RESTRICTED

AERODYNAMICS AND AIRCRAFT DESIGN

A Report of the AAF Scientific Advisory Group

by
H. S. TSIEN
W. R. SEARS   IRVING L. ASHKENAS
C. N. HASERT   N. M. NEWMARK

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A REPORT PREPARED FOR THE AAF SCIENTIFIC ADVISORY GROUP

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The AAF Scientific Advisory Group was activated late in 1944 by General of the Army H. H. Arnold. He secured the services of Dr. Theodore von Karman, renowned scientist and consultant in aeronautics, who agreed to organize and direct the group.

Dr. von Karman gathered about him a group of American scientists from every field of research having a bearing on air power. These men then analyzed important developments in the basic sciences, both here and abroad, and attempted to evaluate the effects of their application to air power.

This volume is one of a group of reports made to the Army Air Forces by the Scientific Advisory Group.

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HIGH SPEED AERODYNAMICS

By

HSUE-SHEN TSIEN

PART I

HIGH SPEED AERODYNAMICS

DECEMBER 1945

THE EFFECT OF COMPRESSIBILITY OF AIR IN AERODYNAMICS

When a body moves through the atmosphere, the effect of its motion on the surrounding air can be considered as that caused by a disturbance. Since any disturbance propagates with the velocity of sound, which itself is nothing but a series of small disturbances, the disturbance caused by the motion of the body is also propagated throughout the medium with the velocity of sound. If the body moves very slowly, then in the time scale of the motion of the body, the velocity of propagation of the disturbance is practically infinitely large. In other words, the disturbance is felt almost instantly (referred to the time scale of the motion of the body). This means that the fluid medium, the air, can be considered as incompressible and hence no appreciable elastic adjustment is present to take up the time of propagation. Therefore, for slow motion, the air can be considered as incompressible and this forms the basis of all classical aerodynamics.

As the speed of motion of the body is increased, the time of propagation necessary for the disturbances can no longer be neglected, i.e., the elasticity or the compressibility of the air must be taken into account. Here it is immediately clear that the measure of the effect of compressibility is the ratio of the speed of the body to the velocity of sound in the fluid, i.e., the Mach number. In other words, if the Mach number is small, the air can be considered as incompressible. But at high Mach numbers, the compressibility of the air must be taken into account when the flow phenomena are studied.

To obtain a proper orientation for the following discussions, Fig. 1 is prepared which gives the corresponding Mach number for various flight speeds in mph. Since the velocity of sound is dependent upon the atmospheric temperature, the Mach number for a fixed value of speed is a function of flight altitude. In Fig. 1, the altitude conditions are assumed to be those of the NACA Standard Atmosphere. It is seen that 764 mph corresponds to Mach number unity at sea level. Therefore, flying at 764 mph means flying with the velocity of sound. Flight below 764 mph speed can then be called subsonic flight, while flight above 764 mph speed can be called supersonic

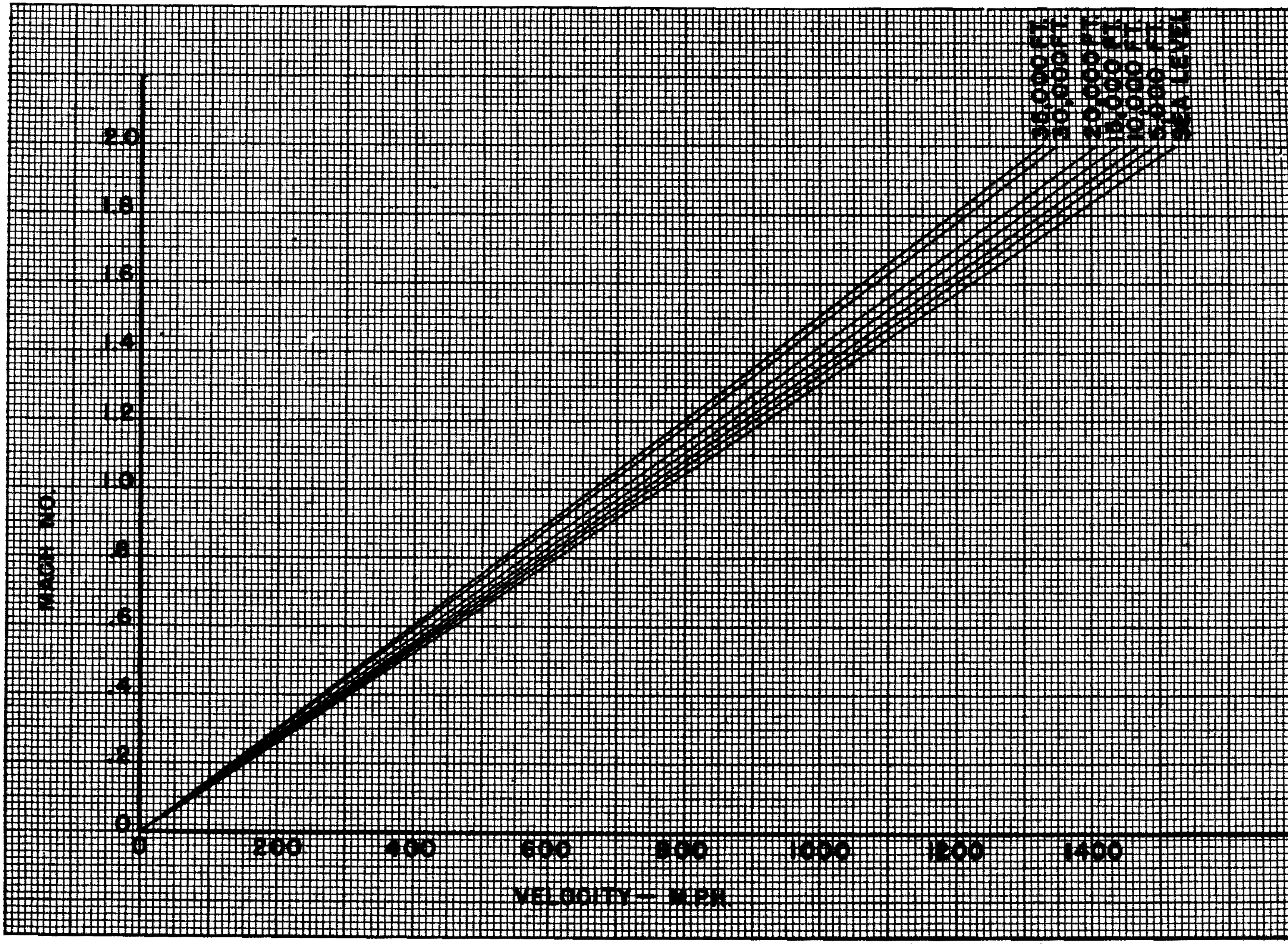
MACH NO.
2.0
1.8
1.6
1.4
1.2
1.0
.8
.6
.4
.2
0
0
200
400
600
800
1000
1200
1400
VELOCITY — M.P.H.
35,000 FT.
30,000 FT.
20,000 FT.
15,000 FT.
10,000 FT.
5,000 FT.
SEA LEVEL

Figure 1

flight. Flight with velocities close to the velocity of sound can be called transonic flight.

If a stream of incompressible fluid is brought to rest, the pressure rise q is found to be equal to

$$q = \frac{1}{2} \rho v^2 \tag{1}$$

where ρ is the density of the fluid in mass units (slugs per cubic feet) and v is the velocity of the stream in feet per second. The pressure rise q is called the dynamic pressure. It is thus convenient to refer to aerodynamic forces to this pressure and render the quantities nondimensional. For example, if L and D are the lift and drag (Fig. 2) of a wing with area S flying through an atmosphere of density ρ with velocity

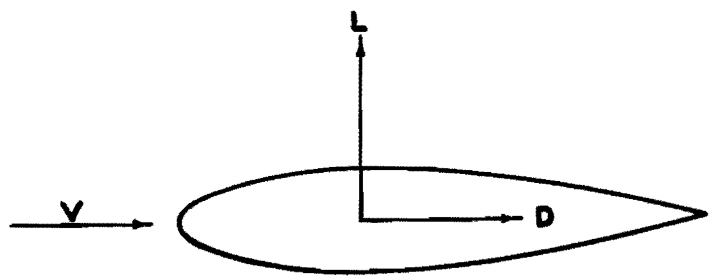

Figure 2

v, then the nondimensional quantities are the lift coefficient C_L and the drag coefficient C_D given by

$$C_L = \frac{L}{\frac{1}{2} \rho v^2 S} \tag{2}$$

$$C_D = \frac{D}{\frac{1}{2} \rho v^2 S} \tag{3}$$

For convenience, this method of reducing to nondimensional quantities is used even for compressible flow or high-speed phenomena where the quantity q is not exactly the dynamic pressure rise although it still has the dimension of a pressure.

As stated previously, the aerodynamic phenomena are functions of Mach number, M. Therefore, for a given wing at a fixed angle of attack α, the lift and drag coefficients, C_L and C_D, should be functions of M. This is found to be the case by both wind-tunnel tests and free-flight tests. Fig. 3 is a set of representative results for a rectangular wing with negligible tip effects of approximately 10% thickness. (Maximum wing thickness is 10% of the chord of the wing.) It is seen that the lift coefficient C_L increases with Mach number at an increasing rate up to a Mach number of approximately 0.7. Beyond that Mach number, the lift coefficient has a sharp drop. The exact behavior of C_L near Mach number unity is not yet clear, due to lack of experimental data. At supersonic velocities, i.e., for Mach numbers greater than unity, it is

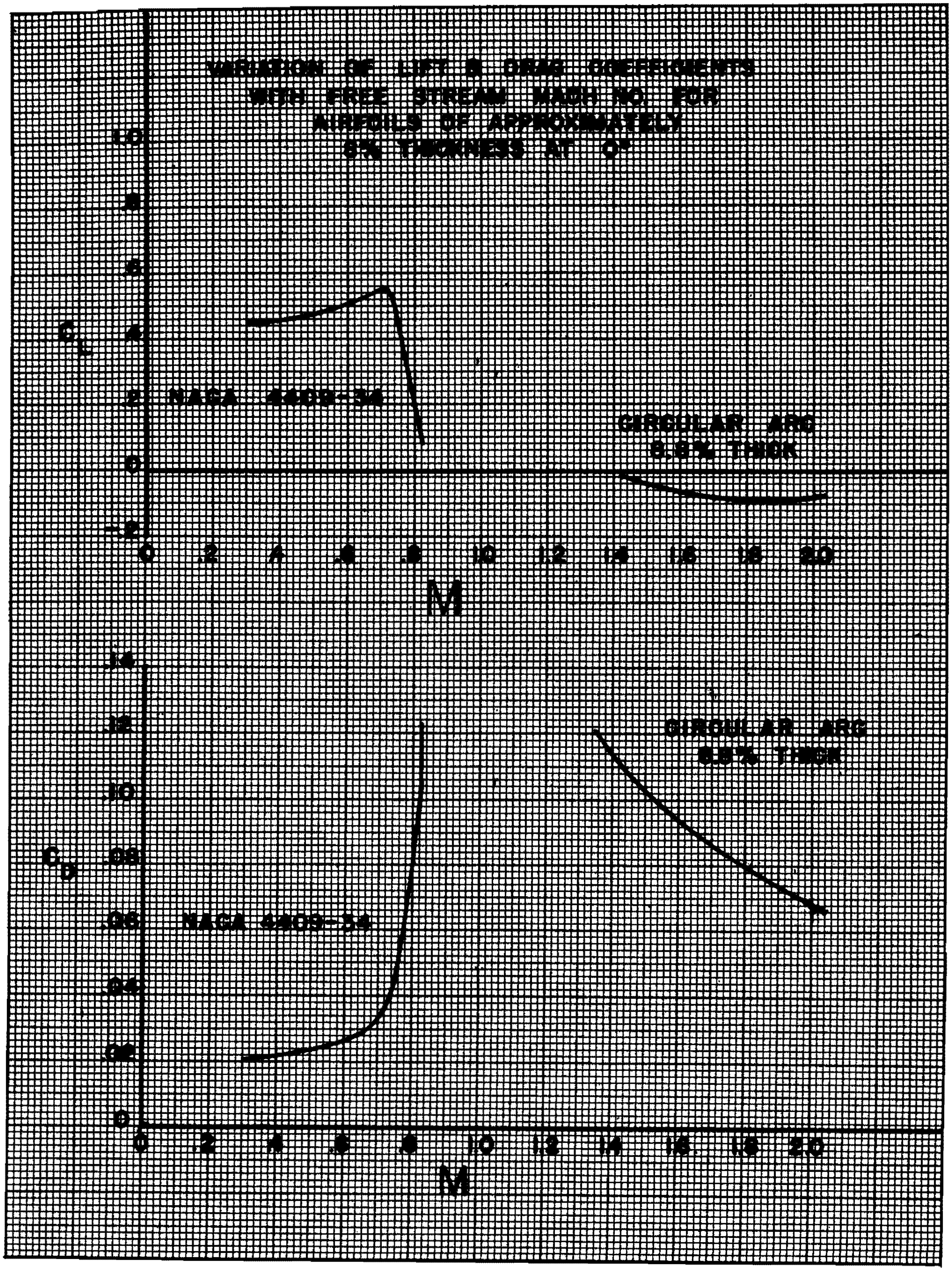
VARIATION OF LIFT & DRAG COEFFICIENTS
WITH FREE STREAM MACH NO. FOR
AIRFOILS OF APPROXIMATELY
8% THICKNESS AT 0°
C_L
NAGA 4409-34
CIRCULAR ARC
8.8% THICK
M
C_D
NACA 4409-34
CIRCULAR ARC
8.8% THICK
M

Figure 3

shown that the lift coefficient steadily decreases with increase in Mach number (Fig. 3). The drag coefficient C_D does not change appreciably for Mach numbers below 0.7. At $M = 0.7$, there is a very sharp rise which continues up to $M = 1.2$, approximately. For still higher Mach number in the supersonic flight, the value of C_D steadily decreases again. In general, the value of drag coefficient at supersonic velocities is much larger than that at subsonic velocities, due to the occurrence of shock loss (as will be explained in the section on shock waves, page 13. Therefore, the efficiency of the wing or the ratio of lift to drag is much smaller for Mach numbers greater than unity. For instance at low speeds, the ratio L/D would be greater than 30, while for supersonic velocities, the ratio is seldom larger than six.

In Fig. 3, the range of Mach numbers between 0.8 and 1.3 is left open as the wind-tunnel test at velocities close to the velocity of sound is not reliable. The reason for this failure of wind-tunnel tests is the strong interference effect of the jet boundary on the flow around the test body. If the velocity of fluid is subsonic throughout the whole field, then the effect of the solid walls of the test section is to increase the velocity over the surface of the body, and the effect of the free jet boundaries of an open test section is to decrease the velocity over the surface of the body. Thus, for such cases, a simple correction in the "effective" free-stream velocity is generally sufficient. However, if the flow velocity is partly subsonic and partly supersonic as is generally the case for the Mach number range between 0.8 and 1.3, the wall effect is not so simple, and no simple free-stream correction factor can be applied, especially with the appearance of shock wave due to the walls. Under this situation, the only method of obtaining aerodynamic data at transonic speeds is by free flight. One method in this category is the dropping of a weighted model from high altitude and then measuring the forces. Preliminary data* by NACA shows that much lower drag coefficient near the sonic velocity prevails than in wind-tunnel tests. For instance at Mach number unity, the drag coefficient of a 6% thick wing is only 0.03. This surprisingly low drag coefficient certainly indicated the importance of further pursuing this type of experiment.

The drop of lift coefficient and the increase of drag coefficient at transonic speeds greatly decreases the aerodynamic efficiency of the airplane. A fast modern jet fighter such as the P-80 is limited in maximum speed by this very fact. However, the difficulty of compressibility of air is not limited to the decrease of lift coefficient and the increase of drag coefficient. The most difficult problem at present is the stability problem. This problem arises out of the fact that when the airplane wing enters the transonic speeds, there is a sudden rearward shift of the center of pressure. Therefore, the airplane tends to dive. This further increases the velocity, and recovery is very difficult. For present-day airplanes, recovery from a dive is effected by dive flaps placed near the leading edge on the underside of the wing. This device changes the pressure distribution on the lower surface in such a way as to increase the lift and restore the forward position of the center of pressure. However, the exact mechanism of the flap action is not yet clearly understood, and further research has to be done.

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* F. J. Bailey, Jr., C. W. Mathews, J. R. Thompson, "Drag Measurements at Transonic Speeds on a Freely Falling Body," NACA, ACR No. L5EO3 (1945).

The rapid increase in drag near the sonic velocity, and the subsequent decrease at supersonic velocities also occur for bodies other than wings. Fig. 4 shows the variation of the drag coefficient C_D for a sharp-nosed body. Here the drag coefficient is defined as

$$C_D = \frac{D}{\frac{1}{2} \rho v^2 A} \tag{4}$$

where D is the drag and A is the maximum cross-sectional area of the body. Therefore, the crucial aerodynamic problem of high-speed flight is the reduction of the drag coefficient peak near the velocity of sound, both in decreasing the body drag and in improving the lift-drag ratio of the wing.

DRAG REDUCTION BY MAINTAINING THE LAMINAR BOUNDARY LAYER

Before discussing in detail the problem of reducing the drag peak near the velocity of sound, it is necessary to examine the origin of the resistance of a moving body. The resistance comes from two sources: (1) the skin friction and (2) the pressure force acting on the surface of the body. The first source will be considered here first.

The skin friction is the result of the viscosity of the air. Due to the viscosity, air molecules tend to stick to the surface of the body, so that the flow velocity immediately adjacent to the surface is zero. But the flow velocity increases rapidly away from the surface and reaches its full value at a very short distance. This layer of flow is called the boundary layer and is generally of the order of one inch in thickness over a wing surface of normal dimensions. The skin friction force τ per unit area is given by

$$\tau = \mu \frac{\partial u}{\partial y} \tag{5}$$

where μ is the coefficient of viscosity, u the flow velocity and y the distance normal to the surface.

The thickness of the boundary layer is a function of the viscosity. Higher viscosity of the fluid gives thicker boundary layer. However, a more exact parameter is the so-called "Reynolds number" which expresses the ratio of dynamic (or inertia) forces and viscous forces of the fluid. For instance, if v is the flight velocity, c the wing chord and ρ the air density, then the Reynolds number R of the wing is

$$R = \frac{\rho cv}{\mu} . \tag{6}$$

This parameter is also nondimensional. It is found that the friction drag D_s is decreased if the Reynolds number is larger. For instance, in case of a flat plate placed parallel to the air stream, the friction drag coefficient C_{D_s} is related to the Reynolds number R as shown in Fig. 5. Here the coefficient is referred to the drag on one side of the plate surface,

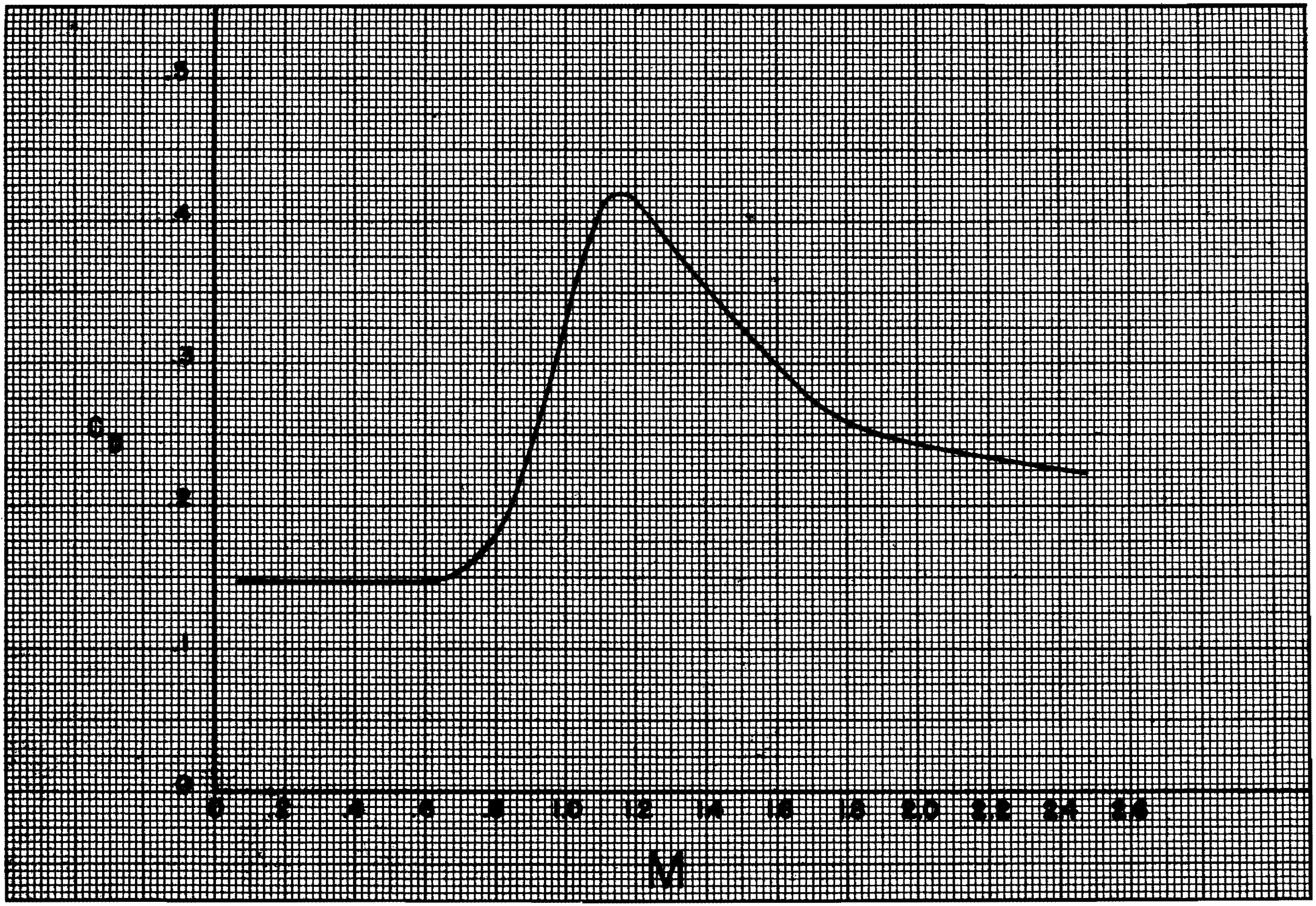
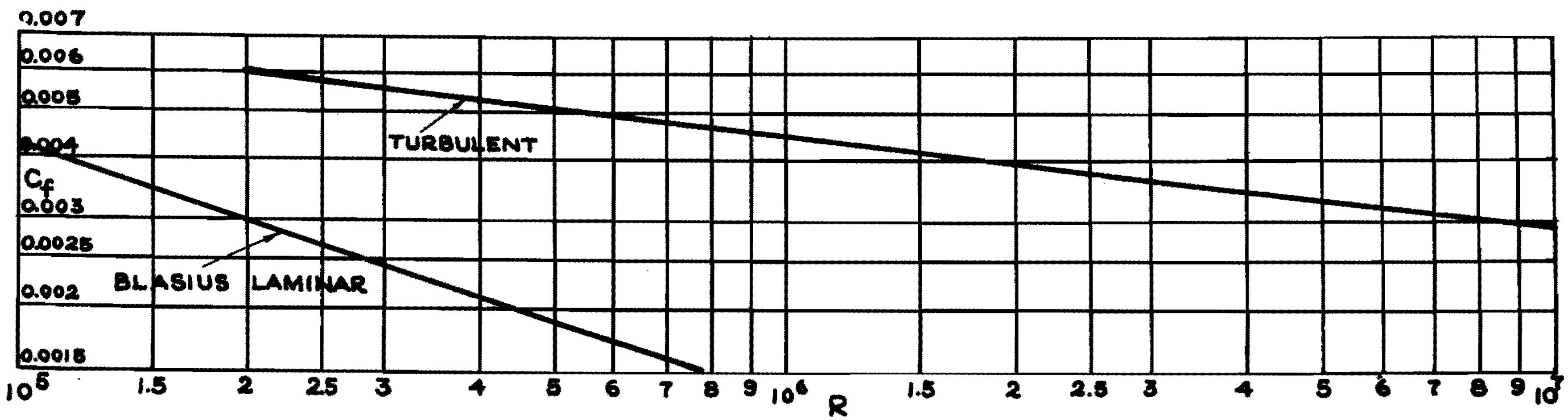


Figure 4



0.007
0.006
0.005
0.004
C_f
0.003
0.0025
0.002
0.0015
TURBULENT
BLASIUS LAMINAR
10^5
1.5
2
2.5
3
4
5
6
7
8
9
10^6
R
1.5
2
2.5
3
4
5
6
7
8
9
10^7

8

Figure 5

$$C_{D_s} = \frac{\text{Friction Drag on One Side}}{\frac{1}{2}\rho v^2 S} \tag{7}$$

and the Reynolds number is referred to the length of the plate in the flow direction.

In Fig. 5 there are two curves, one labelled laminar and the other turbulent. These correspond to two different types of boundary layer flow. In the laminar flow boundary layer, the streamlines follow smooth curves, and only the air molecules have the irregular agitation. The molecular agitation is, however, invisible at macroscopic scale. In the turbulent flow boundary layer, the fluid elements of macroscopic scale follow irregular agitation. Hence the mixing of fluid elements is much more vigorous. Due to this vigorous mixing and agitation, the velocity profiles of the laminar boundary and of the turbulent boundary layer appear quite different as shown in Fig. 6. The velocity for the turbulent boundary layer is the time average velocity. However, even for the turbulent boundary layer there is a laminar sublayer where the skin friction can be calculated as $\mu \frac{\partial u}{\partial y}$. From Fig. 6, it is seen that the slope of velocity curve $\frac{\partial u}{\partial y}$ at the wall is much larger for a turbulent boundary layer than for a laminar boundary layer. Therefore the skin friction is also much larger for a turbulent boundary layer than for a laminar boundary layer. This fact is clearly seen in Fig. 5 which shows the higher skin friction coefficient of the turbulent layer at a given Reynolds number.

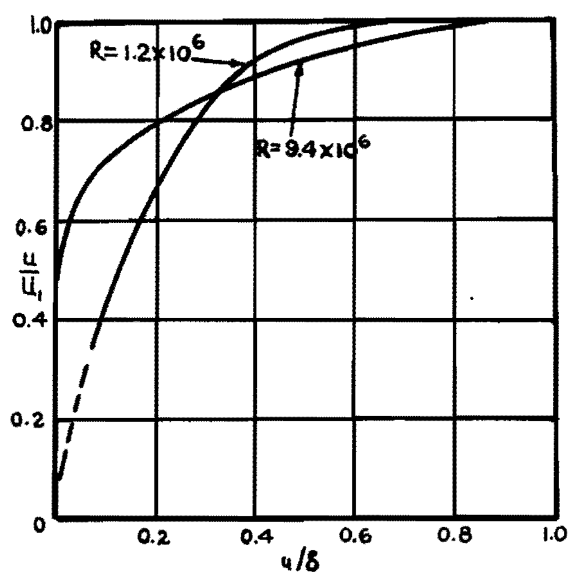


Figure 6 — Velocity Distribution in Turbulent and Laminar Boundary Layers

The problem of reducing drag is then the same as the problem of maintaining the laminar boundary layer. The laminar boundary layer, however, has inherent instability and the preserving of laminar flow or the transition from laminar to turbulent flow depends upon (1) the absence of atmospheric small-scale disturbances, (2) the shape of the boundary layer profile, (3) the Reynolds number of the flow. Over the first, one has no control. But the free atmosphere is fortunately very smooth as far as transition is concerned. The Reynolds number of the flow is fixed by the size and speed of the aircraft and is not within the control of the designer once the specifications of the aircraft are chosen.

