35 CU.IN. ENGINE, MODEL SCALED DOWN.

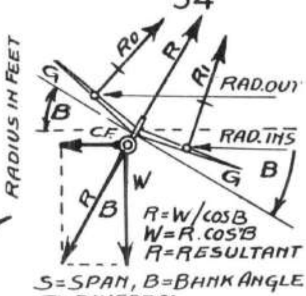


FOR CARRIER DECK EVENT. USE SMALLEST FUSELAGE FOR ENGINE AND MINIMUM AREA.

T.R.208
1924

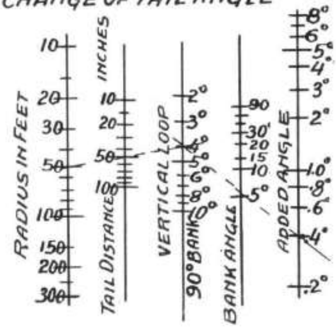


T.R.112



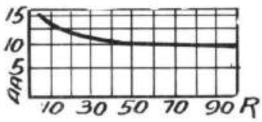
S=SPAN, B=BANK ANGLE
G=DIHEDRAL

CHANGE OF TAIL ANGLE

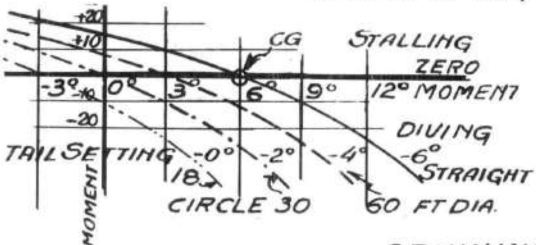


$$AA-\sin(G+B)\frac{T}{RAD}\left[RAD+\frac{S}{4}\cos(B+G)\right]^2=$$
$$AA+\sin(G-B)\frac{T}{RAD}\left[RAD-\cos(B-G)\right]^2$$

TANA = T/R × SIN BANK

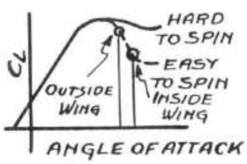


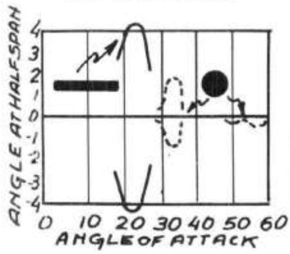
LARGE ANGLE OF ATTACK SLOW GLIDE
SMALL ANGLE OF ATTACK FAST GLIDE



MODEL WITH THRUST LINE BELOW THE C.G. WILL LOOP IF NOT BALANCED.
CIRCLING WILL CREATE NEGATIVE-DIVING MOMENT
MODEL STALLING IN GLIDE SHOULD CIRCLE FOR EVEN GLIDE.

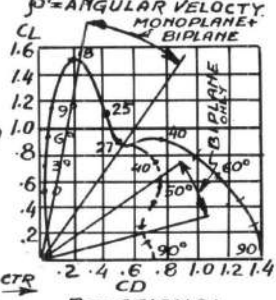
SPINNING





AUTOROTATION ANGLE AT HALF SPAN $= \frac{\rho' \, SPAN}{2 \times VELOCITY}$

ρ' = ANGULAR VELOCTY.



ROLLING M'T BY YAWING M'T

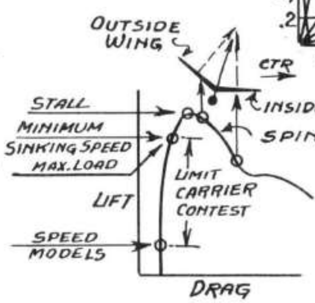
SEE PAGE

$$\frac{ROLL'G\,M'T}{YAW'G\,M'T}=\frac{80\,\Delta\alpha\,AR_w^{2/3}}{[\alpha_w^2+\Delta\alpha^2/4]}$$

FOR DYNAMIC STABILITY DIVIDE BY I = AR FOR SAME AREA

$$RATIO=\frac{80\,\Delta\alpha}{[\alpha_w^2+\Delta\alpha^2/4]\,AR_w^{1/3}}\ ?$$

$\Delta\alpha = BG/57.3$. FOR ALL ANGLES $\tan(\Delta\alpha)=\sin B.\sin G$



ROTATIONAL INSTABILITY.
T.N.633

CENTER OF LATERAL AREA.- Years ago the center or centroid of the lateral area of a model was assumed to be the ultimate reason for its spiral stability if the line through the c.g. and the c.l.a. had a certain angular position with the thrustline . The c.l.a. is the center of the sideview of a model with twice the wing and wheel projections. The displacement axis about which the model should roll is the line drawn through the c.g. and the c.l.a. No fixed rules were given for the c.l.a. as long it was behind the c.g. location. It was suggested to use the c.l.a. for estimating the size of the fin, without the consideration of aspect ratic. Some radio model designers suggest to apply the c.l.a. to obtain a smooth flying. Others again go as far as to deny any effect by placing the fuselage area above or below the c.g. It should be mentioned that any change will affect flying characteristics of models. For clearer perception of the c.l.a. each major unit of a model will be compared with its known values.

A fin with an aspect ratio of 3 is placed vertically gives twice the moment if placed horizontally. The sideview of a stopped propeller will give the right value; a running propeller however will give higher values, because its effect is assumed to be equal to a surface with an aspect ratio of 8, as suggested by the N.A.C.A. Tests on model propellers indicated that the p/D ratio of the propeller influences the moment about the c.g. The effect of the projected area of a wing can be given by an example. The projection of a wing with a 6 degree dihedral has a 10 per cent height of half the span. Assuming that the distance to the c.g. is 1/2 of the chord. The wing woth an aspect ratio of 8 will have its incease at the 2-chord distance and about the same distance to the c.g. Assuming that the aspect ratio of the projected area is 3, then the moments of the wing will be 6 times the one given by the projection. This difference will be greater if the wing is built with tip dihedral.

The projected area of a landing gear can be assumed to follow the c.l.a. value. One unit that may give values in the reverse is an exposed engine if it is compared with a cowled engine. The exposed engine has a CD of 1.0 and if the engine is cowled the side area increases three times the CD however is only .25. This indicates that the error amy be 12 times the assumed one. A similar divergence will be found with pylons, which usually have symmetrical cross-sections.

A big error is made by applying the c.l.a. to fuselages, which have their c.l.a. at the 40-45 per cent distances and their c.p. at the 25 to 30 per cent distances. Two basic designs with equivalent side areas, a square and a round fuselage are shown with their forces at a 10 degree angle. The force for the suqare fuselage is three times higher than one for a round fuselage.

The side area and its c.l.a. has no relation to the assumed rolling moments created in a skidding of a model and its rotating axis about the c.g. and c.l.a. line. All bodies rotate about the c.g. when displaced. There is however an axis about which it will rotate with the least force. The force resisting rotation depends on the moment of inertia (I) of the body. The moment of inertia with respect to a given axis is the sum of the weights or masses times the square of their distances from the axis. Plotting the moment of inertias on various axes the ellipse of inertia is obtained. The axis of the minimum inertia is vertical to the axis of maximum inertia.

Applying the mass or weight distribution to models many observations made by designers and builders will explain some of their flying characteristics. Radio equipment may be located at a disadvantageous place which may make it unstable, because the increase of the inertia required greater control moments.

A certain inertia is required for each free-flight design to slowdown violent angular displacement by the prevailing outdoor air disturbances. Similar to the birds and boats for bad weather and for higher wind speeds the loading per sq.ft. of the supporting area is higher. The moment of inertia of a model can be determined by the bifilar suspension or other methods suggested by books on aerodynamics. In case the length and the spacing on the bifilar suspension is kept the same, the time of the oscillations will give comparative values. The moment of inertia may be estimated mathematically. The value of the inertia will be too low because the air moving with the model when tested is from 5 to 10 per cent higher.

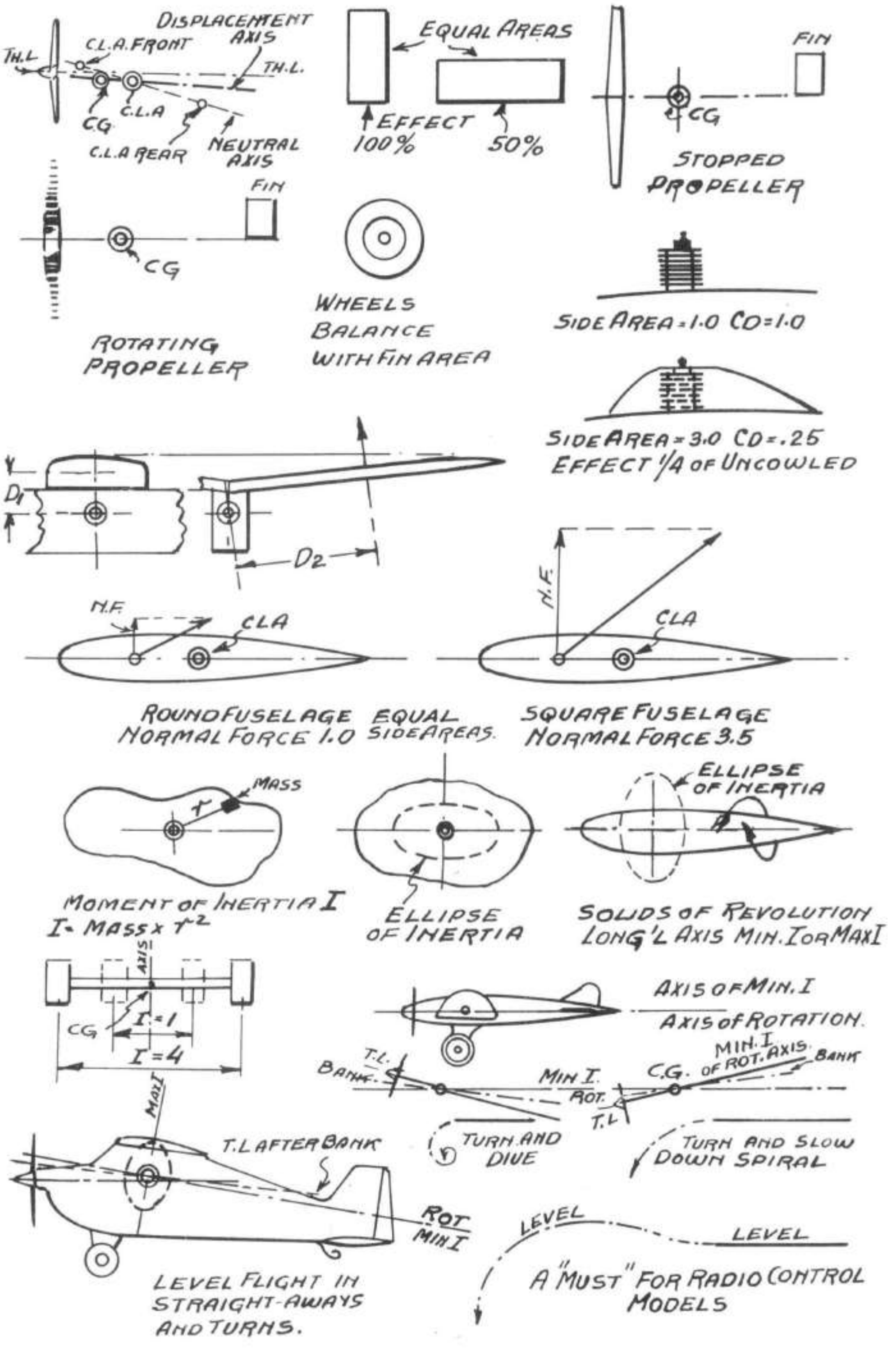
The use of the axis of the minimum inertia instead the neutral and the displacement axes mentioned with the c.l.a. theories (Ch. H. Grant 1941) will follow closer to the mass dynamics than the area distribution. It is obvious that the replacement of the areas by the masses, or weights would have given a closer stability estimate than the one obtained by the c.l.a.

The angular setting of the axis of minimum inertia with the thrust line will give various flight patterns especially a model going into a turn when operated by a radio controlled rudder. The inertia axis may point downward, or be inline or point upward from the thrustline.

In case the axis of minimum moment of inertia is in line with the thrustline and the rudder is operated, the model will skid and the dihedral effect a bank without changing the angle of the thrustline.and if the circling will not increase the angle of attack the model will spiral downward. In case the axis points downward and the rudder is operated the model will turn, bank and the thrustline will point downward and the model will go into a fast spiral dive. In case the axis of inertia points upward and the rudder is operated the model will skid outward, bank and the thrustline pointing outward adds to the skid. The angle of attack increases which balances the lift for a level circling flight. The return to a straight flight will be without violent maneuvers. The advantage of the inertia axis pointing upward is observed by the performances of models with pylon wings and models with their fins below the fuselage. The performances will also be influenced by the location of the c.g., the propeller and the r.p.m. of the engine.

In order to produce a min. I axis that has a positive angle with the thrustline, the units or weights must be distributed about that axis. This may be evolved by moving only the wing higher or placing the tail or the rudder below the fuselage.

CENTER OF LATERAL AREA C.L.A. COUNTER TO MOMENT OF INERTIA



DURATION. Contests for free-flight models are conducted for maximum duration, with specified time limit accepted for each flight. In a tie an additional flight is made without limit to fix the winner. Clipper cargo planes are designed for a maximum time of 40 seconds, which is accepted as an official flight.

The duration depends on the power available and the power required to fly horizontally. This minimum power is also equal to the weight times the minimum sinking speed. The difference between the power available and the min. power is utilized for the climb. For each value of reserve power and weight a certain climbing angle will give a maximum climb distance. A diagram shows the climbing velocities for various angles and for varying thrusts. After the power is shut off the subsequent gliding time is added to the total duration.

If the power delivery is limited, as for gas and jet engines the fastest climb and the slowest sinking speed will decide the duration of the flight. A diagram shows the power required for a model and the power delivered by an 8-4 and 8-6 propellers. The difference indicates that the apparent flying speed does not give the highest climbing speed. The greatest difference of the powers gives the fastest climbing speed.

The duration for a limited engine run and the model climbing on a medium angle is given by a simple equation, which is the ratio of power available and power required for horizontal flight. By interpolation and assuming that the power delivery of an engine is proportional to the cu.in. displacement, an equation is formulated, which uses the function of the area and the weight of a model in estimating the duration.

In some contests the area is not limited and the maximum duration will depend on the ability to build extra light but strong wings. Because the weight of a wing increases with the 3/2 power of the area, there is a limit to the size of a wing a given engine is able to take off. An equation shows the relation of the area for maximum cargo and the displacement of the engine.

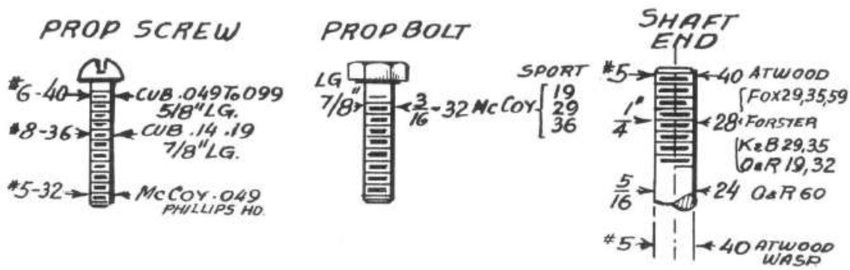
The thrust of the propeller driven by gas engine is usually higher than the total weight of regular free flight models and a steep climb will give maximum duration. The high thrust is divided between lifting the weight and overcoming the drag of the model. The steep climb eliminates the induced drag and by adjusting for circling flight the negative angle of the tail is reduced, which consequently reduces the total drag. The model will climb with the lowest drag possible.

To estimate the climbing speed, the net horse power delivered by the propeller is assumed to be .60×cu.in., the CD equal to the CDpar and the CDmin of the wing. Use the diagram for direct estimate of the climbing speed. For medium climb the ratio of the available power and the required power multiplied by the engine run should be the duration. In estimating the sinking speed, the drag of the stopped propeller should be included. An approximate sinking speed may be estimated by an equation, which is based on a lift coefficient of .80 and a gliding angle of 9.

The same procedure is used in estimating jet driven models. Rubber driven models have a problem by themselves, because the torque varies with the number of turns left on the motor. It has always been the contention to reach a thermal with the fastest climbing speed, without any thoughts about efficiency of performances until the contests were conducted at a time when thermals were supposed not to be present. This started the designs of propellers to fly the model horizontally at the end of the motor run. This increased the motor run close to twice of the former time. No theoretical equation has been developed that uses the varying torque of the rubber motor. Equation can be formulated by assuming that the altitude reached is in direct proportion to the weight of the rubber and that the duration is equal to the weight of the rubber divided by the sinking speed corrected with a constant derived from performances. An equation and a nomogram are given for direct estimate of the duration.

Other equations for rubber driven models are given for hydros, ornithopters, helicopters and autogyros. The areas in all rubber driven models are given in the square root of the areas and therefore the rotors are expressed by their diameters.

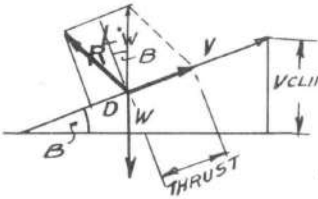
For detailed estimate of performances of rubber driven models the work stored in one ounce of rubber should be taken from 30,000 to 35,000 in.oz. and the efficiency of propellers from 40 to 50 per cent.



CONTROL LINE-SPEED 1/2A~0-.050-35FT, A=.051-.20-52½FT, B=.201-.30-60FT LINE
C~.301-.65 CU.IN.-70FT SIX LAPS 1/2A, 4 LAPS FOR A, 7 LAPS-B. 6 LAPS-C, 3 TIMERS

TEAM RACING 125 SQ.IN MIN. .15-.40 CU.IN., 1.8 CU.IN. TANK 60 FT LINE.
3¾"x2" FUSELAGE. 21-70 LAPS. ENGINE SHUT-OFF 20G PULL.
PRECISION ACROBATICS CLASS A 0-.20, B=.201-.30 C=.301-.65 52½ TO 70 FT.
SCALE CONTROL 4 LB. MAX. 1.25 CU.IN. MAX. 10 LAPS. EXTRA POINT: CON. DETAILS
SCALE FREE FLIGHT .050 CU.IN. 150 oz/CU.IN. MIN.
RADIO FREE FLIGHT (1) RUDDER ONLY, (2) NO RESTRICTIONS.
COMBAT~.35 CU.IN. MAX. 60 FT +-6 IN. STREAMER 8 FT, 2 FT. CORD
U.S. NAVY CARRIER DECK EVENT. 44" MAX. SPAN, A, B, C, D ENG. 60 FT. LINE
TAKE OFF 22 FT-LANDING 22 FT. 100 POINTS NAVY PROTOTYPE. 7 LAPS EACH SPEED DIFF. 3P/MILE
ROUND THE POLE (R.T.P.) 8 oz. MAX.-FOR SPEED RUBBER A-1 oz B-2 oz. (ENGLISH)

DURATION OF FLIGHT



STEEP CLIMB

THRUST = W SIN B + DRAG   DRAG = A CD .0012 V^2

LIFT = W COS B   V SIN B = VCLIMB

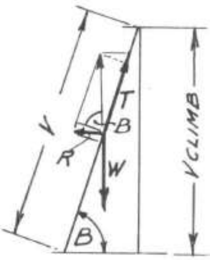
THRUST × V = POWER = W VCLIMB + A CD .0012 V^3

$$\frac{NET\ HP}{W\ oz.} = \frac{V_{CL}}{8800} + \frac{CD_{P} + CD_{M}}{oz/100\ sq\ in} \cdot \frac{V^3}{660.000} \quad \text{STEEP CLIMB}$$

INDUCED DRAG OMITTED. $CD = CD_{PAR} + CD_{MIN}$

NET HP = EFF × HP   EFF = .35–.50

OBTAIN FROM CHART THE CLIMBING SPEED DIVIDE BY SINKING SPEED, ADD 1 AND MULTIPLY BY ENGINE RUN TO GET DURATION



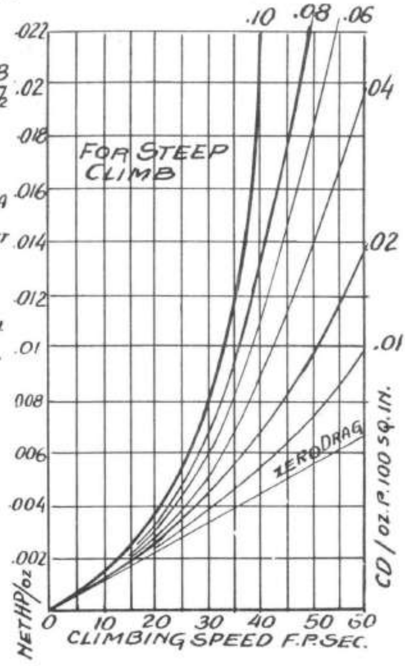
$$V_{SINK} = \frac{W^{1/2}\, CD \cos^{3/2} B}{A^{1/2}\, C_L^{3/2}\, .0012^{1/2}}$$

$W = W_{CON.} + W_{WING}$

$CD_T = (CD + \frac{a}{A}\ 1.28)$

a = PARASITE AREA

ASSUMING CLIMB IS OBTAINED BY LIFT INSTEAD THRUST



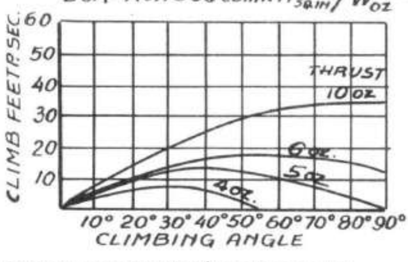
$EFF\,550\,HP = W\frac{D}{L}V + W V_{CLIMB} = W V_{SINK} + W V_{CLIMB}$

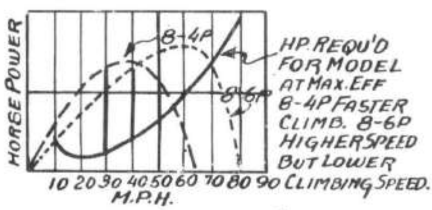
$V_{CL} = EFF.550\,HP/w - V_{SINK}$. $DUR. = V_{CL}/V_{SINK} \times RUN + RUN$.

$DUR. = RUN\ \frac{EFF.550\,HP}{W.\,V_{SINK}}$ $W.V_{SINK}$ = POWER TO FLY ON HORIZONTAL LEVEL

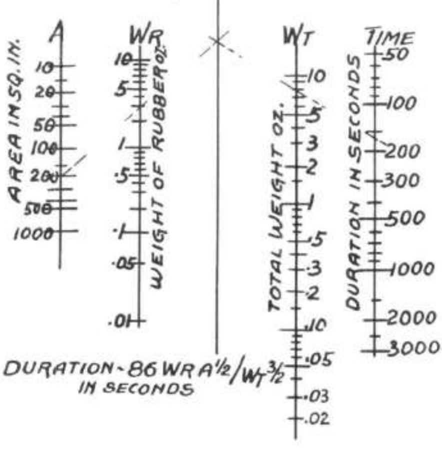
$DUR. = RUN\ \frac{POWER_{AVAIL}}{POWER_{REQU.}}$ W = LB, A = SQ.FT. $C_L = .80$

$DUR = RUN\ 360\ cu.in.\ A^{1/2}_{sq.in}/W_{oz}^{3/2}$





DURATION OF RUBBER MODELS



DURATION ≈ 86 WR $A^{1/2}/W_T^{3/2}$ IN SECONDS

APPROXIMATE SINKING SPEED

$$V_S = 11 \times \frac{W_{oz}^{1/2}}{A_{sq.in}^{1/2}} \ \text{FT.P.SEC.}$$

DURATION FOR HYDRO R.O.W
TIME = 70 WR $A^{1/2}/W_T^{3/2}$ SEC.

DURATION FOR HELICOPTER
TIME = 40 DIA × WR / $W_T^{3/2}$ SEC.

DURATION FOR ORNITHOPTER
TIME = 15–30 WR $A^{1/2}/W_T^{3/2}$ SEC.

DURATION FOR AUTOGIRO:
TIME = 30 DIA WR / $W_T^{3/2}$ SEC.

DETHERMALIZERS. After a free-flight model comes close to its maximum time accepted by the contest rules a mechanical device or fuse should release a dethermalizer unit that increases the sinking speed considerably over the velocity of the riser, but do not damage the structure when landing, and keep the model within the contest area.

The simplest device seems to be a parachute. Analysis by equation shows that a small parachute increases the sinking speed by 100 to 150 per cent. A large parachute will not increase the sinking speed, it will however steepen the glide. The chute should be at least 12 inches in diameter, with a bleeder opening and at least 6 shroud lines to insure opening.

The safest and generally used arrangement is to increase the angle of the wing or the angle of the tail. Increasing the angle of the wing is used on jet models, which have a very thin rear body section for mounting a reliable hinging tail. On regular rubber or gas jobs the tail angle is increased by 35 to 45 degrees, which gives the wing a 60 degree angle of attack, increases the sinking speed 5 to 10 times and the gliding angle from 1:10 to 1:1/2. A diagram shows the balance of forces after the pop-up of the tail and of the wing. It also shows the relative position of the fuselage in the glide. Timers should be used to realease the device and in case fuses are permissible the burning length is adjusted to 1 to 2 inches for each minute.

COMPARING DESIGNS. To compare designs belonging to the same category an equation is derived from the two basic equations for speed and power, which gives a combination of the design units against the combination of the aerodynamic coefficients, the values of power or sinking speed. The power in the equation may be replaced by the weight of the rubber, the weight of the jet pellets or by the cu.in. of the engine. The combinations of the coefficients is applied to selections of airfoil sections for minimum power or for minimum sinking speed. Squaring this ratio a simpler one is obtained, which is the cube of the lift coefficient divided by the square of the drag coefficient. Sections should not be compared without the additional parasite drag coefficient. A numerical example shows the effect of the CL and CD coefficients on the gliding angle and on the sinking speed by having two c.g. locations; one at the 1/3 chord distance and the other at the 2/3 distance. Many assumptions are made in order to obtain an answer to the investigation.

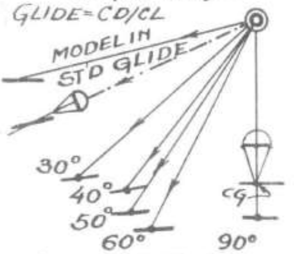
ZERO-LIFT LINE SETTING. Each category of model design will have a specific zero-lift line setting for satisfactory horizontal stability. It will be less for indoor models than for outdoor jobs, especially when windy and gusty weather prevails. The difference of the zero-lift lines is also called the longitudinal dihedral. The increase of the setting moves the c.g. closer to the leading edge and the slope of the pitching moment coefficients becomes steeper. Suggested values for the setting of the zero-lift lines are given on the following page.

DETHERMALIZER

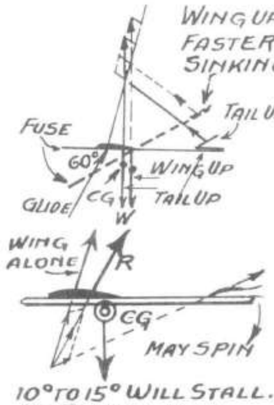
COMPARE SINKING SPEEDS BY $V_S \cong CD/CR^{3/2}$

$CR = (CD^2 + CL^2)^{1/2}$

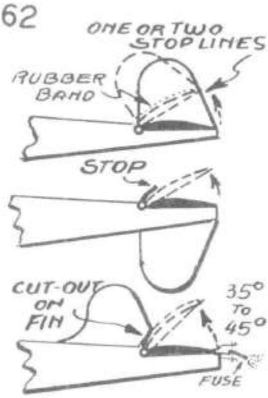
GLIDE = CD/CL



ANGLE OF ATTACK

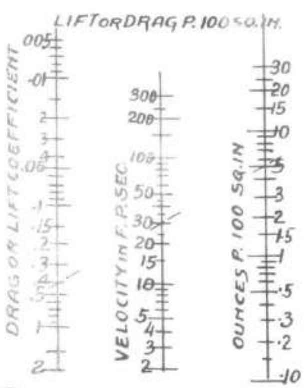


10° TO 15° WILL STALL. NO BALANCE OF FORCES



PARACHUTES
12" DIA - 4" BLEEDER
HEAVY SILK. MIN. 8 SHROUDS.

COMPARING DESIGNS



EXAMPLE: CL=.4, SPEED 30 F.P.S.
ANSWER: LIFT = 5 OZ/100 SQ.IN.

BASIC EQUATIONS

$$550\,HP = W\frac{CD}{CL}V$$

$$V = \sqrt{\frac{W}{A\,CL\,.00119}}$$

AFTER SUBSTITUTION

$$HP = \frac{W^{3/2}\,CD}{A^{1/2}\,CL^{3/2}}\,\frac{1}{19}$$

OR

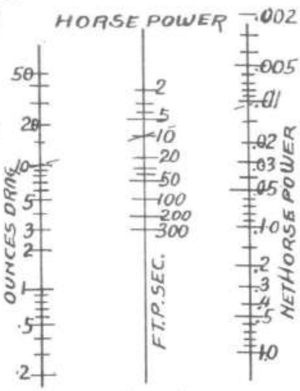
$$\frac{HP\,A^{1/2}}{W^{3/2}} = \frac{CD}{CL^{3/2}}\,\frac{1}{19}$$

HP MAY BE CU.IN., WEIGHT of RUBBER, WEIGHT of ROCKET POWDER.

BEST DESIGN FOR MAX. $CL^{3/2}/CD$.

THE LEFT TERM IS THE POWER LOADING AND THE WING LOADING COMBINED.

HORSE POWER



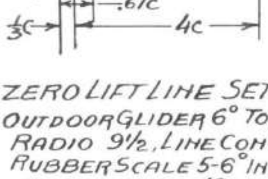
DIVIDE NET HORSE POWER BY THE EFFICIENCY OF PROP TO OBTAIN B.H.P.

PERFORMANCE - TWO CG LOCATIONS w. 1/3 TAIL

SINKING SPEED FUNCTION: $\frac{CD \cdot \cos^{3/2} B}{CL^{3/2}}$

ASSUME $\cos^{3/2} B = 1$

LOWEST SINKING SPEED FOR MAX. $CL^{3/2}/CD$.



$$\frac{CL^{3/2}}{CD} = \frac{(C_{LW} + C_{LT})^{3/2}}{\left(C_{DPR} + C_{DPAR} + \frac{C_{LW}^2}{\pi AR} + \frac{C_{LT}^2}{\pi AR}\right)}$$

CD PROF + CD PAR = .10

C.G. AT 1/3 CHORD

$C_{LW} = .8$ $C_{LT} = 0$ $C_{DIN} = .0252$

GLIDING ANGLE $L/D = 6.38$

$CL^{3/2}/CD = 5.7$

W FOR WING, T = FOR TAIL

INCREASE CL ON TAIL TO .73

CG MUST BE AT .66 CH.

$$\frac{CL}{CD} = \frac{(.8 + 1/3 \times .73)}{(.10 + .0254 + .014)}$$

$CL/CD = 1.04/.14 = 7.45$

$CL^{3/2}/CD = 7.6$

REDUCTION of SINKING VELOCITY $7.6/5.7 = 1.33$ BY 33 PER CENT.

STABILITY MORE SENSITIVE.

ZERO LIFT LINE SETTINGS. INDOOR GLIDER 3° INDOOR RUBBER 4° OUTDOOR GLIDER 6° TOWLINE 7° OUTDOOR RUBBER 4°-6°, GAS 6-8° RADIO 9½, LINE CONTROL: SPEED 2°3° STUNT 0° TEAM RACER 4° RUBBER SCALE 5-6° INDOOR R.O.W. 8° FLYING BOAT 8°-9°

ZERO-LIFT LINE SETTING DEPENDS ON TAIL AREA, C.G LOCATION AND CAMBER RATIO OF WING.

GLIDERS. Nature's best glider is the leaf of the Zanonia Macrocarpa, which carries a seed and has washout tips; it will glide up to 5 miles. Its dimensions were used for construction of a man carrying glider. After many successful glides it was powered but could not take off because the landing gear prevented the increase of the angle of attack for take-off. Another plane with a standard tail, the "Taube" (pigeon) was built which after a short term of success was abandoned, because its spiral stability was poor. An era of biplanes followed which gave way to monoplane, a design aviation started with.

Not many glider models were built before the turn of the century. Some used solid mica wings for scientific testing. Usually the material was stiff paper, silk stretched between bamboo, wire or ratan frames without cambered ribs. For structural connections cork was used. The real interest started when balsa was commercially obtainable for building paper covered or solid balsa gliders.

Gliders are hand-launched or released by a towline. Each has a different problem. Hand-launched indoor gliders should have a minimum drag traveling upward and a maximum lift gliding downward. An equation has been developed for the duration and height of indoor gliders. Diagrams give performances of gliders with various weights and launching speeds. The height reached is in close relation to the weight; the duration however is limited to the initial speed. Contests are won with thin airfoil sections and flat lower surfaces. and polydihedral wings. Better time could be made if the rear section of the wing were made flexible, with an undercamber and a straight dihedral. The glider should climb in a slow right-hand spiral to the ceiling and glide downward to the left. The high launching speed flexes the wing and the spiral climb reduces the tail setting, which add to the height to be reached. The average time for the upward travel is 1 sec.

The material for the wings is 5 to 6 lb.p.cu.ft. balsa, with a hardwood leading edge. For the body a 7 to 8 pound balsa will give the necessary sytength and slenderness. The same weight balsa is used for the tail surfaces. The dihedral break corners are rounded with small fillets. To balance the glider clay should be omitted because the clay increases the drag of the body by 100 per cent. The whole glider is coated with a glider polish, a clear dope plastisized with a few drops of castor oil. This will pervent war ping by slowing down the drying. Some model builders protect the leading edge by placing a steel wire into the nose. The finished glider should weigh between 3/4 to 1 oz. for 100 sq.in. of wing area. The tail setting of the zero lift lines is from 3 to 4 degrees, usually not sufficient for out door flying, especially when risers are present.

The towline glider has only one problem for selecting an airfoil section, the thinnest section suitable for a strong, high aspect ratio wing with the lowest sinking speed. This section may be the N.A.C.A 6406 with an aspect ratio as high as 11 to 12. The wing at the 1/3 length of the fuselage and balanced with weight if necessary. Triangulation for the wing and stab will be required to prevent warping, which also may call for bass wood spars. The auto-rudder construction may have to be balanced by a small fin ahead of the wing.

The location of the vertical control surfaces below the fuselage not only gives the rudder and the fin an undisturbed airflow but also eliminates blanketing by the fuselage and by the stabilizer. A vee-tail will counterbalance the negative rolling moments of the vertical surfaces below the fuselage.

The hook for attaching the twoline for launching should be 60 to 80 degrees ahead of the c.g. depending on the drag of the fuselage and the location of the aerodynamic chord. The hook is attached to the left side if the glider is set for a left glide and placed in the center if an auto-rudder is used. The auto-rudder is connected to a hinged hook in order to set the rudder inline during the launching and after release will move it for a left glide. The tail distance is from 3.5 to 4.5 chord-widths and the tail has an aspect ratio from 5 to 6. The area is from 15 to 20 per cent of the wing area. This low ratio is due to the absence of the moments created by a rotating propeller. It may be also caused by the inclusion of the tail area into the total allowable area.

In case the climb during the launching is too fast, the glider will move on the outside of the circle, which will increase the download on the stabilizer and move the vectors toward the leading edge. This may make the glider laterally unstable.

Two specifications are in use for contests. One is for limited gliders with a maximum total supporting areas of 350 sq.im. and a minimum weight of 10 ounces. The other is for Nordic gliders with a total surface area from 263.5 to 294.5 sq.in. and a minimum weight of 14.46 ozs. or 3200 to 3400 sq.cm. and 400 grams. The limited will give 2.85 oz./100 sq.in. and an approximate sinkinspeed of 2.04 f.p.s. and for a Nordic 3.65 oz./100 sq.in. and a sinking speed of 2.3 f.p.s.

EFFICIENCY OF STABILIZER. The effectiveness of a stabilizer is reduced by the proximity of a fuselage or a fin. This interference depends on the relative size of the portion of the fuselage next to the stab, the aspect ratio and the plan view of the stabilizer. The angle of attack also influences the efficiency. The slipstream and the downwash make additional inroad on the efficiency of the stabilizer.

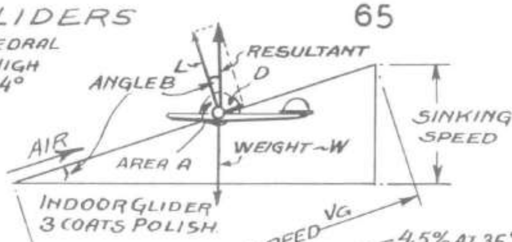
If the elevator is a part of a thick symmetrical section, it may not work for 2 degree deflection, because it is within the turbulent air leaving the airfoil at the separation point. This is corrected by drawing a symmetrical section of a shorter chord and two tangent lines from the trailing edge of the original chord.

Leak is stopped by sealing the hinge gap. A chart gives elevator tests, which indicates that the larger the ratio, the greater the lift. A sketch shwos the connection of flap and elevator on a control line model. The flap angle should be 1/3 of the elevator angle.

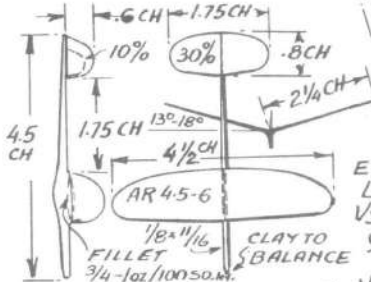
GLIDERS

NATURE'S BEST GLIDER

DIHEDRAL AS HIGH AS 24°



INDOOR GLIDER
3 COATS POLISH.



GLIDER AIRFOILS

EQUATIONS

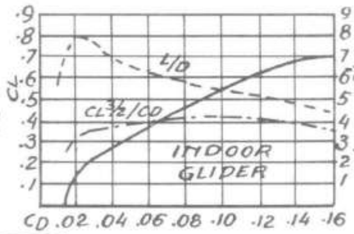
$L = W \cos B$

$V_S = V_G \sin B$

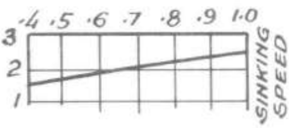
$\rho = .00238$

$V_S = \sqrt{\dfrac{W}{A \times \rho/2}} \; \dfrac{\cos B^{3/2} C_D}{C_L^{3/2}}$

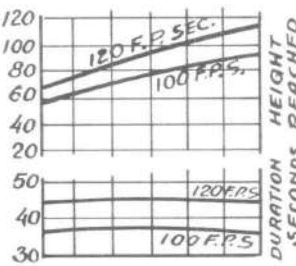
$\dfrac{C_D}{C_L^{3/2}} =$ POWER FACTOR



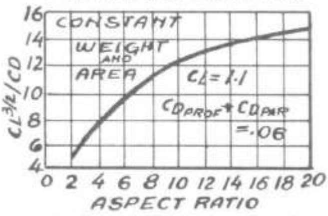
OUNCES PER 100 SQ.IN



V_G F.P.S. $= \sqrt{\dfrac{7550\, W_{OZ}}{A_{SQ.IN}\, C_L}} \cos \tfrac{1}{2} B$



SINKING SPEED + A.R.



HARDWOOD
.006 WIRE

SEMI TRANSPARENT
SHOULD FLEX IN LAUNCH
WHEN USING UNDER CAMBER, STRAIGHT DIHEDRAL.

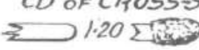
INDOOR GLIDER:

WING	.19 oz
TAIL	.04 "
FUSE	.07 "
30 SQ.IN	.30 OZ

FOR TURN TO LEFT IN GLIDE CLAY ON LEFT TIP OR LEFT L.E. OF STAB UP AND RIGHT T.E. DOWN.

INDOOR: WEIGHT 3/4–1oz/100 SQ.IN
LEADING EDGE: WIRE, HARDWOOD, SCOTCH TAPE.
WING TO FLEX., FUSE. 1/8"
LEAD. EDGE CUBIC PARABOLA
TRAIL. EDGE ELLIPSE
ADD WEIGHT FOR OUT DOOR

GLIDER BODY; TESTED 1950 R.J.H.
CD OF CROSS-SECTION



FOR LARGE ANGLES OF ATTACK 30°-90°
USE $C_R = (C_D^2 + C_L^2)^{1/2}$
COMPARATIVE VALUES FOR V_S
$V_S \cong C_D / C_R^{3/2}$

HANDLAUNCHED GLIDER

MATHEMATICAL DEVELOPMENT of HANDLAUNCHED GLIDER

FORCE REQUIRED $f = m \times a = m\,dv/dt$ DRAG = CONSTANT $\times V^2$ LESS WEIGHT

$m\,dv/dt = -KV^2 - W \quad W = m \times g = m \times 32.2 \quad V = ds/dt, \; 1/dt = V/ds$

$mV\,dv/ds = -KV^2 - W \quad ds = mV\,dv/(-KV^2 - W)$ INTEGRATING

$\int ds = \int m\,dv/(-KV^2 - W) \quad S = -\left[\frac{m}{2K} \operatorname{Log}(KV^2 + mg)\right]_{V_0}^{V} + C$ TO FIND C ASSUME

S=0 INITIAL SPEED V_0 $\quad S = m/(2K) \times \log[(KV_0^2 + mg)/(KV^2 + mg)]$

AT MAXIMUM HEIGHT V IS ZERO. THEN THE HEIGHT REACHED

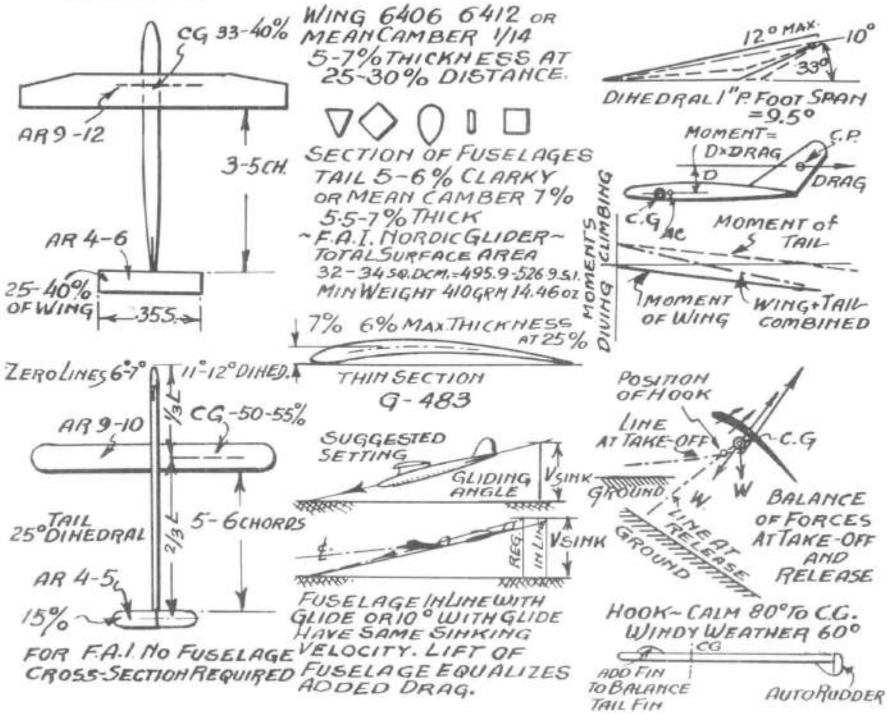
$S = m/2K \times \operatorname{Log}[(KV_0^2 + mg)/mg]$ DURATION = S / SINKING SPEED

S = HEIGHT, $mg = W$, $K = A \times .00119 \times C_{D_{MIN}}$, V_0 INITIAL SPEED FT/SEC

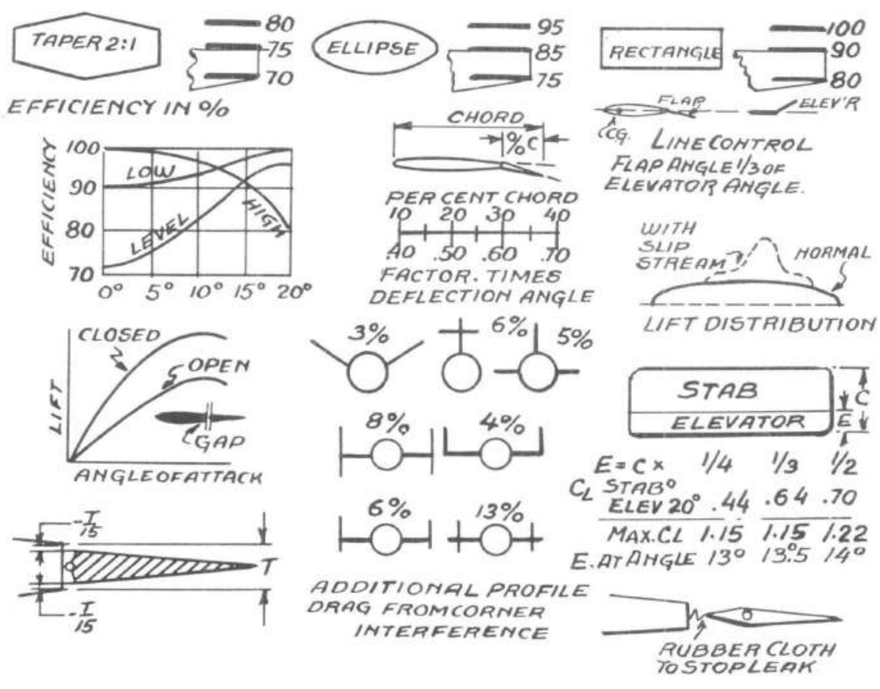
COMBINING WITH SINKING SPEED $V_S = 29\, W^{1/2} \times \cos B^{3/2}\, C_D / (C_L^{3/2} A^{1/2})$

$\cos B^{3/2} \sim 1.0 \qquad$ DURATION $= .0005 \dfrac{W^{1/2} A^{1/2}}{K} \left[\operatorname{Log}_e \dfrac{KV^2 + W}{W}\right] \dfrac{C_L^{3/2}}{C_D}$

GLIDER



HORIZONTAL TAIL



FLOATS and HULLS. Models taking off the water (R.O.W.) use floats or hulls. The floats should be of simple construction with a low drag near to the take-off and in the air. The buoyancy should be 3 to 5 times the load they carry in the normal condition. The volume is estimated for .54 oz. per cu.in. displacement. The weight of the total displaced water will be equal to the weight of the model. The c.g. of the displaced water is a point through which the line of the righting moment passes. Tilting the model and drawing a line through the center of buoyancy parallel to the gravity line will intersect the line of symmetry at a point called the metacenter. Metacenters are determined for various lateral angles and if the metacenters are above the c.g. of the model, the model is hydraulically stable. This investigation is usually applied for checking stability of flying boat hulls.

A nomogram and an equation give the displacement of floats if the length is twice the width and five times the height of the float. The volume of a float or hull can be estimated by the product of the three dimensions and a design factor, which varies from .45 to .70. A flat step gives the best load-drag ratio, but will often porpoise and must be ventilated. A Vee-step is better but hard on construction details. The best all around is the pointed flat step for smooth and greater range of "planing". In case any porpoising appears and no other adjustment helps, a plate placed back of the step at a negative angle will aid in bringing the model to the "top". This plate is often necessary on scale models. The location of the step is 1/5 ahead of the c.g. for normal loads. For top loads the step is slightly forward of the 1/5 ratio. The hull has usually a dead rise of 22 degrees at the bow and reducing to zero at the step. Some modelers like a flat bottom at the bow to prevent diving under water on a steep landing. The step should be an inverted Vee. A small spray strip will stop climbing of water on the side of the hull. The zero-lift line of the wing should be 2 to 3 degree positive with the bottom of the hull. The difference of the zero-lift lines to be 8 to 9 degrees. The thrustline should be set at 4 degrees with the zero-lift line of the wing.

The maximum width or beam of the hull for normal loading is proportional to the square root of the wing area. For higher wing loading the beam should be wider or the power loading should be decreased for a possible take-off. The design of the hull is based on the dimension of the beam. The overall length is from 8 to 12 times the width of the beam. About 45 per cent of the length is used for the fore body of the hull. Models with hulls of large length-beam ratios may have low hydraulic stability by the resulting porpoising.

Models on floats are seaplanes. Floats should be spaced at the same distance the wheels were located, but farther ahead, because the resitance of the floats is higher than the rolling resitance of the wheels. They should be designed to carry the additional torque load of the drive.

LANDING GEAR. The landing gear is designed to take the landing shock of a model, to prevent damage to the propeller, to stop turnover by the torque of the drive, to hinder ground loops and to place tha model into a position of easy take-off.

To take the shock of landing the minimum sinking speed must be known in order to estimate the minimum energy to be absorbed. Also a certain load factor, or safety factor should be assumed for occasional steeper landings. A load factor of 5 to 6 seems a good figure, because load tests on wings gave 36 times the total weight of the models. Part of the kinetic energy is absorbed by the deflection of the integral units of the model and by the flexible support of batteries and radio equipments. Some fuselages are so close to the ground that only the deflection of the tires can be used for absorbing the shock. The front wheel in a tricycle design should have a greater travel than that of the rear wheels.

The energy to be absorbed by the shock unit is equal half the distance the c.g. will travel times the maximum load. A metal landing gear will return the same energy to the model, which may bounce it into the air. If the shock is taken by rubber about 40 per cent of the energy is dissipated. An ideal shock absorber is a hydraulic shock strut controlled by valves and orfices.

The distance of the propeller from the ground should be at least 2 inches in the deflected position of the landing gear. The torque of high speed gas engines cannot upset models with standard wheel spacings. The high initial torque of rubber driven models means ground looping. This is avoided by setting the thrustline at a high angle with the ground in order to be aided by the thrust for a faster take-off. A good launching without pushing depends on the experience of the contestant.

Groundlooping, circling on the ground is often caused by a poor aligment of the tail skid or by the tail wheel. It may be also the result of unequal loads on the wheels caused by the torque of the drive, by a soft ground or by small diameters and widths of the tires. Other causes are the relative position of the wheels ahead of the c.g. and the tread of the wheels. The spacing of the wheels is determined by the height of the c.g. and from the span of the wing. A single front-wheel is offset to the left.

The angle of incidence with the ground depends on the basic design of the model. Many airplanes in the beginning of aviation could not take off because the landing gear prevented the increase of the angle of attack or high grass became entangled with the wire bracing of the gear.

To reduce the drag of the landing gear the struts are streamlined or made of flat dural stock. Cowling balloon tires decreases the drag but should be used only on hard , smooth take-off grounds. Magnesium disk wheels reduce air drag, rolling resistance and weight, which may be final cause for top performances in a load carrying contest.

HYDROPLANES (R.O.W.)

$D = \frac{L}{5}$ — W — $L/2$

VOLUME L×W×D×.72
FOR ABOVE DESIGN
$L_{IN.} \cong 2.88 \sqrt[3]{W oz.}$

BUOYANCY IN OUNCES
.1 .2 .5 1.0 5 10 30 50
1 2 3 4 5 8 10
LENGTH OF SPEC. DESIGN
DESIGN 3 TO 5 TIMES LOAD
T.N. 488 SIMILAR TO H-22

TR 25

6° TO 8°

VIPER BOTTOM

BOW — FOREBODY — AFTERBODY
WATER LINE — WELL ROUNDED

THRUST LINE PARALLEL WITH PLANING SURFACE
C.G.
W
45°
5° TO 6°

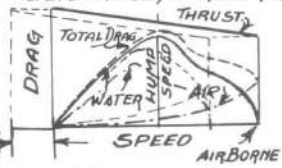
DESIGN FRONT REAR
FLOAT 1 2
LOAD 70% 15% EACH
FLOAT 2 1
LOAD 38% EACH 24%

TN 680
WING FLOATS FOR TORQUE

VENTILATED STEP

BRASS PLATE
FOR LONG'L STABILITY BEFORE PLANING

RESISTANCE/THRUST 4-5



SPONSON

BEAM — BEAM/D = 1.0 TO 1.2
METACENTER
CHINE
D
M
B
DEAD RISE 18°-22°
B CENTER OF BUOYANCY

DRAG/LIFT = 1:5
L — D
PLANING SURFACE

FLOATS OF GOOD DESIGN
33 F.P.S. $C_{D\,MIN} = .15$
$9° \sim C_D = .41$ $C_L = 2.54$
$L/D = 6.25$
BEAM LOADING:
OZ/IN = .7 OZ/100 SQ.IN.

SPRAY STRIP TO PREVENT CLIMBING OF WATER
KEEL

LIFT ON WATER SIMILAR TO AIR = $\frac{\rho_W}{2} \times V_W^2 A_W$
FROUDE LAW:
$F_N = \frac{V_{WATER}}{\sqrt{L.g.}}$

FLAT — VIPER STEP
AR=7-9 BOAT .85 SP.
TAIL 35-40% WING AREA
TAIL DISTANCE 4 CHORD
TAIL THIN CLARK Y
WING CLARK Y
HULL: TN 464, TN 791, TN 470

ZERO LIFT LINES $8\frac{1}{2}°$ TO $9\frac{1}{2}°$
THRUST LINE PARALL. TO STEP. WING ZERO-LIFT LINE 12-15° WITH DECK
LOW ANGULAR SETTING OF WING FLOATS TO PREVENT CROSS-CHINE FLOW.