Therefore the only remaining control is the shape of the boundary layer profile. The shape of the boundary layer profile is influenced by the following factors: (1) the pressure gradient along the flow direction over the surface, (2) the surface smoothness, (3) the partial removal of boundary flow by suction. It is found that the convex profile (Fig. 7) in a negative pressure gradient (decreasing pressure in flow direction) is much more stable than a concave profile (Fig. 7) in a positive pressure gradient (increasing pressure in flow direction). Hence for the maintenance of laminar boundary layer, it is advantageous to make the pressure distribution such that the suction peak is far towards the trailing edge of the wing as shown in Fig. 8. Then the pressure gradient along the surface will be mostly negative. This is the principle of the laminar flow wing (Fig. 8) for low drag. The drag coefficient when plotted against the angle of attack has a sudden jump at a critical angle of attack. This can be easily explained by the change in pressure distribution over the surface of the airfoil. As the angle of attack is increased, the suction pressure peak tends to move forward to the leading edge. Thus the boundary layer flow has to overcome more and more adverse pressure gradient along the surface. Once the boundary layer becomes concave for that reason, transition to turbulent flow will immediately follow and the drag coefficient increases. The maximum value of the ratio of lift coefficient C_L to the drag coefficient C_D of two-dimensional flow (profile drag coefficient) could be as high as 150. For an actual wing of finite span, the lift and drag ratio will be much less than this value due to the addition of the induced drag.

However, in the practical application of this laminar flow principle there are two difficulties: (1) the deflection of the wing in the loaded condition which gives wavy surfaces, and (2) the surface roughness. These two difficulties make the actual realization of the favorable boundary layer profile almost impossible even with a good pressure gradient along the surface. This is a handicap at present in the practical application of laminar flow wings. Thus, here is an example of the influence of constructional method on the aerodynamic characteristics. Hence, to realize a true laminar flow wing, research on constructional methods and on surface finish is necessary to achieve a rigid and smooth wing surface.

The most effective method of maintaining the laminar boundary layer is, however, by partial removal of the flow by suction through slots cut in the wing surface. The pressure distribution over the surface is considerably modified by the suction (Fig. 9) and local favorable pressure gradients are created for laminar flow. Of course, the suction of air requires power. But with carefully designed inlet and ducting system, it is found that the power saving due to reduction in drag far overbalances the power expenditure required by suction and a net gain is achieved. One of the essential points is the recovery of the kinetic energy of the air sucked in by the use of diffusers immediately within the suction slots. J. Ackeret of ETH, Zürich, has shown by using this method that the two-dimensional lift over drag ratio could be raised to as high a value as 260 at large Reynolds number. Furthermore, the power required to drive the boundary layer suction blower is included in this calculation as part of the drag,

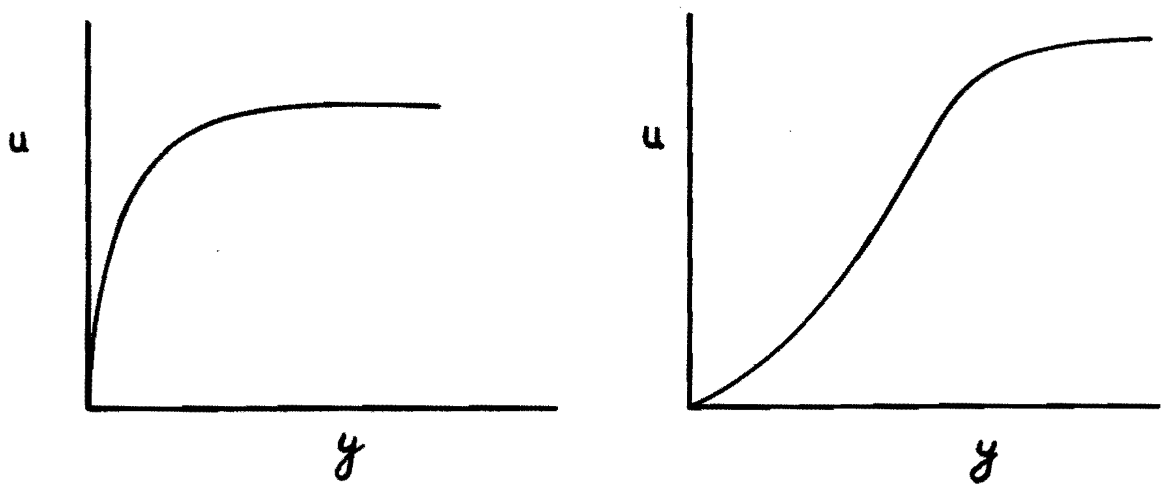


Figure 7

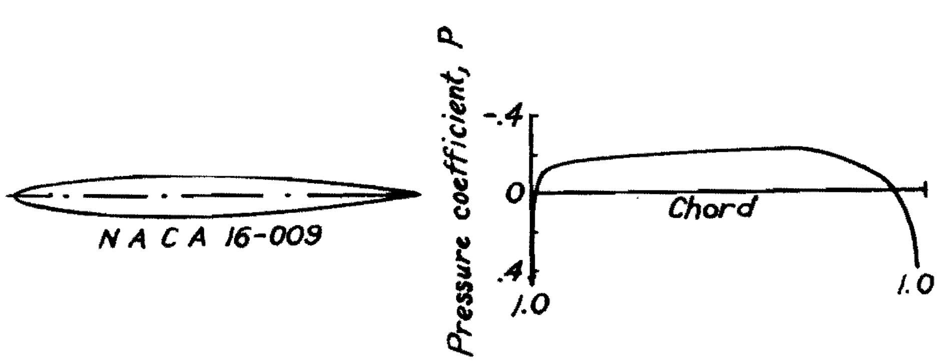


Figure 8

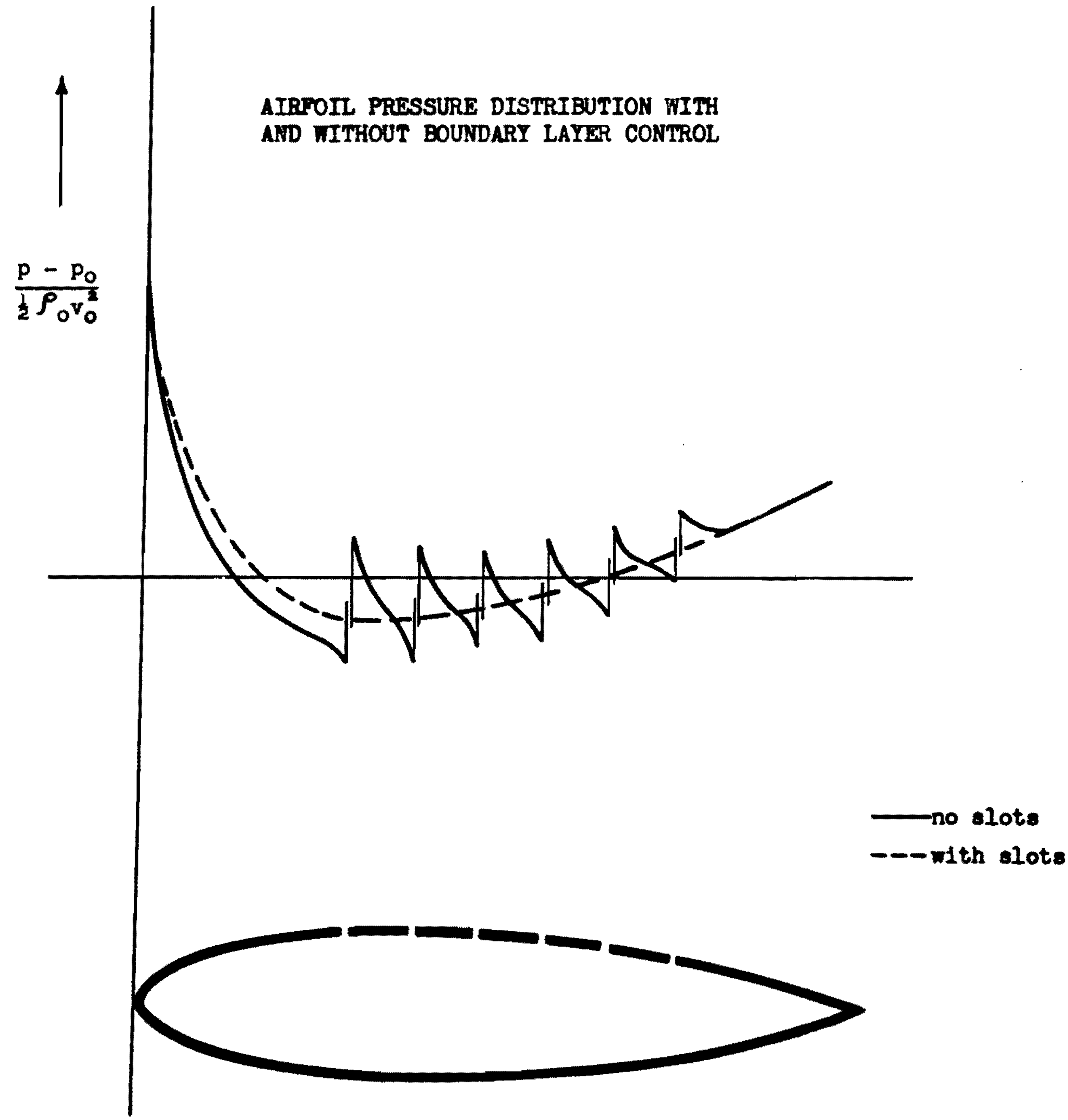


Figure 9 — Airfoil Pressure Distribution With and Without Boundary Layer Control

but with the assumption of 100% compression efficiency. Of course, when the induced drag of a finite span is considered, the lift-drag ratio will be much smaller. However, with the reduction of the profile drag coefficient, the lift coefficient for optimum value of the L/D ratio for a wing of finite span is also reduced. This would bring the cruising condition of the airplane close to the optimum L/D ratio and raise the general performance of the airplane.

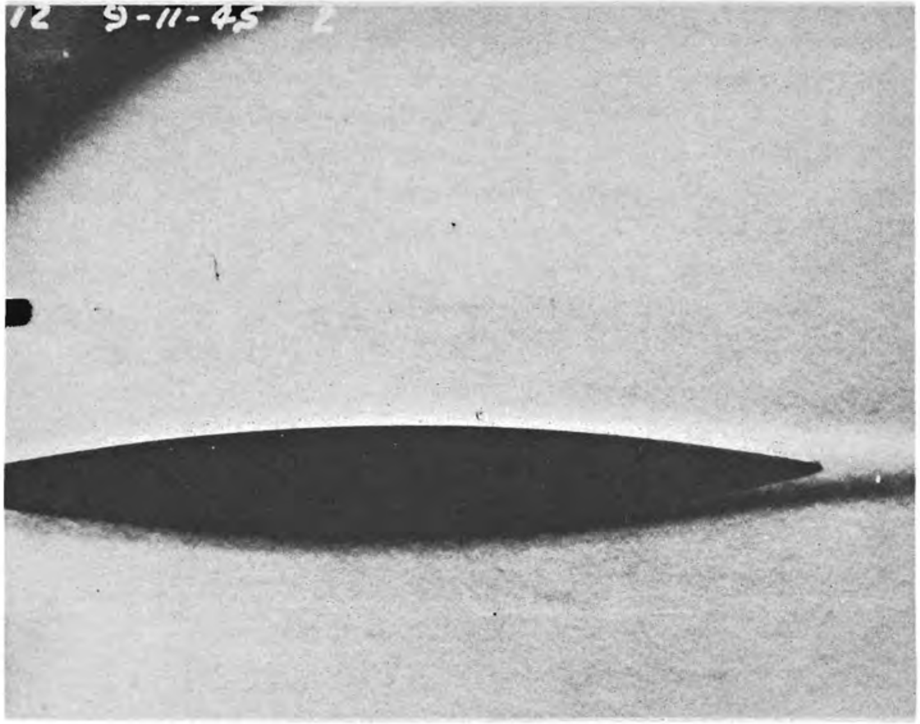
Summarizing, drag reduction through maintaining the laminar boundary layer requires the solution of two particular problems: i.e., (1) the construction of rigid and smooth surfaces, and (2) the application of boundary layer removal by suction and the design of an efficient ducting system to reduce the necessary suction power.

SHOCK WAVE AND THE INTERACTION OF SHOCK WAVE AND BOUNDARY LAYER

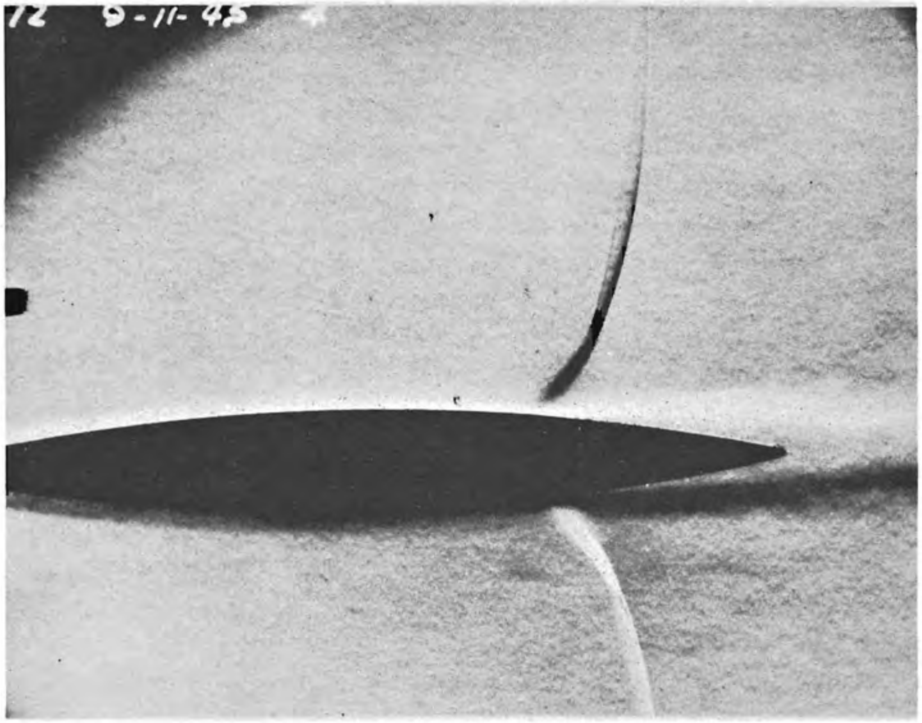
Figures 3 and 4 show that when the velocity of sound is approached, there is a rapid rise in the drag coefficient both for the rectangular wing and for the body of revolution. To understand this important phenomenon, it is necessary to examine in closer detail the flow field.

Figure 10 shows two pictures of the two-dimensional flow over an airfoil taken by the Schlieren method. The light areas are areas of decreasing density in the horizontal direction, or expansion regions. The dark areas are areas of increasing density in the horizontal direction, or compression regions. It is seen that at low Mach number, the flow over the airfoil is smooth with continuous expansion or compression. But at high Mach numbers, a sharp dark line appears in the flow field where a very rapid compression occurs. This is called the shock wave. In a shock wave, the process of compression is not isentropic, i.e., the compression is not as efficient as smooth compression. The result is a loss of mechanical energy through transformation into heat. This loss in mechnical energy and inefficient compression makes the pressure on the downstream part of the body lower than the pressure would be with continuous isentropic compression, if isentropic compression were realized. Therefore, the body is effectively under a backward suction. This is the origin of the increase in drag at high Mach numbers.

With a view toward drag reduction at high speeds, the question is then, can shock waves be avoided? There are many theoretical calculations, neglecting the boundary layer, which show the possibility of shockless flow at very high Mach numbers with continuous isentropic transition from supersonic to subsonic flow. Since the only essential difference between the theoretical solution and the experimental observation is the neglect of the boundary layer, one must naturally conclude that shock wave in the transonic flow is closely related to the boundary layer. This surmise is further substantiated by the examination of flow over a cone at supersonic speeds. Figure 11 shows the comparison of the theory with the spark photograph of an actual conical shell in flight. It is seen that although the theoretical calculations show a

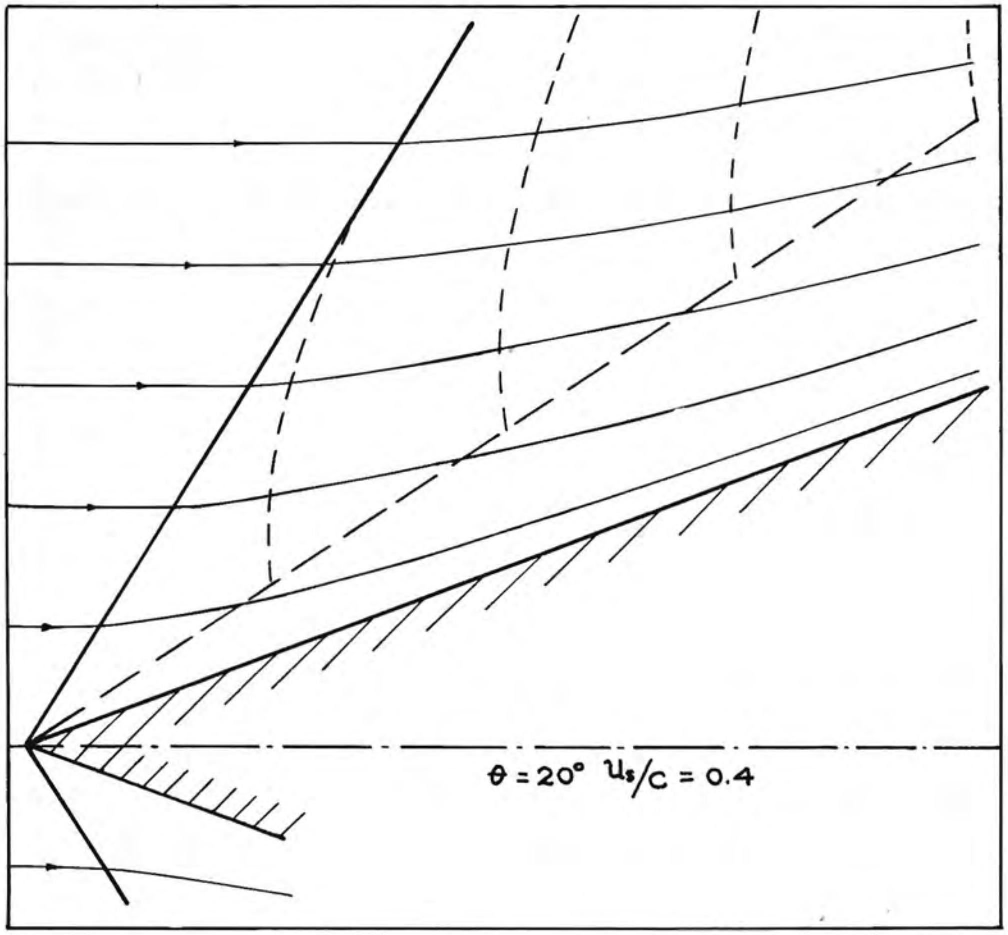


Mach Number = 0.792



Mach Number = 0.860

Figure 10 — Flow Past 3-in. Circular Airfoil of 12% Thickness Reynolds Number = 1.7×10^6
Flow From Left to Right (Taken by H. W. Liepmann of California Institute of Technology)



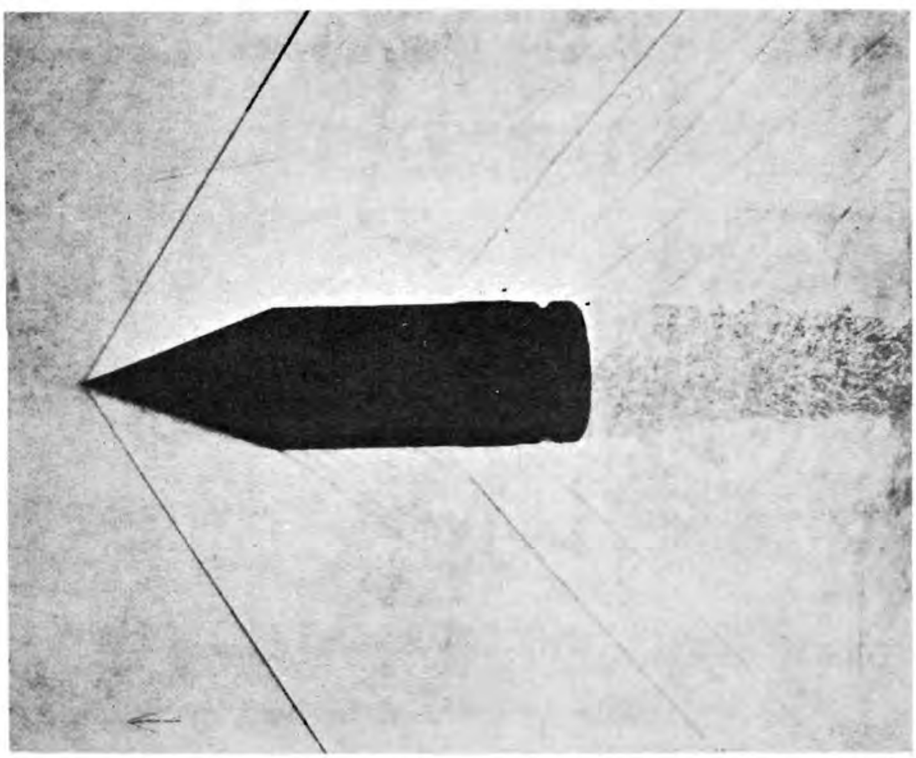


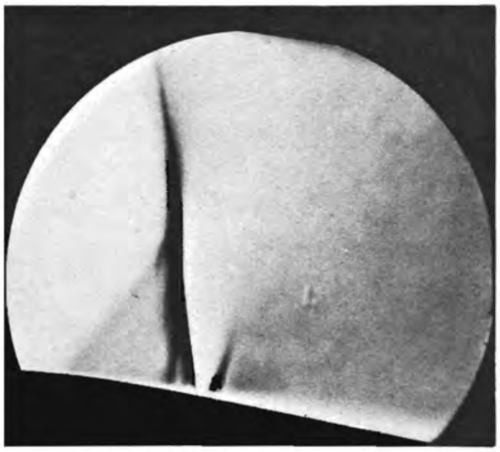
Figure 11 — Flow Over a 20° Cone

transition from the supersonic flow behind the first shock wave to subsonic flow near the surface of the cone, the spark picture does not reveal any discontinuity in this transition region. This means that a smooth compression is actually accomplished. However, here the transition takes place away from the surface and the boundary layer, and thus the flow is not contaminated by the viscosity effect. Therefore, the flow over the cone can be taken as evidence for the destabilizing effect of the boundary layer in transonic flows.

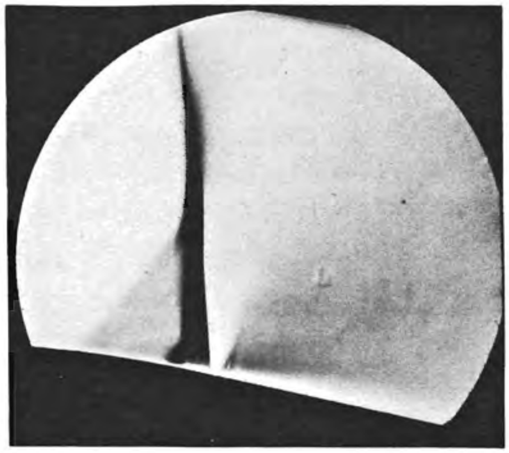
It seems, then, that shock waves and the resultant increase in drag for flight velocities approaching the velocity of sound are closely connected with the existence of boundary layers. Since boundary layers are a Reynolds number phenomenon, this fact means a close interrelation between Reynolds number and Mach number effects. This interaction is most clearly shown by a series of tests initiated by J. Ackeret.* In this series of experiments, the flow over an airfoil was studied at a constant Mach number, but with the density varied over a wide range. Therefore, the tests are made at constant Mach number but variable Reynolds numbers. The results are shown in Fig. 12. The Mach number of the flow just ahead of the shock wave is equal to 1.3. The figures are arranged with increasing density or Reynolds number. At small Reynolds number, the shock-wave system has the characteristic λ formation. The first oblique shock wave thickens the boundary layer but produces no separation. But after the main shock, the separation is quite severe. By increasing the Reynolds number, the first shock finally disappears and the separation after it is relatively mild. It is found that the pressure distribution and the drag of the airfoil are greatly influenced by this change in shock form due to the change in Reynolds number. Furthermore, the first oblique shock is found to be due to the instability of the laminar boundary layer. By introducing artificial turbulence and making the boundary layer turbulent, the first oblique shock can be eliminated and only a single shock remains. This definitely establishes the interrelation between the effects of Mach number and Reynolds number, and accurate results cannot be expected if tests on models of aircraft are made at too small a Reynolds number, even with the correct Mach number.

Further complications in high-speed flows are the effect of the inherent instability of laminar boundary layer on the shock-wave formation, and the effect of condensation of water vapor in the air flow. It is then evident that for the correct understanding of high-speed flow, the following problems must be solved: (1) the stability of laminar boundary layer in relation to shock formation; (2) the stability of turbulent boundary layer in relation to shock formation; and (3) the effect of water vapor condensation.

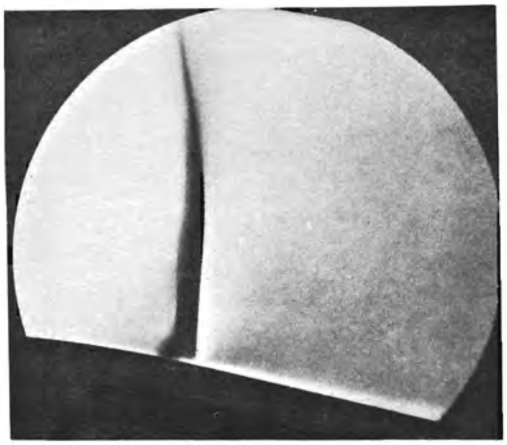
* Communicated through a personal discussion.



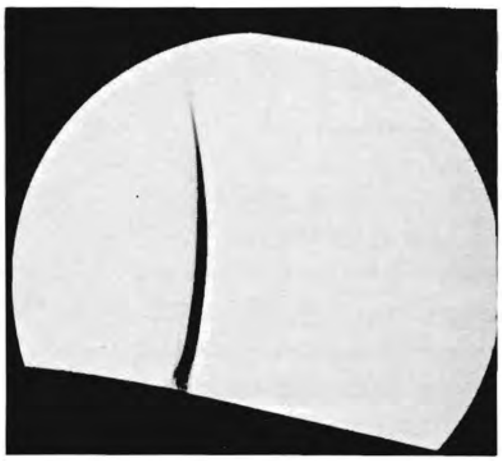
ρ = .0224



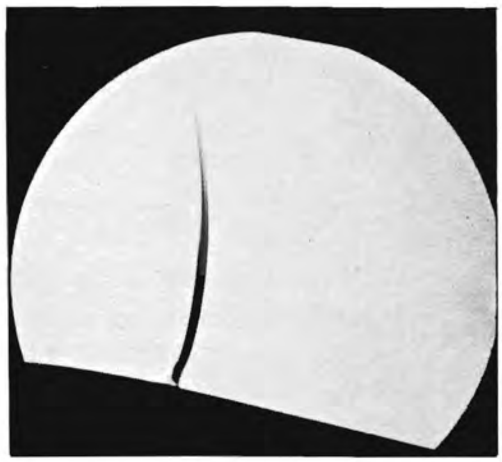
ρ = .0253



ρ = .0287



ρ = .0321



ρ = .0380

ρ = AIR DENSITY IN $kg\,m^{-4}\,sec^{2}$

Figure 12

CONTROL OF THE CRITICAL FLIGHT MACH NUMBER

If an infinitely long wing is placed in a stream of Mach number 0.6 with the span normal to the wind direction, then the flow Mach number is small enough to give shockless flow. Therefore, the drag of the wing is small. Now let the observer of the flow phenomenon move along the direction of the span (Fig. 13) with a velocity corresponding to Mach number 0.5. To this moving observer, not knowing his motion, the flow is the same as that of an infinitely long wing placed in a stream of Mach number $(0.5^2 + 0.6^2)^{1/2} = 0.780$ with a sweepback angle $\tan^{-1}\dfrac{0.5}{0.6} = 39^\circ\ 50'$.

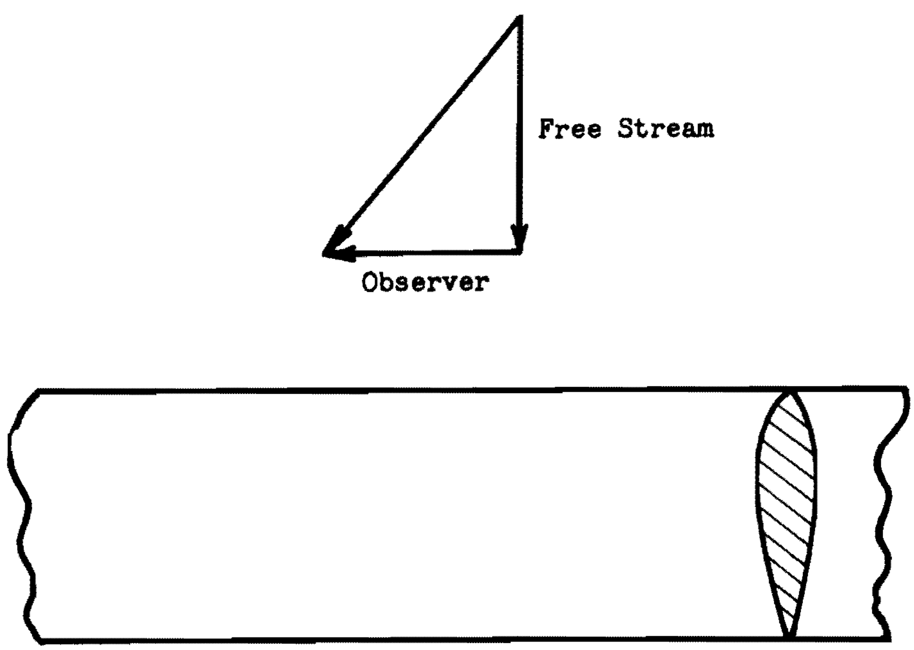


Figure 13

The apparent Mach number of the flow is thus much higher than the critical Mach number for two-dimensional flow, but the physically effective Mach number is only 0.6, since the motion of the observer is only the motion of the reference system and should not change the physical phenomenon. Therefore, by sweeping back the wing, the critical flight Mach number is raised.

Of course the flows in the boundary layer over an actually swept-back wing and a straight wing with moving observer are not the same. However, so long as the danger of shock wave does not exist, the effect of boundary layer on the outside flow is secondary as is well-known in all low-speed flows. Experimentally, this expected superiority of swept-back or swept-forward wings is clearly demonstrated. Due to the end effects of the finite span, the full benefit as calculated by the simple reasoning given above cannot be completely realized. However, the flight Mach number at which there is a sharp rise in drag coefficient can be generally raised by 0.1. Tests* on a series of swept-back wings of 9% thickness (Fig. 14) show that the drag coefficient at a constant value of lift coefficient decreases with sweepback, especially at high Mach numbers (Fig. 15). In fact at Mach number 0.8, the drag coefficient with 45° sweepback is only 1/3 of that of straight wing. The drag coefficient at Mach number 1.2 with 45° sweepback is approximately equal to that at Mach number 0.8 without sweepback. Such drastic improvement in the aerodynamic characteristics at high speeds are indeed most important. The principle is believed to be a fundamental one for all future high-speed airplane design.

The effects of sweepback or sweepforward on the aerodynamic characteristics are not limited to high Mach numbers. Generally the following disadvantages are evident:

(1) The maximum lift coefficient is smaller due to premature stalling.

(2) There is an undesirable shift of the center of pressure near the angle of maximum lift.

(3) The roll stability and the directional stability are reduced at high lift coefficients.

(4) The pitching moment due to the use of flaps is very large.

Thus, much aerodynamic research has to be done before such disadvantages can be eliminated or minimized.

The beneficial effect of sweepback or sweepforward can be seen also from a different point of view: The shock wave over the surface of a body occurs in regions of high flow velocity. Thus, the shock wave on a swept-back wing must necessarily be oblique to the on-coming stream (Fig. 16). It is well known that while a normal shock wave can occur with local Mach number only slightly greater than one, an oblique shock wave can only occur with local Mach number considerably greater than one, depending upon the obliqueness of the wave.

Therefore, the fundamental aerodynamic characteristic of the swept-back wing is oblique maximum velocity line over the surface of the wing. This concept, however, immediately leads to many extensions. For instance, the maximum velocity line over the surface of a wing of very small aspect ratio (say one) is oblique to the stream and thus its critical Mach number must be high. This is actually found to be the case by high-speed wind-tunnel tests. Another possible application of this concept is to make the maximum velocity line over the fuselage of an airplane oblique to the flow direction for raising the critical Mach number of the fuselage. Basically, the essential point of all these innovations is to free the aerodynamic designer from an approximately two-dimensional concept and to think in three-dimensional flow. This additional dimension gives the control of the critical Mach number.

The effective Mach number of a swept-back wing is $M \cos \beta$ where M is the flight Mach number and β is the sweepback angle. Therefore, for very high-speed

* H. Ludwieg, "Pfeilflugel bei hohen Geschwindigkeiten," Lilienthal-Gesellschaft, Bericht Nr. 156, (1942).

Fig. 14

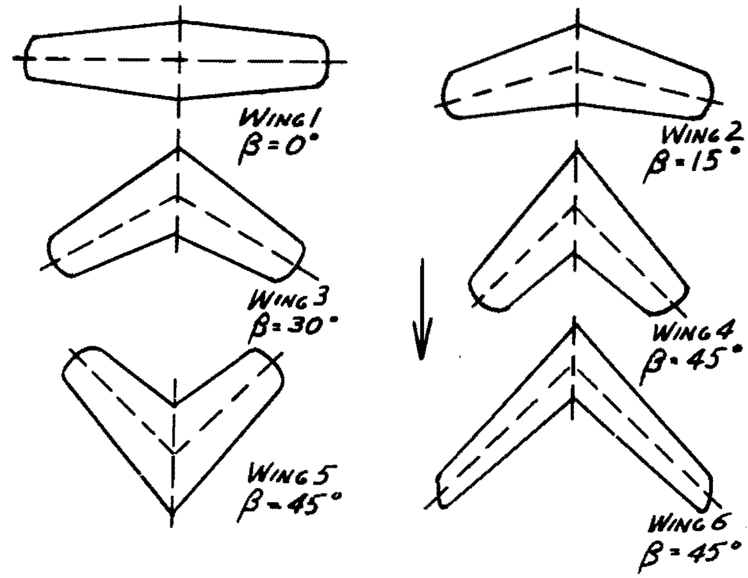


Figure 14

Fig. 15

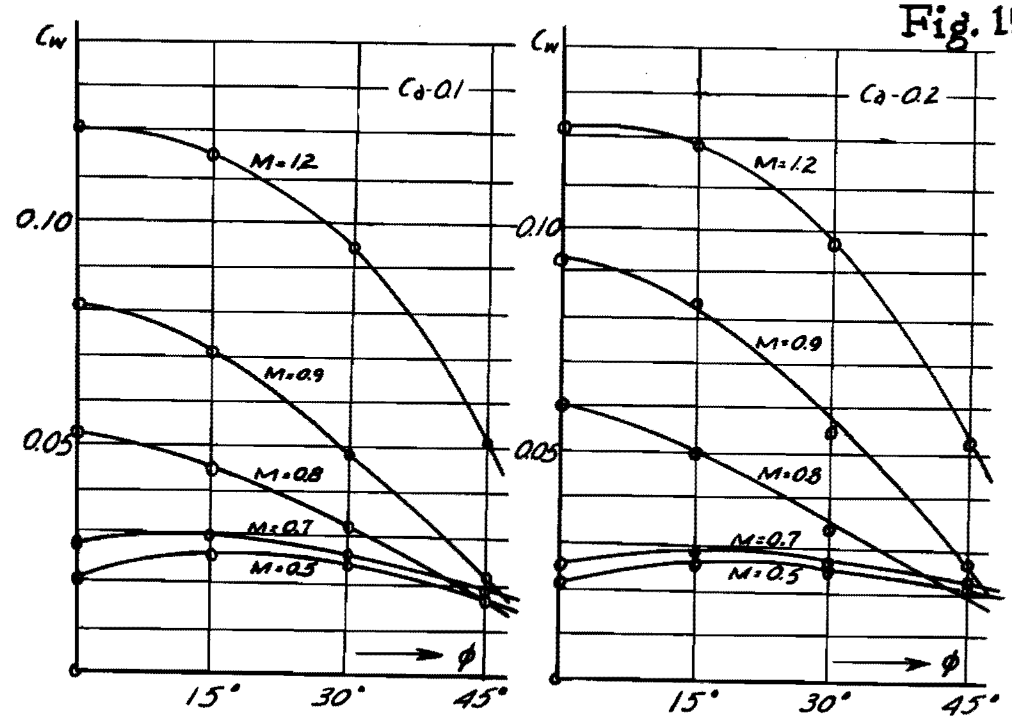


Figure 15

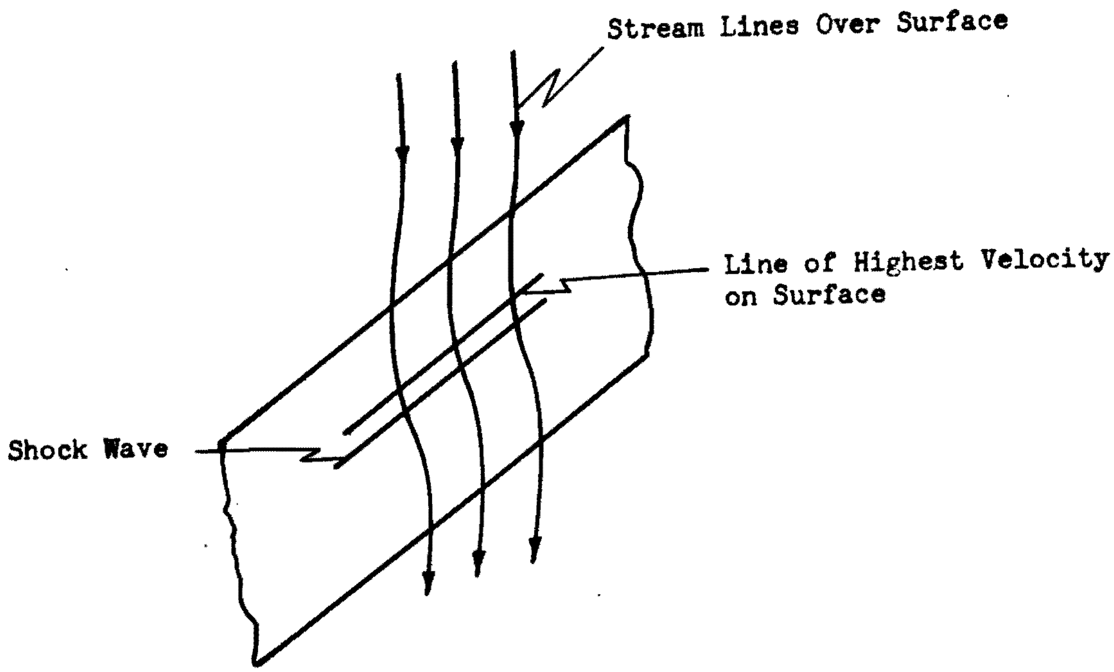


Figure 16

flight with Mach number M-3 say, unless the sweepback angle β is so large as to make M cos β much less than unity, the drag of the wing will be increased by sweepback instead of decreased due to the drag peak near the velocity of sound (Fig. 3). Hence, for very high flight velocities, straight wings may be again the most efficient wing as the possible maximum value of sweepback angle is limited by structural reasons. Of course, the final choice of the wing plan form is closely connected with other elements of the aircraft and can only be decided by careful wind-tunnel tests.

Summarizing then, the adoption of an essentially three-dimensional flow concept allows the possibility of drag reduction at high speeds. However, this new concept also requires exhaustive and detailed high-speed wind-tunnel experimentation before a firm mastering of the tool can be achieved.

INFLUENCE OF JET FROM THE PROPULSIVE POWER PLANT ON THE FLOW AROUND AIRCRAFT

In a general sense, all aircraft propulsive systems are based upon the jet principle. For instance, the propeller slip stream is a jet. However, the use of the recently developed propulsive power plants, such as turbojet, aeropulse, rocket and ramjet, introduces the following new elements:

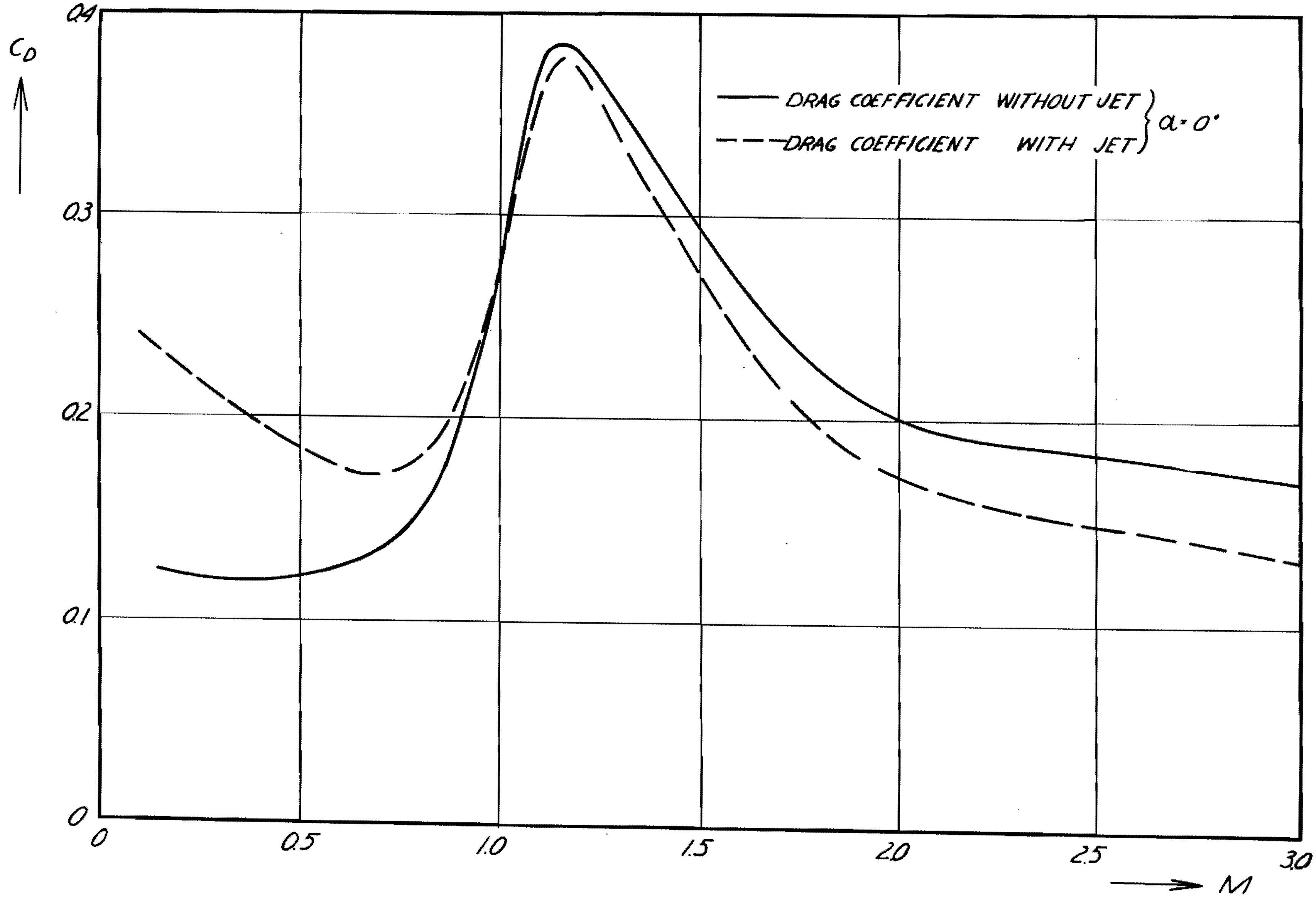
(1) The jet velocity is much higher than in the conventional system.

(2) The jet temperature is also much higher.

(3) During the rapid acceleration period possible with these new power plants, the jet thrust can be many times the drag and thus occupy a much more dominating role in the flow around the body. In fact, very little is known about the mixing, spreading and stability of hot, high-speed jets in high-speed flow.

These factors coupled with the high speed of the aircraft make the interaction between the propulsive jet and the flow around the aircraft much more powerful. There is the problem of optimum air intake and nacelle design for thermal jet engines, especially for swept-back or swept-forward wings due to the essentially three-dimensional character of the flow. The suction effect of a large rocketjet (at the tail of the body) on the flow is also known to be very strong. The German test on a model of the V-2 rocket missile with simulated rocketjet shows that at low speed, Mach number 0.14, there is an increase in drag coefficient of 75%. This increase drops off with rising Mach number and is zero around the velocity of sound. At supersonic velocities, the jet fills up the wake and thus the influence is a decrease in drag coefficient which is larger at higher Mach numbers and becomes 18% at $M = 3$ (Fig. 17). Such strong influence of the jet on the drag of the body must certainly be considered.

Besides the problem of jet action on drag, there is the problem of tail flutter due to jet action. In other words, while by considering the power plant and the aircraft separately, good approximation to the flow problem can be achieved at low flight speeds with less powerful propulsive systems. This convenience in design and in research is lost in the case of high-speed aircraft. Here the aircraft must be considered as a whole. Therefore, many of the design problems of high-speed aircraft have to be solved in high-speed wind tunnels with complete power plant installation and burning of fuel.



C_D
0.4
0.3
0.2
0.1
0
0
0.5
1.0
1.5
2.0
2.5
3.0
M
DRAG COEFFICIENT WITHOUT JET
DRAG COEFFICIENT WITH JET
$\alpha = 0°$

Figure 17

DETAIL DESIGN PROBLEMS OF TRANSONIC AND SUPERSONIC AIRCRAFT

This brief discussion of high-speed aerodynamics would not be complete without mentioning the detail design problems of transonic and supersonic aircraft. One of the outstanding problems is the design of control surfaces. For low-speed aircraft, both spoilers and hinged surfaces are used to vary the lifting force on the control surfaces. However, both devices lose in effectiveness at transonic and supersonic speeds. From the data available at present, the hinged surface still seems to be superior to the spoiler at very high speeds in producing desired control forces. On the other hand, the hinge moment for the conventional design is very high. This is particularly unfavorable for guided missiles as the power of the servomotors must be kept at a minimum to avoid excessive weight. To use aerodynamic balance is also difficult. If the surface is properly balanced at supersonic speeds, then it is aerodynamically overbalanced at subsonic speeds. If the surface is properly balanced at subsonic speeds, then it is not sufficiently balanced at supersonic speeds. It is suggested that all movable surfaces be used, but here there is also the difficulty of large variation in hinge moment in passing through the velocity of sound. Therefore, at present no really satisfactory control surface for transonic and supersonic flight speeds has been designed and much research work has to be done.

In close relation to the control surface problem, there is the problem of the shift of center of pressure of the wing and the body in the transonic range. Of course, with proper design, this difficulty can be solved. But here again exhaustive high-speed wind-tunnel tests are required.

PART II

THE AIRPLANE: PROSPECTS AND PROBLEMS

By

WILLIAM R. SEARS
and
IRVING L. ASHKENAS

PART II

THE AIRPLANE: PROSPECTS AND PROBLEMS

DECEMBER 1945

INTRODUCTION

This report is an attempt to answer the question "What can be expected of the airplane as a military weapon?" To answer this question, we will try to determine:

(1) What are the limitations of the airplane today?

(2) What steps can be taken to remove these limitations?

The characteristics desired of an airplane are determined by its military function, e.g., bombardment or interception; hence, it might be logical to subdivide this report according to function, and to consider the limitations that appear for each type. Upon further consideration, however, it becomes clear that the characteristics desired of military airplanes are substantially alike for all the categories. High speed, fast climb, long range, and great load-carrying ability are wanted for interceptors as well as bombers, and for photo-reconnaissance airplanes as well as transports. Hence, the limitations that we encounter are not likely to depend upon the function of the airplane.

Moreover, the limitations of airplanes are very likely to depend intimately upon the characteristics of their power plants and also upon their flying speeds, which will be greatly influenced by the choice of power plants. Therefore, we have decided to subdivide our discussion according to power plants. We will consider first airplanes with conventional engine-propeller combinations, and then those with newer types of power: turbojects, turbopropellers, rockets, ramjets, and atomic engines.

It will be impossible, in this analysis, to avoid discussing the capabilities and inherent limitations of the power plants themselves, since these will so greatly affect the characteristics of the airplanes. However, these matters do not lie strictly within the scope of the present report, and whenever possible they will be handled by reference to other, independent studies. Many of our conclusions, therefore, must depend for their validity upon the accuracy of these engine data.

AIRPLANES WITH CONVENTIONAL POWER PLANTS

PRESENT ABILITIES

The capabilities of conventional-engine bombardment and pursuit airplanes at the present state of the aeronautical art are adequately set forth in a series of memorandum reports prepared recently by the personnel of NACA (Refs. 1 to 11). The reports present selection charts from which the performance characteristics of airplanes may be read when the basic design parameters, such as power and wing loadings, are given. Since the charts show the values of several performance items as functions of the same independent variables, they can be used to determine what sets of values are consistent, under the assumptions of the calculations.

The charts have been calculated by accepted methods and their accuracy depends only upon the validity of the assumptions made regarding drag, weight, propeller efficiencies, fuel consumption, and, for fighters, aileron effectiveness. These assumptions have been examined and have been found adequately representative of the best airplanes of today.*

Figures 1 and 2 are typical charts taken from these NACA studies (Fig. 1c of Ref. 4, and Fig. 1e of Ref. 11, respectively).

> Selecting an arbitrary case in Fig. 1, for example, we see that it is possible to produce a bomber to have a range of 7300 miles carrying 10,000 lb of bombs, and having a high speed of 400 mph at 35,000 ft, but that greater range cannot be had without sacrificing high speed, and vice versa. Moreover, this bomber requires a take-off run of nearly a mile, and a demand for a shorter take-off would be inconsistent with the other performance items specified.
>
> In the other typical chart (Fig. 2) we see, for example, how the combat performance is affected by the requirement of fighting radius in a fighter. A fighter can have a radius of 400 miles, a high speed of 500 mph, and a rate of climb of 3000 ft/min at 30,000 ft, but if the radius is increased to 1600 miles the rate of climb is reduced to 1500 ft/min, practically regardless of wing loading. (Other charts in Ref. 11, not reproduced here, show that the maximum rate of roll, the take-off run, and the minimum radius of turn are all affected adversely to a serious degree by this increased radius of action.)

We will not consider these selection charts further at this point. They contain such a wealth of data concerning the present abilities of conventional airplanes

* One exception is the assumption of lift coefficient for take-off of bombers (1.3) which appears overly conservative. Therefore, the chart take-off distances for bombers may be somewhat too great. However, this is not a serious point because the influence of these distances on our conclusions is small.

that they cannot very well be compressed or summarized here. They show clearly how almost all the performance requirements affect the others, so that, except very broadly, no concise statements can be made regarding the abilities of present-day airplanes. In view of the existence of these charts, the subject of the present performance of conventional airplanes will be considered a closed one, and we will proceed to consider the possibilities of achieving, in the future and with different power plants, performance outside the indications of the charts.

It is not exactly true that the airplanes represented in these selection charts can immediately be designed and built and their indicated abilities achieved in combat without encountering certain troublesome engineering problems. For example, some of the airplanes shown in the charts employ unusually high wing loadings, and rather effective high-lift devices are probably required to obtain satisfactory landing characteristics in such cases. Moreover, flight at such altitudes as 40,000 ft still introduces difficulties in the operation of many aircraft accessories, turbosupercharger regulators, and the like, and problems of engine cooling. Similarly, flight at 500 mph, as indicated in Refs. 8 and 11, often introduces compressibility difficulties. Finally, it is doubtful that there exist today propellers, wheels, brakes, etc., to satisfy the needs of some of the extreme cases that appear in the charts. But all of these problems are in the engineering stages, i.e., the basic data required for their solution are available to a greater or lesser degree, and we can confidently expect that they can be solved in a relatively short time. Unfortunately, as we proceed to more radical types of airplanes and engines, we will not always find this to be the case, and it will appear that some more serious problems exist.

FUTURE POSSIBILITIES

Let us try to determine here the possibilities of improved performance of conventional-engine airplanes in the future. For bombers, at least, certain interesting data on this subject are available in the NACA studies mentioned above (Refs. 3, 4, 6, and 7).

1. *Aerodynamic Refinement (Drag Reduction).*

In Ref. 4, the effects of drag reduction on the performance characteristics of bombers have been investigated by constructing selection charts with the formula, for the minimum drag coefficient,

$$C_{D_0} = .0090 + .06\ F/S \tag{1}$$

(where F/S is the ratio of fuselage and nacelle frontal area to total wing area), instead of the previously used formula,

$$C_{D_0} = .0120 + .12\ F/S \tag{2}$$

which represents present-day airplanes. Formula (1) may be supposed to represent nearly the ultimate that can be achieved in drag reduction in bombers, probably in the form of tailless or all-wing types, or possibly in the form of conventional airplanes entirely devoid of armament and external radio antennas.

Figure 3 is a reproduction of Fig. 10c of Ref. 4, and shows, by comparison with Fig. 1, the effects of the drag reduction mentioned above.

We see that a six-engine bomber could now have a range of 9500 miles (carrying 10,000 lb of bombs), a high speed of 450 mph at altitude, and quite respectable take-off and climb performance. Alternatively, the 400-mph bomber considered above could be made to have a range of over 10,000 miles, or the 7300-mile bomber could have greatly improved maximum speed and climb.

It is clear that the rewards of aerodynamic refinement are considerable, even in conventional-engine bombers. Since it is possible that the drag coefficient represented by formula (1) can never be achieved in practice, except in the all-wing airplane, we are brought to a consideration of the problems and possibilities of such aircraft.

The greatest experience to date with all-wing airplanes has been that of Northrop Aircraft, Inc., under the direction and with the aid of the Army Air Forces. A number of small airplanes, scale models of large bombardment types, have been built and extensively flown by Northrop during the past five years.