ALL LOWER SURFACES OF WING AND TAIL 8-9 COATS OF DOPE.
FOR DESIGN,
$W.IN = .04\, W^{3/2}/A^{1/2}$
FOR MAX. LOAD CONSTANT ≅ .01

LANDING GEAR.

C.G.
D_I
~DROP TEST~
h
C.G. TRAVEL $D_I - D_{II}$
D_{II}

ENERGY ABSORBED
LOAD
TRAVEL
METAL — RUBBER — HYDRAULIC

DISSIPATED ENERGY CROSS LINED.
METAL L.GR.
ENERGY ABSORBED = LOAD×TRAVEL/2

FREE DROP:
ENERGY = W×h

$TRAVEL = \frac{V^2_{SINKING}}{32.2}/F.S$ F.S.: FACTOR OF SAFETY OR LOAD FACTOR

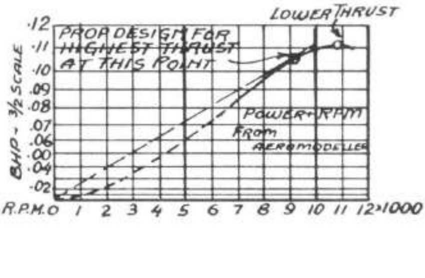
H
.5H F. FLIGHT
.10H RADIO OR LESS

RUBBER BAND
RUBBER BAND
FLAT DURAL

IF CLEARANCE IS SMALL LANDG SHOCK TO BE ABSORBED BY THE TIRES ONLY.

C.G.
H
7H — H/2 — 2H
TRICYCLE.

GRAPHIC METHOD FOR MAX. THRUST FROM BHP-RPM. DIAGRAM.
NET HP. = EFF.×BHP, MAX EFF.~V/RPM.D MAX.
STANDARD EQUATION $D = BHP^{1/4}/(RPM^{1/2}\, V^{1/4})$
NET HP. ≅ $V^{5/4} BHP^{3/4}/RPM^{1/2}$ ~ V ~ CONSTANT
AND SQUARED NET HP. ≅ $BHP^{3/2}/R.P.M.$
MAX $BHP^{3/2}/RPM$. TANGENT DRAWN TO
DIAGRAM PLOTTED ON 3/2 POWER SCALE



FLAPPING WINGS- ORNITHOPTERS. The movement of a bird wing was used for the first successful flying model. Even the first man-carrying gliders had the outline of a bird. Complicated mechanism to imitate the automatic changes of the incidence angle and airfoil sections never produced valuable informations on the aerodynamics of flapping wings, because all changes of angles, airfoils, spreading and closing of the tip feathers are inherently actuated by the direction and strength of the relative airflow; the bird does nothing about it. Mounted birds placed into wind tunnels gave a gliding angle of only 1 to 4.5, and the actual gliding angle of flying birds is more than 10 times the dead bird. Airfoil sections of a bird wing give only laminary (?) flow and any possible turbulent flow is made laminary by nature to grow feathers at the right place, as seen on egrets and pelicans. There two theories to be formulated for the presence of the thumb feather (alula) and for the scalloped trailing edge of a bird.

In 1901 Marey took movies of bird flights and attached instruments to pigeons to measure the power required for flying. He did not obtain the minimum power or the actual gliding angle of birds. Some of the results are given in the diagrams shown. The best information were collected by movies taken of a large rubber band driven model and from wind tunnel tests of a small wing oscillating at high speed. The pictures gave the speed, the angles measured from the outline of the ribs and the power by the rubber used. A twin ornithopter tested in 1912 gave an apparent CL of 1.56 but only a gliding angle of 1.32. The tests indicated that the lift is done by the inner section of the wing and the propelling by the outer section.. Also that the body does not move into the .opposite to the direction of that of the wing, but is 90 degrees off phase. This phase difference reduces the strain on the muscles of the birds.

The tip of the insect wing describes figures of eights with varying tracks in order to move a greater mass of air. Its principle is similar to the action of a bird wing, except the wing has a greater axial rotation, which is due to the centrifugal force. Models with wings describing eights and rotating about their lateral axes gave very high lift. The weight of the mechanism will be too heavy for efficient flying models.

Many other ideas were tested with wings moving in a horizontal or in a vertical plane. Great sums were wasted 50 years ago to build an airplane by oscillating upturned cups up and down. Oscillating wings back and forth gave a lift as estimated with standard equations. Friction and weight always precludes construction of efficient models with oscillating wings. Oscillating solid wings about their 25 per cent lateral axes gave no effect at all. To be effective a wing must have a double action. It must move up and down, and rotate about its lateral axis, which may be actuated by its inherent flexibility. Tests showed that the oscillating plane should be 20 degrees ahead of the vertical and that the maximum flap angle should be 100 degrees. The duration of rubber driven ornithopters is less than 50 per cent of propeller driven models.

Wings are usually flapped by a crankshaft and connecting rods. A tensioner will not increase duration; it may however improve efficiency if the number of strands are reduced. To eliminate oscillation of the fuselage twin wings were tried; they increased the apparent CL to 1.56 but the gliding was reduced to 1.32.

VIBRATIONS. Many kinds of vibration are possible on models. They are greatly dampened by the extensive use of balsa. Vibrations may be caused by poorly balanced engines, propellers or by the vortices in the wake of the downwash. A single cylinder engine cannot be balanced. A great deal of reduction of the vibrations can be achieved by making the piston as light as possible and by balancing the crankshaft with a portion of the weights of the connecting rod and the piston.

Another unit that causes vibrations is the propeller. The gyroscopic couple of a two-blader in circling flight is sinosoidal, which become smoother with three-bladers, which however have lower efficiencies. A hand-sanded propeller is often unbalanced by weight by an unequal thrust. This unequal thrust is present with any propeller flying in a circle. 1/1000 oz. on the tip of a 10 in. propeller gives a 2 oz. centrifugal force running at 15,000 r.p.m. For top performances propellers should be checked for weight balance and for equal pitch distribution on both blades. Engines should be tested mounted in the model to make certain that it will run with the same r.p.m. with the same propeller as tested on the bench.

Another vibration may occur by the resonance of the natural frequency of the blade with the r.p.m. of the engine. To-day propellers have at least twice the natural frequency of the r.p.m. of the engine. Some propellers manufactured years ago had very thin hub sections which gave a frequency of 15,000. The natural frequency of an average wing is about 2500 to 3000, which is too low for resonance. The only time wings may be damaged by flutter is in power dives. Usually the loss of a wing is due to weak rubber bands or shift of the wing on the support.

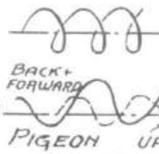
Flutter may occur with statically unbalanced elevators or rudders if the control rods have play in their connections. A weight and an aerodynamic balance will remedy this situation to a certain degree. An unbalanced propeller or engine can cause to flip the escapement and the relay.

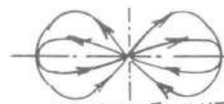
Vibration lock or resonance of a long fuel line may stop the flow or it may even drain the whole line. Long lines should be fastened to prevent resonance.

The turbulent air leaving the trailing edge may start a resonance in very thin tail surfaces. The vortices have a frequency of the speed divided by four times the thickness of the wing. An 8 in. wing moving at 30 f.p.sec. give vortices pulsating at 9000 times per minute, which may be close to the natural frequencies of a thin stab. The frequencies of an average stab is much higher and placing it into the wake may not start any vibrations, but an added slipstream of the propeller, which is also pulsating may reach the frequency of the stabilizer.

Basic equations are given for vibration of beams of uniform cross-section supported on one end.

FLAPPING WINGS - ORNITHOPTER



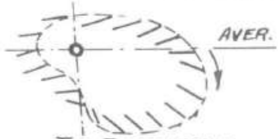


INSECT FLIGHT

? $LIFT = \frac{\rho}{2} A (.205\, M^2 \times FLAP^2/SEC + .36\, SPAN^2\, FLAP/SEC)$

? $POWER = \frac{\rho}{2} A (.205\, M^2\, FLAP^3/SEC + .36\, V)$

MANPOWER - 1/5 H.P.
FOR SHORT TIME - 1.0 H.P.



TIP FEATHERS

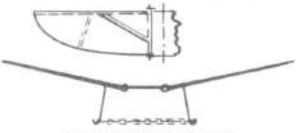
BUZZARD
REF. E.J. MAREY 1873

BIRDS

	A.R.	SPAN	LBS/SQ FT.
BUZZARD	6.5	5.4 FT	1.06
ALBATROSS	18.5	12.5 "	3.12
GULL	11.1	4.1 "	1.03
PELICAN	8.1	9.2 "	1.36
HERON	7	5.45 "	.82

POP. AVIA. 1935
R.J.H.

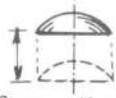
TEST OF FLAPPING WINGS
SINGLE: $C_L = 1.45$ L/D 2.72
TWIN: $C_L = 1.56$ L/D 1.32
TESTED 1912 R.J.H.



RUBBER BAND
NO CHANGE IN DURATION

TAIL SETTING -12° TO -15°
WING AR 8.5 TO 9.5
STAB 25-40%
RUDDER 8°-12°
RUDDER AREA NOT CRITICAL

RUBBER UP TO 50% WT



OSCILLATING HOLLOW CUPS.

RUSSIA 1900
EMPTY WEIGHT
60 SQ.IN. .20 OZ.
DURATION IN SECONDS

$$= 15\sim30\ \frac{W_R A^{1/2}}{W_T^{3/2}}$$

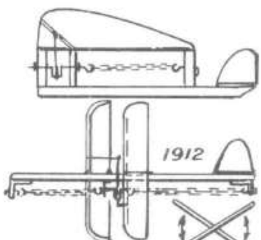
A IN SQ.IN., W IN OZ.

REFERENCES:
KÜCHEMANN & WEBER

OSCILLATING SOLID WINGS - NO OVERALL GAIN.

HELICOPTER BLADES
2 TO 12 OSC. P. REV. NO GAIN.

FLAPS/SEC.
SPAN 60" 2/SEC.
" 18" 6/SEC.

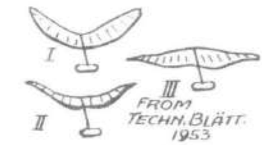


WE = 2 OZ. A = 220 SQ.IN.

NUMBER OF STRANDS

$$N_{1/8} = .85\, W_T^{2/3} \times SPAN^{2/3}$$

WITHOUT WEIGHT TENSIONER



FROM TECHN. BLÄTT. 1953

HIGH LIFT. POOR MECH'L DETAILS.

VIBRATIONS

NOT BALANCED:

$$C.F._{OZ} = W_{OZ}.\ DIST_{IN}\ \frac{R.P.M^2}{35{,}000}$$

ENGINE OR PROPELLER

RELAY MAY VIBRATE WITH ENGINE RUNNING

RELAY WILL STOP VIBRATION IN THIS POSITION.

DEAN HARTOG. 1940.
MECHANICAL VIBRATIONS

RESONANCE:
STRONG WINGS IN RESONANCE WITH ENGINE RUNNING 2500-5000 R.P.M.

VIBRATION LOCK: FUEL LINE TOO LONG MAY STOP OR REVERSE FLOW OF FUEL.

RESONANT VIBRATION OF PROPELLERS.
REG. PROPS AT 34,000 R.P.M.
OLDER DESIGNS W. THIN SHANKS AT 15,000 R.P.M.

FLUTTER:
WINGS IN POWER DIVE MAY FLUTTER.
TOO WEAK RUBBER BANDS

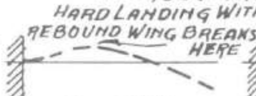
RES. VIBR. T.M. 957

EQUATION FOR FREQUENCY

$$f_R = a_n \sqrt{\frac{E.I\ 32.2}{W.\ L^4}}$$

EI - BENDING STIFFNESS

$a_n = 3.52$



$a_n = 22.4$

$a_n = 61.7$

ROTATING WINGS. The lifting power of rotating wings, cylinders and other objects was demonstrated with models driven by rubber, steam and gas engines. Some rotate the lifting unit about their horizontal axes and others about their vertical axes. With certain arrangements all models lifting by rotation about any axis can glide by self-rotation, auto rotation. The big problem is always stability and how to overcome the gyroscopic precession and the counter torque.

Various ideas were tried, such as levitaion by centrifugal force, accelerating downward the boundary layer of a cup rotating at high speed. The closest to success came the cyclogiro. By that time the autogiro made its first flight, which was soon followed by powering the rotor, the helicopter.

Helicopter models driven by rubber bands are flown since 1880. Most rubber driven models fly up straight without any forward velocity. Lately however gas driven models are adjusted for forward flight, with an added autogiro trim for landings without damage to the model. This combination may be instrumental for a new spot landing contest.

Helicopter models use extensively balsa wood with rotors of solid balsa or tissue covered. Indoor models usually have microfilm covered rotors. Indoor helicopters obtain half of their duration by hovering about half of the diameter above the ground. The height of the room has little influence on the duration because a small prong on the top rotor protects the rotor from touching the ceiling.

Helicopters with a single rotor must have a drag plate placed on a rod or at the tail of the fuselage in order to slow down counter rotation. The loss is about 5 to 10 per cent of the power. Similar reactions are obtained on flying saucers by using the surface friction of the large lenticular bodies. In order to balance precessional forces the angular moment of inertias of the propellers and the rotating bodies must be the same. The slipstream of the rotor blasting on an inclined surface attached to the fuselage is not sufficient to balance the torque. Three 1/7 to 1/10 cambered surfaces however will do the work if the rotor has a low pitch setting and a low solidy. Model helicopters have seldom adjustable tail rotors.

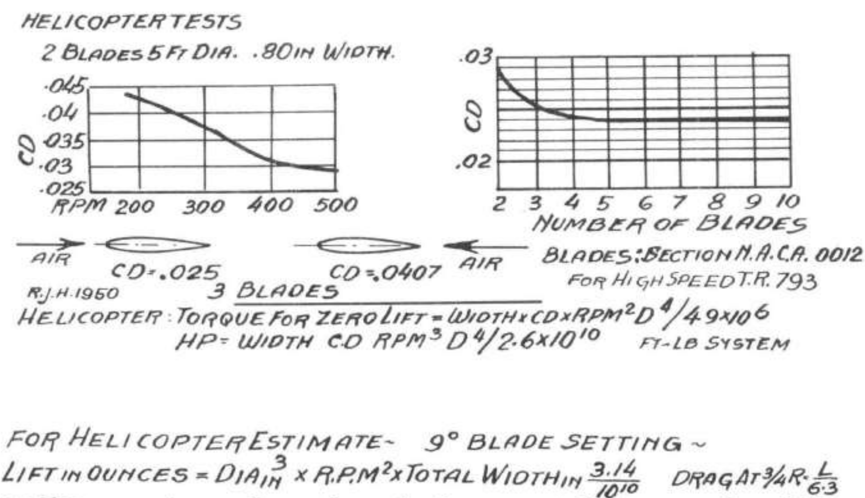
To eliminate counter-torque rotors are driven by placing power drives on outriggers or on the tips of the rotor blades. One design uses powder jets, which works very well; another design has propellers to pull the rotor around. If both rotate in the same direction the precessions are balanced, the gyro forces however are not eliminated.

The location of the c.g. depends on the pitch setting, the gyro forces of the rotor or rotors and the location of the air resultant of the fuselage, usually slightly behind the rotor axis.

Propellers of gas driven helicopters run at 15,000 r.p.m. and the rotors at 600 r.p.m., which will balance the precessional moments if the weights are just right. A tail rotor can correct the difference of the torque and counter torque.

The forward speed is induced by the slipstream blowing on an inclined surface placed on each side of the fuselage. The axis of the rotor slopes slightly to the side in order to balance the eccentric resultant of the advancing blade.

Helicopter designers suggest hinging the propeller and the rotor. Some models have hinged propellers but no hinged rotors. A pitching hinge at the 35 per cent chord distance and the c.g. of the blade at the 25 per cent of the chord distance gives automatic pitch adjustment for flying and autorotation. One design has two fixed blades and two hinged for pitch adjustment for flying at high r.p.m. and autorotating at low r.p.m., controlled by a weight placed ahead of the blades. The pitch for flying should be about 9 degrees and for autorotation zero or negative. Some helicopter models control the pitch by the coning angle, which is the resultant of the centrifugal force and the lift moments. The hinge at the hub should be about 35 degrees measured forward in the direction of the tip of the blade. The total width of the blades should not be over 30 per cent of the diameter. Coning angle improves stability; placing however both rotors below the c.g. will give automatic stability, because they become pusher propellers, which are stable.



HELICOPTER: TORQUE FOR ZERO LIFT = WIDTH x CD x RPM^2 D^4/4.9x10^6

HP = WIDTH CD RPM^3 D^4/2.6x10^{10} FT-LB SYSTEM

FOR HELICOPTER ESTIMATE ~ 9° BLADE SETTING ~

LIFT IN OUNCES = $DIA_{IN}^3 \times R.P.M^2 \times TOTAL\ WIDTH_{IN} \frac{3.14}{10^{10}}$ DRAG AT 3/4 R $\cdot \frac{L}{6.3}$

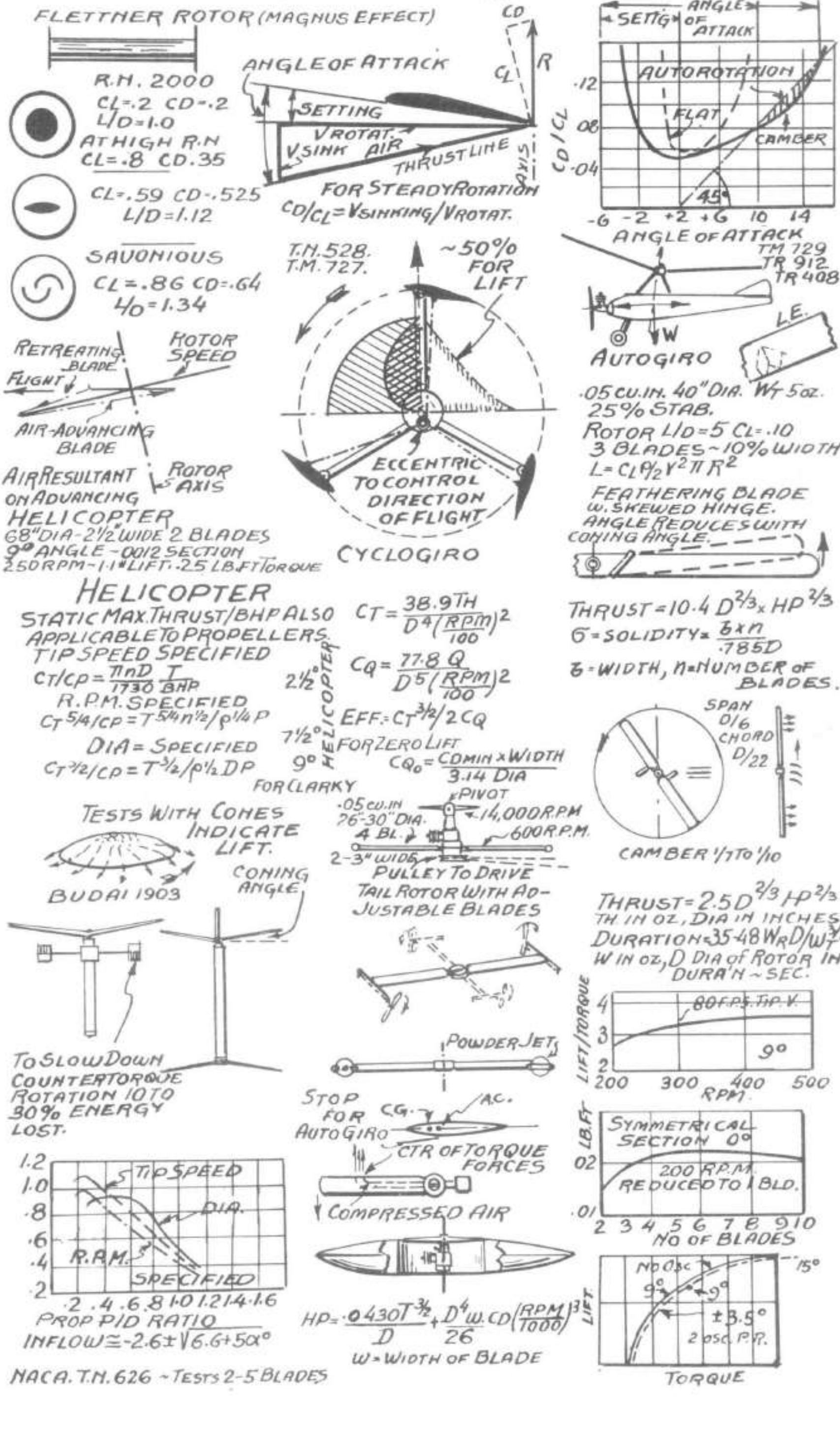
OR ESTIMATE LIFT OF TOTAL AREA AT VELOCITY AT 3/4 RADIUS, C_L = .55

L/D = 6.3 TORQUE AT 3/4 RAD. $LIFT_{LBS} = A_{SQ.FT} V^2 \times .0012 \times .55$

$V = .75R \times 6.28 \times R.P.M/60$ FT. LB. SYST.

HELICOPTERS	INDOOR NO SPECIFI'N.	OUTDOOR	W_T = 1 oz. MIN.
ORNITHOPTERS	" " "	"	W_T = 1 oz. MIN
AUTOGIROS	VANE AREA > WING AREA	SAME FOR OUTDOOR	W_T = 1 oz. MIN.

ROTATING WING



RUBBER. A motor made of rubber strands stores the highest torsional energy per unit weight. It will deliver as high as 55,000 in.oz. of work for each ounce of weight. The best coil spring will deliver only 25,000 in.oz. of energy. Plotting the torque against the number of turns the curves indicate that the winding and the unwinding energy lines are not the same. The difference is called the hysteresis, a characteristic of rubber used in dampening shock and vibrations.

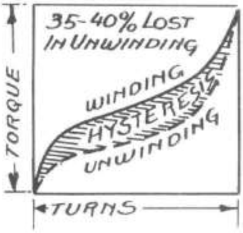
The maximum power is delivered if a rubber power motor is wound up to 80 to 86 per cent of the breaking point and the restoring time for the following winding is close to an hour. The winding is done under a steady tension, starting with 2 to 3 times of the original length of the motor and lubricated with a solution of liquid soap and glycerin. A spring keeps the motor from bunching, which may upset the stability or damage the fuselage. This tehsicner is set to engage the shaft at a stop that allows the prop to fold at a predetermined position. The turns left unwound are not sufficient to keep the model in a horizontal flught.

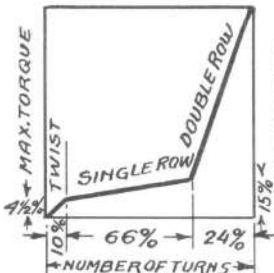
RISER. The rising air increases the duration of a free-flight model. Risers or thermals are always present, especially over an open, level contest ground, usually increasing their speeds up to midday. Due to the rotation of the earth thermals move in a slow spiral to the right and consequently most of the models are adjusted for a glide to the left. Added factor influencing the flight is the strength of the wind, which has an upward slope in a warm front and a downward slope in a cold front. The strength and the slope of the wind is not steady and theories could not supply proofs that dynamic soaring is possible by use of the variation of the strength of the wind.

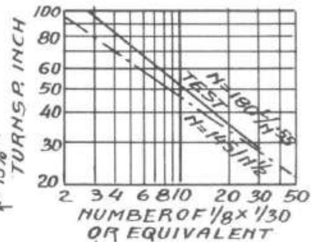
A riser starts horizontally and increases its vertical velocity as it moves toward the center. This results in a curved air flow, a circular flow. A model reaching such an airflow will dip immediatly because the negative tail setting is reduced. This reduction moves the vectors of the resultants toward the trailing edge and the c.g. will be at a lower angle of attack, which increases the gliding speed. Models that have low tail settings such as indoor gliders may go into a dive, because there is no horizontal stability left. Sketches show vector diagrams for both conditions. If the tail setting is sufficient the model may keep within the riser, which is due to the higher air speeds on the periphery of the revolving air.

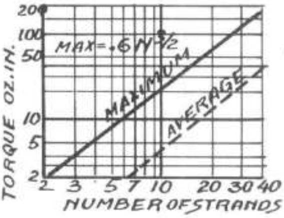
Risers are created by air moving over sloping terrain or hill. Models should be launched on or close to the edge of the hill, in order not to be blown into the region of the down-wind. Large gliders use thermals, cold fronts, ridges or even sea waves to reach rising thermals.

RUBBER

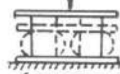








ENERGY STORED
42,000 TO 55,000 IN. OZ
PER OUNCE RUBBER
RUBBER IN TENSION
35,000 IN. OZ/OZ.



BLACK RUBBER PAD IS COMPRESSED 45% BY 4000 LB. P. SQ. IN.

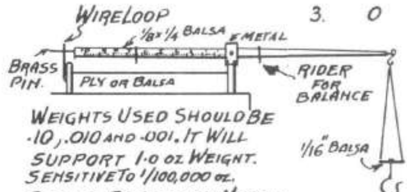
ENERGY:

80% OF MAX. TURNS 130 FT. LB.
90% " " " 200 " "
UP TO BREAKING POINT 280 " "

ASSUMING SIMILARITY IN UNWINDING IN THE AIR AND ON THE TEST STAND

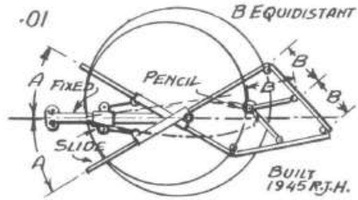
STATIC DURATION FOR SAME RUBBER
TIME = TURNS/TORQUE
TIME = $1/N^2$

	DURATION	TURNS	ENERGY
1.	.15	.24	.68
2.	.85	.66	.31
3.	0	10	.01



WEIGHTS USED SHOULD BE .10, .010 AND .001. IT WILL SUPPORT 1.0 OZ WEIGHT. SENSITIVE TO 1/100,000 OZ.

SCALE FOR INDOOR MODELS
MFG. BY MIDWEST 1939



INSTRUMENT TO DRAW JOUKOWSKY SECTIONS MADE OF BRASS TUBINGS.

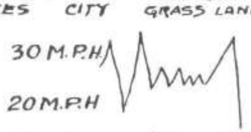
THERMALS, RISERS

GUST TESTS T.N. 731.