At first, serious problems of longitudinal stability and control forces were encountered in their airplanes near the stall. These have now been solved to the satisfaction of the company and the AAF by the use of wing slots and specially developed power-boost control systems. No other serious difficulties of stability or control in smooth air have ever been met, but pilots report that these light tailless airplanes air difficult to manage in a very turbulent atmosphere. It is difficult to predict how far this characteristic is a result of the low wing loadings involved in the dynamically similar flying mock-ups, and how much improvement will be noted in the full-scale bomber by virtue of its increased size and corresponding increased time scale.* Lest this improvement be insufficient, an automatic pilot especially suited for the all-wing airplane, and capable of being used in rough-air cruising and in the bombing run, is being developed and flight tested in anticipation of the B-35 bomber.

It appears, in fact, that the biggest problem of the all-wing bomber is the actual demonstration of the low drag that is predicted for it. The Northrop experience has not been conclusive in this respect, for the power output of the small engines employed has never been a known quantity. Wind-tunnel tests made by the NACA of the Northtop N9M flying mock-up have shown very promising drag coefficients, of the same order as formula (1) above. Nevertheless, the performance estimates for the B-35 do not indicate as great performance as some of the airplanes based on formula (1) in Ref. 4. This is because the wing loading of the B-35 is to have a very modest value (about 40 lb/sq ft) in order to produce satisfactory landing and take-off speeds.

* Wind-tunnel tests have also shown that the directional stability of the full-scale bomber will be considerably greater than that of the flying mock-ups because of the great increase in propeller effects.

It is clear that improvement in the usable maximum lift coefficients of tailless airplanes is needed.

In conclusion, it is our opinion that the all-wing, conventional-engine bomber offers important possibilities for long range and relatively high speeds, and that the existing problems of this type are not insurmountable and are by way of being solved. Future developments of all-wing designs, directed especially toward the use of greater lift coefficients in take-off and landing, should result in the attainment of substantially the performance indicated in Fig. 3.

There is another possibility for the attainment of the drag coefficients assumed in Fig. 3, namely, the maintenance of laminar boundary layers over the external surfaces of the airplane, especially the wing. At very low speeds, or in very small airplanes, such boundary layers can be maintained over a considerable proportion of the surface if the surface is unusually smooth and clean, and the drag coefficients of formula (1) can be attained. At present, however, the same cannot be achieved in airplanes of practical sizes and speeds, even when great care is taken to manufacture and maintain smooth outer skins and pusher propellers are used to avoid slip-stream disturbances. Basic research is being carried out (and should obviously be continued) to explore the behavior of the boundary layer and transition. It is likely that important practical results will accrue from this research within the next 15 years, but it appears unlikely that much control will be exercised over the boundary layer (except possibly by removing it entirely) in the near future. (See 2 below.)

Our discussion here has concerned only bombardment airplanes. What are the possibilities of drag reduction in fighters and interceptors? Of course, if a fighter and bomber can be made to look alike, the same reductions of drag are possible and the same improvements in range and speed are available. But experience has shown that it is difficult to achieve the all-wing ideal in a fighter of reasonable size, because of the excessive size of the engines and cockpit; hence, the drag assumed in Refs. 8 to 11 probably cannot be much reduced. Moreover, any considerable increase of performance above that of these studies, e.g., Fig. 2, brings our fighters into the realm of severe compressibility effects, overshadowing the problems of drag reduction at more modest speeds considered here, and these effects are discussed separately below.

2. *Drag Reduction at High Speeds.*

On the high-speed side, both the charts of Figs. 2 and 3 lead us to flying speeds (e.g., 500 mph at 35,000 ft or above) that are high enough to ensure the existence of compressibility effects on drag and on stability and control. Also, these speeds are so high that compressibility effects must occur on the propeller blades, and the maintenance of good propulsive efficiencies becomes highly questionable. Nevertheless, the authors of Refs. 1 to 11 have not considered these deleterious effects (and, no doubt, that is why they have not drawn curves for any higher speeds) and, therefore, the selection charts for these high-speed regions must be interpreted as showing what can be achieved if the compressibility effects are minimized.

Until recently, the known methods of reducing compressibility effects at high subsonic Mach numbers were, essentially, the use of very thin wings, the avoidance, by careful filleting, etc., of high local interference velocities, and (possibly, for future development) the reduction of shock-wave effects by removal of the boundary layer. Our German contemporaries, however, have recognized for several years another effective method of postponing to higher speeds the effects of compressibility, namely, sweepback or sweepforward of wings, tail surfaces, propeller blades, etc. Since the spring of this year [1945], this effect has been known in America, and a number of investigations are now under way to determine the extent of its effectiveness.

From a study of the German literature on the subject, it is already clear that this sweepback, or sweepforward, is effective enough in pushing the compressibility boundaries upward, e.g., 100 mph, to be adopted by American designers for fighters and high-speed bombers. We repeat here that, so far as conventional-engine airplanes are concerned, this innovation, applied to wings, tails, and propeller blades, serves only to make available the performance estimates of Refs. 4 and 8, e.g., Figs. 2 and 3 herein.

In fact, this adoption of large sweepback or sweepforward will not be without its problems, for the poor low-speed, e.g., stall, characteristics of heavily swept wings are well known. Probably tip slots will be required to improve these characteristics, but it is still doubtful that as high maximum lift coefficients will be obtained as are obtained with nearly straight wings. These are also structural problems, such as the tendency of swept-back wings to load up at their tips thus increasing the bending moments at high angles of attack. Therefore, the selection charts are not strictly accurate, so far as weight estimates and maximum lift coefficients are involved, for airplanes with great sweepback, and immediate attention must be given to both the aerodynamic and structural problems of swept wings in order to achieve the expected airplane performance.

The alternative remedies for subsonic compressibility troubles are less promising. Admittedly, the use of thin wings is effective in raising the compressibility limit* and will surely be combined with the use of sweepback in high-speed airplanes, but it rapidly increases the structural weight of the wing, and thereby invalidates the range and radius figures of the selection charts. Boundary layer removal at shock-stalling speeds has been investigated only superficially (by the British and the Germans), and the indications so far as that the shock stall can be postponed only a few miles per hour by the expenditure of a great deal of power and weight in the blower system. Nevertheless, it is recognized that the most severe compressibility effects, both in drag and in control, arise from the behavior of the boundary layer and not, as was once believed, directly from the existence of the shock wave. Hence, it becomes increasingly important that fundamental knowledge of thé boundary layer be obtained and that this research be extended to compressibility speeds.

3. *Improved Fuel Consumption.*

The effects of improved fuel consumption in conventional engines has been investigated (in Ref. 6) relative to bomber performance. Various assumptions were made regarding the expected improvement in fuel consumption, including the "ideal"

* The difference in critical speeds of wings of 10 and 16% maximum thickness ratios is 50 mph. if the wings have no swing back, it is greater as the sweep is increased.

case of a uniform brake-specific consumption of 0.35 lb/hp-hr at all power settings up to military. The range is the only performance item affected, and a typical graph showing how maximum range is affected, for one arbitrary choice of power and wing loadings, is reproduced in Fig. 4 (Fig. 3 of Ref. 6). We see that the possible gains are appreciable in long-range bombers. Moreover, the possible improvement in range at higher powers, i.e., at higher cruising speeds, is much greater, as would be expected. Similar results are not available for the range or radius of fighters, but it is clear that the improvement would be comparable. Again the range at higher power settings will be most affected, and this would appear to be especially important for night fighters and other interceptors.

Actually, the degree of improvement represented by the test-stand curve in Fig. 4 is relatively modest, since it implies only the carrying over into actual operations of consumptions that have been obtained in ideal test-stand runs, and the extension of this economy to a wide range of power settings. Much progress has recently been made in this direction by the use of improved fuels, and there are AAF airplanes in service today in which normal-rated power can be drawn with very lean mixtures without detonation. Consequently, the performance advantages indicated by the test-stand case may be considered as practically at hand.

On the other hand, the ideal case (uniform 0.35 lb/hp-hr) lies considerably farther in the future; it implies a thermal efficiency of nearly 40%. This will probably be achieved, within the next decade or two, by a combination of antidetonating fuels and improved, temperature-resistant metals, together with fuel injection to the cylinders (to obtain uniform distribution). Appreciable improvement beyond this point appears very unlikely, however, for the next 20 years. There is also little prospect of producing practical fuels with heat values much higher than gasoline, to obtain better specific consumptions at available thermal efficiencies.

4. *Improved Propeller Efficiencies.*

The NACA studies have been made under the assumption of propulsive efficiencies of 85% for bombers and 78 to 85% for fighters. From recent test data on models of the latest propeller designs, it appears that efficiencies about 5% greater than these may be expected in very favorable cases in the future, for flying speeds well below the speed of sound. Approximately, this would increase the range and radius estimates 5%.

Undoubtedly, swept-back propeller blades will be developed in the next few years to improve efficiencies at high Mach numbers. As mentioned above, this will permit our estimated performance values to be obtained, rather than introduce any unforeseen improvements.

5. *Structural Improvements: Reduced Weights.*

The selection charts of the NACA are based on certain assumptions regarding structural and other weights that have been determined from studies of existing airplanes. In Ref. 6 the effect on range was determined for reducing or increasing the structural weight 30%. A typical result is shown in Fig. 5 (Fig. 9 of Ref. 6); the effect on maximum range without bombs is nearly 30%. This improvement might arise from refinement of structural design, use of improved materials, reduction of power-plant weight, reduction of the weight of military equipment, or any combination of these. Let us consider these possibilities briefly:

> The present-day airplane structure is an extremely efficient one, and it is not likely that any great savings in weight will be achieved by elimination of unnecessary structure. It is more probable that structural designers and research workers will be hard-pressed to maintain their present structural weight ratios in spite of new military features, such as pilot ejection, and aerodynamic innovations, such as sweepback, smoother skins, thinner wings, etc., and higher speeds.
>
> Airplane structural materials have been constantly improved by the use of improved heat treatments and the like, but these improvements have been relatively small and have been applicable only to certain parts of the airplane structure, such as spar caps. Specialists in this field expect further increases in ultimate stress of special light alloys, but not enough to alter appreciably the structural weight ratios of airplanes in the next decade or two.
>
> The specific weights of conventional aircraft engines have diminished continually over the years, but the same cannot be said of installed airplane power plants. The weights of superchargers, intercoolers, fuel and oil systems, and other equipment has risen so rapidly as to overshadow the reduction of engine weight. There is no reason to suspect that either of these trends will change radically in the future.
>
> The continual increase of weight of military equipment (armament, radio, radar, flak suits, armor, etc.) during the past decade will also undoubtedly continue. In fact, this is another department in which we find that research and development will be needed to maintain our present position without achieving substantial improvement.

In summary, it appears unlikely that we can obtain structural weight ratios anywhere near the 70%-of-normal assumed in Ref. 6 (Fig. 5 herein). It is more likely that improvements in materials, design refinement, and power-plant weight will be required to offset other innovations and improvements without increasing the proportion of fixed weight.

6. *Higher Wing Loadings.*

It will be noted that both of the NACA selection charts for bombers reproduced here (Figs. 1 and 3) show decidedly diminishing returns due to increased wing loadings above about 80 lb/sq ft. This is, in fact, characteristic of all the charts of Refs. 1 and 7. Maximum speed, for any given power loading, increases with increasing wing loading only up to a fairly reasonable value (such as 80 lb) and then diminishes; maximum range behaves similarly; and rate of climb and take-off distance naturally deteriorate continually with increasing wing loading.

In fact, it appears that the best present-day flap arrangements are nearly adequate to permit the use of wing loadings as great as will be desirable for conventional bombers for some time to come.

> Present-day flap arrangements are capable of providing maximum lift coefficients of about 2.5 (power off) for landing; this permits a stalling speed of 112 mph in an airplane loaded to 80 lb/sq ft. This does not appear excessive in view of the tremendous effects of power and the fact that the bomber should seldom be required to land at full weight. Moreover, present-day flaps permit take-off at a coefficient of about 1.8, while the selection charts are drawn for the value 1.3 at take-off.*

One important exception appears in the case of tailless and all-wing airplanes, where lift coefficients presently available are somewhat lower than are desirable to achieve the performance of Fig. 3. This is a very new field, however, and it is certain that improvement will be rapid.

> Actually, the take-off lift coefficient assumed in Fig. 3 (1.3) is not much greater than that estimated for the B-35 Flying Wing bomber on the basis of wind-tunnel and flight tests. Certain modifications proposed for future development by the manufacturer are very likely to raise the available value to at least 1.3 in a few years.

For the fighters considered in Refs. 8 to 11, the situation is somewhat different, since some increase of wing loading above presently acceptable values, e.g., 50 lb/sq ft, appears to be desirable to increase high speed, albeit at the expense of minimum turning radius, rate of climb, and fighting radius. It may be expected that the use of full-span flaps with retractable ailerons, or the equivalent, will permit fighter wing loadings of 65 to 70 lb/sq ft within a few years. On this basis it is safe to extrapolate the curves of Fig. 2 to higher loadings and corresponding higher maximum speeds.

7. Increased Size.

There are certain indications that a continued increase in airplane size is desirable from the performance viewpoint, at least for bombers and transport airplanes:

> (a) A comparison of 2-, 4-, and 6-engine bombers using 3000-hp engines is shown in Ref. 4. Figures 2a and 2c of Ref. 4, show, for example, general tendencies for the speed and range to increase with size, even when the take-off distance is fixed. This tendency is also found in Ref. 3, where the performance of 1-, 2-, 4-, and 6-engine bombers using 2000-hp engines is compared.
>
> (b) Similar tendencies are observed in Ref. 5, where bombers are compared having four 1200-, four 2000-, and four 3000-hp engines.
>
> (c) Statistical data show a continual increase of bomber weight during the past 20 years, and an accompanying great improvement in performance.

* Note that the take-off distance is inversely proportional to the take-off lift coefficient, approximately.

Let us investigate briefly the significance of these indications. Do range and speed necessarily increase with size? Is there no practical limit to the size of airplanes? Some information is obtained from simple dimensional analysis of performance formulas:

(1) The Breguet formula for the range of conventional-engine airplanes is

$$R = 863 \frac{\eta}{C} \frac{L}{D} \log_{10} \left(\frac{W_1 + F}{W_1}\right) \text{ (miles).*} \tag{3}$$

Hence, considering a geometrically similar family of airplanes of varying size, having propellers of equal efficiency and engines of equal specific fuel consumption, we see that the range does not vary with size if the ratio of fuel weight to landing weight remains constant.

(2) However, if the weight W_1 includes any pay load, such as cargo, crew, armament, or bombs** the range remains constant with these loads held as fixed percentages of the landing weight, i.e., the load-carrying ability of the airplane is increased in proportion to the weight. This simply confirms the fact that bigger airplanes can carry bigger loads over the same distance.

(3) If the wing loading of a family of geometrically similar conventional airplanes is held constant, i.e., the weight varies as the span squared, the landing speed and distance are constant. Also, the cruising speed for best range remains constant and the power loading at cruising (and presumably also the power loading based on full power) is constant. The high speed and the take-off distance are then constant. The only advantage of size, then, assuming constant weight ratios as in (1) and (2) is the ability to carry a bigger bomb load. It is questionable whether a number of small airplanes would not be preferable to a geometrically similar large one, providing they can carry the largest size of bomb required.

(4) On the other hand, if the wing loading of the family is varied in proportion to the span b, so that the landing speed increases with $\sqrt{b}$ and the landing distance with b, then the cruising speed for best range increases with $\sqrt{b}$ and the power loading at cruising decrease with $1/\sqrt{b}$. If this can be accomplished (without destroying the similarity of weight ratios) the range will be constant, the high speed proportional to $\sqrt{b}$, the take-off distance proportional to $\sqrt{b}$, and, of course, the gross weight and pay load proportional to b^3. There is then some advantage in performance to be gained in large airplanes, but only at the expense of landing and take-off distances.

Let us examine more closely the assumption of constant weight ratios made in all the cases above. The weight of structural material employed to resist shear in

* η = propulsive efficiency; c = fuel consumption in lb/bhp-hr; L/D = lift/drag ratio; W_1 = landing weight; F = weight of fuel; R = range in miles.

** The Breguet formula should be modified somewhat to account for bombs dropped at midrange; however, the dimensional relationships remain unchanged.

the geometrically similar airplanes, assuming the same material used in all, will be proportional to the structural cross-sectional area A and the length parameter b. Moreover, for equal allowable stress,

$$A \backsim W \tag{4}$$

$$[\text{Shear structure weight}] \backsim Ab \backsim Wb$$

For bending material, a similar analysis, based on the stress formula

$$\sigma = Mc/I \qquad (5)^*$$

shows that

$$A \backsim (Wb)^{2/3} \tag{6}$$

and

$$[\text{Bending structure weight}] \backsim Ab \backsim W^{2/3}\, b^{5/3}, \tag{7}$$

Hence, we see that for both types of structure the weight tends to increase with size more rapidly than was assumed in the preceding discussion.

(5) For the case treated in (3), $W \backsim b^2$, we find

$$[\text{Shear structure weight}] \backsim Wb \backsim b^3, \tag{8}$$

$$[\text{Bending structure weight}] \backsim W^{2/3}\, b^{5/3} \backsim b^3, \tag{9}$$

i.e., the structural weight ratio must increase in proportion to b, and it is not legitimate to assume constant weight ratios in computing range.

(6) For the case treated in (4), $W \backsim b^3$, we find

$$[\text{Shear structure weight}] \backsim Wb \backsim b^4 \tag{10}$$

$$[\text{Bending structure weight}] \backsim W^{2/3}\, b^{5/3} \backsim b^{11/3}, \tag{11}$$

i.e., again the structural weight ratio must increase, although not quite as rapidly as b, and it is not legitimate to assume constant weight ratios.

Our analysis seems to indicate that the performance of geometrically similar airplanes should deteriorate as the size is increased, in contrast to the trends noted above. The observed improvement of range with size must therefore be attributed to other effects; for example:

(1) Much of the structural material in an airplane is not designed for either shear or bending, but rather by minimum gage requirements, for practicability of assembly or maintenance, or to support pieces of equipment of fixed size. In addition, an increase of size of some structural members permits the designer to improve the efficiency of his structure by introducing refinements that would be impractical on a smaller scale. Hence, the weight of part of the structural material is not controlled by the equations derived above, and the structural weight ratio does not always deteriorate with increasing size.

* σ = stress; M = bending moment; I/c = "section modulus" of the beam (proportional to the linear dimension of the beam, cubed).

(2) The weights of some fixed equipment including part of the military load, such as crew and crew facilities, armament, etc., do not necessarily increase as the airplane size is increased. Therefore, the ratio of take-off weight to landing weight may not deteriorate as rapidly with size as the structural weight ratio.

(3) Increases in size allow the distribution of a greater proportion of the gross weight along the wing span. (Note that wing volume $\backsim b^3$.) The attendant reduction in wing loads makes the foregoing formulas partially inaccurate and permits some reduction in structural weights.

There is little reason to doubt, however, that the deleterious effects of the structural weight ratio will be encountered in very large airplanes. It is difficult to predict at what size the compensating effects mentioned here will cease to counteract the harmful tendencies of the structural weight, but we believe that the NACA selection charts (which are based on constant weight ratios) should definitely *not* be used beyond gross weights of 300,000 lb until further experience with large airplanes (e.g., XB-36, Hughes flying boat, etc.) is available to confirm the assumption made therein.

There are other difficulties in airplane design, construction, and operation that increase with increasing size, but which are even more intangible than the effects discussed above, and are more difficult to predict. Among these are size of hangars, difficulties of handling on airports, design loads on runways, and cost (man-hours or dollars) of design and fabrication. None of these seems to be a basic or insuperable difficulty, however, and undoubtedly means will be devised to cope with all of them if very large airplanes are found to be required for military uses.

One further problem that arises in connection with very large airplanes is that of control-system actuation. This problem arises simply from the fact that whereas control moments tend to increase with the cube of the linear dimension (even at constant airspeed), the control moments available in the pilot's muscular efforts do not depend on the size of the airplane. Several very successful power-boost control systems have been developed and put into production to solve this difficulty, and there seems to be no necessity to consider control forces a primary problem any longer. Similarly, the existing problems in the design of wheels, tires, brakes, and shock struts for outsize airplanes will undoubtedly yield to engineering development or be circumvented by use of multiple wheels, etc.

In summarizing, we conclude that the erstwhile improvement of airplane performance with increasing size is in conflict with certain inherent principles of mechanics and cannot continue indefinitely. Probably the deleterious effects of these tendencies are being felt in the largest airplanes now being designed and built; if not, they must be encountered in larger ones. Other difficulties in the design and operation of large airplanes do not appear to be unsolvable. If very large airplanes are required to transport very heavy bombs or equipment for ground armies, they will undoubtedly be produced; but their performance must suffer as the result of the effects discussed here.

POSSIBILITIES FOR SUPERSONIC FLIGHT

Let us consider briefly the possibilities of the engine-propeller-driven airplane for supersonic speeds.

One question that arises immediately is that of propulsive efficiencies at supersonic speeds. It has long been recognized (and occasionally exaggerated) how the efficiency of a conventional propeller must diminish as the flying speed approaches the speed of sound, so that the existence of compressibility phenomena along most of the blade length cannot be avoided. Nevertheless, we know that reasonably efficient wings can be designed for supersonic flight, and, since a propeller blade is essenessentially a wing following a helical path, it seems reasonable to demand a usable supersonic propeller. But unfortunately no such propeller has ever been built and tested, and, in fact, there exists no adequate theory for its design or the prediction of its characteristics.

Rough estimates, based on known characteristics of supersonic wings and on the assumption that severe induction effects can be avoided, as they can be in wings, indicate that practical efficiencies may be obtained (say 70 to 75%) at supersonic speeds, but that the efficiency will probably fall off rapidly as the propeller deviates from its design condition. The same estimates have been extended to calculate the thrust for take-off of such propeller designed for supersonic operation, and it appears that a fairly respectable amount (such as half that of present-day propellers, per horsepower) will be available.

It is important that the theory of wings for supersonic speeds, which is now in early stages of development, be extended to provide a supersonic-propeller theory. Propeller tests must then be carried out in supersonic wind tunnels, and effects of major parameters, such as sweepback, determined. This program is needed to evaluate the possibilities of conventional-powered airplanes at high speeds, and, more important, to provide for the use of high-output gas turbines in the future (see "Turboprop," p 52).

A questionable point in the application of conventional engines to high-speed flight is engine cooling. The adiabatic temperature rise of the atmospheric air upon being brought to rest (or at least decelerated) at the face of an air-cooled engine or a radiator may be so great as to make cooling inherently difficult.

This temperature rise is given by the formula

$$T = 1.8\ (V/100)^2\ \text{(Degrees F)} \tag{12}$$

where V is the speed in mph. It is clear that the rise is sufficient to give cooling difficulties. (In fact, it already does. It is on account of this rise that cooling at high speeds is often as difficult as cooling in a climb.) Nevertheless, it is not clear that the problem is insurmountable. For example, at 1000 mph, where the rise amounts to 180°F, the cooling-air temperature is still considerably lower than the allowable cylinder-head temperature (about 500°F, for present materials), and since very high ram pressures may be available at the face of the engine, it may be possible to force enough cooling air through the cylinder fins to cool satisfactorily. Whether this can be accomplished without an inordinate expenditure of power is doubtful, and depends on how much of the heat rejected to the cooling air can be utilized in jet propulsion (the so-called "Meredith effect"*).

It is clear that liquid cooling, using fluids that boil at temperatures lower than the cylinder temperature, imposes even greater difficulties at high Mach numbers. It may even be necessary to adopt coolants that can be used at temperatures higher than the cylinders; this, of course, would require an expenditure of mechanical power in a virtual refrigeration system, since the heat would not pass from the cylinder to a hotter substance of its own accord.

In conclusion, it appears that propellers show some promise for use at supersonic speeds (with conventional engines or otherwise) but a whole new theory of supersonic propellers, as well as supersonic wind-tunnel data, is needed to confirm their usefulness. Conventional reciprocating engines do not appear as promising for supersonic use, in view of an inherent difficulty of cooling. It will be seen farther below that this cooling difficulty is only one of several shortcomings of the conventional engine for very high-speed airplanes, which may lead to its discard in favor of newer types of engines.

SUMMARY

The present abilities of conventional-engine airplanes are set forth in a series of selection charts prepared by the NACA. Considerable improvement over present performance can be expected by virtue of drag reduction, improved fuel consumption, and increased propulsive efficiencies, especially if the advantages of the all-wing airplane can be realized. Simultaneous advances in high-speed propeller design must be made if the full promise of these improvements is to be achieved. No great improvements are expected to result from structural refinements nor from further increases of wing loading beyond the highest values now in use. Moreover, there are inherent disadvantages in the structural weight ratios of large airplanes; as a result, if the present trend to larger airplanes; as a result, if the present trend to larger airplanes is continued indefinitely it must involve some sacrifice in performance. The use of conventional engine and propellers for supersonic flight depends on the development of special supersonic propellers, which looks promising, and the solution of the engine-cooling problem, which is more doubtful.

* If the "Meredith effect" is neglected, the cooling power expenditure becomes clearly exorbitant. For present-day engines this power varies as $(\Delta T)^{-5.5}$, where — T is the difference between cylinder and air temperatures. At 1000 mph, according to the result above, ΔT may be only 60% of its low-speed value; this would increase the cooling power loss, 17 fold, i.e , to 85 or 100% of the engine brake horsepower.

AIRPLANES WITH IMPROVED POWER PLANTS

TURBOJET

1. *Introduction.*

By turbojet we denote a jet-propulsion unit powered by a gas turbine. According to the title of this section, we consider such a power plant to be "improved" in relation to a conventional engine and propeller. In what respects do the improvements appear, and to what extent are they inherent advantages of jet propulsion or the gas turbine?

The primary advantage of jet propulsion for airplanes is the elimination of the propeller. As we have mentioned previously, the conventional propeller suffers from compressibility phenomena at high speeds, and its efficiency drops off. Every jet-propulsion device operates like a propeller in that it produces thrust by imparting a rearward velocity to a mass of air (or other working fluid) In the turbojet (as well as the turbofan* and motor jet**), however, the propelling means is housed within a duct so that the conditions under which its blades operate are at least partially controllable. The Mach number of the blades is therefore not necessarily as great as the Mach number of flight, and the associated compressibility losses can be avoided to some extent.

At lower speeds, on the other hand, where compressibility losses are not severe, the jet suffers in comparison to the propeller by reason of its reduced diameter.

The ideal efficiency of a propeller or jet is equal to $\frac{1}{1 + \frac{1}{2}\Delta V/V}$ where V is the flying speed and ΔV is the slipstream velocity increment imparted by the propeller or jet. Neglecting changes in density, $\Delta V/V$ is roughly proportional to $T/V^2 D^2$, where T is the thrust and D the disk or jet diameter. Thus, for the same thrust, the ideal efficiency of the large-diameter propeller is always greater, but the advantage tends to disappear, even aside from compressibility effects, as the speed increases.

For example, the ideal efficiencies of 2- and 12-ft propellers (the former representing approximately a jet device) are compared below, assuming that both put out 1000 lb thrust:

Flying speed (mph)	200	400	600
Efficiency of 2-ft propeller	.54	.77	.87
Efficiency of 12-ft propeller	.96	.99	.995

The actual maximum efficiency of a propeller is usually about 85 or 90% of its ideal efficiency.

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* A combination of gas turbine and internal fan.

** A combination of conventional engine and internal fan, arranged for jet propulsion.

It is clear that the advantage of the jet at high speeds, as well as its disadvantage at low speeds, is not related to the use of a gas turbine as the primary source of power. If a conventional engine were used, the same characteristics would ensue.