RISER TO THE RIGHT

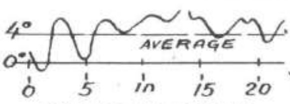
RISER PULSATING

COLD FRONT

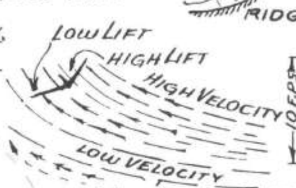


RECORDED VELOCITY CHANGE OF WIND

WARM FRONT

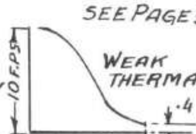


TIME IN SECONDS
INCLINATION OF WIND
CLOSE TO GROUND



MODEL WILL STAY IN RISER IF TAIL SETTING IS ADEQUATE

SEE PAGE 54



ENGINES. Engines used for models are of the two-stroke cycle system. The two cycle operates by creating a partial vacuum in the crankcase on the up-stroke of the piston; this draws in an air-gas vapor mixture through the side, through the hollow shaft, through a rear rotor valve or through a reed valve. In the up-stroke the piston compresses the mixture, which is ignited by a spark, by a glow plug or by the high compression itself. The explosion drives the piston downward, compressing the mixture in the crankcase. The exhaust ports open and after that the by-pass opens, which allows the mixture to rush toward the top of the chamber. Some of the mixture may escape through the exhaust openings. The timing of a two-cycle engine is shown in a diagram, which varies with the ratio of the length of the connecting rod and the stroke. The higher the power output is the harder it is to start the engine. Usually starters are used on high compression engines.

The spark plug ignition allows a more flexible running than the glow plug or the compression ignition. The compression engine is called a diesel, which is a misnomer because the original DIESEL is a four-stroke with a fuel ignition at the end of the up-stroke. The efficiency of a model engine is usually compared with its delivery of h.p. per cu.in. displacement. The thermal efficiency is based on the power output (heat power-B.T.U.) of the fuel. The indicated efficiency is the work done by the explosion pressure on the piston area and the volumetric efficiency is based on the volume of the mixture in the chamber. The volumetric efficiency depends on the scavenging of the exhaust gases and may be assumed to be the h.p. output per cu.in. displacement. Its maximum may be as high as 66 per cent and the h.p. equal to 1.2 to 2 times the cu.in. displacement. Deducting the h.p. delivered at the end of the shaft from the indicated h.p. the mechanical power losses are obtained. These are usually friction losses. The indicated h.p. are calculated from the indicator diagram. The average pressure on the piston during the down-stroke is from 40 to 80 p.s.i.

Improving the power output should come under the heading "souping" up an engine. Some of the improvements may be incorporated in future engine designs. A souped-up engine however may not stand up under steady power delivery and may be good for only one contest. These doctored engines are usually carefully cowled to remove the additional heat passing through the cylinder walls. Spray bars are replaced by spray nozzles to reduce the drag of the intake. A very small hole drilled close to the jet should aid vaporization of fuel by the added air bubbles. Baffle plates on pistons follow uniform curves. Smoothening all passages, removing all ragged edges left from machining, tapering slots on cylinder sleeves, enlarging intake passages, increasing the compression ratio by turning down cylinder and crankcase, Replacing flat gaskets by small round gaskets will reduce blow-outs.or omit gaskets.

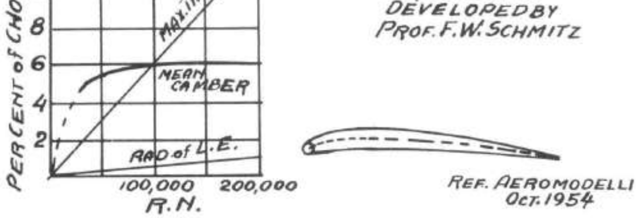
The average crankshaft is poorly balanced, which is partly due to the manufacturing costs involved. Reducing vibrations by balancing the crankshaft will add power to engines mounted in models. Old-timers in the auto business suggest that the crankshaft of a single engine should balance with the end of the con rod and with about 1/7 to 1/2 of the remainder of the con rod and the piston placed on the crank pin. The crank pin is made hollow and solder is added to the lower end of the counter weight. No perfect balancing is possible with a single cylinder engine.

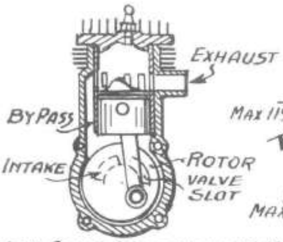
Cowling an engine not only reduces the drag but also dissipates the heat generated by a high compression engine. The sides of the engine are tapered, the cooling fins are filed to a sharp taper. A very close cowl will force the air over the fin surfaces and to prevent air to escape at the plug the opening is filled with sponge rubber and only the plug point is accessible. Thus the slight vibrations will not damage the engine or the fuselage. The hot air leaves the fuselage with the aid of the centrifugal force. The exhaust stack is cut on a bias so that the gases pass into the direction of the flight. The hot air leaving the fins should not be used for intake, because the power of a gas engine increases with cooler air. A special screened opening is provided close to the venturi for fresh air. The spinner for the prop should not be over 25 per cent of the diameter. All these improvements may ruin a few good engines before the correct doctoring is found.

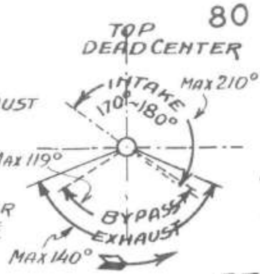
Testing engines for h.p. is made on a torque stand shown in a sketch. The engine is bolted to a swinging support, which has a weight sliding on a horizontal rod. The engine is balanced and the attached propeller run at full throttle, the weight moved on the rod until the running engine is again balanced. The distance the weight was moved, the weight and the r.p.m. are used in estimating the h.p. Varying the diameters and the pitch of the propellers many readings are obtained, which are plotted, the torque and the h.p. against the revolutions. The connecting lines must give smooth curves to prove that the results are acceptable. The revolutions are measured with a reliable tachometer. A simple home-made instrument that uses the resonance effect at maximum amplitude, can be constructed with a 1/32 in. wire sliding on a rod and calibrated with the dimensions estimated from the equation. This instrument may not be suitable above 17,000 r.p.m.

The horse power output cannot be estimated from the design of a propeller and its r.p.m. A propeller should be kept for each engine, which will give a fixed r.p.m. if working correctly. A piece of 1/32 in. wire fastened to the engine mount is trimmed until it vibrates with the maximum amplitude at the revolutions the best propeller should run.

The weight problem of the smooth running four stroke cycle may be lessened by the contest regulations that uses only 60 per cent of the displacement for estimating the power loading. A sketch shows how to hook up an engine operating with spark plug.







GENERAL DESIGN DATA:
STROKE BORE RATIO .90-1.14
CON.ROD STROKE RATIO = 1.90
EXHAUST/CU.IN = .98
INTAKE/CU.IN. = .10
BY PASS/CU.IN. = .39
PISTON SPEED IN FT.P.MIN. = $STROKE_{IN} \times \frac{R.P.M}{6}$
FUEL CONSUMPTION:
FOR STUNTS 1.45-1.70 OZ/CU.IN/MIN.

IGN. COILS OPERATE AS HIGH AS 23,000 R.P.M.
DELIVER 7000 VOLTS .5 MILLIAMP
DRAIN 1.1 AMP ~ 2.9 VOLTS
CONDENSERS USED .03 MICROFARAD
B.H.P = 1.3 ~ 2.00 × CU.IN. ~ 12,000 R.P.M.

B.T.U-HEATING VALUE P.LB.

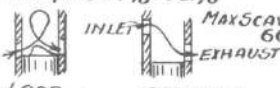
HYDROGEN	60,640
METHYL ALCOHOL	10,260
ETHYL ALCOHOL	13,160
KEROSENE	20,500
BENZINE	18,160

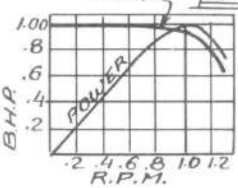
MAX. THERMAL EFFICIENCY
GAS ENGINES 22% WITH PROPS 12%
JET ENGINES 1% TO 2%

TESTING FOR BHP WITH PROPS

$B.H.P \cong \frac{W \times DIS. \times RPM}{1,000,000}$

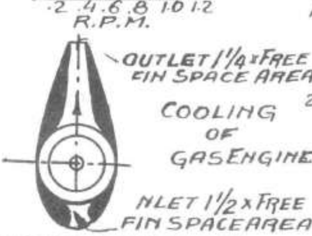
FULL THROTTLE WITH VARIOUS PROPS.





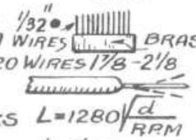
CHANGE OF POWER WITH ALTITUDE
3% P. 1000 FT. UP TO 10,000 FT.
R.P.M. LOSS 4% FOR EACH 1000 FT.

T.N. 674
T.M. 783



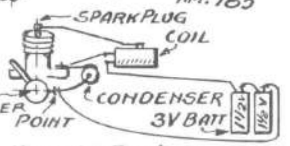
COOLING OF GAS ENGINES

R.P.M. BY RESONANT WIRE



$L = 1280\sqrt{\frac{d}{RPM}}$

L, d IN.



HOOK-UP FOR IGNITION

TWO SPEED CONTROL FOR L/C MODELS

OPERATED BY A THIRD LINE OR BY A RELAY + INSULAT-ED CONTROL LINES

COMBUSTION VELOCITIES 12-25 F.P.S.

FLAME PROPAGATION
GASOLINE 40 F.P.S.
8" DIA 3 FT. P. SEC.
NO PROPAGATION
5/32 TUBE - METHANOL + AIR MIXTURE.

COMBUSTION VELOCITY T.M. 668.

FUEL CONSUMPTION ≅
VAPORLOCK: GASOLINE 20 IN. METHANE 20 IN BENZOL 60 IN.

EFFECT OF INTAKE PIPE ON POWER OF ENGINE T.M. 501.

INLET AREA FOR COOLING ENGINES ON SPEED JOBS
CU.IN/2, SQ.IN. OUTLET CU.IN/3 SQ.IN.

H.P. REQUIRED

IF CD IS KNOWN
BEST IF ZERO THRUST

$H.P. = D^4_{FT}\ WIDTH\ CD\left(\frac{RPM}{1000}\right)^3 / 12.8$

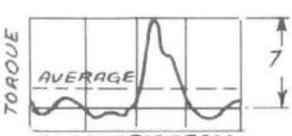
$TORQUE_{LB.FT} = D^4_{FT}\ WIDTH\ CD\left(\frac{RPM}{1000}\right)^2 / 244$

FUEL MIXTURES:
GLO FUEL: METHANOL 65%
NITROMETHANE 20%
CASTOR OIL 25%
FOR HOT WEATHER:
58%, 28%, 13% RESP.
1% AMYL ACETATE.

NITROMETHANE 40%
METHANOL 40%
CASTOR OIL 20%
DIESEL FUEL: ETHER 30% GASOLINE 38% CASTOR OIL 30% AMYL NITR 2%

1 FLUID OZ = 1.8 CU.IN

IF FUEL LINE IS LONG REDUCE VENTURY (INTAKE) AREA. (PLUG RESTRICTOR)



TORQUE DIAGRAM FOUR-CYCLE ENGINE

SPECIFIC WEIGHT:

METHY ALCOHOL	.78
GASOLINE	.69
OIL	.91

THREADS + HEX ON PLUGS
1/4-32 HEX 5/16 - 3/8, 3/8-24, 1/2

THREADS ON SHAFTS
6-32, 8-32, 3/16-32
1/4-28, 5/16-24 3/8-20

JETS- The principle of jets is the accelerated ejection of gases at high speeds. The higher the acceleration the lower the efficiency of the power unit. The low efficiency of a jet cannot be compared to the higher efficiency of a propeller drive, which gives a low acceleration to a great mass of air.
The ejection of ignited air- gas mixture is incorporated in the pulse jet and the ram jet. Ram jets will work only if the ram pressure of the air reaches a certain height, this requires a velocity out of reach of model speeds. A two-pound ramjet gave a 2 ounce thrust at the stand-still, which is too low for operation.

The pulse jet is the only jet available for steady high speed operation of models. Its principle is based on the resonant vibration of the inlet valve with the frequency of the gas column in the attached tube. The frequency of the ignited gas column is equal to the speed of sound divided by twice the length of the tube. The length of the tube is made longer than the valve oscillations would calculate in order to have the flame front not too close to the valves. The speed of the ejected gases is about 3000 ft.p.sec. which indicates that the speed of a model will reduce the thrust of the jet only slightly. This great difference of the ejected gases and the speed of the model gives the pulse jet a 1 to 2 per cent efficiency which is low compared with the efficiency of a gas engine driven propeller of 12 per cent of the B.T.U. supplied by the fuel.

A combination of a gas engine driven blower with an attached tube for the jet effect will give a very low thrust at higher speeds. Such blowers are effective for helicopter designs based on the ducted fan priciple. Many useful equations for basic thermal relation of fuel and velocity are given.

A pulse jet with a 3.5 pound static thrust traveling at 200 ft. per sec. delivers 1.8 net horse power. Pulse jets are only used for speed and in cities they are not favored by club members due to the occasional anti-noise law enforcement.

Rockets, selfcontained jets do not need the outside air for operation. The CO_2 capsules use the expansion of the carbon dioxide in the steel cylinders. Slow burning gun powder jets (Jetex) are lighter than the CO_2 cylinders. The speed of the exhaust gases are from 1200 to 1400 ft.p.sec. A few notes on powder jets are given.

The design of jet models is based on the thrust and the weight of the unit. The area of the wing is 40 to 45 sq.in. to one ounce of thrust and the total weight 3 to 4 times the thrust. For highly streamlined models the ratio is increased. The airfoil section is a medium cambered section, because of the absence of torque and propeller forces. The dihedral is from 6 to 8 degrees. The thrustline should pass below the c.g., adjusted for a climb to the right and a glide to the left. The adjustment is made by an aluminum rudder is the stream of the jetblast. For straight climb the thrustline is slightly below the c.g. Slow spiral climb will be the best. Twin fins are used in case the jet is above the fuselage. Sufficient fuselage area is placed below the c.g. in order to eliminate violent stability recovery. In all stability problems the recovery force should pass close to the c.g.

TESTING. Simple aerodynamic tests can be conducted by gliding a model and measuring the heights at two points and the distance between the two pints. These three dimensions will give only the gliding angle. If the time of the glide between the two points is taken, the gliding speed, the sinking speed and the aerodynamic coefficients can be evaluated. A more scientific method is to glide the model in front of a camera and taking two exposures on the same film. The camera on a tripod, a dark background and a plumbline simplifies the procedure. The time is measured by the ticking of a watch which is fives times a second. Knowing the length of the model, the area and the weight all aerodynamic coefficients can be obtained. Changing the weight and the location of the c.g. the coefficients for other angles can be determined. The advantage of this kind of tests is that the model flies in an undisturbed air, which a wind tunnel cannot supply. These glide tests may eliminate poor designs and may show the direction in which improvements can be made. A good practice for junior aero engineers.

Balsa propellers can be tested by pulling a carriage by a rubber driven propeller up on an inclined wire for about 60 feet and stopping the propeller at the top. The drag of the carriage is obtained by loading it with two different weights and timing their down travels. Two equations will give the rolling and air coefficients, which are used to deduct the drag from the thrust of the test results. The torque values of the unwinding motors are plotted first. A certain number of turns are given to the motor and the whole unit allowed to climb to the top and the time measured with a stop-watch. The power delivered by the propeller is the weight of the unit, times the height it traveled in unit time. The energy used by the propeller is the number of turns, times the average torque, times 6.28 and less the energy used to overcome the drag of the carriage. The efficiency of the propeller is then the energy delivered divided by the energy used.

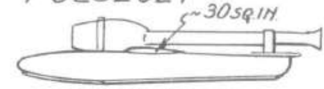
Tests conducted by this method in 1930 showed that the highest efficiency was obtained by 1.3 pitch-diameter ratio balsa propellers. The number of tests were over 2000. This method can be improved by placing flashers on two points of the wire.

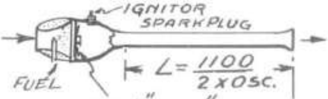
To test materials for bending strength, rectangular or round strips of the material is fastened on one end and the other end is loaded up to the breaking point. By substituting the max. load and the dimensions for the terms in the equation the bending stress (modolus of rupture) is found in lb.p.sq.in. (psi). Many other simple tests can be made if the limits of measuring dimensions and loads are known. Most of these simple tests give only comparative results and not coefficients which could be used in standard equations.

FUSELAGE EVALUATED FROM R+M #112 ~ TEST 40 FT. P. SEC. L = 24"

SQUARE		ROUND	
CROSS SECTION	CD = .133	CROSS SECTION	CD = .109
10°	CD = .208	10°	CD = .142
	dCL/da = .042		dCL/da = .0178
SIDE AREA	CD = .0202	SIDE AREA	CD = .0173
10°	CD = .0314	10°	CD = .0224
	dCL/da = .00635		dCL/da = .00227

PULSE JET





.004"–.007" THICK STAINLESS STEEL

LENGTH OF TUBE

$= \frac{\text{SPEED OF SOUND}}{\text{2 OSC. FR'Y OF VALVE}}$

DIA = 1/10 L +

THERMAL EFF. 1½–2%

EXHAUST 3000 F.P. SEC. 3400°F ABS.

2.5 LB/HOUR GAS FOR EACH LB. THRUST

GAS B.T.U. ~ 20,000/LB.

EQUATIONS:

$B.T.U. = .171(T_2 - T_1)M$, $T_1 = 520°F$

$(T_2/T_1)^{5/2} = V_1/V_2$, $THRUST = \frac{W \times a}{32.2}$

$VALVE\ FREQU. = K_1 TH/L^{3/2}$

LIMITING EFFICIENCY

$= \frac{2}{2+R}$ R = RATIO OF SLIPSTREAM

FREQUENCY VALVE VIBR

$= K \times TH/L^{3/2}$

2 LBS THRUST FOR 1 SQ. IN. TUB CROSS SECT.

$\eta = \frac{2}{1 + (C/V)}$ C EXHAUST V. V FLYING V.

$HP = MASS(C-V)V/550$

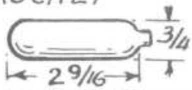
V, PULSE JET

150 GAL GAS, 20 MII.

600 HP, TUBE 10 FT LG.

VALVE FREQU. 40/SEC

ROCKET



CARBON DIOXIDE CAPSULE

1.2 OZ.

THRUST DURATION

5 OZ. ~ TEN SEC

2 OZ ~ THIRTY "

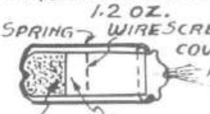
$DURATION = 140 \frac{A^{1/2} W_{FUEL}}{W_{TOT.}^{3/2}} SEC.$

EQUATION

THRUST = MASS × ACC'N

$a = 1200$ F.P. SEC.

$THRUST = \frac{W_F}{32.2} \times \frac{V}{TIME}$ IN LBS



REGULAR GUN POWDER 7000–8000 F.P. SEC.

SPAN OF MODEL

4 OZ. THRUST 32"–54"

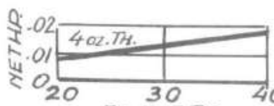
.6 OZ " 12"–20"

4 SEC LOST IN START

JETEX

W_T	W FUEL	THR	DUR.
.40	.2	.6 OZ	12 SEC
2.5	.4	4 OZ	12 SEC

AUGMENTOR TUBE INCREASE THRUST WITH LARGE JETEX ONLY



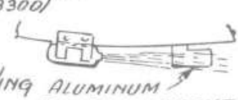
$DURATION_{SEC} = \frac{.083 A^{1/2} TH. V - (C_{DP} + C_{DO}) V\ 1.1/10^5}{(W_B + \frac{A^{3/2}}{3300})^{3/2}}$

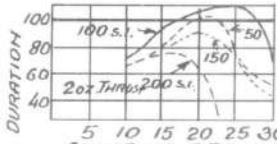
$V_S = 12\sqrt{\frac{W_{OZ}}{A_{SQ.IN}}}$ F.P.S.

TH = THRUST – OUNCES

V = CLIMB F.P.S.

W_B = WEIGHT OF MODEL LESS WING





TESTING

THRUST = W SIN α + DRAG

TESTED 1930

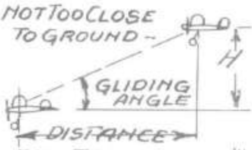
FOR ESTIMATING DRAG OF CARRIAGE

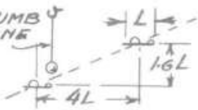
$W_C \sin\alpha = W_C \cos\alpha C_1 + V^2 C_2 = DRAG$

ROLLING DOWNWARD, USE TWO WEIGHTS TO EVALUATE C_1 AND C_2. USE STOPWATCH FOR TIMING DISTANCE L.

$\eta = THRUST\ V / W. V_{CL}$

~ NOT TOO CLOSE TO GROUND ~





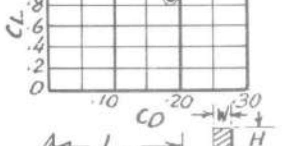
FROM THRUST.

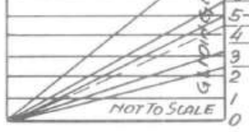
V = L/TIME

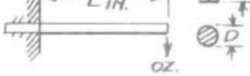
V_{CL} = H/TIME

V/ND AND η = EFFIC'Y ARE OBTAINED

ENERGY USED = No OF TURNS TIMES AVER. TORQUE × 6.28 IN FT. LBS







$LB.P.SQ.IN = \frac{.375\ LOAD}{W \times H^2}$

$LB.P.SQ.IN = \frac{.625\ LOAD}{D^3}$

L, D, H, W = IN. LOAD = OZ.

STRENGTH OF MATERIAL. A material used in model building is balsa wood. It is easily shaped and glued with any kind of cement. Its weight varies from 4½ to 20 lb. per cu.ft. and even higher. The lighter balsa is used for indoor glider wings, leading edges and details where dimensions or stresses are not specific. Fuselage stringers, spars or thin control surfaces use 8 yo 9 lb./cu.ft. balsa. Comparing balsa with spruce or bass wood shows that basswood is superior to balsa in construction of beams and longerons; The disadvantages are poor glue joints obtained in fast work. Balsa with a weight of 13 lb./cu.ft. will have the same structural weight as basswood. Basswood should be double glued. First coated with a thin surface, which is allowed to dry; then glued again and pressed together. Lately hardwood is specified for many structures with limited dimensions.

By removing all moisture from wood the strength increases by 50 per cent, but will stabilize to about 15 per cent moisture contents. There may be a method to prevent the moisture from returning without increasing the weight. Treating wood with chemicals will add only weight. Impregnating however the wood with resin and compressing it increases the strength with the same ratio as the weight increases. The added advantages are a smooth surface and a very high shear strength.

Steel wire is used for control lines. Usually the steel has a SAE 1045 specification with a 140,000 lb.per sq.in. strength. Other steel wire go as high as 250,000 p.s.i. Many modelers use such wire if only a pull test is specified. This kind of wire comes in coils and should be relieved by heating it to 350 degree F. for one hour. Steel wire landing gears are used on most of the models. Steel wire will bend by its own weight to a certain distance if supported on one end. If the same length is used the deflection will give the diameter without the use of a micrometer.

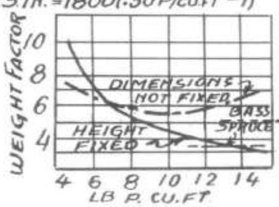
Aluminum alloys are used for engiens, landing geras, bell cranks wheel hubs, spinners and for parts that must have a high strength-weight ratio. Heat treated they may go as high as 90,000 p.s.i. The specific weight is 1/3 of that of steel.

Bending tests on wings gave the breaking poads as high as 40 times the total weight of the model. This high strength may not be neccessary, but the additional torsional forces may justify the construction. The covering adds about 20 per cent to the bending strength and 80 per cent to the torsional strength. The shrinking of the tissue prestresses the structure by 2 to 5 pounds per inch run. This stress will suport a load of 2 to 3 times the weight of the model without noticeable deflection.

Data for strength of material for models are given on page 85.

BALSA

PROPERTIES (A.S.T.M)
5 – 15 LB/CU.FT
COMP. ⊥ GRAIN 46 – 75 P.S.I
= " 450(.56#/CU.FT – 1)
BEAM: (BENDING
P.S.IN. = 1800(.30 P/CU.FT – 1)



SPRUCE = 12.8 LB. BALSA
BASS 12.5 LB. BALSA

COLUMN STRENGTH
USE COMPRESSION = TO GRAIN
THICKNESS TO LENGTH NOT OVER 1:16

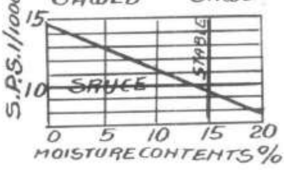
UNDER COMPRESSION BASS WILL BE LIGHTER THAN BALSA. THE DIMENSIONS MAY BE TOO SMALL.

BASSWOOD .40 SP. GRAV.
BENDING 8600 LB/SQ.IN
COMP. ∥ 3300 " " "
" ⊥ 530 " " "

SPRUCE .40 SP. GRAV
BENDING 9400 P.S.I.
COMP ∥ 5000 " "
" ⊥ 750 " " "

BIRCH .62 SP. GRAV
BENDING 14,000 P.S.I
COMP ∥ 9500 "
" ⊥ TO GRAIN 1340 "

PLYWOOD MAX P.S.I. OF WOOD.

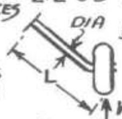


IMPREGNATION:
STRENGTH INCREASE 12%
WEIGHT " 120%
COMPRESSED TO 33%
STRENGTH INCR. 265%
SHEAR INCR. 150%

STEEL

DIA IN 1/1000

3.5	6	8	10	12	14	16
2.2	6.5	12	20	28	39	50



$DIA = .05\sqrt[3]{ARM \times W}$

$DEFL_{IN} = L^2/1800\ DIA.$

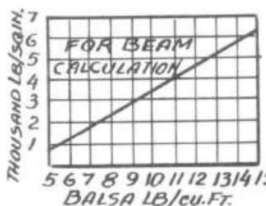
WIRE IN TENSION
$LOAD = (d/1000)^2 \times .20$

SHAFT DIA. FOR RUBBER
$DIA_{IN} = .012\ N^{1/2}$
N = NUMBER OF 1/8 × 1/30 STRANDS OR EQUIV'T.

REG. WIRE S.A.E 1045
140,000 LBS/SQ.IN
PIANO (MUSIC) WIRE
SAE 1095, 225,000 – 350,000

TISSUE
LIGHT = 1 COAT .48 LB/IN
ADD 1.6# FOR EACH COAT
HEAVY = 1 COAT 19 LB/IN
ADD 1.9# FOR EACH COAT

MICROFILM
2.5 PART COLLODION
1.5 AMYLACETATE
1.0 CASTOR OIL



MOD. of ELAS. = 229,000(.36#/CU.FT – 1)

STRENGTH OF WINGS:
FRAME WITH COVER 20% STRONGER THAN FRAME WITHOUT COVER.
COATED TISSUE 2000 – 7000 P.S.I. CROSS-SECTIONAL AREA.