On the contrary, the apparent virtue of the gas turbine for aircraft is its ability to take advantage of the flight ram at high speeds, in the form of increased power output and decreased fuel consumption, and this characteristic is obtained whether the turbine is employed in a turbojet, turbofan, or turbopropeller. This property of the gas turbine enables it, in a typical case, to increase its power output from 5750 bhp at 200 mph to 7700 bhp at 500 mph at sea level, with an accompanying decrease of specific fuel consumption from 0.80 to 0.67 lb/bhp-hr, due only to the increase of entrance pressure and density, at constant rpm, and under the restriction of constant turbine-inlet temperature.*

But even in this respect the advantage cannot be considered inherently peculiar to the gas turbine. An unsupercharged reciprocating engine installed so as to be "supercharged" by flight ram, would have its manifold pressure increased from about 31 to about 40 in. mercury in going from 200 to 500 mph at sea level, and its power output at constant rpm would be increased in about the same ratio as that of the typical turbine described above. The important distinction between the two power plants is easily recognized, nevertheless, in the numerical values quoted for the turbine; there are no reciprocating engines today that can put out 7700 bhp (or even 5750).

Thus we conclude that the greatest virtues of the gas turbine for aircraft do not arise from an inherent aerothermodynamic advantage over the reciprocating engine, but rather from its characteristics of simplicity and lightness, and its ability to produce enormous amounts of power unrestricted by detonation and cooling. Actually, these characteristics are so attractive as to justify the use of the turbine in aircraft, even in the absence of any inherent aerothermodynamic advantage, and speculation regarding the use of other kinds of engines for jet propulsion is probably unfruitful.

2. *Present Abilities.*

The improvement offered by the turbojet, as discussed above, can be summarized for present-day machines, in the following approximate characteristics:

(a) Thrust nearly independent of speed at any given rpm and altitude.

(b) Fuel consumption nearly constant in pounds of fuel per hour per pound of thrust.

It should be clear from the discussion above that these properties are not inherent features of the turbojet but rather empirically determined characteristics resulting from the interplay of several effects under the practical restriction of constant turbine-inlet temperature. Nevertheless, they are exact enough to justify an analysis of the aerodynamic characteristics of airplanes whose power plants are so characterized. From the results of such ananalysis we may determine (1) what design features should be incorporated in turbojet-propelled airplanes to obtain the best performance,

* This temperature, which is determined by the compressor-outlet conditions and the fuel-air ratio, constitutes a practical limitation of the power output, in view of the limited temperature toleration of the turbine materials.

(2) what are the present abilities and limitations of such airplanes, and (3) what future possibilities lie in the ultimate application of the correct design principles.

We begin with a discussion of airplane range:

The Breguet range formula (Formula 3), which teaches all the principles of economy in the case of conventional-engine airplanes, is no longer applicable, since it is based upon a uniform consumption in pounds per horsepower-hour. The analogous formula for a turbojet-propelled airplane is*

$$R = \frac{30}{c'} \sqrt{\frac{b_e}{\sigma}}\, f^{-1/4} \sqrt{W_1} \left\{1 - \sqrt{\frac{W_2}{W_1}}\right\} \text{ (miles)} \tag{13}$$

or

$$R = \frac{30}{c'} \sqrt{\frac{W_1/S}{\sigma} \frac{A_e}{C_{D_p}^3}}\, 1/4 \left\{1 - \sqrt{\frac{W_2}{W_1}}\right\} \text{ (Miles)} \tag{14}$$

The following characteristics are implied:

a. The range increases with increasing altitude, instead of being nearly independent of altitude as in conventional airplanes.

b. The range increases with increasing wing loading (W_1/S), even though the weight ratio W_2/W_1 is constant. This is in vivid contrast to the conventional case.

Both of these new features clearly result from the property of fuel rate proportional to thrust, which implies increasing economy at high speeds. With both increasing altitude and increasing wing loading the airplane flies faster, and takes advantage of the better economy.

There is a limit, of course, to the altitude at which any given airplane can fly at the maximum-range condition assumed in deriving formulas (13) and (14). This is the altitude at which the drag in level flight at the maximum-range condition is just equal to the thrust available; i.e., it is the ceiling for flight at the best range condition.** When this condition is assumed in calculating range, the following formulas are obtained:

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* Here R = range (miles); c' = fuel consumption (lb/hr/lb-thrust); b_e = effective span (ft); σ = air density ratio; $f = C_{D_p} S$ = parasite area (sq ft); W_1 = initial gross weight (1b); W_2 = final gross weight (lb); S = wing area (sq ft); $A_e = b_e^2/S$ = aspect ratio; and C_{D_p} = parasite drag coefficient. The condition for maximum range is found to be $C_{D_p} = SC_{D_i} = 3C_L^2/\pi A_e$. ($C_{D_i}$ = coefficient of induced drag; C_L = lift coefficient.)

** This condition occurs when

$$T = \frac{D}{L} W = \frac{(4/3)\, C_{D_p}}{C_{D_p}\, \pi\, A_e/3} \cdot W = \frac{4}{\sqrt{3\,\pi}} \sqrt{\frac{C_{D_p}}{A_e}} \cdot W, \tag{15}$$

where D = drag, L = lift, W = gross weight, and T = thrust. In this report we assume that the variation of turbojet thrust with altitude is such that T is proportional to $\sigma^{0.7}$ up to the tropopause and to σ in the stratosphere. When these relations are combined, an expression is obtained for σ in terms of airplane parameters and T_o, the sea-level static thrust.

$$R = \frac{35.9}{c'} \frac{b_e}{f} \sqrt{T_o} \log_{10} \left(\frac{W_1}{W_2}\right) \text{ (Miles)} \qquad (16)$$

or

$$R = \frac{35.9}{c'} \frac{\sqrt{A_e}}{C_{D_p}} \sqrt{\frac{T_o}{S}} \log_{10} \left(\frac{W_1}{W_2}\right) \text{ (Miles)} \qquad (17)$$

or

$$R = \frac{2.05}{c'} \left(\frac{L}{D}\right)_{max} V_o \log_{10} \left(\frac{W_1}{W_2}\right) \text{ (Miles)} \qquad (18)$$

where $(L/D)_{max}$ is the maximum lift/drag ratio for the airplane and V_o is the approximate maximum speed of the airplane at sea level, neglecting compressibility effects on drag.

It now appears that: (a) The best range is determined by much the same factors as for conventional-engine airplanes; but in addition, (b) the best range is affected by thrust loading in such a way that, all else being the same, the greater an airplane's top speed, the greater its range.

In order to illustrate these effects and to show how the range, high speed, and take-off distance are interrelated, we have prepared selection charts (Figs. 6 and 7) for families of airplanes powered by turbojets of 8000 and 16,000-lb sea-level static ratios. In preparing these charts, we have made the following assumptions:

(1) The fuel consumption is 1.2 lb/hr/lb-thrust.

(2) The range is obtained from formula (16) above, neglecting climb and glide and with no allowance for take-off.

(3) Weights have been estimated from statistical data based on analyses of military airplanes.

(4) Drags have been corrected for compressibility effects on the assumption that sweepback of 45° (measured at the 50%-chord line), is employed. This results in no correction to low-speed drag values throughout the cruising range, but substantial increments of drag at Mach numbers of .95 and above.

(5) Take-off is performed at a lift coefficient of 1.4.

The airplanes represented in Figs. 6 and 7 are supposed to be bombers; nevertheless, the range shown is for zero bomb load. It is believed that this range is at least representative of the range performance of the machines. Similarly, the high-speed performance is believed to be indicated by the sea-level top speed alone, and the take-off requirements are believed to be indicated by the ground run. In general, we believe that the calculations leading to Figs. 6 and 7 are somewhat optimistic, and that the charts represent about the best present possibilities of turbojet airplanes having the thrust ratings stated.

Inspection of the charts discloses that the top speeds available are considerably greater than those of the conventional-engine airplanes previously treated. The ranges, on the other hand, are considerably shorter. There is a definite advantage, according to Figs. 6 and 7, to doubling the power of a turbojet airplane, even at constant wing and power loeadings.* An analysis of our calculations shows that this improve-

* Figure 7 has been included primarily to illustrate this effect and therefore is not nearly as complete a chart as Fig. 6.

ment in the larger airplanes results from (1) relative suppression of the fuselage as regards both drag and weight (because its volume increases faster than its area, with increased dimensions), and (2) reduction of the percentage of total weight in fixed equipment. The improvement is achieved in spite of a definite tendency for the structural weight ratio to increase with size, as has been discussed in the section on page 30, "Airplanes with Conventional Power Plants." Just as for conventional-engine airplanes, the deleterious effects of increased size are not felt in airplanes of the sizes considered here, and it is not clear at what size they will become serious. It cannot be concluded from Figs. 6 and 7 that continual further increase of size (e.g., total rated thrust) will continually improve the performance.

It appears from Fig. 6 that, whereas top speeds approaching the speed of sound are theoretically available, these would be achieved only in rather short-range airplanes. What might be called high supersonic speeds (over 1000 mph), do not seem to be available in turbojet airplanes at present. These conclusions would not be appreciably altered by the assumption of sweepback angles greater than 45°, since none of the cases represented in the charts involve large compressibility corrections to the wing drag.

Furthermore, the use of these airplanes as pilotless missiles, with take-off accomplished by external launching devices, would not provide appreciably higher speeds, since all of the faster airplanes in Fig. 6 are already characterized by short take-off runs.

3. *Future Possibilities.*

a. IMPROVED FUEL ECONOMY. It is to be expected that the fuel consumptions of turbojet engines will be improved during the next decade, as compressor and turbine efficiencies are increased by refinement in design and increased compression ratios, and as the temperature limitations of materials are raised by metallurgical developments (see Ref. 12). Experts in the turbojet field expect that the value 1.0 lb/hr/lb-thrust will be reached and perhaps bettered. Since range is inversely proportional to specific fuel consumption, and since the value 1.2 was assumed in our selection charts, it is clear that substantial increases in range will result.

b. AIRPLANE IMPROVEMENTS. The effects of airplane-design refinements such as reduced drag, reduced structural weights, and increased size are all analogous to the effects on the performance of conventional-engine airplanes, as discussed earlier. Also, their respective prospects of accomplishment are the same as before.

> For example, the problems and the potentialities of the all-wing airplane appear to be as important for turbojet airplanes as for conventional. It has already been mentioned that the suppression of nacelles and fuselage is a leading factor in the observed improvement of performance with increased size. In the all-wing design this can be carried to practically complete elimination of these appendages. Moreover, some of the features of the all-wing, such as extreme sweepback with its associated lower lift coefficients, are likely to be incorporated in conventional airplanes as well, thus diminishing the margin of deficiency of the all-wing on these counts.

A mathematical analysis of the most efficient turbojet airplane, i.e., the one whose speed is greatest for a given turbojet engine and given gross weight, reveals that its wing area should be chosen in a certain relation to the fuselage (and/or nacelle) frontal area. This, however, is without regard for the requirements of volume to house fuel and useful load. If the analysis is repeated on the assumption that a fixed volume is required, the result is obtained that the fuselage (nacelles) should be very small and the configuration of a flying wing should be approached for best efficiency.

As this report is written, further calculations are being made of the performance of flying-wing turbojet airplanes, for comparison with Figs. 6 and 7. The initial results show high-speed and range values equal to those of Figs. 6 and 7, for airplanes of substantially lower wing loadings. This implies considerable improvement in take-off performance at no sacrifice of speed and range.

The reward in range due to reduction of airplane drag is greater for turbojet airplanes than for conventional-engine airplanes; this is seen by comparison of formulas (16), (17), and (18) with the classical Breguet formulas. For the turbojet case the range varies as ${C_{D_p}}^{-1}$, while for conventional airplanes it varies as ${C_{D_p}}^{-\frac{1}{2}}$. Hence, ranges much greater (perhaps 50% greater) than those of Figs. 6 and 7 should be considered as potentially available as the refinements discussed here are brought to practical realization.

Similarly, the reward in top speed is greater in the turbojet airplane, by virtue of the characteristics of constant thrust; the maximum speed varies as ${C_{D_p}}^{-\frac{1}{2}}$ rather than ${C_{D_p}}^{-\frac{1}{3}}$ as is conventional. Thus, improvements in the basic aerodynamic cleanness or in the effects of compressibility on drag are more effective in increasing the speed.

c. INCREASED THRUST; TAIL-PIPE BURNING. There appears to be little possibility of attaining high supersonic speeds with turbojets whose output is of the order now available. We have already seen that the speeds of these airplanes (Fig. 6) do not appreciably exceed the speed of sound. From the few data available it appears that a present-day turbojet nacelle cannot even propel itself, without wing or other additional drag, at very high speeds.

There are practically no test data available to show the drag coefficients of well-shaped nacelles or fuselages at high speeds. The most reliable seem to be certain values obtained by the Germans on a wind-tunnel model of their V-2 rocket projectile (Ref. 14). Using these values for the turboject nacelle neglecting the drag increment due to air entrance, and a thrust rating of 500 lb/sq ft frontal area at sea level for the turbojet, it is easily verified that the drag exceeds the thrust at all speeds above 800 mph. The thrust rating used for the turbojet is believed to be applicable to both subsonic and supersonic speeds, with good entrance design.

A similar calculation for higher altitudes yields the same result; the maximum speed is about 800 mph.

If these figures are correct, it is clear that the present-day turbojet cannot propel an airplane at very high speeds. There are definite possibilities for a change in this situation in the near future, however:

(1) Studies, (Ref. 12) have indicated that large increases in thrust output may be obtained by use of higher turbine-inlet temperatures than are now allowable. For example, in a typical case, an increase of turbine-inlet temperature from 1600° to 2600°F results in over 60% more thrust.

(2) Tests and calculations show that large increases of thrust may be obtained, at a sacrifice of fuel economy by using a second combustion chamber behind the turbine, so-called "tailpipe burning." This would be employed in take-off and in combat, but not for economical cruising. Ref. 13 gives quantitative values.

(3) Tests run at the NACA and elsewhere have shown increases of thrust due to water injection in turbojets having centrifugal compressors. The water apparently acts primarily to cool the air during and after compression. (It is reported that this improvement is not realized to the same extent in axial-flow compressors.)

d. POSSIBILITIES FOR SUPERSONIC FLIGHT. In the light of these good prospects for great increase of the thrust output of turbojets, our earlier conclusion, based on present engines, that the turbojet is not attractive for supersonic flight, requires re-examination.

In calculating the equilibrium speed of a turbojet nacelle a thrust rating of 500 lb/sq ft was assumed. Even without increased turbine temperatures, the ratings predicted for the turbojet with tail-pipe burning would permit supersonic flight of the V-2 body whose drag values were assumed above. In fact, a comparison of the drag and revised thrust curves shows a small excess thrust at transonic speeds, which increases with increasing Mach number up to $M = 2$, at least.

In order to obtain an idea of the future possibilities of turbojet airplanes for supersonic speeds, we have prepared a set of charts (Figs. 8 to 10) showing the estimated supersonic performance of a family of airplanes powered by turbojets with tail-pipe burning (or other means to increase thrust) providing the following improvement in thrust compared to present-day machines:

50% more static thrust (for same engine frontal area)	180% more thrust at $M = 1.6$
60% more thrust at $M = 0.8$	240% more thrust at $M = 2.0$
120% more thrust at $M = 1.2$	245% more thrust at $M = 2.4$

Variation of thrust with altitude at constant M:

Thrust proportional to $\sigma^{0.7}$ below 35,000 ft

Thrust proportional to σ above 35,000 ft

Fuel consumption: 2.7 lb/hr/lb-thrust.

For the purpose of these charts, wing drags have been estimated according to the best available theory, which is not completely adequate. (There are practically no reliable test data.) Since the theory indicates that this drag is greatly influenced by wing thickness, we have assumed a maximum thickness-chord ratio of .05. For fuselage drag the German V-2 test data mentioned above have been assumed. This is probably optimistic, since the V-2 model had no air in-takes, excrescences, or leakage, and very little interference from tail fins; hence, our results can be expected to represent only very clean fuselages of large fineness ratio.

Component weights and available fuel supplies have been estimated by use of statistical data based largely upon conventional military airplanes, but modified where possible to account for the special features of these machines.* In this case, the wing and thrust loadings given do not represent take-off values, but values at the beginning of the supersonic flight period. It is assumed that fuel carried in drop tanks is used for take-off and acceleration, and that rocket boost, if used, is in the form of self-contained rockets that are dropped before the supersonic flight period is begun. Drags have been estimated for straight, rather than swept-back wings.

The results presented in Figs. 8 to 10 require some discussion. First, it is seen that respectable supersonic flying speeds are obtained, although the duration of flight at such speeds is severely limited. The great advantage of the turbojet for this application is that its good abilities at subsonic speeds are still available, and its extreme ratings for supersonic flight (by use of tail-pipe burning) can be withheld, to conserve fuel, until needed for short periods in combat. However, our calculations indicate that airplane drags are very large near the sonic speed, and some auxiliary source of power, such as rockets, will probably be required to accelerate through this region when supersonic perforamcne is desired. Although acceleration by means of the turbojet alone might be possible, it would be very slow and would consume an extravagant amount of fuel. A dive from high altitude might also be used to provide the needed acceleration.

If the thrusts required in Figs. 8 to 10 are obtained solely by use of tailpipe burning in turbojets having present-day characteristics, it cannot actually be assumed that the subsonic performance of Figs. 6 and 7 will still be fully available with the tail-pipe combustion chamber inoperative. Even if the tail-pipe burners, inoperative, do not reduce the turbojet performance, it must be admitted that the structural weights assumed in Figs. 6 and 7 are too optimistic to correspond to the thin wings and long fuselages visualized here. On the other hand, it seems reasonable to assume that some improvement in turbojet fuel economy will be achieved by the time such supersonic airplanes are built. Hence, it will be assumed that subsonic range performance as good, or nearly as good, as in Figs. 6 and 7 will be available in the supersonic turbojet airplane.

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* In this regard, the experience of the Douglas Aircraft Company has been used as a guide.

We have assumed no sweepback, for several reasons. In the first place, the advantages of sweep at supersonic speeds are less well documented than at subsonic speeds; theoretically there should be little advantage until the sweepback angle is greater than the angle $\cos^{-1}(1/M)$; even for greater angles, e. g., greater than 57° at $M = 2$, there is considerable doubt, in the absence of adequate supersonic wing theory or test data, as to the improvement to be obtained. Moreover the angles of attack required for landing and take-off are so great for wings with very large sweepback and for triangular low-aspect-ratio wings, as to make them practically unusable for airplanes that must take off and land on landing gears.

The validity of the results in Figs. 8 to 10 is very much dependent upon our assumptoins regarding body drags. For all supersonic speeds the margins of thrust over drag for the airplanes are so small as to be actually within the order of our ignorance on this subject.

> Estimates made at the Douglas Aircraft Company have resulted in body drags for complete fuselages with air intakes, etc., that are only about two-thirds as great as the V-2 drags of Ref. 14. Using these estimates, we obtain supersonic performance considerably better than shown in Figs. 8 to 10.
>
> Estimates made at the Northrop Company, representing somewhat more blunt bodies, have resulted in values nearly twice as great as the V-2 drags. If these estimates are used in computing turbojet airplane performance with tailpipe burning, it is found that no supersonic speeds are available, i.e., drag exceeds thrust at all supersonic speeds.

Clearly, the supersonic performance calculated above is of questionable accuracy. It is to be hoped that further research will be undertaken to clear up the wide area of doubt.

e. SUPERSONIC MISSILES. The performance of supersonic pilotless airplanes, turbojet-powered, can be predicted from the results of Figs. 8 to 10. The missiles are assumed to use the same power units as the supersonic airplanes represented in those figures, and, in fact, to differ from these airplanes only in the elimination of certain weight items, such as crew, armament, landing gear, and 50% of the miscellaneous equipment, and in the replacement of these weights by warhead. The warhead weights have been determined according to these assumptions, and are tabulated below for use in Figs. 8 to 10. As in the original application of these figures, no allowance has been made for fuel consumed in launching, since it is assumed that external means are used to accelerate the missiles through the transonic regime.

W/T_0 in Figs. 8 to 10	1	1.5	2	2.5	3	3.5
Warhead weight (lb) of Corresponding Missile	2300	2900	3500	4100	4700	5100

It is seen that considerable advantage can be obtained over piloted airplanes, especially by the use of wing loadings that might be objectionable for take-off and landing. It is possible that further improvement in performance might be achieved by use of low-aspect ratio or highly swept-back wings. However, as mentioned above, the advantages are difficult to assess until better theoretical and/or test data are available.

f. CEILINGS. The gas turbine is not essentially a high-altitude engine, since its power diminishes rather rapidly in the stratosphere.* The ceilings of the airplanes in Figs. 6 and 7 vary with wing loading and power loading; to give an idea of the range of altitudes available, the absolute ceilings have been estimated for a few of the airplanes and are tabulated below:

Figure Reference	Wing Loading (lb/sq ft)	Thrust Loading (lb/lb-thrust at S.L.)	Ceiling (ft)
6	25	2.5	55,000
6	60	2.0	55,000
6	60	4.5	42,500
6	100	2.0	53,000
6	100	4.5	39,000
7	90	5	39,000
7	125	4	42,500
7	125	6	31,000

The ceilings of the supersonic airplanes and supersonic missiles considered in this section have not been calculated, but the limitations of altitudes of such craft are indicated by comparison of Figs. 8, 9, and 10. It is seen that the performance at both sea level and 50,000 ft is definitely inferior to that at 35,000 ft. Thus, the best possibilities for supersonic flight appear to be restricted to the middle altitudes. This conclusion is in agreement with the findings of other investigators.

TURBOPROP

1. *Introduction.*

The turboprop consists of a gas turbine driving a conventional propeller. Our earlier discussion of the charactersitics of the gas turbine in an airplane holds good for this application as well as for the turbojet. Again the turbine is installed so as to be affected by flight ram, and its power output and efficiency are favorably affected by increasing flight speed as before. On the other hand, the propeller provides much greater propulsive efficiencies than a jet at subsonic speeds, so that the improvement in thrust power and fuel economy with increasing speeds are not as spectacular as for the turbojet. In fact, as a reasonable approximation, the specific fuel consumption in pounds per brake-horsepower-hour can be taken to be constant for varying speeds and altitudes. The greatest advantage of the gas turbine with propeller is its ability to produce great amounts of power for its weight, particularly at low altitude.

2. *Present Abilities.*

To indicate approximately the performance characteristics available (or soon to be available) in turboprop airplanes, one of the selection charts of Ref. 4 has been reworked to account for some assumed differences between conventional and turboprop power plants. It is assumed that:

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* See second footnote on page 45.

a. The turbine puts out twice as much power at sea level (static) as a conventional engine of equal installed weight and frontal area.

b. At cruising and top speeds at 35,000 ft, the turbine puts out only 50% as much power as it does at sea-level (static), i.e., the same power as a supercharged reciprocating engine of equal weight and frontal area.

c. The specific fuel consumption of the turbine is just enough (5%) poorer than that of the conventional engine to account for the reduction of propulsive efficiencies in Ref. 4 for cooling horsepower.

Based on these assumptions, which are reasonably valid for present-day turbines, the selection chart shown in Fig. 11 is obtained. It is seen that very long ranges can be obtained with short take-off distances and high initial rates of climb, but that the relationship between range and high speed at altitude is not changed, compared to conventional airplanes. The present-day turboprop is therefore particularly attractive for transport airplanes.

There are certain characteristics of the turboprop airplane that are not shown in Fig. 11:

a. Its high speed at sea level is higher than that of a conventional bomber of comparable altitude performance. However, this feature may be only hypothetical, if gust loadings or other structural limitations prevent the use of such great speeds at low altitude.

b. Its speed for best range is higher than that of a corresponding conventional airplane, since the cruising power of the turbine at any altitude is not much less than the maximum power. This feature is probably of great military importance; long-range cruising can be assumed to occur at speeds just under those indicated as maximum speeds at 35,000 feet in Fig. 11.

3. *Future Possibilities.*

a. IMPROVED FUEL ECONOMY. The specific fuel consumptions of turboprops can be expected to be improved by the same means and to about the same extent as those of turbojets.* This will bring the gas turbine to thermal efficiencies as good or better than those of reciprocating aircraft engines and will provide maximum ranges some 20% greater than shown in Fig. 11.

b. AIRPLANE IMPROVEMENTS. The rewards of drag reduction in turboprop airplanes are the same as for conventional-powered airplanes, page 43.

c. INCREASED THRUST; BOOSTERS. It should be possible to increase the jet thrust of a turboprop installation in the same proportion as that of a turbojet, by use of tail-pipe burning. However, the jet thrust is usually a fairly small fraction of the total thrust of the turboprop and the gain due to such burning would be relatively small.

An alternative scheme has been proposed (Ref. 16) in which a booster unit would be used for increased thrust for combat or emergency. The booster consists of a compressor, gear-driven by the gas turbine, and a combustion chamber. It is

* See Ref. 15.

estimated that the total thrust of the turboprop installation could be increased 50% at a Mach number of .67 by use of a booster. The total fuel consumption with booster in use would then be somewhat less than two pounds of fuel per hour per pound of total thrust available. Larger boosters might be used to give still greater increments of thrust, but thise would be at the cost of greater fuel consumptions.

An increase of 50% in net thrust available could produce increments in maximum speed of about 25% in Fig. 11. This would clearly make the turboprop bomber superior to conventional-powered bombers on all counts. If we visualize further improvements in its performance by virtue of drag reductions, etc., we must expect this bomber to provide high subsonic top speeds (in the order of 600 mph) and at the same time very long ranges at relatively high cruising speeds.

d. POSSIBILITIES FOR SUPERSONIC FLIGHT. The possibilities of the turboprop for supersonic flight lie somewhere between those of the conventional power plant and the turbojet. The prospects of supersonic propellers have already been discussed above ("Possibilities for Supersonic Flight," page 40).

It appears that the propulsive efficiencies of propellers must always be inferior to those of jets at supersonic speeds. Thus, the turboprop must be handicapped in comparison with the turbojets at such speeds. Also, its weight is necessarily greater.

It is our conclusion that the turboprop's prospects at very high speeds are doubtful, unless further research shows that the most optimistic supersonic drag values mentioned above are the correct ones; this would enable the turboprop with boosters to perform successfully in the supersonic field.

e. CEILINGS. The power drop-off of the turboprop with increasing altitude is similar to that of the turbojet. Hence, it is not essentially a very high-altitude power plant; it will provide airplane ceilings approximately the same as those shown on page 52.

RAMJET

1. *Introduction.*

A late addition to the family of improved power plants is the ramjet (athodyd), which depends entirely on flight ram for its compression. Its characteristics are not very well known at present, but have been predicted by several investigators and checked very roughly be a few tests. Its performance is very poor at low speeds. Its fuel consumption is expected to be higher than that of the other power plants discussed above, at least below a Mach number of about 2.

2. *Supersonic Ramjet Missile.*

In view of these characteristics, we believe that the ramjet will be used only for supersonic performance and probably only in pilotless missiles.* Accordingly, we have prepared a set of selection charts for supersonic ramjet missiles carrying varying amounts of explosives. Some details of these calculations are as follows:

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* An exception may occur in the use of this device in composite-powered aircraft, such as are discussed briefly in "Composite-Powered Airplanes," p. 57.

a. The thrust values of Ref. 17 (as presented in Ref. 18) are assumed for the ramjet. The maximum frontal area of the ramjet is arbitrarily selected as 15.5 sq ft. This is the same as has been assumed for the turbojets of both Fig. 6 and Figs. 8 to 10.

b. Fuselage-drag coefficients are again taken from the V-2 tests of Ref. 14. (The uncertainty involved here has already been suggested in the preceding pages.)

c. The missiles carry the same amount of equipment and controls as the supersonic turbojet missiles considered above. In this case, however, a power-plant weight was chosen according to page 25 of Ref. 18. The design speed is arbitrarily chosen as 1500 mph (M-2) at sea level.

d. Fuel consumptions are estimated from Ref. 17.

e. The variation of ramjet thrust with altitude is taken from Ref. 17.