DURAL

HINGES FOR FOLDING PROPS.

FLAT STOCK FOR L. GEAR
$WIDTH = \dfrac{LOAD_{OZ}\ L_{IN}}{TH^2_{IN}\ 16,000}$
EXAMPLE:
18 OZ, TH. 3/32" WIDTH 1½, L 4.5"
18 OZ, TH 1/16" WIDTH ¾ L 2½"
TAPER OVER 1" WIDTH
$DEFLECTION = L^2/105\ TH^2$

ALUM. ALLOYS:
17ST. 62,000 P.S.I. 20%
24ST 68,000 P.S.I.

AVERAGE CEMENT –
TENSION 2000
SHEAR 1000

ROLLING RESISTANCE
= WEIGHT .010 TO .015
LOCKED WHEELS
= WEIGHT × .5 TO 1.5

DURAL: 1.6 OZ. CU. IN.
BRASS: 2.245 OZ " "
MAGNESIUM: 1.04 OZ " "
DIA OF WIRE $d = .01\ N^{1/2}_{1/8}$

BALSA PROP HUB

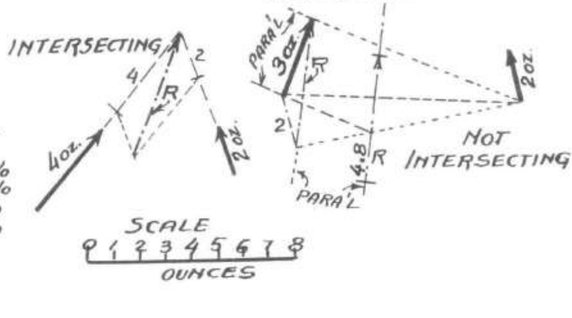
$h = .07\ N^{1/2}_{1/8''}$

CELL. ACET. – CELL. ACET. BUTYR.
SPEC. GRAV. 1.37
TEN. STR. 3000 – 9900 P.S.I.
COMP. 7500 ~ 30,400 " "
SOFTENS 140 ~ 250°F.
ELONG. 5 – 48%
$MOD.\ OF\ ELAS. = 2.0 – 3.5 \times 10^5$

RESIN IMPREGNATED AND COMPRESSED WOOD
PRESSURE 500 TO 1500 P.S.I.
SHEAR INCREASED 7 TIMES
DENSITY ONLY TWICE.

HOW TO FIND RESULTANT OF TWO FORCES

FORCES GIVEN BY DIRECTION AND MAGNITUDE IN ANY SCALE.



WEIGHT ESTIMATE. The weight of a finished model should be known in order to come within the regulation of a contest. Except for indoor models no itemized data are available. The modeler should know the expected weight of each component unit, because the requirements are only for minimum weight and maximum wing area. The weight of each part and its location about an axis will soon be realized by modelers as a decisive factor for top performances. Indoor microfilm covered models are the most trying to build because each part must have a minimum weight within 1/10,000 oz. to come close to the record time of duration.

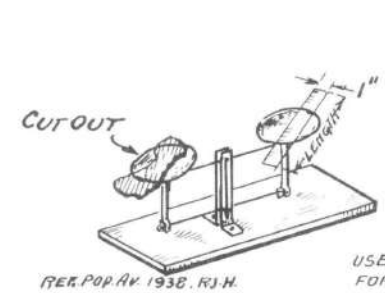
The total weight of a radio model may be the only design that can be estimated from the units that do not add strength to the structure. These units are the radio equipment and the power plant. The total weight is found by multiplying the sum of the units by 2. In the future the constant will be reduced by closer stress estimate.

A few charts give distribution of the total weights of typical designs. Usually the weight of the wing is determined first. Three nomograms give the expected weights of wings for gas, rubber and indoor models. Another nomogram is for estimating the weights of regular wood or compressed wood gas propellers. Data are given for estimating the weight of light or heavy tissue and for each doping. The weight of engines is estimated from the displacements, and the weight of a rubber motor is found by a nomogram from the length and the number of strands of 1/8 or its equivalent.

A nomogram gives the weight of 36 inch long balsa strips if the specific weight per cu.ft. of the balsa is known. It also gives the weight of metal blocks of a volume less than one cubic inch. Weights are given for control lines, telescoping aluminum and brass tubing with 1/64 in. (.015) wall thickness. For modelers who own sentive scales the weights of coins are given in grains. 437½ grains make one ounce.

The weight of a 7/8 in. diameter solid rubber wheel is .15 oz. and a 2-3/8 in. diameter wheel weighs 1 oz. One 2 in. semi-pneumatic wheel weighs .70 oz. and a 3 in. diameter weighs 2 oz. Streamlined wheels weigh less.

One 1.25 in. diameter plastic spinner weighs .30 oz.

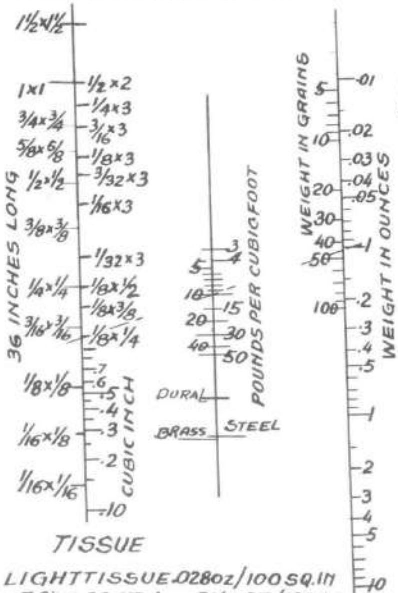


FOR FINDING AREAS OF IRREGULAR SHAPES

AREA OF CUT OUT IS EQUAL TO THE LENGTH OF STRIP OR STRIPS IF WIDTH IS 1".

USED AT THE 1940 NATIONAL, CHICAGO. FOR FUSELAGE CROSS-SECTION.

WEIGHT ESTIMATE



EQUATION FOR 36 IN LENGTHS

$W/CU.FT = 3.00\ W_{OZ}/AREA\ SQ.IN \times TH.$

$W/CU.FT = 1728\ W_{LB}/WIDTH \times TH \times LENGTH$

$W/CU.FT = 108\ W_{OZ}/WIDTH \times TH \times LENGTH$

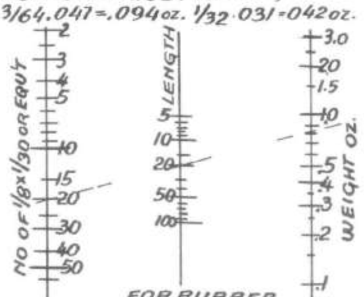
RUBBER SPEC. WEIGHT .50 oz/CU.IN

640 FT 1/8×1/30 . 320 FT 1/4 × 1/30
PER POUND

40 FT ~1/8 × 1/30, 20 FT 1/4 × 1/30
PER OUNCE

$WEIGHT_{IN\ OZ.} = L_{IN}\ N_{1/8} / 480$

STEEL WIRE P. FT IN OZ. 1/8" .125 = .672
3/32 .0937 = .38, 5/64 .785 = .26 OZ
1/16 .0625 = .168, .055 = .129,
3/64 .047 = .094 OZ. 1/32 .031 = .042 OZ.

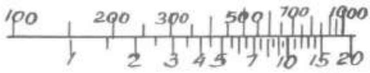


FOR RUBBER ESTIMATE

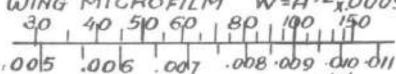
TISSUE

LIGHT TISSUE .028 OZ/100 SQ.IN
EACH COAT + .014 OZ/100 S.I.
HEAVY TISSUE .0625 OZ/100 SQ.IN.
EACH COAT + .087 OZ/100 SQ.IN.

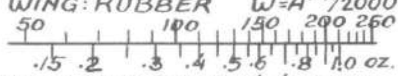
WING GAS $W = A^{3/2}/1740$ AR<8



WING MICROFILM $W = A^{1/2} \times .0009$

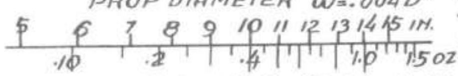


WING: RUBBER $W = A^{1.41}/2000$



215 SQ.IN. WEIGHT 2.2 OZ. BRK'G LOAD 12 LBS.
DEFLECT .20"/LB.

PROP DIAMETER $W = .004 D^2$



SILK (LIGHT) = .04 OZ/100 SQ.IN.

WEIGHT OF ENGINE = 22.5 CU.IN.
COIL: 1.2–2.5 OZ CONDENSER: .5 OZ

GAS		OUTDOOR RUBBER	
POWER PLANT	28%	FUSELAGE	45%
WING	22"	WING	24"
STAB+FIN	12"	TAIL	9"
FUSELAGE	38"	PROP+NOSE	22"

INDOOR RUBBER			MAX RUBBER WEIGHT EMPTY
STICK 24	FUSE 34		NEW WAKEFIELD SAME AS ABOVE.
WING 34	~	28	
TAIL 12	~	13	
PROP 30	~	25	

WEIGHT PER CU.IN ~ DURAL 1.6 OZ.
STEEL 4.65 OZ, BRASS 4.85 OZ.,
LEAD 6.15 OZ FUEL .415 OZ
WATER .576 OZ.

DIA. TUBE	1/16	3/32	1/8	5/32	3/16	7/32	1/4	9/32
ALUMINUM	.0036	.0056	.0076	.0096	.011	.013	.0156	.0175
BRASS	.011	.017	.023	.029	.035	.041	.047	.053

PER IN. TELESCOPING, WALL .015 APPROX.

WEIGHT OF GAS TANKS
1.5 TO 2.0 OZ. PER FLUID OUNCE
WEIGHT OF BATTERIES
.65 OZ TO 1.35 OZ P. CU.IN. OF BATT.
BELL CRANK .10 TO .30 OZ.
WITH MOUNTING

LINE CONTROL WIRE FOR TWO LINES

DIA.	.006	.008	.010	.012	.014	.016
52 FT	.162	.29	.453	.65	.89	1.16
70 FT	.218	.39	.610	.875	1.2	1.50

$W_{OZ} = d^2 L_{FT} .000087\ d_{IN\ \frac{1}{1000}IN}$, 2 WIRES

FLYING BOAT: $W_T = 100$
HULL = 28%, FLOATS 5%
ENG. 35%, WING 23%
TAIL 9%

3½" DIA WHEEL 2.75 OZ., HOLDS 10 TO 20#

TIMER: .25 – .60 OZ.

SPINNER DURAL 1¼" DIA ~ .30 OZ, 1½" ~ .50 OZ
WITH PLATE, SHAFT,

TOW LINE: WING 48%
FUSELAGE 36% STAB+FIN 16%

COMPRESSED AIR TANK 3" DIA × 24 LG – 140#/SQ.IN 15 OZ,

MED. FUEL LINE ~ .015 OZ./IN.

1/8" = .125, 3/32" = .09375, 5/64" = .078125, 1/16" = .0625, .055, 3/64" = .046875, 1/32 = .03125

COIN: CENT 50, DIME 40, NICKEL 80, 25¢ 100, 50¢ 200, DOLLAR 400 IN GRAINS

CONSTRUCTION DETAILS. Two pages, 90 and 91, are covered with many construction details, which are mostly selfexplanatory. The designs of warp-free wings will be examined of their relation to the shrinkforces, beam distribution and to the airforce resultants. The shrinkforces developed by subsequent coatings are assumed to be evenly distributed over the outline of the airfoil section, with the resultant passing through the centroid of the outline. Another characteristic of a wing to be considered is the flexural center or center of twist, which is a point on a wing, where a vertical force causes only vertical and parallel displacement of the structure without rotation.

In an ideal wing the resultant air forces pass through the a.c. which is also the flexural center and the centroid of the shrinking forces. This may be obtained with a symmetrical section by careful dimensioning the beams and locating them at the right places. To eliminate higher mathematics for determining the flexural center an approximation will give satisfactory answer if the center of balance of the cross-sectional areas of the beams is used in the estimate. To move the centroid of the shrink forces from its normal .42 position toward the leading edge, the covering and the coating is increased toward the leading edge. Due to the varying location of the c.p. on cambered sections the balance of the forces cannot be obtained. If the areas of the lateral members are closer to the top surface, the wing will warp downward; if the areas are closer to the lower surface, the wing will warp upward. The hygroscopic characteristic of the coating varies the shrink force, which can be taken from 2 to 4 pounds for one inch width. The design of standard wings, which provides a 30 to 40 load factor, gives sufficient rigidity to all unbalanced forces against warping. Double glued joints are necessary for full strength of the construction.

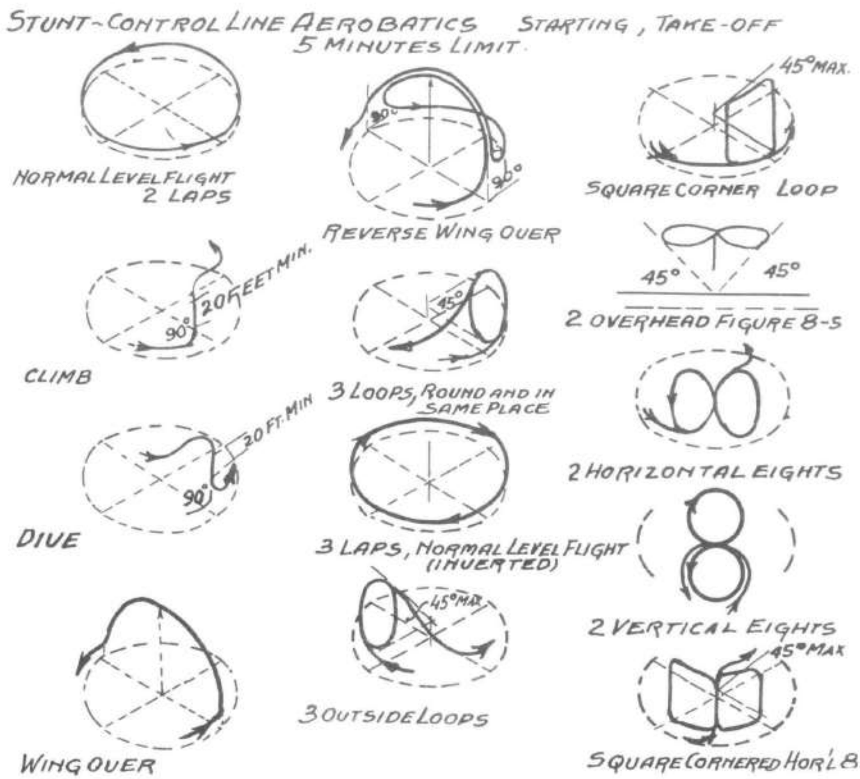
Triangulation gives exceptional high strength-weight ratio and for very thin sections the thin beams may be made of basswood instead of balsa, except if made of 12 lb./cu.ft. balsa. Ribs should be in line with the flight direction to reduce turbulency. Trailing edges should be notched and also gussetted if the sections are thin and the material is light weight balsa. The strength of such balsa is lower than that of the cement. The weakest connections are butt jointed beams, and if made of light balsa the strength is only 1/3 the strength of solid beams. For full strength use a two-strap construction or a lap joint of three times the thickness of the beam. The center section of the wing is usually sheeted in order to distribute the holddown forces exerted by the tension of the rubber bands.

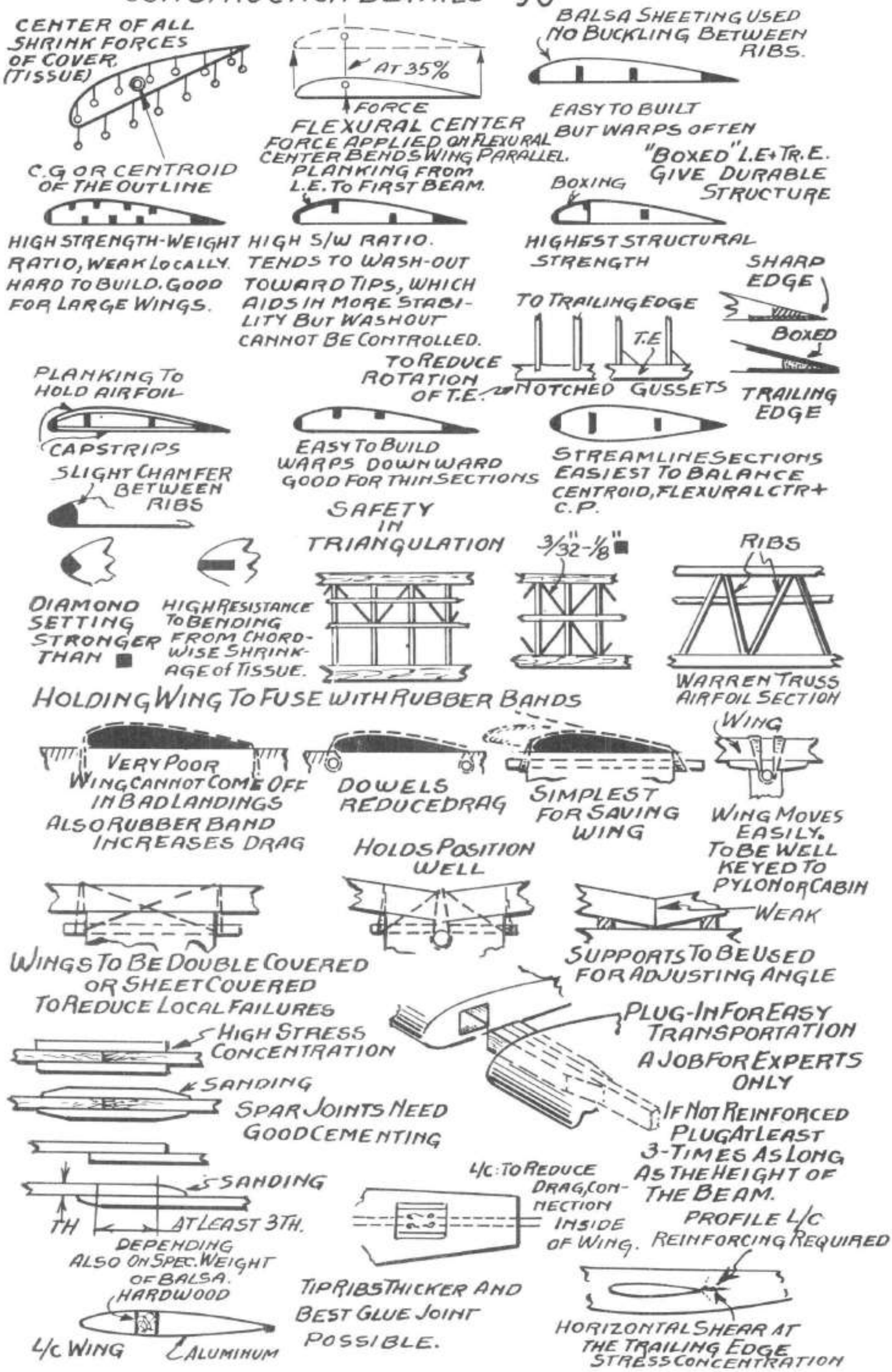
The high torsional moments of spiraling models are carried over to the wing by the platform of a pylon job or by the top of the fuselage. The width of the platform should be at least half the chord length. To prevent shifting or misalignment the wing should be connected to the platform with key and channel or pegs. If a dihedral break is at the center of the wing a close fit of platform and wing must be made, and the beam straps should be hard balsa or plywood.

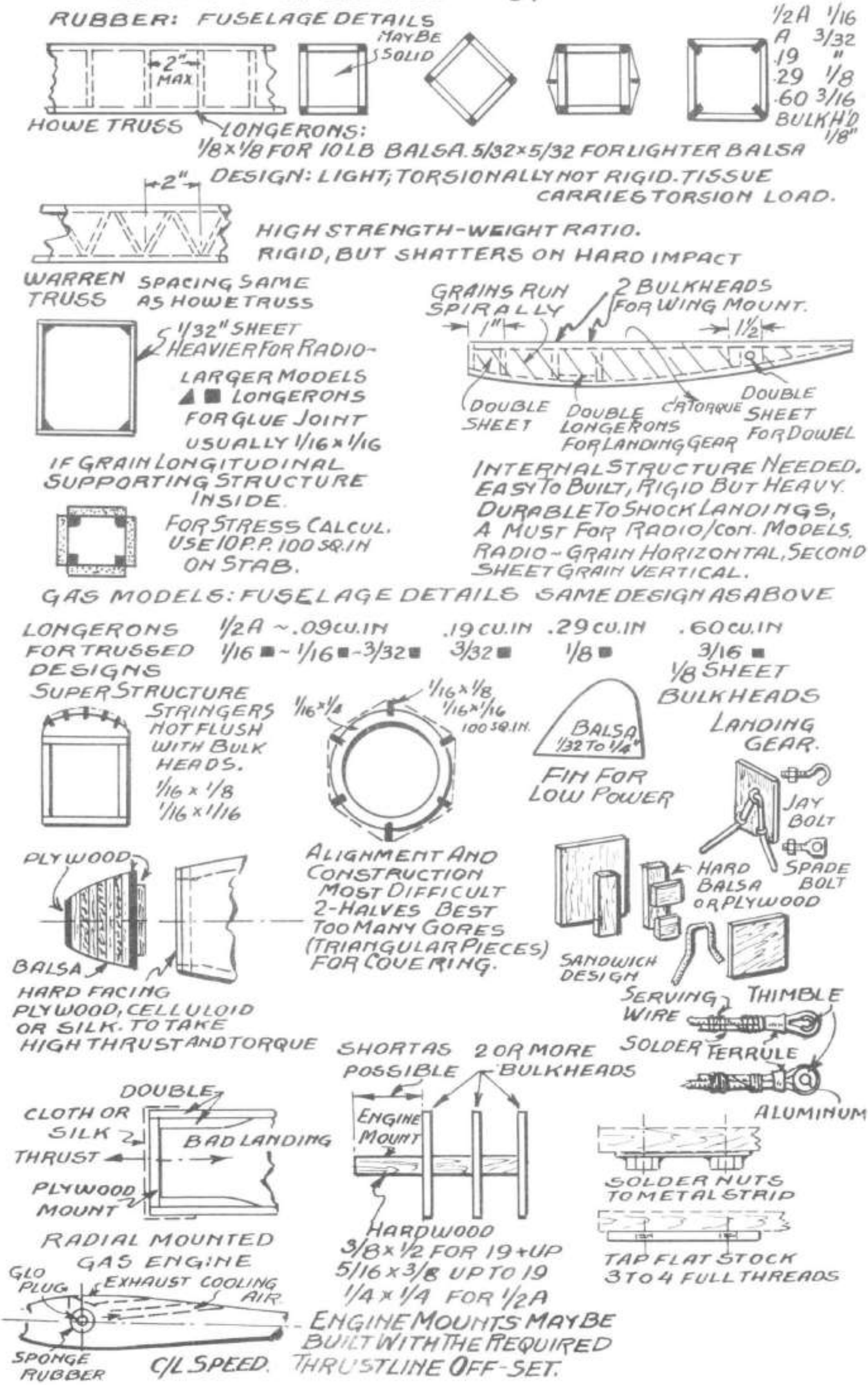
Equation for estimating the number of 1/8 rubber strands to hold down wings on platforms or fuselages is given on page 28. Use twice the estimated number if only two pegs are used.

A fuselage is designed to carry the torque of the drive, its gyro forces, the stabilizing forces of the tail and the landing loads. The framed structure follows the design of a Howe trussing, the Warren trussing or a full balsa planking. Often a combination of the three are used in one model. The Howe trussing carries vertical members and the torsional stresses are absorbed by the tissue covering, which makes it flexible. The Warren trussing has balsa diagonals, fully triangulated and tissue covered, which gives this kind of design a rigidity that may shatter in a bad landing. Planked fuselages are easily built, very strong for their weights and especially favored for models that do not require a minimum of cross sectional areas. Engine mounts should be of hardwood and the load distributed on two or more bulkheads. Solid fuselages used for control line models often split at the trailing edge of the wing, which is due to the low cleavage strength of balsa. Reinforcing this portion of the fuselage or drilling a relief hole at the trailing edge will prevent splitting. Solid balsa wings, stabilizers or fins are used to save weight and for extra saving the doping is omitted (?). Sharp corners should be avoided, smoothened and tapered to diversify the horizontal shear, which is very low for balsa.

Sharp corners should be avoided. Smoothened and tapered connections will distribute horizontal shear, which is very low for balsa.




CENTER OF ALL SHRINK FORCES OF COVER. (TISSUE)
C.G OR CENTROID OF THE OUTLINE
AT 35%
FORCE
FLEXURAL CENTER
FORCE APPLIED ON FLEXURAL CENTER BENDS WING PARALLEL.
BALSA SHEETING USED NO BUCKLING BETWEEN RIBS.
EASY TO BUILT BUT WARPS OFTEN
PLANKING FROM L.E. TO FIRST BEAM.
"BOXED" L.E.+TR.E. GIVE DURABLE STRUCTURE
BOXING
HIGH STRENGTH-WEIGHT RATIO, WEAK LOCALLY. HARD TO BUILD. GOOD FOR LARGE WINGS.
HIGH S/W RATIO. TENDS TO WASH-OUT TOWARD TIPS, WHICH AIDS IN MORE STABILITY BUT WASHOUT CANNOT BE CONTROLLED.
HIGHEST STRUCTURAL STRENGTH
SHARP EDGE
TO TRAILING EDGE
T.E
BOXED
TO REDUCE ROTATION OF T.E.
NOTCHED
GUSSETS
TRAILING EDGE
PLANKING TO HOLD AIRFOIL
CAPSTRIPS
EASY TO BUILD WARPS DOWNWARD GOOD FOR THIN SECTIONS
STREAMLINE SECTIONS EASIEST TO BALANCE CENTROID, FLEXURAL CTR + C.P.
SLIGHT CHAMFER BETWEEN RIBS
SAFETY IN TRIANGULATION
3/32"-1/8"
RIBS
DIAMOND SETTING STRONGER THAN
HIGH RESISTANCE TO BENDING FROM CHORDWISE SHRINKAGE OF TISSUE.
WARREN TRUSS AIRFOIL SECTION
HOLDING WING TO FUSE WITH RUBBER BANDS
WING
VERY POOR
WING CANNOT COME OFF IN BAD LANDINGS
ALSO RUBBER BAND INCREASES DRAG
DOWELS REDUCE DRAG
SIMPLEST FOR SAVING WING
WING MOVES EASILY. TO BE WELL KEYED TO PYLON OR CABIN
HOLDS POSITION WELL
WEAK
WINGS TO BE DOUBLE COVERED OR SHEET COVERED TO REDUCE LOCAL FAILURES
SUPPORTS TO BE USED FOR ADJUSTING ANGLE
HIGH STRESS CONCENTRATION
PLUG-IN FOR EASY TRANSPORTATION
A JOB FOR EXPERTS ONLY
SANDING
SPAR JOINTS NEED GOOD CEMENTING
IF NOT REINFORCED PLUG AT LEAST 3-TIMES AS LONG AS THE HEIGHT OF THE BEAM.
SANDING
TH
AT LEAST 3TH.
DEPENDING ALSO ON SPEC. WEIGHT OF BALSA.
L/C: TO REDUCE DRAG, CONNECTION INSIDE OF WING.
PROFILE L/C REINFORCING REQUIRED
HARDWOOD
L/C WING
ALUMINUM
TIP RIBS THICKER AND BEST GLUE JOINT POSSIBLE.
HORIZONTAL SHEAR AT THE TRAILING EDGE STRESS CONCENTRATION


RUBBER: FUSELAGE DETAILS
MAY BE SOLID
2" MAX
1/2A 1/16
A 3/32
.19 "
.29 1/8
.60 3/16
BULKH'D 1/8"
HOWE TRUSS
LONGERONS:
1/8×1/8 FOR 10 LB BALSA. 5/32×5/32 FOR LIGHTER BALSA
DESIGN: LIGHT; TORSIONALLY NOT RIGID. TISSUE CARRIES TORSION LOAD.
2"
HIGH STRENGTH-WEIGHT RATIO.
RIGID, BUT SHATTERS ON HARD IMPACT
WARREN TRUSS
SPACING SAME AS HOWE TRUSS
1/32" SHEET
HEAVIER FOR RADIO-
LARGER MODELS
▲■ LONGERONS
FOR GLUE JOINT
USUALLY 1/16×1/16
IF GRAIN LONGITUDINAL
SUPPORTING STRUCTURE INSIDE.
FOR STRESS CALCUL. USE 10 P.P. 100 SQ.IN ON STAB.
GRAINS RUN SPIRALLY
2 BULKHEADS FOR WING MOUNT.
1"
1 1/2
DOUBLE SHEET
DOUBLE LONGERONS FOR LANDING GEAR
C/R TORQUE
DOUBLE SHEET FOR DOWEL
INTERNAL STRUCTURE NEEDED.
EASY TO BUILT, RIGID BUT HEAVY.
DURABLE TO SHOCK LANDINGS,
A MUST FOR RADIO/CON. MODELS.
RADIO - GRAIN HORIZONTAL, SECOND SHEET GRAIN VERTICAL.
GAS MODELS: FUSELAGE DETAILS SAME DESIGN AS ABOVE
LONGERONS FOR TRUSSED DESIGNS
1/2A ~ .09 CU.IN
1/16 ■ ~ 1/16 ■ - 3/32 ■
.19 CU.IN
3/32 ■
.29 CU.IN
1/8 ■
.60 CU.IN
3/16 ■
1/8 SHEET BULKHEADS
SUPER STRUCTURE
STRINGERS NOT FLUSH WITH BULKHEADS.
1/16 × 1/8
1/16 × 1/16
1/16×1/4
1/16×1/8
1/16×1/16
100 SQ.IN.
BALSA 1/32 TO 1/4"
FIN FOR LOW POWER
LANDING GEAR.
JAY BOLT
SPADE BOLT
PLYWOOD
BALSA
HARD FACING PLYWOOD, CELLULOID OR SILK. TO TAKE HIGH THRUST AND TORQUE
ALIGNMENT AND CONSTRUCTION MOST DIFFICULT
2-HALVES BEST
TOO MANY GORES (TRIANGULAR PIECES) FOR COVERING.
HARD BALSA OR PLYWOOD
SANDWICH DESIGN
SERVING WIRE
THIMBLE
SOLDER
FERRULE
ALUMINUM
SHORT AS POSSIBLE
2 OR MORE BULKHEADS
ENGINE MOUNT
DOUBLE
CLOTH OR SILK
BAD LANDING
THRUST
PLYWOOD MOUNT
RADIAL MOUNTED GAS ENGINE
HARDWOOD
3/8×1/2 FOR 19+UP
5/16×3/8 UP TO 19
1/4×1/4 FOR 1/2A
SOLDER NUTS TO METAL STRIP
TAP FLAT STOCK 3 TO 4 FULL THREADS
GLO PLUG
EXHAUST
COOLING AIR
SPONGE RUBBER
C/L SPEED.
ENGINE MOUNTS MAY BE BUILT WITH THE REQUIRED THRUSTLINE OFF-SET.

DRAFTING EQUATIONS. Advancing scientific engineering is usually induced by formulating equations which conform to the laws of physics or by assuming a certain relation of design details and performances. Plotting variables of models of the same design will generally result in empirical equations that will indicate the direction in which improvements may be made. Most equations have one or more constants that are obtained by substituting the variables of well performing models. Systematic tests is the solution. In case the conception of the forces acting on the model is not recognized a few simple tests may clarify their actions. Substituting model components in equations used for airplane designs will result in equations that need only their conversion to the inch-ounce system and the establishing of the constants from top performing models. Performances aided by thermals should not be used.

The development of equations for rubber driven models is shown. The diameter is evolved from the equation used for airplane propellers with the assumption that the torque is equal to the moment one wing will give carrying the total weight of the model. The speed is presumed to be the one for the minimum power or the gliding speed at the minimum sinking speed. An equation for the minimum number of strands should be developed from the given area and weight of the model.

Many modelers observed the fact that the more rigid the wings were built the more they shattered at hard landings. The shock of any moving object is absorbed by the distance it travels times half of the maximum stress. The longer the distance a shock absorbing unit will travel the lower will be the stress imposed on the structure. Two beams of the same dimensions are compared for their shock absorbing qualities by subjecting them to the decelerated forces of their own weights. Applying standard equations only comparative values will be considered. The beam placed vertical is 4 times stronger than the one placed horizontal. The deflection of the vertical beam will be only 1/4 of that of the horizontal beam. The kinetic energy absorbed by the beams is equal to the load multiplied by the deflection, which is the same for both beams. So this was a poor trial for comparing rigidity.

Some equations are developed by empirical methods. Taking variables that are assumed to be functional to each other and plotting them on logarithmic paper. If the line connecting the points is straight the equation for the variables is a simple power function expressed by the slope of the line. To eliminate mathematical operations the equations are replaced by nomograms. To simplify the construction of alignment charts the scales are made equidistant. Five log scales will suffice for most of the charts to be made. See scales on page 25.

DEVELOPING EQUATIONS FOR RUBBER DRIVEN MODELS.

$V_{SINK} = 12 W_{oz}^{1/2}/A_{sq.in}^{1/2}$ ~F.P.S $TORQUE_{MAX} = .6 N^{3/2}$ OZ.IN. N = NUMBER ~ 1/8 × 1/30 RUBBER

$V_{GLIDE} = 95 W_{oz}^{1/2}/A_{sq.in}^{1/2}$ F.P.S. $TORQUE_{AVE} = .0585 \times N^{3/2}$ FOR ESTIMATE.

MAX TURNS/IN = $145/N^{1/2}$ POWER REQU. = EFF. REV.P.S × 6.28 × TORQUE = $W_{oz} V_{SINK} \times 12$

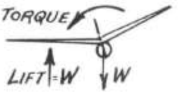
REV.P.S = $394 \times W^{3/2}/(A^{1/2} \times N^{3/2})$ DIA = $3000\sqrt[4]{HP/(REV.MIN^2 \times MPH)}$ USING C FOR ANY CONSTANT POWER/6.28 R.P.S = TORQUE, DIA = $C\sqrt{TORQUE/R.P.S \times V_{GL}}$

DIA = $C A^{1/4} N^{3/4}/W^{1/2}$ USE DIA. OF RECORD MODELS C = 1.15

PITCH = $V_{GL} \times 12/R.P.S.$ BY SUBSTITUTION $P = 2.82 N^{3/2}/W_{oz}$, $P/D = 2.5 \frac{N^3}{W^2 A}$

CONSTANTS FOR PITCH AND P/D TOO HIGH, TO BE CORRECTED.

TO ESTIMATE No OF STRANDS: ONE WING SHOULD COUNTERACT TORQUE

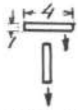


MAX. TORQUE = $.6 N^{3/2} = W \times SPAN/4$ $N = .55 W^{2/3} SPAN^{2/3}$

SUBSTITUTING N IN DIA $DIA = .635 A^{1/4} \times SPAN^{1/2}$

CONSTANT TO BE INCREASED

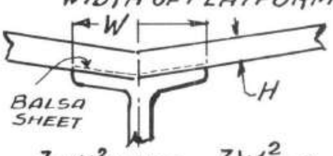
INVESTIGATION OF LANDING SHOCK.



WEIGHT	LOAD	DEFLEC.	ENERGY
4	4	1	4
4	16	1/4	4

SAME ENERGY ABSORBED FOR BOTH POSITIONS.

WIDTH OF PLATFORM FOR WING



ASSUMING STRENGTH OF WING PROPORTIONAL TO bh^2 OF THE BEAMS AND BEARING SUPPORT PROPORTIONAL TO $WIDTH^2 \times b$.

STRESS FOR THE BEAM ~10 LB. BALSA 3200 P.S.I AND FOR THE PLATFORM ⊥ TO GRAIN 50 P.S.I.

$$\frac{b \times H^2}{6} \times 3200 = \frac{bW^2}{6} \times 50$$

$$\frac{W^2}{H^2} = \frac{3200}{50} = 64$$

$\frac{W}{H} = 8$ ASSUMING $H = \frac{CHORD}{12}$

NUMBER OF RUBBER STRANDS FOR HOLDING WING TO PLATFORM PROPORTIONAL TO STRENGTH OF WING.

W = 8H, W = 8 × CH/12 = .67 CHORD = WIDTH CAN BE REDUCED IF LOAD IS DISTRIBUTED BY PLANKING OR USE FULL WIDTH AT THE PLACE OF THE BEAM.

~ EXAMPLE FOR USE of LOG. PAPER ~

1954 NATIONAL

1/2 A	85.6
A	130.19
B	136.93
C	159.37

SPEED = $CU.IN^{x} \times C$

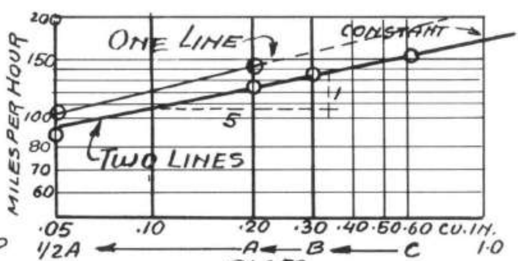
X = SLOPE 1/5 = .2

C AT CU.IN = 1 ~ 174 THEN

$SPEED_{M.P.H} = 174\ CU.IN^{.2}$

LINE INDICATES SPEED FOR 1/2A SHOULD REACH 96 M.P.H. (?)

1954 NATIONAL SPEED CONTROL LINE



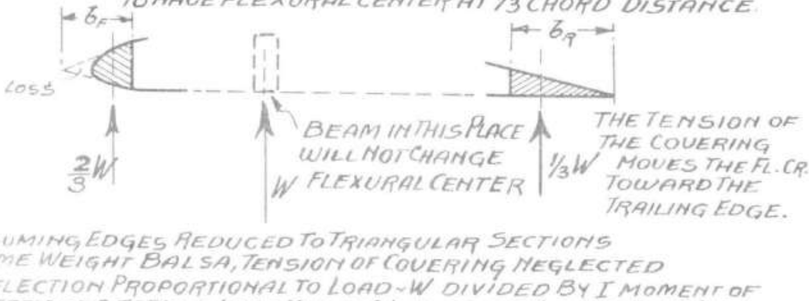
PLOTTED ON LOG. SCALES.

PROPELLER MACHINES. Manufactured propellers advanced interest in airplane models. The first ones (1929) were machine-cut balsa propellers, followed by the machine-carved gas props. All plastic or plastic impregnated, compressed wood propellers are ready for use without additional manufacturing propcesses. Illustrations show basic principles used in designing machines or fixtures for cutting or carving propellers. Balsa props are cut with a bandsaw adjustable to a P/D ratio as low as 1.2. The saw-blade is narrow with a large teeth setting. The twist or pitch is generated by an arm on the right side sliding on a sloping bar. This bar is adjustable for pitch and diameter. The thickness of the prop blank is controlled by a stop. The thickness of the tip is not sufficient to carve a cambered blade, which is overcome by gluing a small piece of balsa to the leading edge. Machine-cut basswood props were tried but the cost of broken bandsaw blades made the product nonprofitable.

Gas props are manufactured by a series of operations. Rectangular blanks with the shaft holes drilled are shaped to their outlines by a single fixture or fed to a half-automatic carver. Then two cutters are used to finish the carving. A straight one for the flat side and a profile cutter for the top side. The profile cutter is a combination of all the shapes of the blade. The pitch is formed by the sides of a box into which the blank is placed and pressed against the ball bearings next to the cutter. A better design uses the pitch bar similar to the one used for cutting of balsa props. Another machine adds a shift of the carver to the rotation of the blank for special prop designs. All carved props are hand-sanded and static-balanced. Lacquering will finish the product. Props are checked for warp and are kept off the market if pitch varies by 1 inch.

A diagram gives the pitch-diameter ratio by placing the end of a prop block on the chart and the line passing through the diagonal of the block is the ratio. To obtain the slopes of the other stations the radius of the prop is marked on a horizontal line and a vertical drawn at the axis. The slope obtained from the diagram is drawn from the tip; it will intersect the vertical at a point that is used to connect with any station along the radius to get the slope or the angle at that station. This method will give the same pitch at all station or a uniform pitch distribution.

ESTIMATING SIZE OF LEADING AND TRAILING EDGES
TO HAVE FLEXURAL CENTER AT 1/3 CHORD DISTANCE.



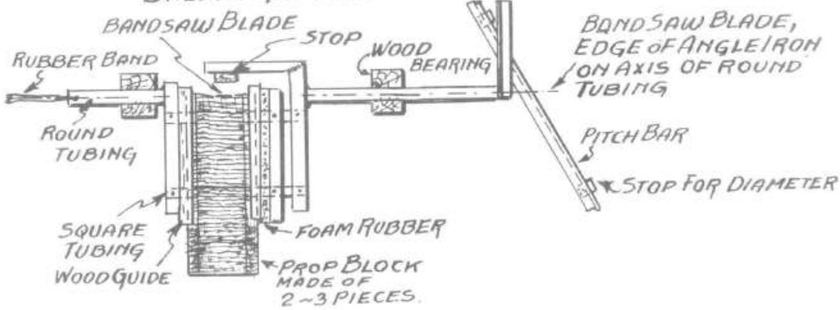
ASSUMING EDGES REDUCED TO TRIANGULAR SECTIONS
SAME WEIGHT BALSA, TENSION OF COVERING NEGLECTED
DEFLECTION PROPORTIONAL TO LOAD - W DIVIDED BY I MOMENT OF INERTIA OF SECTION. LOAD FRONT 2/3 W, REAR 1/3 W. $I = b \times h^3/48$
FRONT BEAM $H = .665b$ REAR BEAM $H = .25b$ BOTH DEFLECTION EQUAL

$\frac{2}{3} W \times b_F \times .665^3 b_F/48 = \frac{1}{3} W \times b_R \times .25^3 b_R/48 \qquad 2 \times b_F^4 \times .295 = b_R^4 .156$

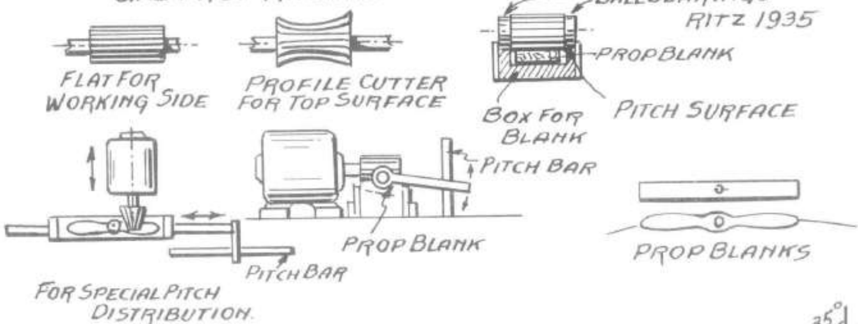
$.59\, b_F^4 = .156\, b_R^4 \qquad \frac{b_F}{b_R} = \sqrt[4]{\frac{.156}{.59}} = .264^{1/4} = .72 \qquad b_F = .72\, b_R$

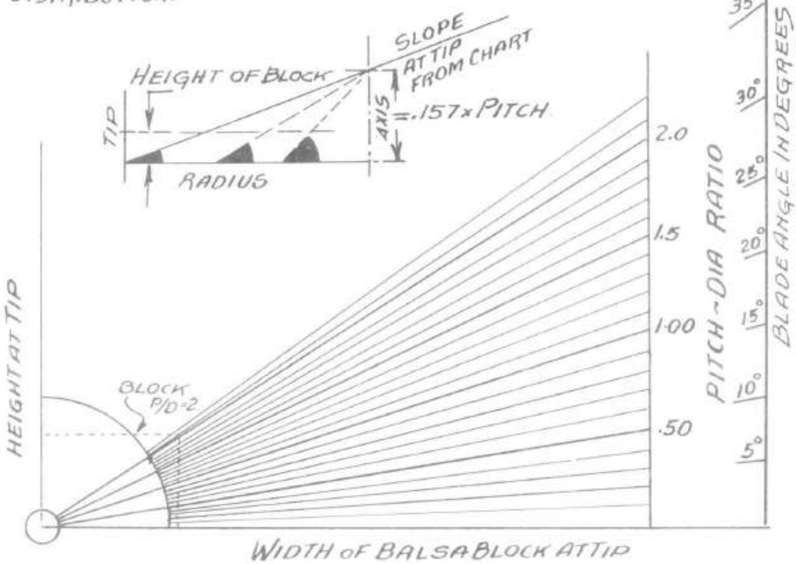
WITH NOSE RADIUS $b_F = .55\, b_R$

BALSA PROP MACHINE



GAS PROP MACHINE





Lay a straight edge across two known values and read the answer on the third scale.

MULTIPLY: one number on scale 1 with second number on scale 2, read answer on scale 3.

DIVIDE: dividend on scale 3 divisor on scale 1, read answer on scale 2.

AREA of CIRCLE: dot over scales 2,6 and diameter on scale 4, read on scale 3.

Scale 1 are squares of scale 1. Scale 2 are square roots of scale 1. Scale 6 are cubes of scale 2. Scale 2 are cube roots of scale 6. First section of scale 3 ten times the logs on scale 5; second section 100 times sines on scale 5; third section 1000 times tangents on scale 5.

~ ADDITION TO MODEL AERONAUTICS PUBLISHED JUNE 1955 ~

IN ORDER TO ADVANCE MODEL AERODYNAMICS, FIRST A THEORY MUST BE FORMULATED, WHICH IS FOLLOWED BY SYSTEMATIC EXPERIMENTS FOR VERIFICATION. ALSO EQUATIONS GIVEN TO USE IN DESIGNING MODELS.

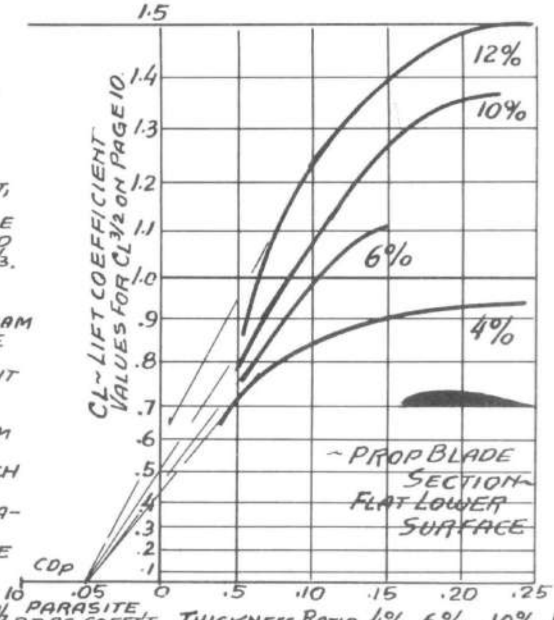
COMPARING AIRFOILS FOR STEEP CLIMBING GAS MODELS.

THE DIAGRAM ON PAGE 60 IS CORRECT FOR CLIMBING ESTIMATE, WHICH INCLUDES THE WEIGHT OF THE MODEL. DISREGARDING THE WEIGHT, THE POWER REQUIRED IS PROPORTIONAL TO THE CUBE OF THE SPEED OR THE SPEED PROPORTIONAL TO THE $CD^{1/3}$.

THE SINKING SPEED IS FUNCTIONAL $CL^{3/2}/CD$

THE ACCOMPANYING DIAGRAM IS PLOTTED ON A $CL^{3/2}$ SCALE AND THE TANGENT TO THE CURVE FROM THE CD_P POINT GIVES THE COEFFICIENTS FOR THE LOWEST SINKING SPEED OR FOR THE MINIMUM POWER REQUIRED FOR HORIZONTAL FLIGHT, SUCH AS FOR R/C MODELS.

ALWAYS ADD THE PARASITE DRAG COEFFICIENT TO THE DRAG COEFF'T OF THE AIRFOIL.



THE CLIMBING FACTOR WILL BE $1/(CD_{MIN}+CD_{PAR})^{1/3}$ AND THE FACTOR FOR SINKING SPEED. $CL^{3/2}/CD$ DIVIDING THE TWO FACTORS WILL BE INDICATIVE OF THE DURATION OF THE MODEL

THICKNESS RATIO	4%	6%	10%	12%
CD MIN	.008	.0102	.020	.025
FOR LOWEST SINKING SPEED CL	.72	.985	1.10	1.23
FOR LOWEST SINKING SPEED CD	.05	.10	.10	1.03
BEST L/D WING ALONE	27	23.2	18.8	17

$$F_{DUR.} = \frac{\text{CLIMBING SPEED}}{\text{SINKING SPEED}}$$

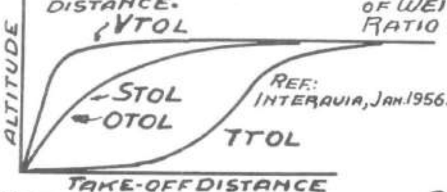
OR

$$\frac{CL^{3/2}}{(CD_{MIN}+CD_{PAR})^{1/3}(CD+CD_{PAR})}$$

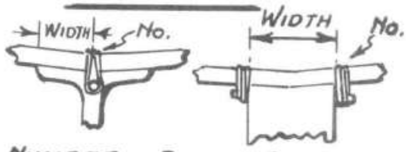
THICKNESS RATIO	4%	6%	10%	12%
FOR MIN. POWER R/C	6.1	6.5	8.5	8.9
FOR STEEP CLIMBING FREE FLIGHT.	15.8	16.5	18	21.2

ABOVE DOES NOT TAKE INTO CONSIDERATION THE VARIATION OF VELOCITIES, WHICH INFLUENCE THE COEFFICIENTS, AND THE INCREASE OF WEIGHT WITH INCREASE OF THE THICKNESS RATIO OF THE WING.

RESEARCH IN DESIGNS TO REDUCE TAKE-OFF DISTANCE.



VTOL - VERTICAL TAKE-OFF AND LANDING. USUALLY "TAIL SITTERS"
STOL - SHORT TAKE-OFF AND LANDING. USING SLIPSTREAM.
OTOL - OBLIQUE TAKE-OFF AND LANDING.
TTOL - TANGENTIAL TAKE-OFF AND LANDING. CLASSICAL 3-POINT LANDING.



~ NUMBER OF RUBBER STRANDS 1/8x1/30 TO HOLD DOWN WING. ~

MANY GOOD MODELS DISINTEGRATED IN THE AIR OR LOST A CONTEST BECAUSE INSUFFICIENT NUMBER OF STRANDS ALLOWED THE WING TO SHIFT OR LIFT FROM THE SUPPORT. THE TENSION OF THE STRANDS TIMES THE WIDTH OF THE PLATFORM SHOULD BE EQUAL TO THE MOMENT CREATED BY THE VIOLENT MANEUVERING AND THE MASS ACCELERATION OF THE MODEL. ASSUMING THAT THE MOMENT SHOULD BE TWICE THE STATIC LOAD, ACTING ON ONE SIDE ONLY. 1/8"x1/30" RUBBER 16 OZ.

$$\text{MOMENT OF WING} = W_{OZ} \times SPAN_{IN}/8$$

$$\frac{W_{OZ} \times SPAN \times 2}{16 \times 8} = No_{1/8} \times WIDTH_{IN.}$$

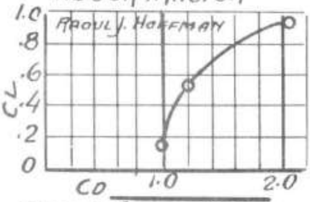
$$No \cong SPAN_{IN} \times W_{OZ.}/64 \times WIDTH_{IN.}$$

MAIN FACTORS IN DESIGNING MODELS:

1) ZERO-LIFT LINE SETTING OF WING AND TAIL 2) LOCATION OF C.G LONGITUDINALLY 3) LOCATION OF C.G IN RELATION TO THE THRUST LINE. 4) ZERO-LIFT LINE IN RELATION TO THE THRUST LINE. 5) DISTRIBUTION OF WEIGHTS 6) DIA AND PITCH OF PROPELLER 7) CHECK ON SLIPSTREAM, TORQUE AND GYRO FORCES.

KITS FOR PRODUCTION: SMALL VARIATIONS SHOULD NOT CHANGE FLYING CHARACTERISTICS.

GLIDE TEST MADE IN 1915 ~ SCHWEITZER (I.M.A.C) INDOOR TRACTOR



GLIDER TESTED BY THE MISSISSIPPI STATE COLLEGE BY GIL. HOFFMAN

CATAPULT LAUNCHING

DRAG COEFFICIENT

RE.No. 82,000 A.R. 5.

REF.: BRUNO ECK-1953.

THE TESTS INDICATE THAT THICK AIRFOIL SECTIONS ARE INEFFICIENT AT LOW SPEEDS (R.N.) WIRE WILL HELP BUT IT WILL NOT IMPROVE THIN CAMBERED SECTIONS.

GYRO MOMENTS OF PROPS

G.M. = R.P.M × D^4/(T×3.500,000) OZ.IN. OR
= R.P.M. × DIA^4/($V_{F.P.S.}$ RAD_{FT} × 22,000,000 ~ D = DIA. IN. OF PROP. T-TIME of 1 LAP IN SEC.

VALUES FOR 9" DIA, 12,000 R.P.M

VELOCITY F.P.SEC. \ RAD —	10	25	50	75	100	125	150 FT	
25	9.7	3.9	1.94	1.3	.97	.78	.65	
50	19.4	7.7	3.9	2.6	1.94	1.56	1.3	PAGE
100	38	15.5	7.76	5.2	3.88	3.1	2.6	46,
150	58	23.2	11.6	7.8	5.9	4.6	3.88	48
200	77	31	15.5	10.4	7.7	6.2	5.2	49

FOR 6" DIA, 12,000 R.P.M TAKE 20%

CENTRIFUGAL FORCE: MULTIPLY VALUES BY. W.