The results of this calculation are presented in Figs. 12 to 14. It appears that the ramjet offers very good prospects as a power plant for supersonic missiles. Although the performance at 50,000 ft is superior to that at 35,000 ft in some respects, this improvement with altitude cannot be expected to continue much farther, in view of the rather rapid drop-off of power mentioned above.

In evaluating Figs. 12 to 14, it should be borne in mind that several practical problems of take-off or launching and acceleration have been ignored. Moreover, in the diagrams and data relating to performance at altitude, the question of fuel consumption during the climb has been avoided. We are confident that these are, indeed, practical operating problems rather than basic difficulties, and that they will be solved by one expedient or another in order to achieve the supersonic performance shown. Such expedients may be rocket assistance, mechanical catapulating, or even release from carrying airplanes at altitude. Our neglect of such problems in this discussion does not imply that they are unimportant. Our aim is to investigate the prospects of airplane performance under the most favorable circumstances; if these prospects are attractive, the operating difficulties must be attacked.

3. *Future Possibilities.*

The ramjet is so much in its infancy that predictions of its future are largely speculative. There will probably be improvements in both thrust and economy, but it is difficult to say how the results will compare with the characteristics assumed here. These improvements will be achieved by improved combustion-chamber design, better heat-resisting materials, and more efficient diffusers.

Aerodynamic refinements reducing the drag of supersonic airplanes and missiles are certain to come as the result of research, particularly in large supersonic wind tunnels. Such refinements, probably directed especially toward the suppression of shockwaves and the disturbances they cause, will significantly increase the speed of such airplanes or their ranges for given airspeeds.

ROCKET

1. *Introduction.*

The rocket power plant is unique in that its thrust is nearly independent of altitude. Since it is not dependent upon the atmosphere for combustion, it has no ceiling and provides unsurpassed performance at very high altitudes.

A typical rocket fuel having high values of specific impulse per pound and per gallon (lb/sec/lb of fuel and lb/sec/gal of fuel) is nitromethane. Since this fuel is available commercially it is chosen to present an example for rocket-powered missile performance.

2. *Supersonic Rocket Missile.*

The limitations on performance of a rocket-powered missile are imposed by the high fuel consumption involved. In order to provide an example, the following characteristics of a family of missiles is considered:

a. Propellant: Nitromethane; Specific impulse, 220 lb/sec/lb.

b. Warhead weight, 2000 lb.

c. Wing and body drag coefficients are estimated by the same methods as have been used above.

d. Power-plant weight, installed, is assumed to be 0.2 lb/lb of thrust.

e. The missile is assumed to be launched by external means at the cruising speed (i.e., the speed along the climbing flight path).

f. The missile is assumed to climb at such a rate that the dimensionless wing loading (wing loading divided by atmospheric pressure) does not deviate far from the conditions for best L/D. Specifically, the initial wing loading is taken as 400 lb/sq ft, and it is assumed that the missile climbs at such a rate that the precentage decrease in weight (due to consumption of fuel) is approximately one-half the percentage decrease of atmospheric pressure.

This is believed to represent a typical family of antiaircraft missiles, or ground-to-ground missiles during their initial ascent.* The results are shown in Fig. 15. According to our assumptions, the altitude attained depends only on the fuel-weight ratio W_f/W_1; the essential effect of varying the cruising speed is simply to change the angle of climb and thereby the horizontal range.

The range of rocket missiles in horizontal flight, with corrections for acceleration and climb, is calculated in Ref. 19. The equations of that reference have been applied to the present case for comparison, and good agreement has been noted.

3. *Future Possibilities.*

The future of rocket motors depends largely on developments of propellants. This subject has been investigated by members of the Scientific Advisory Group and is reported in Refs. 20 and 21.

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* The range obtained in gliding could easily be estimated.

ATOMIC ENERGY

The form that will be assumed by the nuclear-energy engine, when and if it appears, cannot be predicted at this time. It seems clear, nevertheless, that if such energy is made available for industrial or military applications, it may well be used first in aircraft, where the advantages of reducing the fuel weight are great.

It has been predicted that atomic energy will be put to practical use in the form of heaters replacing the combustion chambers of heat engines, and that the fuel consumption of such engines will be negligible. Should this be the case, the problem of range of airplanes will vanish, for any airplane will be capable of flying any desired distance at any speed and weight at which it can rise from the ground and fly at all.

This amazing state of affairs will apparently not be achieved without the expenditure of rather large amounts of power-plant weight, perhaps including considerable shielding material to confine radiation. It is interesting in this connection to investigate what installed power-plant weights will be acceptable for airplanes. To do this, use can be made of the various selection charts included here (and in Refs. 1 to 11):

> Any point on a selection chart represents a certain combination of performance features that are available at a certain wing loading, power loading, total power, etc. In calculating such performance features, (including range, it has been necessary to estimate the breakdown of gross weight into structure, power plant, equipment, fuel, and useful load.
>
> Now, if the total weight assigned to power plant and fuel is imagined to be devoted to the nuclear power plant, approximately the same speed, climb, and take-off performance will be available, and for all ranges. The only approximation involved is the assumption that the new power plant will occupy nearly the same space as the previous one (plus its fuel), so that the dimensions and aerodynamic drag of the airplane remain practically unchanged.
>
> A comparison of this total available weight and the power required results in a value for the maximum permissible weight of the atomic engine in pounds per brake horsepower for the performance represented at the point under consideration.

If this analysis is made of the various selection charts herein, successively, it can be expected that the requirements imposed on the atomic engine, in pounds per brake horsepower or pounds per pound of thrust, will increase in severity as we progress from conventional-engined bombers to high-speed and supersonic missiles. For this reason we have put our attention on the low-speed bomber, in order to obtain the highest practical allowable weight. The result is presented in Fig. 16, where contours of allowable power-plant weight are drawn on the selection chart of Fig. 1. Naturally, if the weights shown can be improved upon, the airplane performance will be correspondingly bettered by installation of larger power plants.

COMPOSITE-POWERED AIRPLANES

No attempt will be made here to investigate the possibilities of airplanes powered by more than one of the power-plant types described. Such composite types are already being developed, in an effort to combine the high-speed characteristics of turbojet airplanes, for example, with the better range and take-off characteristics provided by the conventional engine and propeller or the turboprop.

In general, such composites are created only at the expense of increased weight and complication, and some compromise of the most desirable features of both types. Nevertheless, estimates show that by this method certain gaps in the performance characteristics of the pure types can be filled.

SUMMARY

The improved types of aircraft power plants are superior to conventional engine-propeller combinations by virtue of greater power output at high flying speeds, reduced weight, and, in several cases, elimination of the propeller. Most of them are characterized by inferior fuel economy, at least at low flying speeds.

The *turbojet* can provide high subsonic speeds and relatively good range. At present it does not appear to provide supersonic performance. When equipped for tail-pipe burning it will offer attractive possibilities for supersonic flight, at least at medium altitudes, provided that airplane drag values at supersonic speeds are approximately as assumed here.

The *turboprop* will permit substantial improvements in the performance of long-range bombers. Its use of a propeller makes it of doubtful value for supersonic flight.

Ramjets appear to provide the most spectacular supersonic performance, provided that means are provided for launching the aircraft (probably a missile) and accelerating it through the transonic region.

The *rocket* is the most likely source of power for launching and accelerating such missiles. In addition it can be employed as a power plant for supersonic flight, but only over rather restricted ranges.

The *atomic engine* may provide present-day take-off, speed, and climb performance in large bombers, together with unlimited range and endurance, provided that its installed weight is not greater than two or three times that of a conventional engine of equal brake horsepower. The corresponding allowable power-plant weights for other applications of such engines have not been investigated.

UNCONVENTIONAL AIRCRAFT*

INTRODUCTION

Among unconventional aircraft, i.e., aircraft with other than fixed wings, only the various forms of rotary-wing aircraft are considered promising enough to be worth examination. Flapping wing devices such as the ornithopter will not be investigated.

In the field of rotary-wing aircraft, the principle varieties are the helicopter, autogyro, gyrodyne, and cyclogryo. At present, the helicopter is the most advanced of these. The autogyro is no longer of any interest, while the gyrodyne and cyclogyro may be promising possible successors or complements to the helicopter, although practically no development effort has been directed to them as yet.

In viewing the field of rotary-wing aircraft, it is important to remember that their position is a unique one. This type of aircraft rarely competes directly with conventional, fixed-wing aircraft in functions for which the airplane is now used, but on the other hand is suitable for certain missions for which the airplane cannot be used. In other words, the outstanding characteristic of the aircraft is their ability to fly at zero forward speed, while range and speed, though desirable, must always be secondary. For long-range transport and high-speed flight it is inconceivable that a system of rotating wings will ever be as efficient as a fixed wing. Hovering, vertical take-off and landing are distinct characteristics of rotary-wing aircraft, and all other performance must be subordinated, desirable though it may be. If use of a rotary-wing aircraft is not dictated by the necessity for these zero forward speed characteristics, then a fixed-wing aircraft could probably be used to better advantage.

Before efficient rotary-wing aircraft designs can be considered, some knowledge of their function, must be had, and it is desirable to investigate the possible military missions to be performed by rotary-wing aircraft. For example, rotary-wing aircraft are more ideally suited for short-range and short-endurance missions, and if definite requirements of short endurance are laid down, the utility of rotary-wing aircraft can be greatly increased over present-day helicopters.

POTENTIAL MILITARY MISSIONS FOR UNCONVENTIONAL AIRCRAFT

1. *Airborne Operations.*

They may be used to transport and deliver personnel and supplies, especially under conditions of difficult terrain, such as mountains, forests, jungles, etc.

a. For short-term operations where it is extravagant of manpower and effort to clear landing strips for fixed-wing aircraft.

b. Prior to the completion of landing strips for fixed-wing aircraft or when there is insufficient time vaailable for clearing landing strips.

* This section has been prepared by Captain Chester N. Hasert.

c. Where the terrain is too difficult to clear regardless of time available.

d. For very short-range air transport over natural obstacles and barriers, i.e., as "flying cranes" to carry troops and supplies across rivers in lieu of bridges or boats. They could be made available at the desired location and ready for operation more quickly than boats or temporary bridges.

e. Tow devices using rotors, analagous to conventional fixed-wing gliders, but capable of safe descent (vertically) into much more difficult terrain. These may be in the form of complete "glider" aircraft (towed autogyros) housing the cargo internally, or they may be simply rotors attached to large pieces of equipment and towed "as is."

f. Drop devices, using rotors in place of parachutes, to drop supplies carried in fixed-wing aircraft for delivery to ground forces.

2. *Liaison Duties.*

They may be used for liaison duties similar to those performed by the light planes of the recent war, but are capable of operation from even more difficult terrain and more direct service.

a. Transport of small numbers of personnel for short ranges to relatively out-of-the-way locations.

b. Observation.

c. Air evacuation of medical cases from forward areas.

3. *Miscellaneous Special Duties.*

a. Rescue and supply of small isolated groups in isolated areas of difficult terrain.

b. Possible altitude stations for television and radar to increase their range.

c. As "flying-cranes," across rivers, as mentioned in 1-d.

d. Tow and drop devices mentioned in 1-e and 1-f.

e. Navy convoy patrol work.

f. Aircraft for storage in small space, for example, the German autogyro "observation kite" operating from a submarine.

It is desirable to keep these functions in mind when considering rotary-wing aircraft performance, as the optimum design of rotary-wing aircraft is even more critically dependent on designing specifically for one function and one set of performance requirements than is the case with fixed-wing aircraft. For example, for short range, considerable savings in power plant weight can be realized and large increases in useful load obtained by use of the improved power plants mentioned here.

CHARACTERISTICS OF VARIOUS ROTARY-WING AIRCRAFT

1. *Autogyro.*

The autogyro can be dropped from further consideration, except as mentioned in (1) of the preceding section, as it cannot take off vertically and has poor lift-drag ratios in forward flight at cruising and high speed. In short, an autogyro is no better than a well-designed conventional aircraft designed for the same function.

In the low-speed range where the autogyro is most efficient, a fixed-wing aircraft with the necessary low wing loading and high-lift devices is more efficient.

However, the autorotating rotor may have a few applications due to its simplicity and compactness, when the blades are folded. A set of rotor blades may be folded and stored in much less space than the equivalent fixed wing performing the same function, as illustrated by the German autogyro observation kite operating from a submarine.

2. *Gyrodyne.*

The gyrodyne employs the lifting rotor in the best configuration for high speed and range. The autorotating lifting rotor, as in the autogyro, must be pulled through the air at a backward tilt, while the lifting rotor if used for forward propulsion (i.e., as in the helicopter) must be tilted forward to a considerable angle at high forward speeds, and loses efficiency rapidly, i.e., a low L/D ratio is obtained from the rotor. The logical step is, thus, the gyrodyne, where forward propulsion is supplied by independent means, such as the normal propulsive airscrew, and the rotor is used for lift only and may therefore be operated at its optimum angle of attack, maximum L/D, and the high speed and/or cruising thereby improved. Thus, if emphasis is to be on a high-speed or long-range rotary-wing machine still capable of vertical flight and hovering, the gyrodyne is clearly superior to the helicopter. Since there has been practically no emphasis on high speeds of rotary-wing aircraft, there has been practically no development work on this type of machine, and its practical potentialities only slightly explored.

The optimum power plant for forward propulsion presents exactly the same problem as for the fixed-wing aircraft, and in view of the probable low cruising speed will probably be found to be a turboprop or reciprocating engine and propeller, depending on the range and endurance.

3. *Cyclogyro.*

Relatively little work has been done on the cyclogyro, although some theoretical analyses and wind-tunnel model tests have been made by the University of Washington and the ATSC. Studies made by the ATSC (Ref. 22) indicate that the cyclogyro is potentially superior to the helicopter in high speed, and is still capable of hovering flight. This analysis also indicates that the efficiency of the cycloidal propeller (as a propulsion and lift device) of the cyclogyro is superior or at least comparable to the conventional screw propeller and fixed-wing system, giving greater ratios of high speed to landing speed. However, as the weight of the total propulsive and lifting system in proportion to the gross weight tends to be greater for the helicopter than for conventional aircraft, so it may be that the cyclogyro may pay for its higher aerodynamic efficiency by having a still higher weight of propulsive and lifting system. This is particularly indicated by the severe structural and vibration problems. The weight penalty involved in solving these problems is difficult to predict without further actual development work.

It is reasonable to expect that the cyclogyro would be more efficient than the helicopter in forward flight, from the nature of its lifting surfaces. A conventional rotor in forward flight always has regions of reversed flow near the hub, varying in magnitude with the forward speed, as well as variations in velocities and induced angles of flow along the blade, also varying with forward speed, so that it is impossible to design a rotor blade for best efficiency under all conditions. However, for the cycloidal rotor, the entire span of the blade has the same air velocity over it, and the angle of attack throughout the cycle can be regulated by the pitch mechanism such that reversed flow is never encountered, but the blade sections always move into the wind. Also, conditions along the blade are more uniform, and so it is possible to design cycloidal propeller-rotors with higher efficiencies than helicopter rotors for high forward speeds.

In the actual design of a cyclogyro aircraft, however, there are certain losses in efficiency to be expected. For example, the magnitude of the interference in a tandem rotor arrangement, which would probably be a desirable configuration, is not known exactly, and although these effects have been estimated, they have not been confirmed by tests as yet.

As previously mentioned, it may be that the apparently high aerodynamic efficiency of the cyclogyro is paid for by increased weight of the cycloidal rotor system. Preliminary structural analysis and weight estimate of the ATSC cyclogyro proposal indicated a rotor specific weight of about 1.5 to 2 lb/hp, or a total weight of power plant and rotor system of 3.5 to 4 lb/hp for lift and propulsion. This is the same order as conventional gear-driven helicopters (the weights of jet-driven rotors are potentially much lower). However, until one is actually built, these weight estimates are to be considered speculative.

As an illustrative case of cyclogyro design performance (probably optimistic), a proposed high-speed cyclogyro (Figs. 17 and 18) in the same class as the XP-77 airplane, i.e., with same engine, armament, and approximate gross weight was designed at ATSC and compared with the XP-77. With regard to this particular design, however, it should be mentioned that at present, high-speed cyclogyros are not considered the most likely direction of immediate development, due to the increased severity of the vibration and structural problem, and the trend is more likely to be towards cyclogyros designed for the low-speed range. It should be noted that the ATSC cyclogyro considered here was not designed for hovering flight.

The performance comparison is as follows:

	XP-77	*Cyclogyro*
High Speed at Sea Level	327 mph	340 mph
High Speed at 27,000 ft	420 mph	428 mph
Max. Rate of Climb, Sea Level	3050 ft/min	2900 ft/min
Max. Rate of Climb, 27,000 ft	2020 ft/min	2000 ft/min
Speed Range (V_{max}/V_{min})	4.5	8.5
Gross Weight	3700 lb	3900 lb

(The high speeds quoted are estimated performance, without compressibility corrections, and may not agree with flight-test results of the XP-77, but they are calculated on the same basis for both cyclogyro and XP-77 and should be valid for comparison purposes.)

This comparison is given as the only existing example of what a cyclogyro should be expected to do on the basis of existing knowledge. This cyclogyro design is based on rather meager data and represents only very preliminary design work. It may be that considerable optimism was incorporated in these design studies.

In summary, it appears from the work done so far (mostly studies by ATSC (now Air Materiel Command) supported by some wind-tunnel data from the University of Washington) that the development of the cyclogyro should at least be carried through a little further until definite conclusions can be reached as to its actual potentialities.

4. *Helicopter.*

a. INTRODUCTION. Present-day helicopters possess a relatively poor performance, i.e., maximum speeds of 100 mph and short ranges. Even as to the helicopter's principal distinction, its ability to hover and perform vertical take-off and landings, it cannot, in some cases, even hover except at low gross weights and low altitudes, and its hovering performance decreases rapidly with overload gross weight and altitude. In addition to its poor performance, its stability and control characteristics also leave much to be desired, and unsatisfactory vibrations are still present. There has been little real improvement since the first practical helicopter in this country (Sikorsky's) and most of the present effort has been spent largely on mechanical difficulties, the vibration, and transmission problems.

The relatively poor performance of the present-day helicopter is to some extent to be expected in view of the relatively early stage of its development. Development during the war was only slightly encouraged and was handicapped by low priority.

Application of advances in high-speed aerodynamics to the rotor are on the whole not as promising as might be expected. Improved blade-tip design may result in increases in the allowable rotor tip Mach number with some improvements in performance. Higher disk loadings will increase the high speed, and at the same time tend toward wider, and thus smoother and stiffer rotor blades. The use of boundary layer control to delay blade stalling and to decrease the drag is also potentially promising in improving the performance of the rotor. Such a system may be more adaptable in conjunction with a jet-propelled rotor.

One of the immediately available improvements in helicopters appears to be the adoption of generally higher disk loadings, especially for improvements in cruising and high speed. Analagous to fixed-wing aircraft, a higher disk loading, like a higher wing loading, results in an increase in induced power at low speeds, but with a saving in parasitic drag at high speeds. Present-day helicopters have been held to extremely low disk loadings (2 to 3 lb/sq ft) in order to keep the hovering power requirements low and correspondingly to keep the power-off rate of vertical descent to a safe value.

In addition to the advantage of lower parasitic power requirements, the higher disk loadings allow wider rotor blades, with accompanying increase in blade stiffness and decrease in profile drag and vibrations associated with blade twisting and airfoil deformation. In the case of jet-propelled rotors with hub intakes, there is also the advantage of wider blades allowing larger duct cross sections and correspondingly smaller losses.

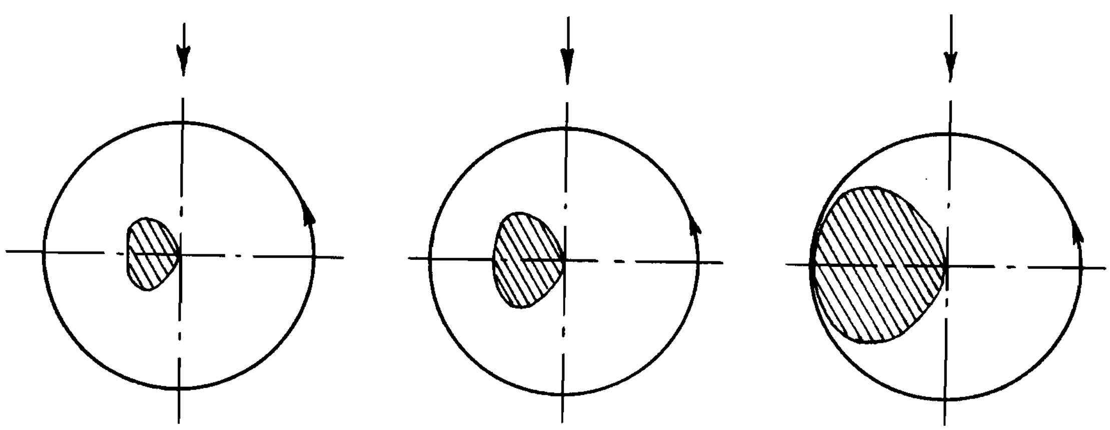
The limited performance of rotary-wing aircraft can be ascribed to several factors:

(1) Large horsepowers are required for vertical take-off and this is accompanied by high weight of the propulsion system, in percentage of the gross weight. This results in low useful load, percentagewise. More attention to optimum power plant selection will in many cases decrease the total power plant weight and improve the precentage of useful load.

(2) The actual performance in rate of climb, high speed, and range, is limited in ways peculiar to rotary wings, and more indirectly than is the case with conventional aircraft. In high-speed flight, for example, one of the limits on rotor is stalling of the retreating blade, which is operating at a high angle of attack, and partly in reversed flow while the forward blade is operating at extremely high velocities.

Rotary-wing aircraft must possess their prime characteristic of low-speed flight, vertical take-off, hovering, etc., while at the same time, insofar as this prime characteristic is not compromised, their speed and range must be improved as much as possible.

b. THE MAXIMUM FORWARD SPEED OF HELICOPTERS. The high-speed limit of a helicopter depends on several factors not as simply related as in the case of a fixed-wing aircraft. If the forward speed is increased while the rotor tip speed is held constant, then the retreating blade will have a greater region of reversed flow. At an advance ratio (forward speed/tip speed) of 0.35, customary on today's helicopter, a region of reversed flow extending from the hub out to 35% of the blade radius is encountered (covering approximately $\left(\frac{35}{2 + 100}\right)^2 = 3\%$ of the rotor disk area), while at an advance ratio of .50, the reversed flow region extends out to midspan on the downwind blade and covers an area of $\left(\frac{1}{2 + 2}\right)^2 = \frac{1}{16} = 6\%$ of the rotor disk area.



In addition to these areas of reversed flow, there are still larger areas of stalled flow over the blades, as in the area immediately adjoining the reversed flow the low tangential velocities on the blades, in conjunction with the induced flow velocities through the disk, produce large angles of attack beyond the stalling angle of the blades.

In these regions of reversed flow and stalled conditions, the efficiency of the rotor is very low, naturally, and the greater the extent of these areas (which increase as the square of the advance ratio) the lower the efficiency (L/D ratio) of the rotor. In addition, these areas of reversed flow and stalled blades aggravate the vibration problems. Still further, the dissymmetry of flow tends to produce much greater lifts on the up-wind side than on the down-wind, so that compensating devices, either a flapping blade or cyclic variation of pitch, must be stronger in their action, usually at a sacrifice in efficiency and range of operation (in pitch) if trim is to be maintained.

Thus it is clear that there must be some limit placed on the maximum advance (forward velocity/tip speed) ratio, to avoid excessively inefficient rotors with severe vibration problems.

There is the possibility, of course, that tip speed ratios far in excess of today's value may be used in conjunction with a different type of blade construction. Such a rotor would have much higher rotor blade weights per square foot, and much higher power requirements. This is in line with higher disk loadings, and, in general, tends to approximate a conventional screw-type propeller.

If there is to be a limit to the advance tip-speed ratio, then an increase in high speed must come with an increase in tip speed. Present-day tip speeds (approximately from 400 to 600 ft/sec) are already high enough so that even at the low forward flight speeds of helicopters, the tip of the advancing rotor blade, travelling at a speed equal to the sum of rotational tip speed plus forward speed, runs into high Mach numbers and corresponding compressibility troubles.

The actual Mach number of the tip, for example, is

$$M_{rot} + M_{fwd} = (1 + \lambda) = M_{fwd}\left(1 + \frac{1}{\lambda}\right)$$

Thus, a 600 ft/sec tip-speed rotor, travelling at an advance ratio of $\lambda = .35$ would have a total tip speed of 810 ft/sec, or a Mach number of .725. The high Mach number at the tip under such conditions produces the usual large increases in drag, and, occurring at the tip with large moment arm, results in large increases in rotor torque and horsepower required.

From experience in high-speed aerodynamics of the airplane, and from propellers, there are suggested the various means of increasing the critical Mach number of the tip, by thinning the blade and sweeping back the tip portion of the rotor blade. Swept-back blades, however, are difficult to use on rotors due to the desirability of keeping zero moments about the blade axis to prevent twist, etc., although a more radical blade construction (high disk loading and high solidity) may be able to take care of this.

c. ROTORS WITH SUPERSONIC TIP SPEED. With large enough powers available, e.g., with jet propulsion, the possibility suggests itself, from the airplane analogy, of driving the rotor at tip speeds in excess of the speed of sound for obtaining high forward speeds at the expense of higher powers. This expense in power would be somewhat compensated for by the improvement in specific fuel consumption if tip-located ramjets were used as the power plant, a likelihood which increases with higher Mach number tip speeds.

Under such condition of a high-speed helicopter (the helicopter itself in all likelihood would still be travelling as well under the speed of sound, but perhaps two or three times the speed of present-day helicopters) with rotor blade tips operating at supersonic speeds, say a Mach number between 1 and 2, we would have the picture of the rotor blade on the advancing side well above the speed of sound, with a shock wave of varying Mach number along the radius. As the blade moved cyclically to the downstream position, it would again be in subsonic flow. It would seem that much research might be needed in this field before such a rotor could be built.

d. POWER PLANTS FOR HELICOPTER. One way in which the helicopter suffers in comparison to fixed-wing aircraft is its useful load/weight-empty ratio. This is due largely to the high value per horsepower of the weight of total lifting and propulsion mechanism, i.e., engine, rotor, gears, contratorque devices, etc. Conventional power plant installations in helicopters run up to about 4 lb/hp for total of engine, rotor, gearing and antitorque devices, and approximately 6 lb/hp including fuel for three to four hours duration. The usual power loadings are only 10 lb/hp, so that from 40 to 60% of the weight is taken up by the rotor and its power plant and fuel alone, leaving only the remaining 60 to 40% for fuselage, landing gear, fixed equipment and pay load. The advantage offered by jet propulsion in application to the helicopter is not so much a speed advantage as decreased power plant weight and increased useful load. Compared to the 4 lb/hp of conventional design, it is estimated (optimistically) that as little as 1 lb/hp for the weight of total engine, rotor, gearing, and contratorque devices is possible with a ramjet-powered helicopter. Since the functions for which rotary-wing aircraft are most advantageously employed are frequently those of short endurance and range, definite improvements are to be had with lighter fixed power-plant weights, even if the fuel consumptions are higher.

e. JET PROPULSION FOR HELICOPTERS. As mentioned above, jet propulsion for helicopters *does not* (as with fixed-wing aircraft), offer appreciable increases in high speed or performance directly, but it does greatly reduce vibration troubles, increase pay loads for short flight durations, and is more favorable to the use of higher disk loadings, which in turn yield some increases in performance.