VELOCITY F.P.SEC \ RAD —	5	10	25	50	75	100 FT.	
25	3.9	1.9	.78	.39	.26	.19	
50	15.5	7.8	3.1	1.55	1.03	.78	PAGE
75	35	17.5	7	3.5	2.33	1.75	48,
100	62	31	12.4	6.2	4.1	3.1	54
150	140	70	28	14	9.3	7.0	
200	249	124	49.6	25	16	12	

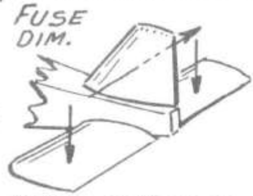
CF = V^2/(32.2R) PER UNIT WEIGHT - MPH = .682 F.P.SEC.

COMPARING GYROMOML ON A CIRCING MODEL WITH THE SAME POWER, SPEED AND CIRCLE DIAMETER. THE R.P.M. OF THE TWO ENGINES HOWEVER ARE DIFFERENT.

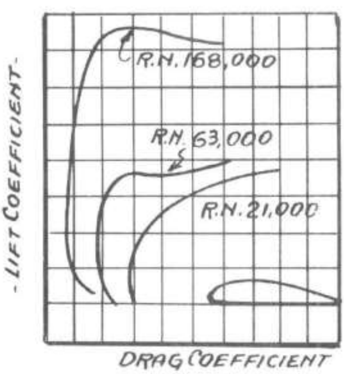
THE EQUATION ON PAGE 48 SHOWS THAT THE G.M. IS A FUNCTION OF R.P.M × D^4. THE EQUATION FOR DIA. ON PAGE 18 GIVES THE VARIATION WITH 1/$RPM^{1/2}$.

SUBSTITUTING DIA IN THE G.M. VALUE, THEN G.M = f.(1/RPM^2 × RPM) = f. 1/R.P.M. THIS INDICATES THAT THE HIGHER THE R.P.M. THE LOWER THE G.M. IF ONLY THE R.P.M VARIES THE POWER WILL BE THE SAME.

FUSE DIM.



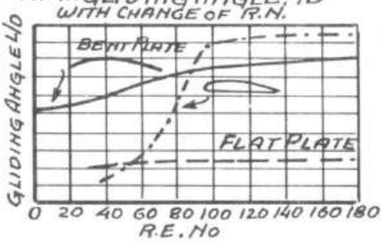
EMPIRICAL METHODS USED IN AIRPLANE DESIGNS (1930) REDUCED THE FUSE DIMENSIONING TO ASSUMING A CERTAIN LOAD PER SQ.FT. ON THE STAB AND RUDDER. THE LOADING FOR MODELS SHOULD DEPEND ON THE CATEGORY OF THE DESIGN.

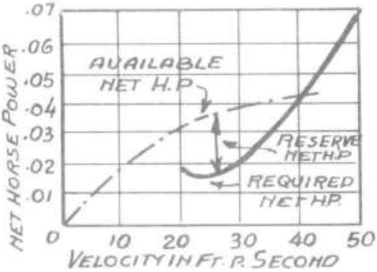


DRAG COEFFICIENT

- REF.: BRUNO ECK. -

MAX. GLIDING ANGLE, L/D WITH CHANGE OF R.N.





FOR SIMPLE ESTIMATE OF CLIMBING SPEED THE DIFFERENCE OF THE AVAILABLE AND THE REQUIRED H.P. DIFF. IS .02 H.P.

CLIMBING SPEED = $\frac{550 \text{ RES. H.P}}{W_T}$

$= \frac{550 \times .02}{4.2} = 2.6$ F.P. SEC.

OR 156 FT. PER MIN.

FUSELAGE MOMENT COEFFICIENT PER DEGREE.

$= \frac{N_{AV}}{q\, A_F\, L_F} = .0193$

A_F = FUSELAGE CROSS-SECTION

L = LENGTH OF FUSE.

T.R. 705, 1941.

ROLLING MOMENT OF WING DUE TO YAW –

$= -.33\, C_{D_{eMIN}} + .043\, C_{D_i}$

T.R. 589.

EXAMPLE:
LONG'L STABILITY ESTIMATE:

WING 640 S.I. 6,4 A.R.
STAB 234 S.I. 3.7 A.R.
36 PERCENT OF WING –
STAB DIST FROM A.C ~ 3.2
WT 32 OZ., RUDDER 38 S.I.
STAB EFF. 90, $1 - d\varepsilon/d\alpha = .80$
$C_{L_W} = .0725\alpha$, $C_{LT} = .060\alpha$
ESTIMATE FOR .65, .85 C.G. CHORD DIST. WING AREA 1.00, TAIL AREA .36, MODEL TO BE TRIMMED AT 15° ABS.

FOR .65 CG DIST., TAIL DIST 2.85
HIGH WING, LOW C G. CMAC = –.047
AC AT .30 C SEE PAGE

ANG.	0	5°	15°	20°
CMAC	–.047	–.047	–.047	–.047
–.0256α	0	.128	.384	.512
WING	–.047	+.081	+.337	+.465

TAIL MOMENT –.0445α + .055 SET
THIS MOM TO BALANCE M_W +.337
AT 15° ABS. .337 – .667 + .055 S = 0
SETT = (.667 – .337)/.055 = 6°

STAB ~	0°	5°	15°	20°
.055 SET	.330	.330	.330	.330
–.0445α	0	–.222	–.667	–.890
	+.330	+.108	–.337	–.560

ADDING WM + TM ~

ANGLE	0	5	15	20
WM =	–.047	+.081	+.337	+.465
TM =	+.330	+.108	–.337	–.560
	+.283	.189	0	–.095

ESTIMATE FOR CG .85%

WING α	0	5	15	20
A.C.	–.047	–.047	–.047	–.047
LIFT .040α =	0	.200	.600	.800
TOTAL	–.047	+.153	+.553	+.753

TAIL DIST 3.2 – .55 = 2.75
TAIL – –.413 + .051 SETTING
TO FIND SETTING FOR TRIM AT 15°
.553 – .0413 × 15 + .051 SET = 0
.553 – .620 + .051 SET. = 0
SET = .620 – .553/.051 = 1.31°
CM TAIL = +.067 – .0413α

ANGLE	0	5	15	20
SETT	+.067	+.067	+.067	+.067
–.0413α	0	–.206	–.620	–.826
TAIL	+.067	–.139	–.553	–.759
ADD WING	–.047	+.153	+.533	+.753
TOTAL	+.020	+.014	0	–.006

THE RESUL'T MMT COEFF. ARE TOO LOW TO STABILIZE THE EFFECT OF PROP + FUSE TO BE CONSIDERED FOR FREE FLIGHT GAS MODEL. IT MAY GLIDE IN CALM AIR, BUT MAY SPIN IN A FAST THERMAL.

– HOW EQUATION FOR DIA. OF RUBBER DRIVEN PROPS HAS BEEN DEVELOPED. –

EQUATION FOR DIA. ON PAGE 18

$$DIA = 3000\sqrt[4]{\frac{HP}{RPM^2 \times V}}$$

ALL CONSTANTS ARE OMITTED AND ONLY FUNCTION (f) OF FACTORS INVOLVED ARE CONSIDERED.

$$HP = W_T \times \frac{D}{L} \times V = f\left(\frac{W \times W^{1/2}}{A^{1/2}}\right) = f\,\frac{W^{3/2}}{A^{1/2}}$$

ALSO HP = f(TORQUE × R.P.S)

TORQUE = $f\,N^{3/2}$ (PAGE 77)

$$DIA = \sqrt[4]{\frac{TORQUE}{R.P.M. \times V}} = f\sqrt[4]{\frac{N^{3/2} A^{1/2}}{R.P.S \times W^{1/2}}}$$

$R.P.S = f\left(\frac{W^{3/2}}{A^{1/2} N^{3/2}}\right)$ THEN

$$DIA = f\sqrt[4]{\frac{N^{3/2} \times A^{1/2} A^{1/2} N^{3/2}}{W^{1/2} W^{3/2}}}$$

$$DIA = CONST.\ \frac{N^{3/4} A^{1/4}}{W_T^{1/2}}$$

SEE PAG 18

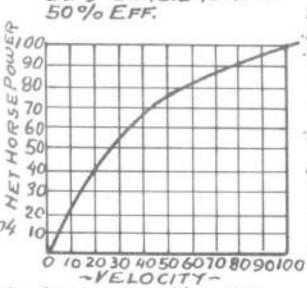
$$V = \sqrt{\frac{7560\, W_{oz}}{A\, C_L}} = \frac{24.6}{\sqrt{C_L}}$$

$$HP = \frac{W \times V}{8800\, C_L/C_D} =$$

$$H.P. = \frac{V}{205 \times C_L/C_D}$$

~ PROP CALCULATION ~

.09 CU. IN. MAY DELIVER .11 B.H.P. AT 14.000 R.P.M. TO PREVENT DEAD STOP IN LOOPS THE ENGINE IS RUN AT 9300 R.P.M., DELIVERS .08 B.H.P. THE SPEED IS 35 F.P.S. USING DIAGRAM ON PAGE 20 THE PROP WILL BE 9" DIA, 3.2 P/D AND 50% EFF.



CHANGE OF NET H.P. WITH VARIATION IN SPEED

SIMPLIFIED EQUATION FOR FIN AREA. ~

EQUATION GIVEN ON PAGE 104 CAN BE SIMPLIFIED BY ASSUMING THAT FOR THE SAME CATEGORY OF DESIGN, α_W, AR_{FIN}, AR_{WING} TO BE FIXED. THEN THE RATIO

$$\frac{\text{SPAN} \times \text{WING AREA}}{\text{FIN AREA} \times \text{DIST. TO FIN}}$$

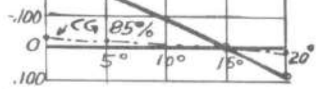
SHOULD BE CONSTANT. ~

CONSTANTS
FOR F.F. GAS 46
" F.F. RUBBER 20
" TOW GLIDER 30
" INDOOR GLID. 14

POWER CURVE

FOR 550 SQ. IN. 44 OZ. TOTAL WEIGHT .09 CU. IN. 9300 R.P.M. CLARK Y AIRFOIL SECTION. THE LIFT OF THE TAIL NOT CONSIDERED. USE COEFFICIENTS FOR CLARK Y ON PAGE 7 AND ADD PARASITE DRAG COEFF'T CD_{PAR} = .05 TO CD COEFF'TS.

ANGLE	–2°	0	2	4	6	8	16
CL	.253	.395	.54	.683	.828	.97	1.25
V —	48.7	39.2	33.4	29.7	27	25	22
$C_{D_0} + CD_p$	.070	.074	.081	.092	.105	.123	.210
CL/CD	3.6	5.35	6.65	7.45	7.9	7.9	6.95
HP —	.066	.035	.024	.019	.017	.015	.015



THE SLOPE OF THE STABILITY COEFFICIENT SHOULD BE INDICATIVE FOR EACH BASIC DESIGN.

EQUATIONS FOR ALTITUDE
HEIGHT IN FEET = H
$H = 32{,}000 \log_{10} \frac{HP\ AVAILABLE}{HP\ REQUIRED}$
$\frac{NET\ H.P.\ AVAIL.}{NET\ HP\ REQU'D} = \frac{31.000 - .43H}{31.000 + .37H}$
REQUIRED H.P. IS MIN. H.P. FOR HORIZONTAL FLIGHT- A PLANE FLYING AT ITS MIN. HP AT VELOTIES, THAT ARE INDIRECTLY PROPORTIONAL TO THE SQUARE ROOTS OF THE AIR DENSITIES. THE POWER OF INTERNAL COMBUSTION ENGINES IS IN DIRECT PROPORTION TO THE AIR DENSITY. COMBINING THESE RELATIONS FOLLOWING EQUATION IS FORMULATED

$$\frac{MAX.\ NET\ HP}{MIN.\ NET\ HP} = \left(\frac{DENSITY\ GROUND}{DENSITY\ ALTITUDE}\right)^{3/2}$$

FROM WHICH THE CEILING, VELOCITIES AND POWER REQUIRED AT ANY ALTITUDE LEVEL CAN BE CALCULATED IF THE POWER CURVE AT ANY OTHER LEVEL IS KNOWN.
THE ANGLE OF ATTACK DURING THE ESTIMATE DOES NOT VARY.
THE EQUATION DOES NOT INCLUDE THE EFFICIENCY OF THE PROPELLER.

REF.: N.A.C.A T.R 368 (NOT 638 AS GIVEN ON PAGE 2.)

STATIC THRUST / BHP
FULL LINES N.A.C.A WOOD PROPELLERS
BLADE WIDTH DIA/10,
DOTTED LINES METAL
PITCH AT .75 R_E

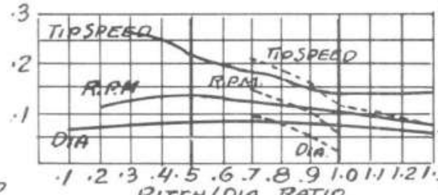
$C_T^{5/}/C_P = \frac{T^{3/4}}{BHP} \frac{n^2}{10^2}$

n = REV. P. SEC. SPECIFIED

$C_T^{3/2}/C_P$ = THRUST$^{3/2}$/BHP × 19 × DIA

DIA. SPECIFIED.

SUBSTITUTE THIS DIAGRAM + EQUATIONS FOR THE ONES ON PAGE 75.



$C_P = \frac{P}{n^3 D^5 \rho}$

$C_T = \frac{T}{n^2 D^4 \rho}$

$\frac{C_T}{C_P} = \frac{THRUST}{BHP} \frac{n \pi DIA}{1727}$

$n \pi D$ = SPEC TIP SPEED

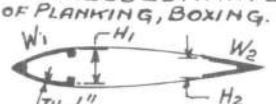
P/D	.3	.5	.7	.9	1.1	1.3	WOOD
C_T	.0015	.100	.142	.168	.179	.181	PROPS
C_P	.0234	.0477	.077	.1032	.1242	.1458	

EQUATION ON PAGE 62
$\frac{HP.A^{1/2}}{W^{3/2}} = \frac{CD}{C_L^{3/2}} \frac{1}{19}$ IS IN THE FT-LB SYSTEM. THE H.P IS THE NET H.P. DELIVERED BY THE PROPELLER.
CONVERTING THE EQUATION INTO THE OZ.-SQ.IN. SYSTEM
$\frac{101\ H.P. A^{1/2}}{W^{3/2}} = \frac{CD}{CL^{3/2}}$
ASSUMING THAT THE H.P DELIVERED BY AN ENGINE IS PROPORTIONAL TO THE CU.IN. DISPLACEMENT. THE ACTUAL HP VARIES FROM 1.2 CU.IN TO 2.00 CU.IN. THE LOWER VALUE SHOULD BE USED. RADIO CONTROLLED MODELS USE ABOUT 3/4 R.P.M. OF THE REVOLUTIONS OF MAX. POWER DELIVERY. THIS PREVENTS THE MODEL FROM FLYING ON GYROSCOPIC FORCES.
THE BASIC EQUATION WILL BE
CU.IN. = $C \times W^{3/2}/A^{1/2}$, C A CONSTANT, WHICH VARIES WITH DESIGN.
ON PAGE 24 FOR CARGO THE CONSTANT IS GIVEN AS .006. USING THE 1955 CHAMPION CARGO MODEL: A = 600 S.IN. W_T 55 3/4 OZ .049 CU.IN.
THE CONSTANT WILL BE .003.
THE ENERGY STORED IN RUBBER IS IN DIRECT PROPORTION TO THE DURATION.
USING THE SPECIFICATIONS OF A RECORD INDOOR MODEL W_T = .0686, W_{RUB} = .0396 AREA = 148 S.IN. THE EQUATION WILL BE:
$DURATION = 70 \frac{W_R \times A^{1/2}}{W^{3/2}}$ SEC.

9.6" DIA 5" PITCH PROP TESTED
AT GOTTINGEN LAB 1932.

R.P.M.	13,170	10,490	8820	7430
MAX. EFF	70.7%	69.4%	61%	57%
AT V/nD	.49	.48	.46	.45
R.N AT 75% R	141,000	112,000	94,000	80,000

STRESS ESTIMATE OF PLANKING, BOXING.



THICKNESS 1/16, 7-8 LB. BALSA 2000 LB. P. SQ IN. PAGE 85. BENDING STRENGTH.
MOM. = WIDTH × TH × H/3 LB. IN.
APPROX MAX MOM. AT ROOT OF WING. SPAN 60", W_T 4.2 POUNDS.
$MOM_{MAX} = W_T \times SPAN/8$
MOM = 4.2 × 60/8 = 31.5 LB. IN.
ASSUMING THAT 2/3 OF THE LOAD IS CARRIED BY THE FRONT BEAM AND 1/3 BY THE REAR BEAM AND A LOAD FACTOR WITH A MARGIN OF SAFETY TO BE 10.
THE MOMENT CARRIED BY THE BOXING.
$MOM = \frac{1}{3} W \times TH \times H \times$ LBS/S.IN.
IF THE WIDTH IS NOT KNOWN THE SLOPE H/W IS USED
FRONT BEAM H/W = .5
REAR BEAM H/W = .35
$MOM_F = 1/3\, W \times .5 W \times TH \times 2000$
$MOM_R = 1/3\, W \times .35 W \times TH \times 2000$

SEE PAGE 84 ON WING TESTS.

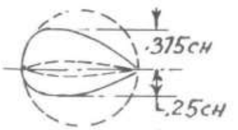
HOW TO ESTIMATE THE LOCATION OF ANOTHER ENGINE IF THE ENGINE ON SPOT IS EXCHANGED, KEEPING THE SAME C.G.

$NEW\ DIST = \frac{WEIGHT_1 \times DIST_1}{WEIGHT\ NEW\ ENG.}$
WEIGHT = ENGINE + PROP
W + D. IN OUNCES + INCHES.

FRONT BEAM M = 10 × 21
$210 = \frac{1}{3} W^2 \times .5 \times 1/16 \times 2000$
$W^2 = 10.08$ W = 3.18"
REAR BEAM M = 10 × 10
$W^2 = 3 \times 100/(.35 \cdot 1/16 \times 2000)$
$W^2 = 6.85$, W = 2.62"
STRINGERS ADDED TO PREVENT CRINKLING

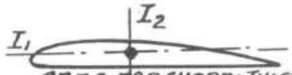
THE LOAD FACTOR FOR STEEP CLIMBING F.F. MODELS SHOULD BE GREATLY INCREASED.

EFFECTIVE A.R. WITH ENDPLATES.



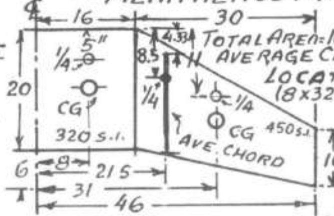
WITHOUT 3.54 A.R, WITH DISC EFF. A.R= 4.98, WITH TRAPEZOIDAL ENDPLATES 4.50

WITHOUT 6.00 A.R. WITH DISC 7.8 WITH TRAPEZOIDAL ENDPLATES 6.8



AREA = .738 CHORD x THICK
$I_1 = .0464$ CHORD x THICK^3
$I_2 = .0464$ CHORD^3 x THICK

~ MEAN AERODYNAMIC CHORD ~



TOTAL AREA $= 16 \times 20 + (20+10)/2 \times 30 = 770$
AVERAGE CHORD $= 770/46 = 16.75$
LOCATION OF AV. CHORD
$(8 \times 320 + 31 \times 450)/770 = 21.5"$
TO FIND LOCATION OF THE L.E OF AV. CHORD USE 1/4 POINTS. ADDING DISTANCES TIMES AREAS DIVIDED BY TOTAL AREA GIVES 1/4 POINT
$(5 \times 320) + (11 \times 450)/770 = 8.5$

DIST OF L.E of AV. CHORD $= (8.5 - 16.75/4) = 4.33"$

THE HEIGHT of THE AVERAGE CHORD IN THE ELEVATION IS THE SUM OF THE AREAS TIME THE HEIGTS OF THE CG-S DIVIDED BY THE TOTAL AREA.
ASSUMING A 14° TIP DIHEDRAL THE HEIGHT EQUALS $(320 \times 0 + 450 \times 15 \sin 14°)/770 = 2.1"$

DETHERMALIZER

HOW SINKING SPEED WITH TAIL-UP IS COMPARED:
USE DIAGRAM FOR 360° ON PAGE 8 AND ADD CD PAR .15 TO THE CD COEFFICIENTS.
FOR SINKING SPEED SEE PAGE 65
$V_S = W^{1/2}/.0012 A^{1/2} \times (C_D \cos^{3/2} B / C_L^{3/2})$
FOR SMALL B ~ GLIDING ANGLE ~ $\cos^{3/2} B \cong 1.00$

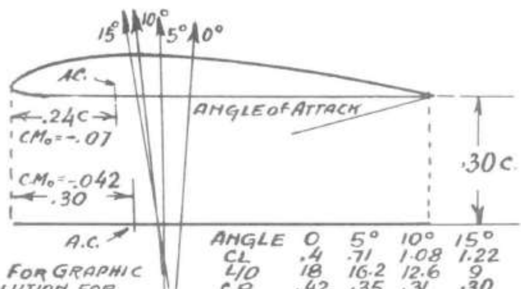
IN GLIDE W BALANCES R.
THEN $\underline{L = R \cos B}$ OR $C_L/\cos B = C_R$
$C_R = \sqrt{C_L^2 + C_D^2}$ $\quad V_S \cong C_D / C_R^{3/2}$

ANGLE	40°	50°	60°	65°	70°
CL	.8	.72	.60	.50	.40
CD	.75	.85	.95	1.08	1.15
CD+.15	.90	1.00	1.10	1.23	1.30
CR	1.2	1.23	1.25	1.39	1.47
$C_R^{3/2}$	1.315	1.365	1.4	1.64	1.78

$V_S = f \dfrac{.90}{1.315} \quad \dfrac{1.00}{1.36} \quad \dfrac{1.10}{1.40} \quad \dfrac{1.23}{164} \quad \dfrac{1.30}{1.78}$

$= f .665 \quad .735 \quad \underline{.785} \quad .750 \quad .730$

THE HIGHEST VALUE .785 - OR 60°



ANGLE	0	5°	10°	15°
CL	.4	.71	1.08	1.22
L/D	18	16.2	12.6	9
CP	.42	.35	.31	.30

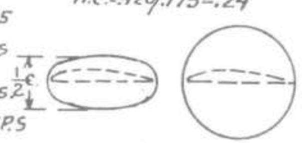
FOR GRAPHIC SOLUTION FOR A.C. SEE PAGE 12.
EQUATION FOR A.C.
$C_P = AC - CM_0/C_L$
AND COEFF. FOR 0°+10°.
$.420 = AC - CM_0 \times 2.50$
$-\underline{(.310 = AC - CM_0 \times .925)}$
$.110 = -CM_0 \times 1.575$
$CM_0 = -.110/1.575 = -.070$
$.420 = -2.50 \times .070$ A.C.
$A.C. = .420/.175 = .24$

C.P. ON REFERENCE LINE .30C BELOW BASE LINE USED FOR PYLON WING

ANGLE	0	5	10	15
C.P.	.404	.357	.338	.34

FOR 30% BELLOW BASE
$.404 = A.C. - C.M_0 \times 2.500$
$-\underline{(.338 = A.C - CM_0 \times .925)}$
$.066 = 0 - CM_0 \times 1.575$
$CM_0 = -.066/1.575 = -.042$
$AC = .404 - .105 = .299$
OR ~ .30

BALANCE $A_W .45 \times 1.25 = .35\ A_W \times C_V\ 2.25$
$C_V = 1.25 \times .45/(.56 \times 2.25) = .446$
$C_L = C_V/\sin 30° = .446/.50 = .892$
THE AR of THE STAB 3.6 $C_L/\alpha = .0595$
$.892/.0595 = 15°$
DEDUCTING 15° FROM 57° GIVES THE POP-UP ANGLE of 42°
EXAMPLE: 8 oz. 200 s.i.
THE SINKING SPEED IS 13.5 F.P.S.
$V_S = \sqrt{\dfrac{7560 \times 8 \cos B}{200\ C_L}} \times \sin B = 13.5$ F.P.S

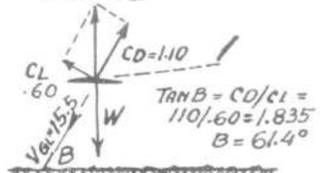


CORRECTION FOR AIRFOIL CHARACTERISTICS GIVEN FOR INFINITE ASPECT RATIO.

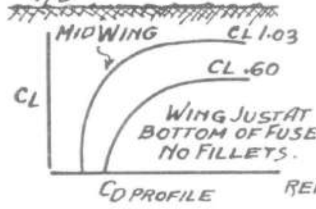
AR = 2.66
$C_L/\alpha = .057$ STALLS AT 20°
WITH 1/2 CH ENDPLATES
$C_L/\alpha = .062$, STALLS AT 18.3°
WITH FULL DISK ENDPLATES
$C_L/\alpha = .0645$ STALLS AT 17°

AR = 1.33
$C_L/\alpha = .035$, STALLS AT 30°
WITH 1/2 CH. ENDPLATES
$C_L/\alpha = .053$, STALLS AT 30°
WITH FULL DISK ENDPLATES
$C_L/\alpha = .056$, STALLS AT 25°
LARGE GAINS ON LOW ASPECT RATIOS ONLY.

N.A.C.A REPORTS ARE USUALLY PLOTTED ON TWO DIAGRAMS.
EXAMPLE: CL = 1.0 AT 12°, TO BE CORRECTED FOR AR = 6.
THE INDUCED ANGLE (SEE PAGE 14) $= 18.24 \dfrac{C_L}{A.R.}$
$= 18.24 \times 1/6 = 3.04°$, WHICH IS ADDED TO 12°
$12 + 3.04 = 15.04°$.
CD = .02 FOR INF. AR, WHICH IS THE PROFILE DRAG. $C_D = C_{D_0} + \dfrac{C_L^2}{\pi \times AR}$
$C_D = .02 + 1/\pi \times 6 = .073$.
$C_L^2/\pi AR$ IS THE INDUCED DRAG COEF'T.



$\tan B = C_D/C_L = 1.10/.60 = 1.835$
$B = 61.4°$



REF. E.P. WARNER 1936

UP-GUST

THE RESULTANT OF A CIRCLING MODEL IS ASSUMED TO PASS THROUG THE PLANE OF SYMMETRY

THE OUTER WING TRAVELS FASTER THAN THE INNER WING, WHICH HAS A LARGER ANGLE of ATTACK.