Its performance advantage lies chiefly in reduction of the weight of the total lifting device, i.e., not only of the power plant itself, but also in elimination of transmission and antitorque devices. Despite its inefficiency, or high-fuel consumption, its saving in fixed wieght gives it lower weights of total propulsive system for reasonable endurance.

Jet-propulsion power plants applied to rotary-wing aircraft assume somewhat special forms, although they are basically the same as those discussed for fixed-wing aircraft propulsion. Thus we may have forms of the ramjet, the intermittent ramjet, and the compressor thermal jet, with compressor driven by either reciprocating engine or turbine.

Although the jet exhaust will in all cases be located at the tip, the other essential parts of a jet unit may be placed in several other possible locations. The air intake may be located at either the tip or at the hub, since the centrifugal compression of the air as it travels through the blade to the tip is theoretically equal to the available ram of a tip intake. Thus, the duct losses of the air travelling through the blade are exactly analagous to the diffuser intake losses. Due to the large yaw angles encountered on a rotor tip in forward flight, the hub intake may be preferred. Also, the location of small turbojets or engine-driven compressor jet units at the tip would be undesirable due to the large centrifugal accelerations at the tip and the gyroscopic forces on the rotating parts. Thus, the optimum configurations would be those with hub intakes, and compressors, if any, located at the hub. A study of all these possible configurations has been made in Ref. 23 and quantitative estimates of the relative weights, performance and fuel consumptions are given. These results are based largely on theoretical analyses of the power plants unconfirmed by experiment, but are the best available data and should at least be satisfactory for a comparison of power plant applications.

Analagous to the studies of optimum power plant for fixed-wing aircraft, total weight of fixed power plant plus fuel for a given endurance could be studied. However, since the power plant of a jet-propelled rotor is so inseparably tied up with the blade design and the rotor itself, as well as the fact that comparisons with conventional engines must allow for contratorque devices and considerable transmission gearing, a better criterion is the total weight of power plant, rotor, and fuel. This value has been plotted as a function of endurance, in Fig. 19, for the conventional helicopter using a reciprocating engine to drive the rotor, for a rotor driven by ramjets at the tip, for a rotor driven by intermittent jets at the tip, for a rotor driven by thermal jets employing a compressor powered by a conventional engine of various sizes, and finally, for a rotor driven by thermal jets employing a compressor geared to the rotor, i.e., analagous to the turbocompressor jet where the jet-powered rotor replaces the turbine. Figure 19 is drawn for endurance at full take-off power and Figure 20 for cruising (60% take-off power).

A summary of the results is as follows:

Ramjet. The ramjet-powered rotor is lighter than a gear-drive (conventional) rotor for endurances up to 1 hr, full throttle or 2.5 hr cruising (60% power).

Intermittent Jet. The intermittent jet drive is lighter than the gear drive for endurances up to 2 hr full throttle or 3.5 hr cruising.

Rotor-Driven Compressor. This system is lighter than the conventional gear drive for endurances up to 2 hr full throttle and 4 hr cruising. Thus it is somewhat superior to the intermittent jet and distinctly superior to the ramjet, for the tip speeds assumed (M_o = 0.75).

Engine-Driven Compressor. The use of a small engine driving the compressor for a thermal jet system produces a power plant and rotor which is lighter than the conventional gear-driven rotor at all practical cruising endurances and even at full throttle up to 4 hr duration.

Since the engine-driven compressor system appears to be so important, and since its efficiency varies so markedly with the ratio of compressor power to power supplied in the form of heat release, as well as being greatly dependent on tip Mach number, the variation of these parameters is illustrated in Fig. 21.

These results are based on an assumed helicopter having the following characteristics:

Disk Loading	3 lb/sq ft
Power Loading	10 lb/hp
Tip Speed Mach Number	0.75

SUMMARY

Present-day helicopters possess relatively poor performance.

In the future, we may expect only moderate gains in performance for the present configuration of helicopter, incorporating the advancements in jet propulsion and high-speed aerodynamics. In general, they should be quite satisfacotyr for the requirements peculiar to their type, a field in which fixed-wing aircraft will never compete.

Other configurations of rotary-wing aircraft, such as the gyrodyne and cyclogyro show promise of somewhat greater improvements in performance, and should be investigated further.

Improvements in power plant, i.e., jet propulsion, tend more toward increasing the useful load at fixed range than improving the high speed.

SUMMARY OF DESIGN PROBLEMS

In this report we have investigated the future possibilities of the airplane, based on certain predictions regarding the future characteristics of aircraft power plants. Our conclusion is that there are prospects for great advances over present-day performance, including practical realization of supersonic flying speeds.

It cannot be overemphasized that these possibilities are not available today, and cannot be achieved in the future, unless aeronautical research keeps pace with the demands of the designer. There are obstacles confronting the designer today that practically prevent him from further progress. Unless these are removed or overcome, airplane performance must remain substantially where it now stands.

The most troublesome problems have to do with high speeds, subsonic, transonic and supersonic. An examination of the state of our knowledge in this field today, and a consideration of past progress in aeronautical research, leads to the conclusion that solution of these problems depends primarily upon wind-tunnel testing. Such testing contributes to progress in more than one way: It provides test data that may be of immediate design use; it directs the paths of theorectical research and tests the validity of theoretical results; it provides a backlog of experimental data to guide and promote the inventive abilities of the airplane designer.

It has often been suggested, somewhat wishfully, that the effects of compressibility (Mach number) and scale (Reynolds number) in airplanes and in wind tunnels might be independent of one another and, therefore, might be determined separately. This would enable us to test small models at very high speeds and to correct our results for scale effect by means of data obtained on large models or full-scale airplanes at lower speeds. Unfortunately, this happy state of affairs apparently does not exist. It appears necessary for us to discard the idea of applying scale corrections to high-speed test data from small wind tunnels, and to face the expensive and rather overwhelming task of building very large high-speed tunnels, even in the supersonic class.

It may be argued that tests in flight would be less expensive than tests in such wind tunnels. The argument is incorrect, for flight testing cannot provide all of the information desired and available in the wind tunnel.*

The principle categories of problems that must be attacked in the high-speed region can be deduced from the preceding pages. Some of them are listed here:

1. Drag determination and drag reduction at subsonic, transonic, and supersonic speeds.
2. Maintenance of lift, and control of moments, at transonic speeds.
3. Design of supersonic air intakes.

* An exception must be admitted in the case of tests in the immediate neighborhood of the velocity of sound, where wind tunnels cannot be used and flight test of some sort must be resorted to.

4. Determination of the characteristics of wings of finite span and of various shapes, in supersonic flow.

5. Investigation of the possibilities of supersonic propellers.

6. Collection of data regarding air loads for structural design, for supersonic and especially transonic flight.

There are urgent structural problems to be solved to make high-speed flight possible. Several of these involve structural vibrations and the existence of nonstationary air forces on wings and control surfaces. These can hardly be attacked successfully except in high-speed wind tunnels, and again, *there can be no compromise on scale effect.*

The need for supersonic wing theories has already been mentioned. It is necessary that theoretical studies be encouraged in connection with experiments, in order that correct interpretation be made of test data and that information be available outside the range of test data.

In the low-speed region, there are still aeronautical problems whose solutions can profoundly affect the airplane's future. Typical of these is the problem of the low-speed properties (e.g., for landing and take-off) of configurations suitable for high-speed flight. This includes the development of high-lift devices for supersonic airfoils, and improvement of the characteristics of highly swept-back wings.

There are other classes of aeronautical problems whose solutions are vital to the future of airplanes; functioning of equipment at extreme altitudes, design of flight controls powerful enough for high-speed flight, means of safe egress of crew from high-speed aircraft, and development of pilot accomodations suitable for high accelerations and very high altitudes. It will be necessary for manufacturers of such equipment and designers of such facilities to become aware of the extreme altitudes and speeds that we contemplate.

APPENDIX

MAXIMUM RANGE OF AIRPLANES WITH FUEL CONSUMPTION PROPORTIONAL TO THRUST

1. Let R = Range, W = gross weight, V = flying speed, T = thrust, and c' = specific fuel consumption in pounds per unit time per pound of thrust (assumed constant with varying speed and altitude). Then

$$dR = VdW/c'T \tag{1}$$

Assume level flight: T = D

$$= \left(C_{D_p} + \frac{C_L^2}{\pi Ae}\right)\frac{\rho}{2}SV^2 \tag{2}$$

where C_{D_p} = parasite drag coefficient, C_L = lift coefficient, Ae = effective aspect ratio, ρ = air density = $\sigma\rho_o$, S = wing area. Hence

$$\frac{dR}{dW} = \frac{1}{c'}\frac{V}{D} = \frac{1}{c'}\frac{2}{\rho SV}\left(C_{D_p} + \frac{C_L^2}{\pi Ae}\right)^{-1}$$

$$= \frac{1}{c'}\sqrt{\frac{2}{\rho SW}}\left(C_{D_p}\,C_L^{-1/2} + \frac{C_L^{3/2}}{Ae}\right)^{-1} \tag{3}$$

The maximum value of dR/dW (miles per pound) occurs at the minimum value of the expression in brackets. It is easily verified that this occurs when

$$C_{D_p} = 3C_L^2/\pi A_e \tag{4}$$

and the maximum value of dR/dW is

$$\frac{dR}{dW} = \frac{1}{c'}\sqrt{\frac{2}{\rho SW}}\,\frac{3}{4}\left(\frac{\pi Ae}{3}\right)^{1/4} C_{D_p}^{-3/4} \tag{5}$$

The formula for maximum range is obtained by integrating from the initial to the final gross weight. With proper constants, the result can be written in the following form:

$$\begin{aligned} R &= \frac{30}{c'}\sqrt{\frac{b_e}{\sigma}}\,f^{-3/4}\sqrt{W_1}\left(1 - \sqrt{\frac{W_2}{W_1}}\right) \text{ (miles)} \\ &= \frac{30}{c'}\sqrt{\frac{W_1/S}{\sigma}}\left(\frac{Ae}{C_{D_p}^3}\right)^{1/4}\left(1 - \sqrt{\frac{W_2}{W_1}}\right) \text{ (miles)} \end{aligned} \tag{6}$$

where b_e is the effective span ($\sqrt{AeS}$), f is the parasite area (C_{D_p} S), and c' is now measured in pounds per hour per pound of thrust.

This is the formula for maximum range at a given altitude.

2. Since the range improves with increasing altitude in the formula above, it is desirable to obtain the expression for the maximum range for all altitudes. This will occur at the highest altitude at which the best range-flight condition $C_{D_p} = 3C_L^2/\pi Ae$ can be maintained by use of full throttle.

The thrust at full throttle is given approximately by

$$T = T_o \sigma^{0.7} \text{ for } 0 \leqq h \leqq 35{,}000 \text{ ft} \tag{7}$$

$$= (.31)^{0.7} T_o \frac{\sigma}{.31} \text{ for } h \geqq 35{,}000 \text{ ft}$$

where T_o is the sea-level static thrust at full throttle.

We can assume that the altitudes concerned are above 35,000 ft; the second formula can then be used. Equating drag and thrust, we have

$$T = 1.42\, T_o = \frac{D}{L} W$$

$$= \frac{\frac{4}{3} C_{D_P}}{\sqrt{\frac{\pi\, Ae\, C_{D_P}}{3}}} \quad W \tag{8}$$

or

$$\sigma = .92 \frac{W}{T_o} \sqrt{\frac{C_{D_P}}{Ae}} \tag{9}$$

for maximum range.

Substituting this value in the expression for dR/dW above, we obtain

$$\frac{dR}{dW} = \frac{.79}{c'} \sqrt{\frac{2}{\rho_o}} \sqrt{\frac{T_o}{S}} \sqrt{\frac{Ae}{C_{D_p}}} \frac{1}{W} \tag{10}$$

and finally

$$R = \frac{35.9}{c'} \sqrt{\frac{T_o}{S}} \sqrt{\frac{Ae}{C_{D_p}}} \log_{10} \left(\frac{W_1}{W_2}\right) \text{ (miles)}$$

$$= \frac{35.9}{c'} \sqrt{T_o}\, \frac{b_e}{f} \log_{10} \left(\frac{W_1}{W_2}\right) \text{ (miles)} \tag{11}$$

Approximately, neglecting compressibility effects and assuming the induced drag to be negligible, we can replace T_o in trems of the high speed at sea level:

$$T_o \frac{1}{2} \rho_o V_o^2 f \tag{12}$$

The range formula can then be written

$$R = \frac{1.82}{c'} V_o \frac{b_e}{f} \log_{10} \left(\frac{W_1}{W_2}\right) \text{ (miles)} \tag{13}$$

(V_o in mph). Also, since $(L/D)_{max} = \frac{\sqrt{\pi}}{2} \frac{b_e}{f}$; this can be written

$$R = \frac{2.05}{c'} V_o \left(\frac{L}{D}\right)_{max} \log_{10} \left(\frac{W_1}{W_2}\right) \text{ (miles)} \tag{14}$$

These formulas give the maximum range of an airplane, considering all possible altitudes. It will be noted that the altitude increases during the flight in order to satisfy the formula for σ (formula 9) as the weight diminishes.

3. Suppose that the parasite area f is given by the sum of a constant part f_1, necessary for a given power plant and the wing parasite area C_{D_o} S. What is the best choice of wing loading (i.e., of S for given W_1) for range?

For range at constant altitude, the best range will occur at the minimum value of $\left(C_{D_0} + f_1/S\right)^2 S^2$. This occurs when $S = \frac{1}{2}\frac{f_1}{C_{D_0}}$. (15)*

For range at varying (optimum) altitude, the best range will occur at the minimum value of $\left(C_{D_0} - f_1/S\right)^2 S$, which occurs at $S = f_1/C_{D_0}$. (16)

4. On the other hand, suppose that a certain volume is required in the airplane. How should it be proportioned between wing volume and fuselage volume, for best range?

For range at varying (optimum) altitude, the maximum will occur at the minimum of $C_{D_p}^2 S$ or f^2/S. Let v_w be the wing volume, v_n be the fuselage or nacelle volume, and v be the sum, a constant. We can assume v_w proportional to the 3/2-power of the wing area, i.e.

$$v_w = \left(\frac{2S}{k_w}\right)^{3/2} \tag{17}$$

where k_w is a constant. Also, we can assume that the parasitic drag of each component is proportional to its wetted area:

$$f = C_{D_p}; S = C_f\,(S_n + 2S) \tag{18}$$

where C_f is the effective skin-friction coefficient and S_n, the nacelle wetted area, is equal to $k_n v_n^{2/3}$. Then the expression to be minimized is

$$\frac{(k_n v_n^{2/3} + 2S)^2}{S} = \frac{\left\{k_n\left[v - \left(\frac{2S}{k_w}\right)^{3/2}\right]^{2/3} + 2S\right\}^2}{S}$$

$$= \left\{k_n\left[vS^{-3/4} - \left(\frac{2}{k_w}\right)^{3/2} S^{3/4}\right] + 2S^{1/2}\right\}^2 \tag{19}$$

This minimum occurs when

$$S = \frac{k_n}{2}\frac{v + v_w}{(v - v_w)}\frac{1}{3} \tag{20}$$

or since $S = \frac{k_w v_w^{2/3}}{2}$

$$\frac{k_n}{k_w}\left(\frac{v}{v_w} + 1\right) = \left(\frac{v}{v_w} - 1\right)^{1/3} \tag{21}$$

For airplanes with thin wings it appears that k_w will have a value about 20 to 25. The fuselage or nacelle will have a value of k_n between 4.8 (for a sphere) and 8. As an extreme case, favoring the nacelle, let us investigate $k_n / k_w = .20$. We find solutions $v / v_w = 1.065$ and 9.0. It can be ascertained that the former gives a maximum range, while the latter gives a minimum. Hence, for best range, the needed volume should be obtained almost entirely in the wing.

PERFORMANCE OF ROCKET MISSILE IN CLIMBING FLIGHT

In our curves for L/D as function of M, l_w/p, and f_n/S, we find that the value of L/D does not vary greatly with l_w/p if this variable is restricted to a certain range near

* This result has been pointed out by Lush in Ref. 24.

$(L/D)_{max}$. In order to maintain L/D near maximum (and to simplify the calculations), we will assume

$$\frac{dl_w}{l_w} = \frac{1}{N}\frac{dp}{p} \tag{22}$$

with varying altitude. Now in an isothermal atmosphere $dp/p = \text{constant} \times dh = -Adh$, say. For flight in the troposphere we can choose a mean value for dp/p and denote it by $-A\,dh$.

If V is the velocity along the flight path and θ is the inclination of this path to the horizontal,

$$\frac{dh}{dW} = \frac{-kV\sin\theta}{T} \tag{23}$$

where k is the specific impulse (lb-sec per lb of fuel) and T is the thrust. Also $T = \left(\frac{D}{L} + \sin\theta\right) W$, hence

$$\frac{dh}{dW} = \frac{-kV}{W}\frac{\sin\theta}{\frac{D}{L} + \sin\theta} \tag{24}$$

$$\frac{dW}{W} = -\frac{\frac{D}{L} + \sin\theta}{kV\sin\theta}\,dh = \frac{1}{N}\frac{dp}{p} = \frac{-1}{N}A\,dh \tag{25}$$

or

$$\frac{1}{N}AkV = 1 + \frac{D}{L}\cos\theta. \tag{26}$$

It is our assumption that by restricting dW/W to 1/N times dp/p, we are holding D/L approximately constant. Hence, one solution to the above equation is

$$V = \text{constant and } \theta = \cos^{-1}\frac{L}{D}\left(\frac{1}{N}AkV - 1\right) \tag{27}$$

The final relations for altitude and horizontal range are then

$$h = \frac{N}{A}\log_e\left(\frac{W_1}{W_2}\right) \tag{28}$$

and

$$R = h\cot\theta. \tag{29}$$

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2. Brevoort, M. J., Stickle, G. W., and Hill, P. R.; "Selection Charts for Bombers Using Four 3000-Horsepower Engines;" NACA Confidential Memoradum Report for the Army Air Corps; 27 February 1942.

3. Brevoort, M. J., Stickle, G. W., and Hill, P. R.; "Generalized Selection Charts for Bombers Powered by One, Two, Four, and Six 2000-Horsepower Engines" and "Supplement I. — Capacity and Economy;" NACA Confidential Memorandum Report for Army Air Forces, Materiel Command; 6 July 1942, and 19 September 1942.

4. Brevoort, M. J., Stickle, G. W., and Hill, P. R.; "Generalized Selection Charts for Bombers Powered by Two, Four, and Six 3000-Horsepower Engines;" NACA Confidential Memorandum Report for Army Air Forces, Materiel Command; 13 August 1942.

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7. Hill, P. R., and Crigler, J. L.; "The Effect of Altitude on Bomber Performance;" NACA Confidential Memorandum Report for Army Air Forces, Materiel Command; 15 October 1943.

8. Ivey, H. R., Stickle, G. W., and Brevoort, M. J.; "Generalized Performance Selection Charts for Single-Engine Pursuit Airplanes;" NACA Confidential Preliminary Memorandum Report for the Army Air Forces, Materiel Command; 6 February 1943.

9. Ivey, H. R., Stickle, G. W., and Brevoort, M. J.; "Fire Power Versus Performance;" NACA Confidential Memorandum Report for the Army Air Forces, Materiel Command; 24 June 1943.

10. Ivey, H. R., Stickle, G. W., and Brevoort, M. J.; "Pursuit Possibilities with 3000 Horsepower;" NACA Confidential Memorandum Report for the Army Air Forces, Materiel Command; 9 December 1943.

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18. Williams, G. C., and Quinn, J. C.; "Ramjet Power Plants;" Report for Commander in Chief, U. S. Fleet, No. 174; May 1945.

19. Gilliland, E. R.; "Rocket-Powered Missiles;" Report for Commander in Chief, U. S. Fleet, No. 198; May 1945.

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21. Hammett, L. P.; "Solid Propellants for Rockets and Other Jet Devices;" AAF Scientific Advisory Group Report; 26 June 1945; published in SAG Report "Aircraft Fuels and Propellants;" May 1946.

22. Hage, R. E., and Heuver, H. M.; "Analytical Study of the Performance of a Cycloidal Propeller and a Preliminary Survey of a Fighter Type Design;" ATSC, Engineering Memorandum Report Eng-51, p 706-84, dated 22 July 1943.

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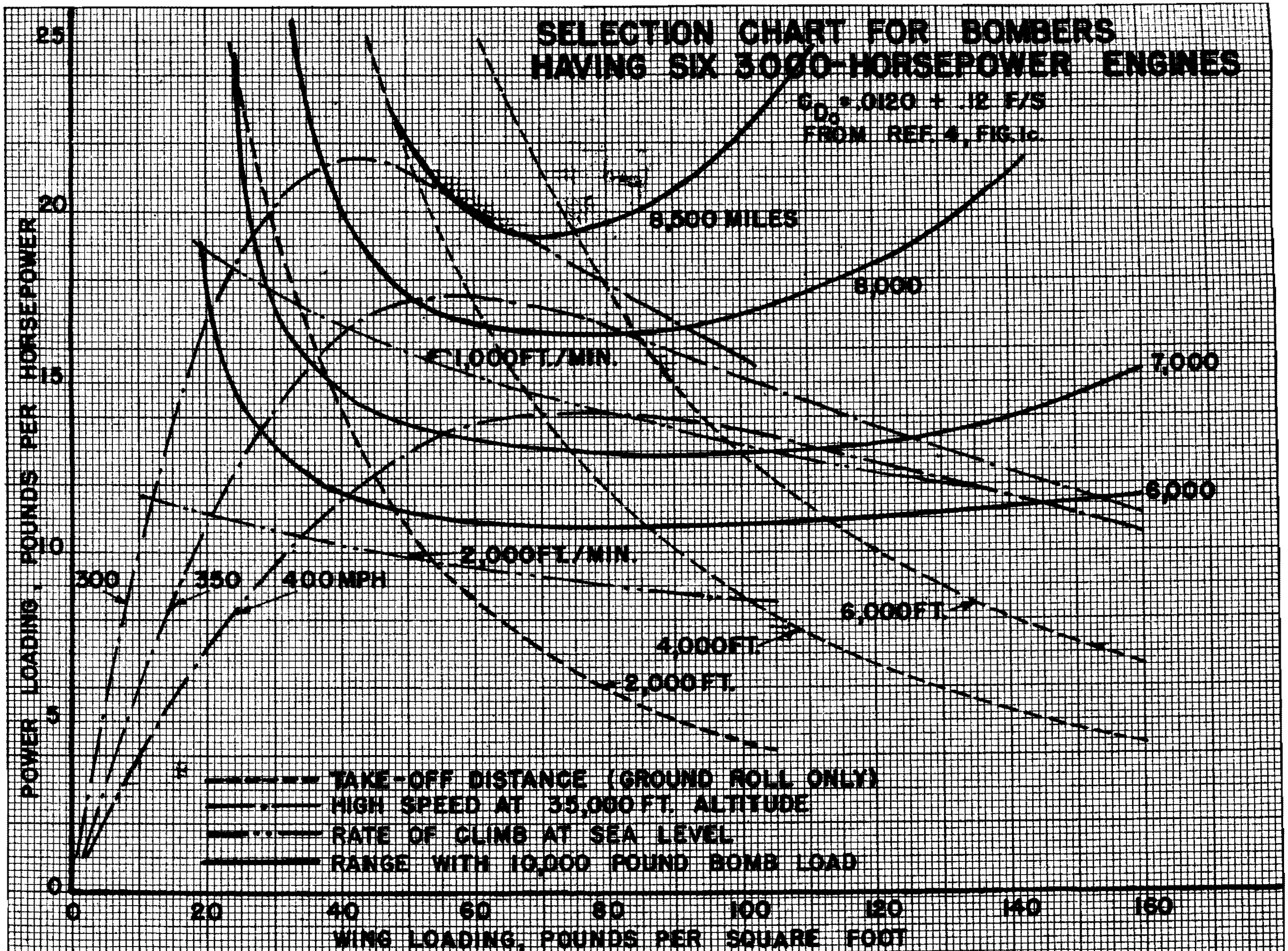
24. Lush, K.; "Characteristics of Jet-Propelled Airplanes;" British A.R.C. Publication 5662.