FOR EXAMPLE:
THE VELOCITY OF THE OUTER WING IS 20 F.P.S. AND THE ANGLE of ATTACK 7°, AND THE VELOCITY of THE INNER WING 15 FPS AND AN ANGLE OF ATTACK 12.5. THE MODEL ENCOUNTERS A 2 F.P.S UP-GUST. THE ANGLE ON THE OUTER WING INCREASES BY 2/20 OR 5.6° AND THE INNER WING BY 2/15 OR 7.6°. THE LIFT ON THE INNER WING 7.6/12.5 OR BY 61%, ON THE OUTER WING THE INCREASE IS 5.6/7° OR 80%. THE MODEL WILL INCREASE ITS BANK, SIDE SLIP AND IF STABLE IT WILL TURN SLIGHTLY TOWARDS THE DISTURBANCE.

~SWEEP BACK~ MAX. CL REDUCES 1/2% FOR EACH DEGREE. CD MIN DECREASES BY 1% FOR EACH DEGREE.

B.T.U.

SMOKELESS POWDER	2000
BLACK POWDER	1132
C + O TO CO_2	3832
$2H + O_3$ TO H_2O = 183,000	

2H ATOMIC HYDROGEN
O_3 TRIATOMIC OXYGEN~OZONE UNSTABLE.
STABILIZER TO BE INVENTED

~FLYING BOATS~

STANDARD EQUATION USED:

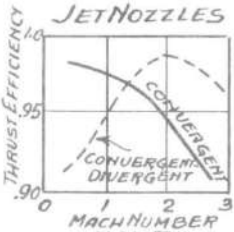
$$\text{BEAM LOADING} = \frac{W_{LBS}}{64B^3_{FT}} = 1.0$$

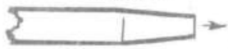
NEW FUNCTION OF FORE BODY LENGTH L AND BEAM (WIDTH) B IS USED IN EQUATION

$$\frac{W}{64L^2B} = .0836$$

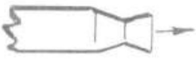
THIS RESULTS IN A LONG NARROW HULL COMPARED WITH THE EARLIER BEAMY ONE.
FULLY FAIRED STEP, LONG AFTER BODY, WHICH PREVENTS PORPOISING.
REF.: AEROPLANE 1-6-1956
EQUATION FOR FLYING BOAT MODELS TO BE FORMULATED.

JET NOZZLES





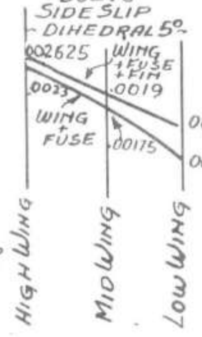
CONVERGENT



CONVERGENT-DIVERGENT.
REF.: AIR PICTORIAL ~DEC. 1955.~

MAX LIFT + MAX DRAG COEFFICIENTS
$C_{L MAX} = 3.0$ (?)
$C_{D MAX} = 4.0$ (?)
AS GIVEN BY THE FOKKER AIRCRAFT CO
BASIC SECTION NACA 64-421 21% THICK
USED ON THE AIRPLANE "F27 FRIENDSHIP"
A.R = 12
REF: AIR PICTORIAL DEC. 1955.

T.R. 730
ROLLING MOMENT DUE TO SIDE SLIP
~DIHEDRAL 5°~



C.G. LOCATION WITH RELATION TO THE THRUST LINE FOR SLOW-CLIMBING MODELS - R/C.

THREE POSITIONS:

1.) C.G. THROUGH THE THRUST LINE

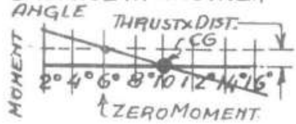
RUDDER ONLY.

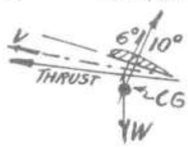
A BALANCE FOR HORIZONTAL FLIGHT IS OBTAINED AT MIN H.P. IF THE C.G IS LOCATED FOR MIN SINKING SPEED. THERE IS HOWEVER AN EXESS OF POWER FOR TAKE-OFF AND CLIMB.

THIS REDUCES THE ANGLE OF ATTACK. ADDING A NOSE-UP UNTIL THE ANGEE of ATTACK IS RESTORED. PART OF THE WEIGHT IS LIFTED BY THE THRUST.

THE THREE MAIN FORCES MEET AGAIN AT THE C.G, IF THE SLIPSTREAM WILL NOT AFFECT STABILITY.

2) THE C.G IS BELOW THE THRUST LINE, WHICH CREATES A DIVING MOMENT THAT GIVES A BALANCE AT ANOTHER ANGLE



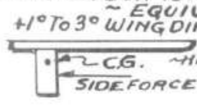


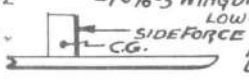
3) THE CG IS ABOVE THE THRUST LINE. THIS CREATES A CLIMBING MOMENT THAT GIVES A BALANCE AT A LARGER ANGLE AS SHOWN IN THE DIAGRAM.

BEST GLIDE AT 10° WITH THRUST IT WILL BALANCE WITH THE RESULTANT OF 14°.

ROLLING MOMENT OF FUSELAGE DUE TO SIDE FORCE
NO EXACT ANALYTICAL METHOD AVAILABLE.
TESTS INDICATED THE MOMENTS ARE SMALL IN COMPARISON TO THE EFFECT OF WING DIHEDRAL.

~ EQUIVALENT EFFECT ~
+1° TO 3° WING DIHEDRAL — -1° TO -3° WING DIHEDRAL.





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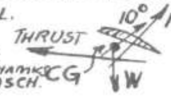


FORCES NOT TO SCALE.

R.P.M. FOR MAX. THRUST. – EXAMPLES FOR PAGE 69

B.HP + R.P.M. TAKEN FROM AER. MOD.

B.H.P	R.P.M.	DIA	V/nD	EFF	NET.HP	FACTOR $\times 10^4$
.51	13,000	9.2	.56	.58	.296	.280
.50	12,000	9.5	.59	.595	.298	.295
.473	11.000	9.85	.62	.61	.288	.296
.44	10.000	10.1	.66	.62	.272	.292

DIA ESTIMATED FOR 70 M.P.H. K&B35

EQUATION FOR $DIA = 3600\sqrt[4]{BHP/(RPM^2 V)}$

FACTOR $BHP^{3/2}/RPM$ TO BE REVISED.

ALLEN MERCURY. ~

B.H.P	R.P.M	DIA	V/nD	EFF	NET H.P	FACTOR $\times 10^4$
.26	11.400	9.86	.328	.46	.117	.116
.25	10.000	10.4	.355	.47	.1175	.125
.23	9,000	10.73	.383	.487	.112	.1225
.21	8000	11.15	.416	.516	.1085	.12

DIA. ESTIMATED FOR 35 M.P.H.

TOWING TESTS

SAILPLANE R.J.5
$CD_{MIN} = .0055$
CESSNA 120
WING 2412
$CD_{MIN} = .030$ –
BELLANCA CRUISER
POWERED FLIGHT
$C_L = 0$ C_D .04
GLIDE w. COWL SEALED.
$CD_{MIN} = .025$
GLIDE: $C_L = 1.0$
$C_D = .095$
WITH POWER
$C_D = .134$.
CONDUCTED BY AUGUST RASPET
MISS. STATE COLLEGE
SEE PAGES: 55, 56 57, 28.

– MAXIMUM DURATION –

THE VALUE IS THE MINIMUM OF $CD/CL^{3/2}$

$$CD = CD_{PAR} + CD_e + CL^2/\pi AR$$

$$CD/C_L^{3/2} = \frac{CD_P + CD_e}{C_L^{3/2}} + \frac{CL^{1/2}}{\pi AR}$$

DIFFERENTIATING WE OBTAIN THE MINIMUM

$$\frac{dCD}{dCL} = -\frac{3}{2}\frac{CD_P + CD_e}{C_L^{5/2}} + \frac{1}{2}\frac{C_L^{-1/2}}{\pi AR} = 0$$

$$\text{THEN } CD_P + CD_e = \frac{1}{3}\cdot\frac{C_L^2}{\pi AR}$$

THIS EQUATION IS USED ONLY IF THE $CD_{PAR} + CD_e$ IS VERY LOW. IT WILL NOT WORK FOR AN AVERAGE FREE FLIGHT MODEL.

$CL = \sqrt{3(CD_P + CD_e)\pi . AR}$

EXAMPLE: MODEL $CD_P = .04$
$CD_{MIN} \approx .03$ AR 7
$CL = (3(.03 + .04)3.14 \times 7)^{1/2} = 2.15$
TOO HIGH. FOR HIGH SPEED AIRPLANE
$CD_P = .02$, $CD_e = .006$ AR = 3
$C_L = [3(.02 + .006)3.14 \times 3)^{1/2} = .86$
O.K.

WEIGHT DISTRIBUTION FOR STABILITY.

~ FREE FLIGHT – RADIO ~

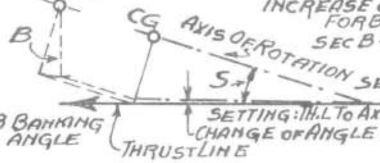
BANK ANGLE	0°	5°	10°	20°
C.F OF W.	0	.085W	.176W	.36W

TO BALANCE THE WEIGHT THE LIFT MUST INCREASE

BY	0	.0038	.0154	.0654

THE HORIZONTAL COMPONENT IS THE C.F.
ALL 3 FORCES W, L AND C.F PASS THROUGH THE C.G.

THE INCREASE OF LIFT CAN BE EFFECTED BY INCREASING THE SPEED OR THE ANGLE OF ATTACK. ON THE RUDDER-ONLY MODELS THE ANGLE OF ATTACK SHOULD INCREASE IN A BANK, IF THE WEIGHTS ARE DISTRIBUTED IN A CERTAIN WAY, WHICH GIVES THE AXIS OF EASIEST ROTATION AN ANGULAR POSITION WITH THE THRUST LINE. A GRAPHIC CONSTRUCTION ANALYZED AND AN EXAMPLE WILL CLARIFY THE PROBLEM. (?)



INCREASE OF LIFT = SEC B – 1.

FOR BALANCE

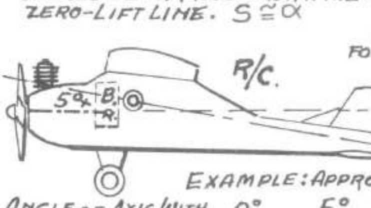
$$SEC\,B - 1 = \frac{\alpha \text{ INCREASE}}{\alpha}$$

$$SEC\,B - 1 = \frac{1 - COS\,B}{RAD\,\alpha \times SEC\,S}$$

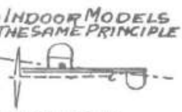
$$RAD\,\alpha = \frac{1 - COS\,B}{(SEC\,B - 1) \times SEC\,S} = \frac{1 - COS\,B}{SEC\,B - 1} \times SIN\,S$$

$\frac{RAD\,\alpha}{SIN\,S}$ = WITHIN PRACTICAL LIMITS = $\frac{1 - COS\,B}{SEC\,B - 1} \sim = 1.00$

THE AXIS OF EASIEST ROTATION ~ MIN. MOMENT OF INERTIA – SHOULD BE PARALLEL WITH THE AXIS OF THE ZERO-LIFT LINE. $S \cong \alpha$



GOOD INDOOR MODELS FOLLOW THE SAME PRINCIPLE



– ASK ANY CUB – LIGHT PLANE – PILOT

EXAMPLE: APPROXIMATE VALUES

ANGLE OF AXIS WITH THRUST LINE	0°	5°	10°	15°
WING	1854	1853	1851	1848
STAB	179	167	173	197
POWER PLANT	8	7	9	15
LANDING GEAR	110	114	116	120
FUSELAGE	2245	2238	2245	2261
BATT. + RADIO	15	15	16	20
TOTAL	4411	4394	4410	4461

VALUES ESTIMATED FOR OUNCES AND INCHES
MIN. VALUE FOR 5° ~

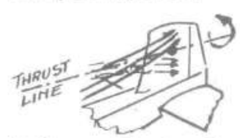
LOOPING OF RUDDER-ONLY R/C MODELS.

SOME RADIO CONTROL MODELS LOOP ONLY WITH THE WIND (DOWN-WIND) AND CANNOT LOOP INTO WIND (UP-WIND). THE REASON IS THAT THE WIND BLOWS AT AN UP-WARD ANGLE.

SEE PAGE 77

MODEL IS AIDED FROM A TO B DUE TO THE UPWARD DIRECTION OF THE WIND.

LOCATION OF FIN ON R/C MODELS.



TO REDUCE THE SIDE FORCE RESULTING FROM THE ROTATIONAL SLIPSTREAM THE FIN IS LOCATED APPROXIMATELY WITH THE THRUST LINE PASSING THROUGH THE CENTER OF FIN AREA.

FLAPPING WINGS

G.T. WALKER 1923

ANGLE OF ATTACK

	INNER	MIDDLE	OUTER
DOWN BEAT	12°	12°	12°
UP BEAT	12°	6°	0°

DURATION ESTIMATE OF AN INDOOR MODEL

GLIDING TESTS CONDUCTED WITH AN INDOOR MODEL (1940) GAVE A SINKING SPEED OF 6" PER SEC. AND A GLIDING ANGLE OF 1:5. THE AREA WAS 147 SQ.IN., A TOTAL WEIGHT .1375, THE WEIGHT OF THE RUBBER WAS .0575 OZ.

THE DURATION FOR INDOOR MODELS, WITH NON-FOLDING PROPS, IN SECONDS IS $= 70\, W_R \cdot A^{1/2} / W_T^{3/2}$ IN OZ. AND SQ.IN. THIS WOUD GIVE $70 \times 12.1 \times 0575 / .51 = 955$ SEC. THE BEST FIGHT WAS 736 SEC.

ANOTHER METHOD FOR ESTIMATING THE DURATION IS THE PROP ENERGY CONTAINED IN THE RUBBER DIVIDED THE ENERGY REQUIRED FOR CLIMBING FLIGHT. THE RUBBER IS WOUND 80% OF THE BREAKING POINT, PROP EFF. 50%, THE CLIMBING POWER IS TWICE THE SINKING SPEED TIMES THE WEIGHT OF THE MODEL.

$$\text{DUR. IN SEC} = \frac{55{,}000 \times .80 \times 50}{2 \times 6 \times .1375}$$

13300 SEC FOR 1 OZ RUBBER OR $1330 \times .0575 = 765$ SEC

SEE AERONUT BULLETIN MARCH 1940. PAGE 3.

DURATION ESTIMATE OF A STEEP CLIMBING GAS MODEL. (PAGE 60)

200 SQ.IN. .049 CU.IN. $W_T = 5$ OZ. PARASITE COEFF. .10 (PAGE 6) $C_L = 1.15$ FOR LOWEST SINK'G SPEED. CD_{MIN} (CORR'D) = 02 SINKING SPEED BY EQUATION [PAGE 60] $V_S = W^{1/2} CD / A^{1/2} CL^{3/2} .0012$ = 1.87 F.P.S. THE OTHER EQUATION $V_S = 11 W_{OZ}^{1/2} / A^{1/2}$ 1.74 F.P.S. THE .049 ENG. DELIVERS ~ .05 HP. THE PROP USED HAD A 6" DIA. 3" PITCH P/D ~ .50 SHOULD GIVE 50% EFF OR .025 NET HP. NET HP/OZ = .025/5 = .005 CD/OZ, P. 100 S.I. = .12/2.5 = .048 USING DIAGRAM THE TWO VALUES WILL GIVE A CLIMBING SPEED OF 29 F.P.S. ENGINE RUN 20 SEC, ALTITUDE REACHED = 20 × 29 = 580 FT. DURATION = 580/1.74 + 20 332 + 20 = 352 = 5 MIN + 52 SEC. THE THEORETICAL EQUATION DUR = RUN × POWER AVAL/POW R. BECAUSE THE FLYING SPEED IS ONLY 11.5 FT. P. SEC AND THE EFF. OF THE PROP WILL REDUCE TO 25%.

DUR = 20 × .0125/.001 = 250 SEC.

THE EQU. $DUR = 20 \times 360 \times 049 \dfrac{A^{1/2}}{W^{3/2}}$ WILL GIVE 440 SEC. TOO HIGH.

IDEAL SLOTS FOR BOUNDARY-LAYER CONTROL.

FOR PRESSURE FOR SUCTION

WING:

ASSUMING THAT THE PROF. DRAG IS CONSTANT IN A YAWED WING, THE INDUCED DRAG HOWEVER WILL DIFFER DUE TO THE CHANGE IN THE ANGLES, OF THE LEFT AND RIGHT WING CAUSED BY THE DIHEDRAL IN A SKID.

$$MOM_W = A_W q \frac{SPAN}{4} \frac{\left[(\alpha + \Delta\alpha)^2 - (\alpha - \Delta\alpha)^2\right](.039 AR^{1/3})^2}{\pi AR_W} \beta$$

β SKID ANGLE α ABS. ANGLE $\Delta\alpha = DIHED \times \beta / 57.3$ SEE PAGE 48.

$$MOM_F = A_F q\, DIST_F \cdot .039 AR_F^{1/3} \beta$$

COMBINING THE TWO EQU'S

$$\frac{A_F}{A_W} = \frac{SPAN\, \alpha_W\, DIHED}{DIST_F\, AR_F^{1/3} AR_W^{1/3}} \frac{1}{4750}$$

ASSUMING THE AVER. EFF. A.R. OF A FIN IS 2.5, THE EQUATION WILL BE

$$\frac{A_F}{A_W} = \frac{SPAN}{DIST_F} \cdot \frac{DIHED}{AR_W^{1/3}} \cdot \frac{\alpha_W}{6450}$$

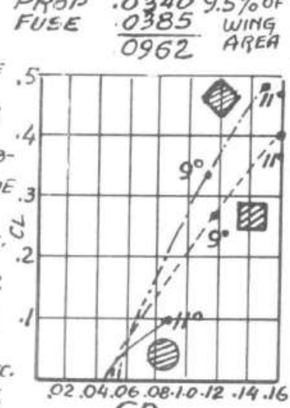
EXAMPLES: RADIO: $\alpha = 15°$, SPAN 50", AR 5, $DIST_{FIN}$ 20" DIHEDRAL 6°, PROP DIA 9" $DIST_P = 11$". FUSE CROSS SECT. 22", CG TO CP 1" $W_A = 500$ R/C D/L = 1/5

WING:	0219	
PROP:	0215	OR
FUSE:	0136	5.7% OF
TOTAL	0570	WING AREA.

WAKEFIELD: SPAN 48" AREA 200 S.I. AR 11.5 DIST TO FIN 23 DIST TO PROP 19" CG TO CP = 8" FUSE CROSS-SECT. 9 S.I. D/L = 1/5 DIHEDRAL 11°, $\alpha_W = 15°$

FIN AREA OF WING AREA

WING	.0237	OR
PROP	.0340	9.5% OF
FUSE	.0385	WING
	0962	AREA



COEFFICIENTS BASED ON THE MAX. CROSS SECTION OF THE FUSELAGES.

RUDDER CALCULATION:

CALCULATION OF FIN AREA RESISTS EXACTNESS, BECAUSE THREE INTEGRAL PARTS: FIN FUSELAGE AND PROP ARE CALCULATED SEPARATELY. THEIR TOTAL WILL NOT BE EQUAL IF TESTED IN A COMBINATION, BECAUSE THE AIR FLOWS FROM ONE UNIT TO THE OTHER.

PROPELLER: FOLLOWING N.A.C.A. TR 820 YAWED PROP ACT LIKE AS A FIN A.R. = 8 PROJECTED PROP AREA, AERODYNAMIC PRESS. CREATED BY THE THRUST THE MOMENT ACTING AT THE TIP OF THE PROP.

THE AERODYNAMIC PRESS. IS ESTIMATED BY THE THRUST ON THE DISC AREA, AND THE THRUST A FUNCTION OF THE WEIGHT = D/L DURING THE FLIGHT. USUALLY 1/5 TO 1/7 ON GLIDERS MAY BE UP TO 1/10, BUT NO PROP.

$$\frac{A_{FIN}}{A_W} = .056 \frac{DIST.PROP}{DIST_{FIN}} \times \frac{1}{AR^{1/3}}$$

FOR D/L = 1/7

$$\frac{A_{FIN}}{A_W} = .040 \frac{DIST_P}{DIST_F} \times \frac{1}{AR^{1/3}}$$

ASSUMING AN AVE. EFF. AR FOR THE FIN TO BE 2.5 THE EQUATIONS THEN ARE

$$\frac{A_F}{A_W} = .041 \frac{DIST_P}{DIST_{FIN}} \sim 1/5$$

$$\frac{A_F}{A_W} = .029 \frac{DIST_P}{DIST_F} \quad \text{FOR } \frac{D}{L} = 1/7$$

DISTANCES IN INCHES

STOPPED PROP

$$\frac{A_{FIN}}{A_W} = \frac{(1.16 - .44\, P/D)\, A_{PR} \times DIST_P}{2.25\, AR_T^{1/3}\, A_W \times DIST_F}$$

NO PROP EST. IN FAST CLIMB.

FUSELAGE: THE C.P OF THE FUSELAGE FORCES IS ASSUMED TO PASS THROUGH THE 25% OF THE LENGTH. MOMENTS ARE BASED ON THE MAX. CROSS-SECTION WHICH DOES NOT INCLUDE THE PYLON. THE C.P. OF A PYLON IS USUALLY AT THE C.G.

$$\frac{A_F}{A_W} = \frac{DIST\ TO\ 25\% \times A_{FUSE} \times 8.4}{DIST_{FIN} \times AR_F^{1/3}\, A_{WING}}$$

FOR AV. RECTANG FUSE

VALUES FOR $AR^{1/3}$

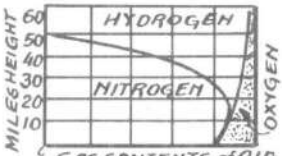
AR	1	1.5	2	2.5	3	3.5	
$AR^{1/3}$	1	1.14	1.26	1.36	1.44	1.52	
	4	4.5	5	5.5	6	6.5	7
	1.59	1.65	1.71	1.76	1.82	1.87	1.91
	7.5	8	9	10	11	12	
	1.85	2.0	2.08	2.15	2.22	2.29	

INDEX

REFERENCES ABBREVIATIONS

T.R. = NACA Technical Report
T.M. = NACA Technical Memo
T.N. = NACA Technical Notes
G= Goettingen E = Eiffel

(Index Tabulated by Charles Sotich)



GAS CONTENTS of AIR

of AIR .07651 LB/CU.FT.
" HYDROGEN .005021 "
" HELIUM .01056 "
" SATURATED AIR .073 "
PRESSURE, STD 29.42" MER
14.7 LBS. P. SQ.IN.

THOUSANDS OF FEET						
HEIGHT	0	2	4	6	8	10
DENSITY	1	.94	.89	.84	.78	.737
PRESSURE	1	.93	.86	.80	.74	.68
1/DEN 3/2	1	1.10	1.19	1.3	1.45	1.56
RESERVE HP. FOR CLIMB ON GROUND		9%	16%	22%	31%	40%

BY R.J. HOFFMAN
1908 T.R. 638

HUMIDITY:
RELATIVE: PERCENTAGE OF SATURATION
ABSOLUTE:
WEIGHT OF WATER PER UNIT VOL. OF AIR.
SPECIFIC: MASS OF WATER VAPOR IN UNIT MASS of AIR

RADIO
SPEED RADIO WAVES
186,000 MILES P. SEC.
300,000 MET/SEC.
MEGA - 1,000,000
MICRO - 1/1,000,000
KILO - 1000, DECI 1/10
MILLI 1/1000
WAVELENGTH = $\frac{\text{SPEED}}{\text{CYCLE}}$
FOR 27.255 MEGACYCLE
186,330x5280/27,255,000
= 36 FT = 1/4 ≅ 9 FT.

NEWTON'S LAW - $F = Ma$
FORCE = F, M = W/g = MASS
g = 32.2, a = ACCELERATION
WORK = FORCE x DISTANCE
POWER = WORK IN A GIVEN TIME
BHP = 550 FT.LB/SEC
TORQUE = HPx550/n x 2π
π = 3.1415.... n = REV P. SEC.

SPEED OF SOUND:
ABOUT 1100 FT/SEC.

INCH = 2.54 CM OR 25.4 m.m.
CU.IN = 16.39 cu.cm. LB = 7000 GR.
FT.P.S = 1.467 M.PH FT = .305 MET
GRAIN = .0648 GRAM GAL = 231 C.IN.
OZ = 31.1 GRAM SQ.IN = 6.4 SQ.CM.
FLUID OZ. = 1.805 CU.IN
POUND (AVOIR) 16 OZ. QT = .946 LT.
" (TROY) 12 OZ.

NAUTICAL MILE ~ 6080.2 FT
1 KNOT = 1 N.M/HR = 1.689 F.P.S
MILE = 5280 FT
1 KILOMET./HR = 1.609 MIL./H.

OHM'S LAW FORMULAS:
W = WATTS = EI, I^2R, E^2/R
E = VOLTS = IR, $\sqrt{WR}$, W/I
I = AMPERES = E/R, $\sqrt{\frac{W}{R}}$, $\frac{W}{E}$
R = OHMS = E/I, E^2/W = W/I^2

ENERGY =
POTENTIAL ENERGY IF RELEASED CAN DO WORK.
BODY HAS KINETIC ENERGY IF IT IS MOVING
KIN. EN = $M \times V^2/2$
MOMENTUM = $M \times V$
ANGULAR MOMENTUM = $I \times \omega$
ω - ANG. VELOC. IN RADIANS
1 RADI. = 57.3°
360° = 2π
I = MOMENT OF INERTIA.

BERNOULLI'S THEOREM
TOTAL HEAD = POTENTIAL, PRESSURE AND VELOCITY HEADS, MUTUALLY CONVERTIBLE.

GREEK ALPHABET

ALPHA	Α α	NU	Ν ν
BETA	Β β	XI	Ξ ξ
GAMMA	Γ γ	OMICRON	Ο ο
DELTA	Δ δ	PI	Π π
EPSILON	Ε ε	RHO	Ρ ρ
ZETA	Ζ ζ	SIGMA	Σ σ s
ETA	Η η	TAU	Τ τ
THETA	Θ θ ϑ	UPSILON	Υ υ
IOTA	Ι ι	PHI	Φ φ ϕ
KAPPA	Κ κ	CHI	Χ χ
LAMBDA	Λ λ	PSI	Ψ ψ
MU	Μ μ	OMEGA	Ω ω

RAUL J. HOFFMAN THROPHY

Raul J. Hoffman was technical adviser to the Chicago Aeronuts until his death in 1956. The Club has perpetuated his name by sponsoring a throphy bearing his name. This throphy is annually awarded to the modeler, regardless of age, whose model accumulates the highest three flight total in Class "A" Free Flight Gas Event at the Nationals. A plaque is also given to the winner to keep.