FIGURES



SELECTION CHART FOR BOMBERS
HAVING SIX 3000-HORSEPOWER ENGINES
$C_{D_0} = .0120 + .12\ F/S$
FROM REF. 4, FIG. Ic
8,500 MILES
8,000
7,000
6,000
1,000 FT./MIN.
2,000 FT./MIN.
300
350
400 MPH
6,000 FT.
4,000 FT.
2,000 FT.
TAKE-OFF DISTANCE (GROUND ROLL ONLY)
HIGH SPEED AT 35,000 FT. ALTITUDE
RATE OF CLIMB AT SEA LEVEL
RANGE WITH 10,000 POUND BOMB LOAD
POWER LOADING, POUNDS PER HORSEPOWER
WING LOADING, POUNDS PER SQUARE FOOT
0
5
10
15
20
25
0
20
40
60
80
100
120
140
160

Figure 1

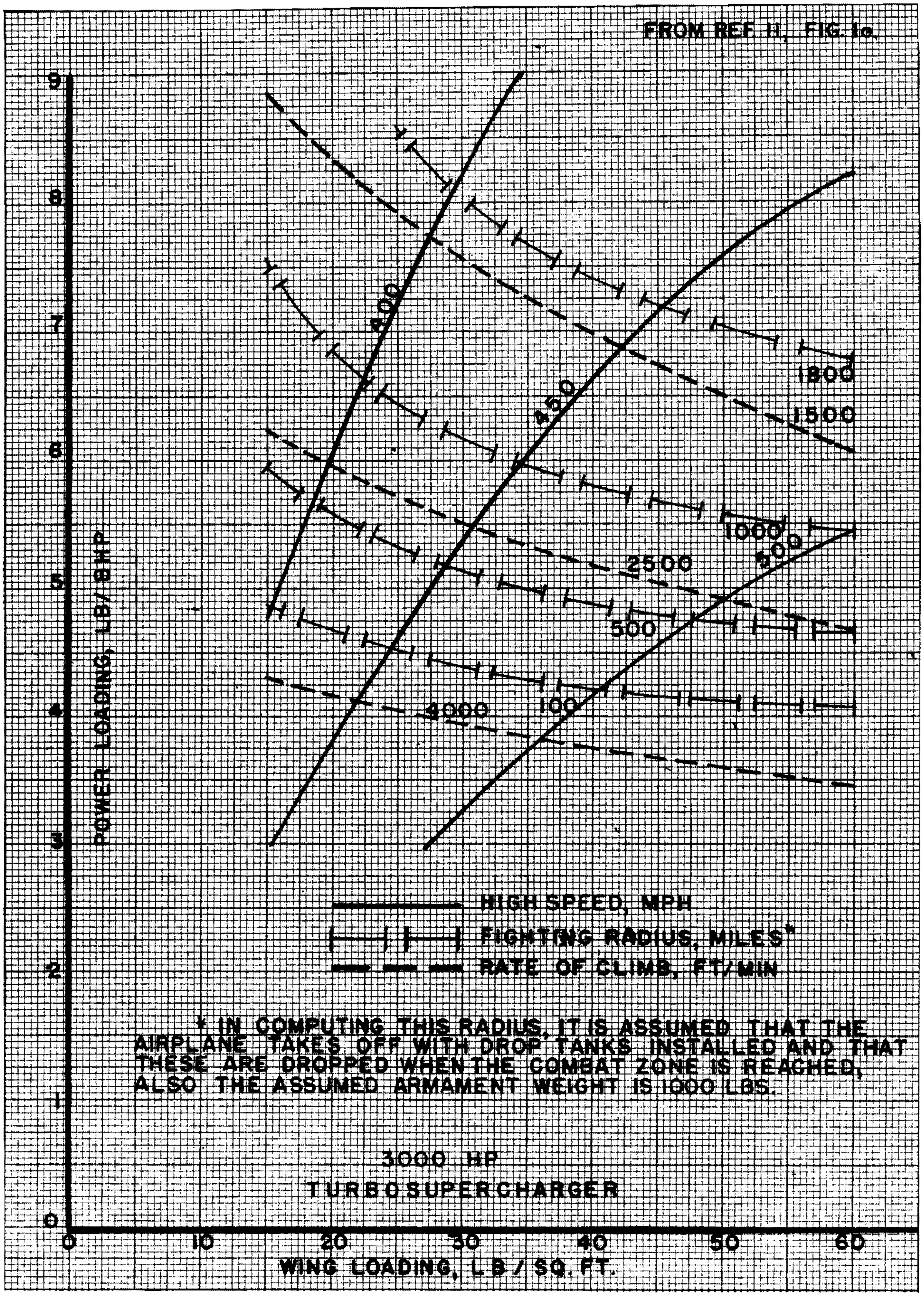
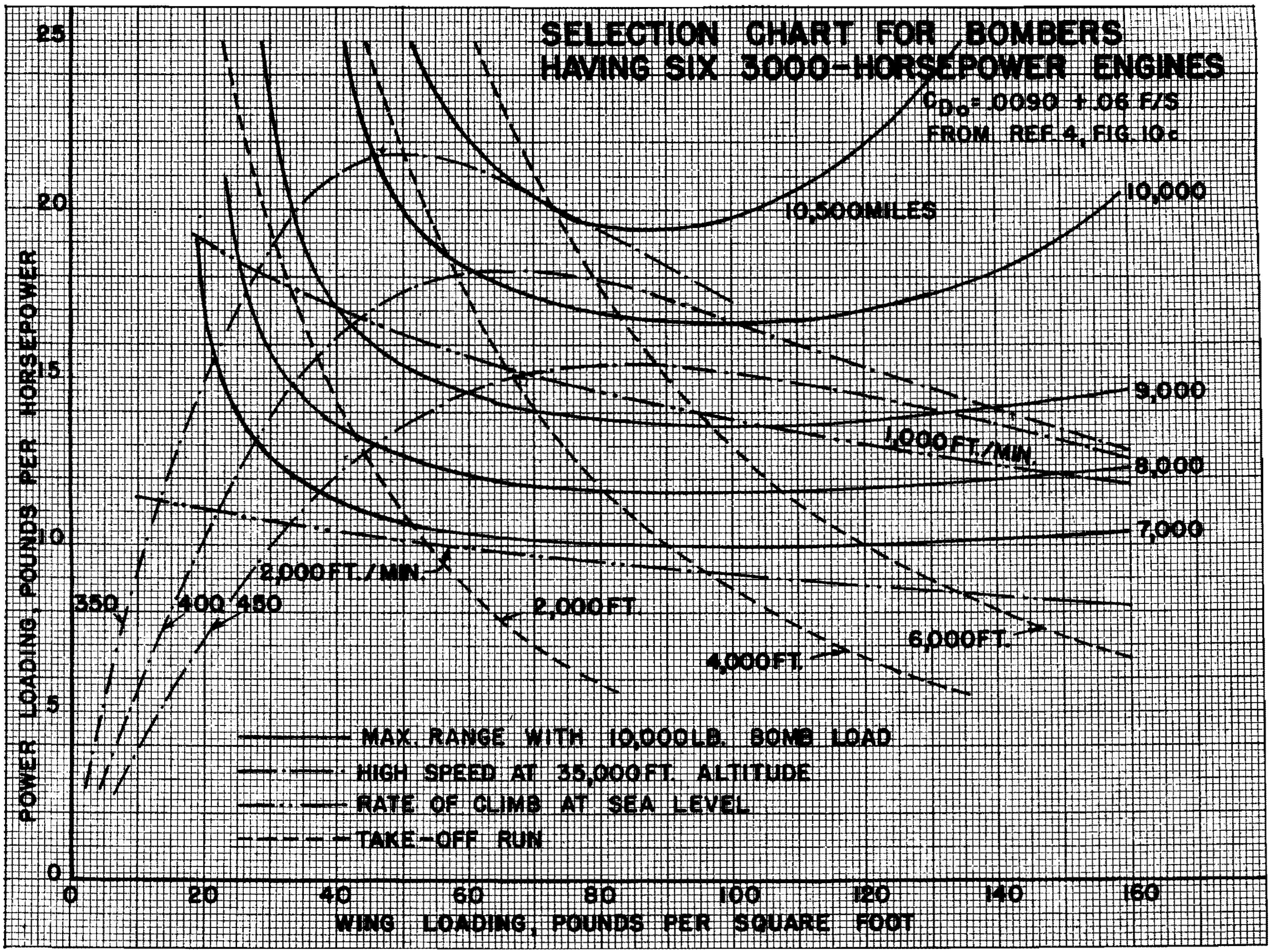


Figure 2 — Selection Chart for 3000-Horsepower Pursuit Airplanes at 30,000 Feet Altitude

80



SELECTION CHART FOR BOMBERS
HAVING SIX 3000-HORSEPOWER ENGINES
C_{D_0} = .0090 + .06 F/S
FROM REF 4, FIG. 10c
10,500 MILES
10,000
9,000
8,000
7,000
1,000 FT./MIN.
2,000 FT./MIN.
350
400
450
2,000 FT.
4,000 FT.
6,000 FT.
MAX. RANGE WITH 10,000 LB. BOMB LOAD
HIGH SPEED AT 35,000 FT. ALTITUDE
RATE OF CLIMB AT SEA LEVEL
TAKE-OFF RUN
POWER LOADING, POUNDS PER HORSEPOWER
WING LOADING, POUNDS PER SQUARE FOOT
0
5
10
15
20
25
20
40
60
80
100
120
140
160

Figure 3

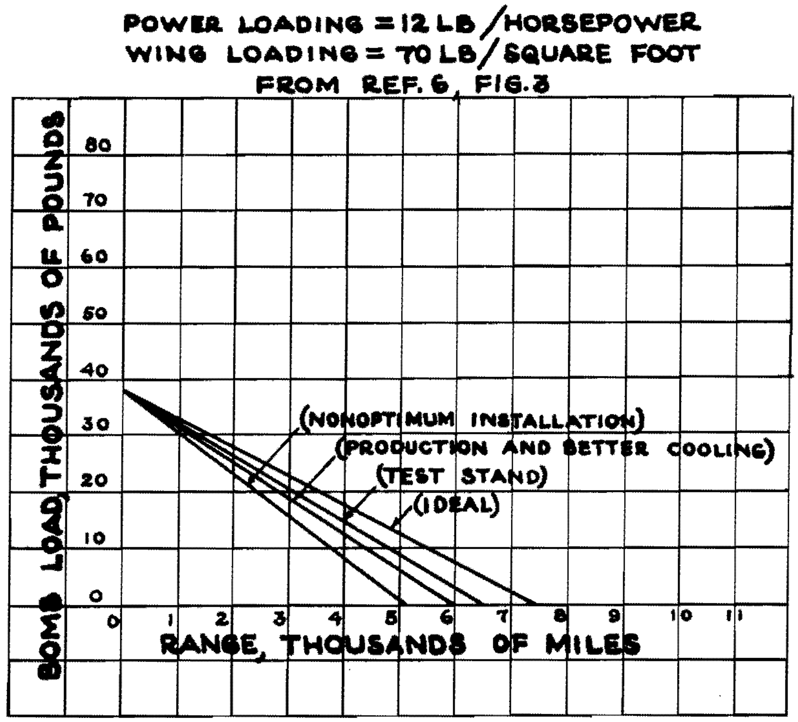


Figure 4 — Effect of Specific Fuel Consumption on Range of Four-Engine Bombers

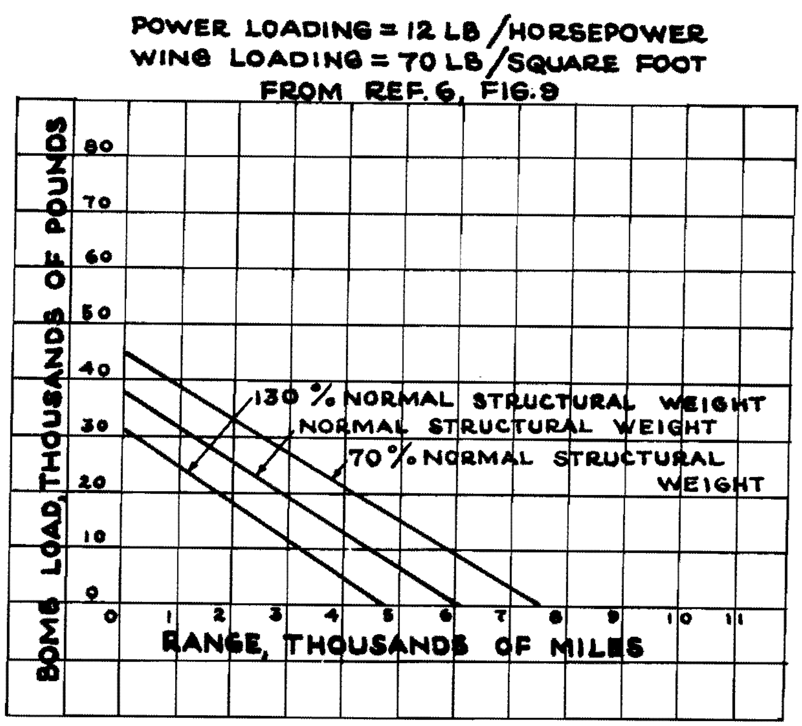


Figure 5 — Effect of Structural Weight on Range of Four-Engine Bombers

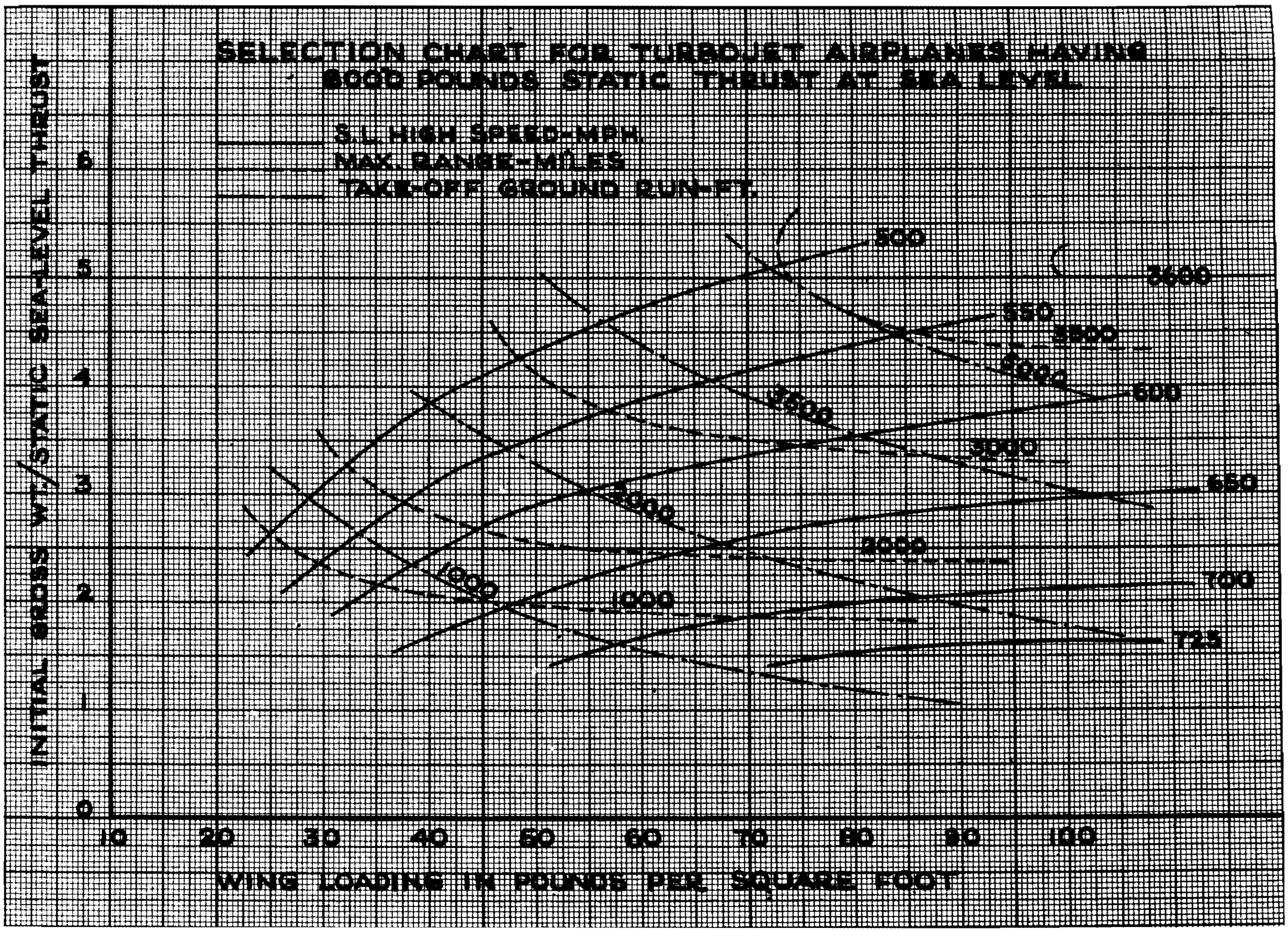

SELECTION CHART FOR TURBOJET AIRPLANES HAVING
8000 POUNDS STATIC THRUST AT SEA LEVEL
S.L. HIGH SPEED-MPH.
MAX. RANGE-MILES
TAKE-OFF GROUND RUN-FT.
INITIAL GROSS WT./STATIC SEA-LEVEL THRUST
WING LOADING IN POUNDS PER SQUARE FOOT
0
1
2
3
4
5
6
10
20
30
40
50
60
70
80
90
100
500
550
600
650
700
725
3600
3500
3000
2500
2000
1000

Figure 6

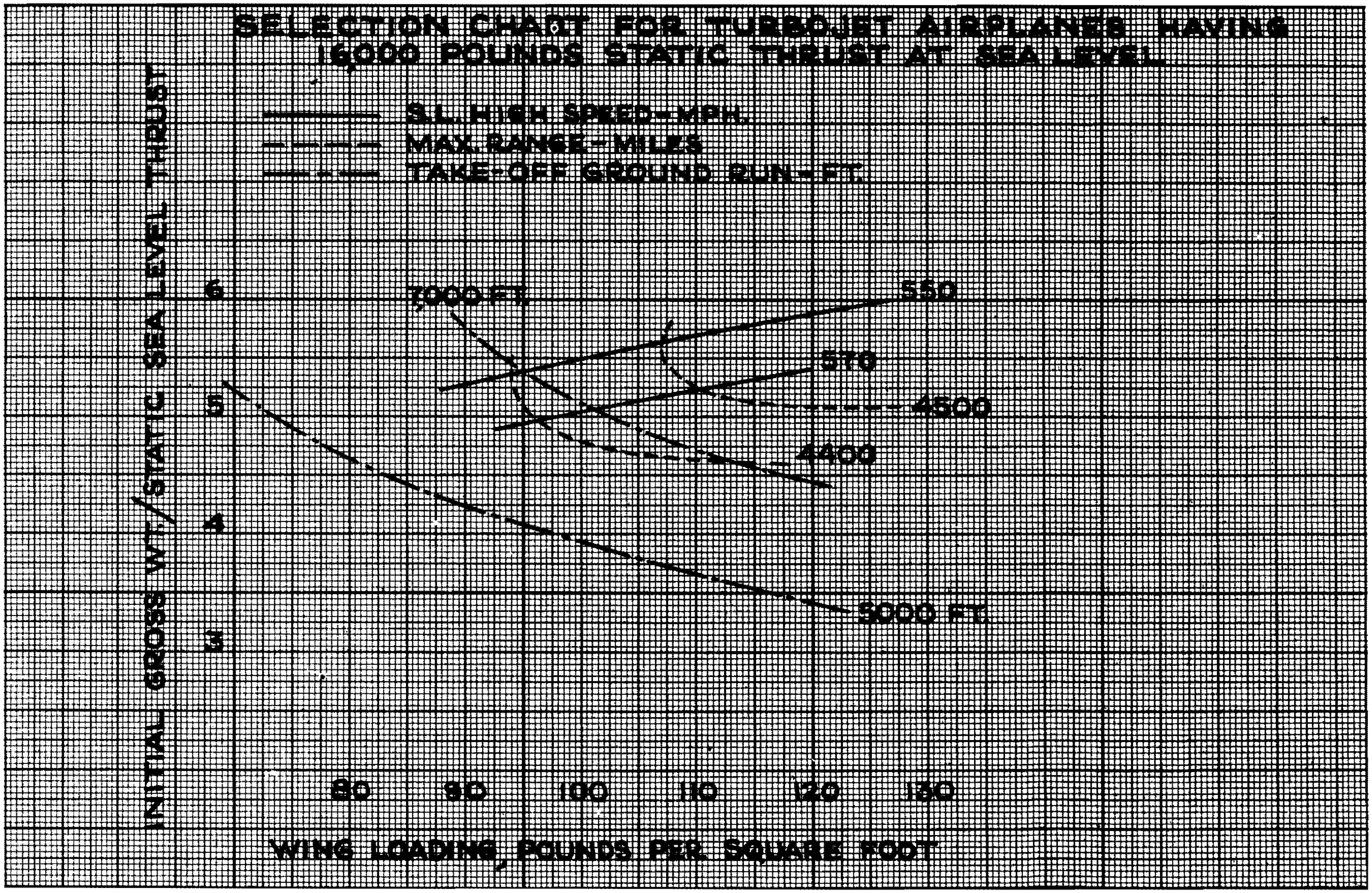

SELECTION CHART FOR TURBOJET AIRPLANES HAVING 16,000 POUNDS STATIC THRUST AT SEA LEVEL
S.L. HIGH SPEED-MPH.
MAX. RANGE-MILES
TAKE-OFF GROUND RUN-FT.
7000 FT.
550
570
4500
4400
5000 FT.
INITIAL GROSS WT./STATIC SEA LEVEL THRUST
6
5
4
3
80
90
100
110
120
130
WING LOADING, POUNDS PER SQUARE FOOT

Figure 7

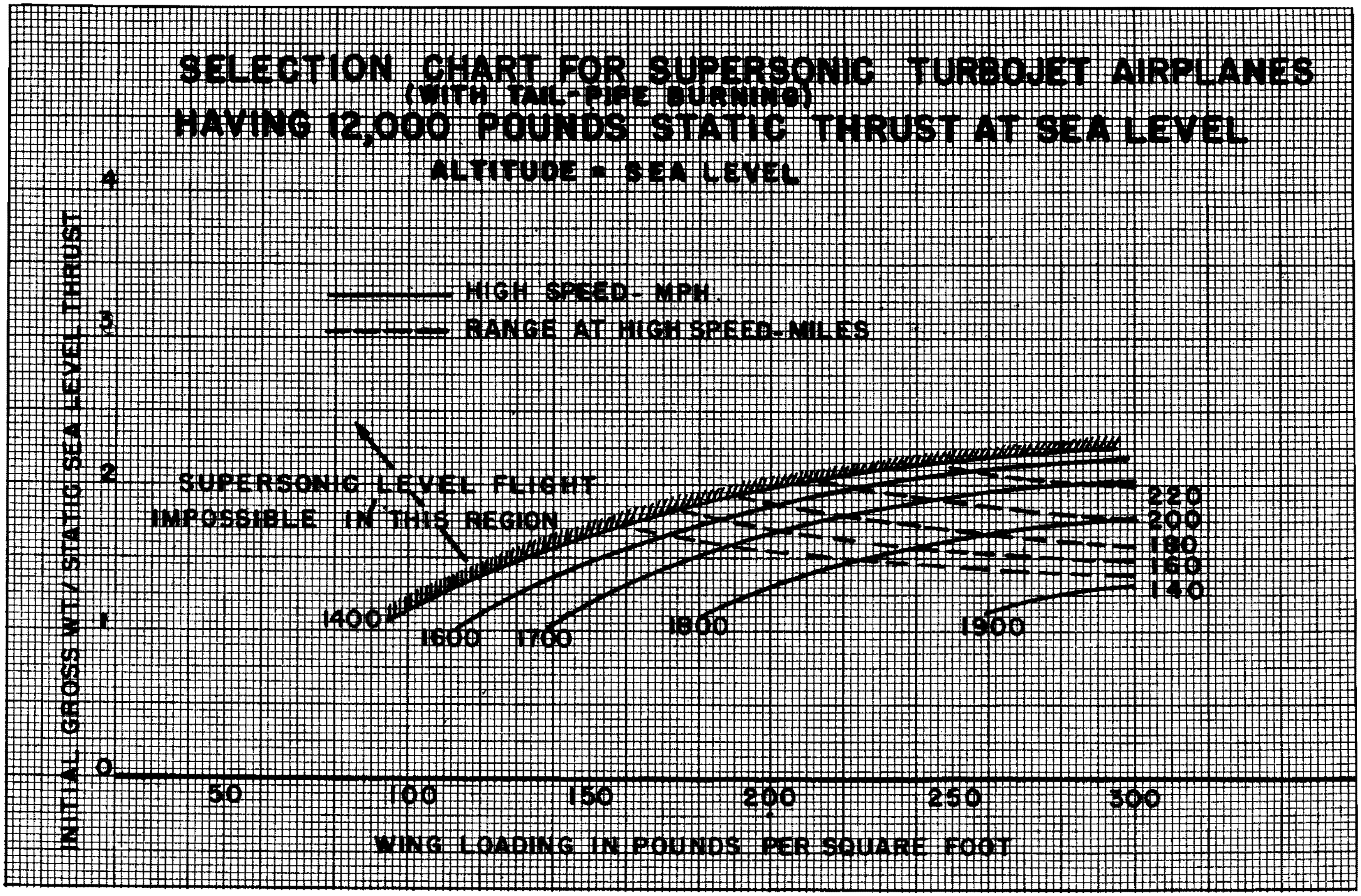

SELECTION CHART FOR SUPERSONIC TURBOJET AIRPLANES
(WITH TAIL-PIPE BURNING)
HAVING 12,000 POUNDS STATIC THRUST AT SEA LEVEL
ALTITUDE = SEA LEVEL
HIGH SPEED- MPH.
RANGE AT HIGH SPEED-MILES
SUPERSONIC LEVEL FLIGHT
IMPOSSIBLE IN THIS REGION
1400
1600
1700
1800
1900
220
200
180
160
140
INITIAL GROSS WT./ STATIC SEA LEVEL THRUST
0
1
2
3
4
50
100
150
200
250
300
WING LOADING IN POUNDS PER SQUARE FOOT

Figure 8

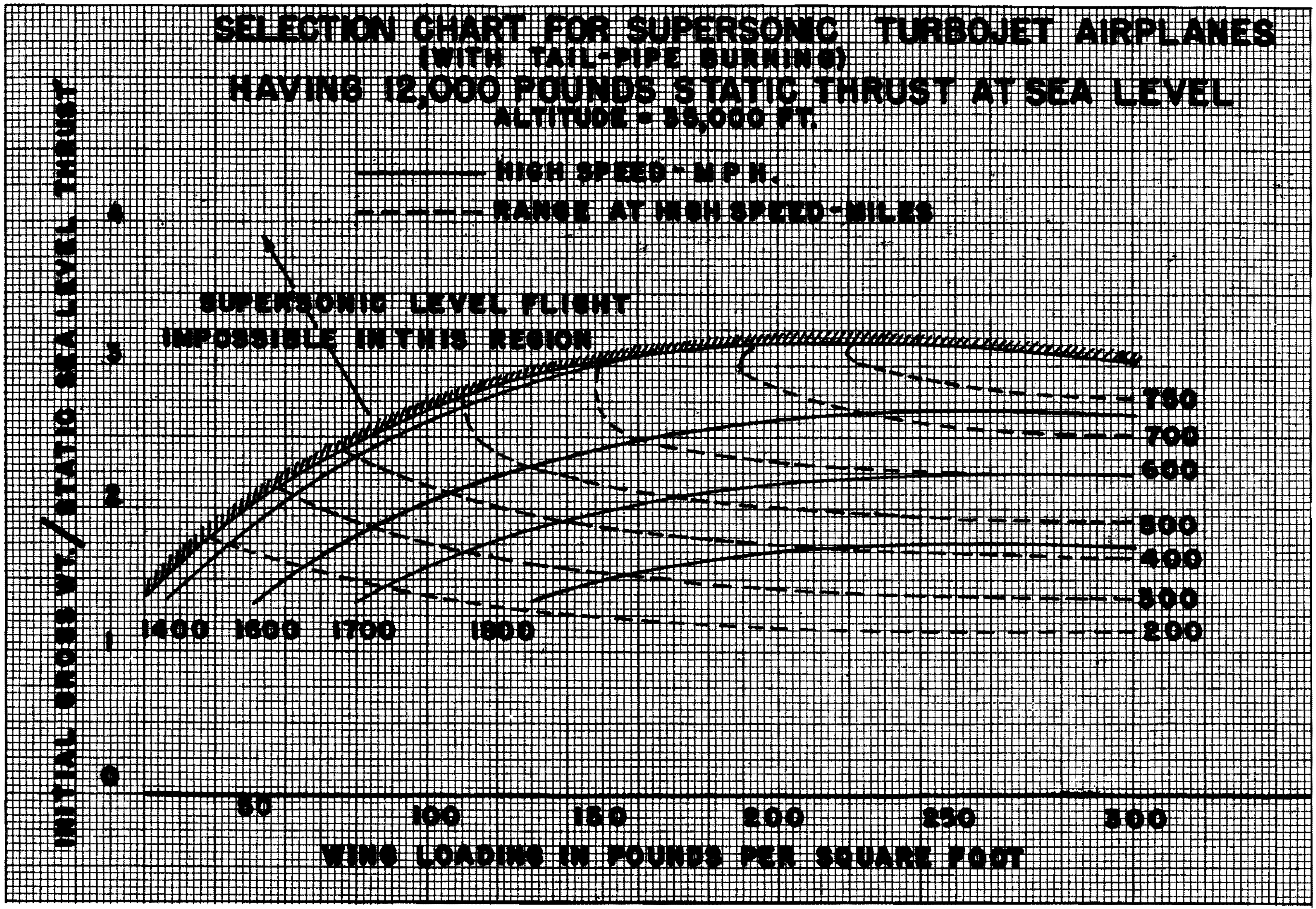

SELECTION CHART FOR SUPERSONIC TURBOJET AIRPLANES
(WITH TAIL-PIPE BURNING)
HAVING 12,000 POUNDS STATIC THRUST AT SEA LEVEL
ALTITUDE = 35,000 FT.
HIGH SPEED - M.P.H.
RANGE AT HIGH SPEED - MILES
SUPERSONIC LEVEL FLIGHT
IMPOSSIBLE IN THIS REGION
INITIAL GROSS WT./STATIC SEA LEVEL THRUST
0
1
2
3
4
1400
1600
1700
1800
750
700
600
500
400
300
200
50
100
150
200
250
300
WING LOADING IN POUNDS PER SQUARE FOOT

Figure 9

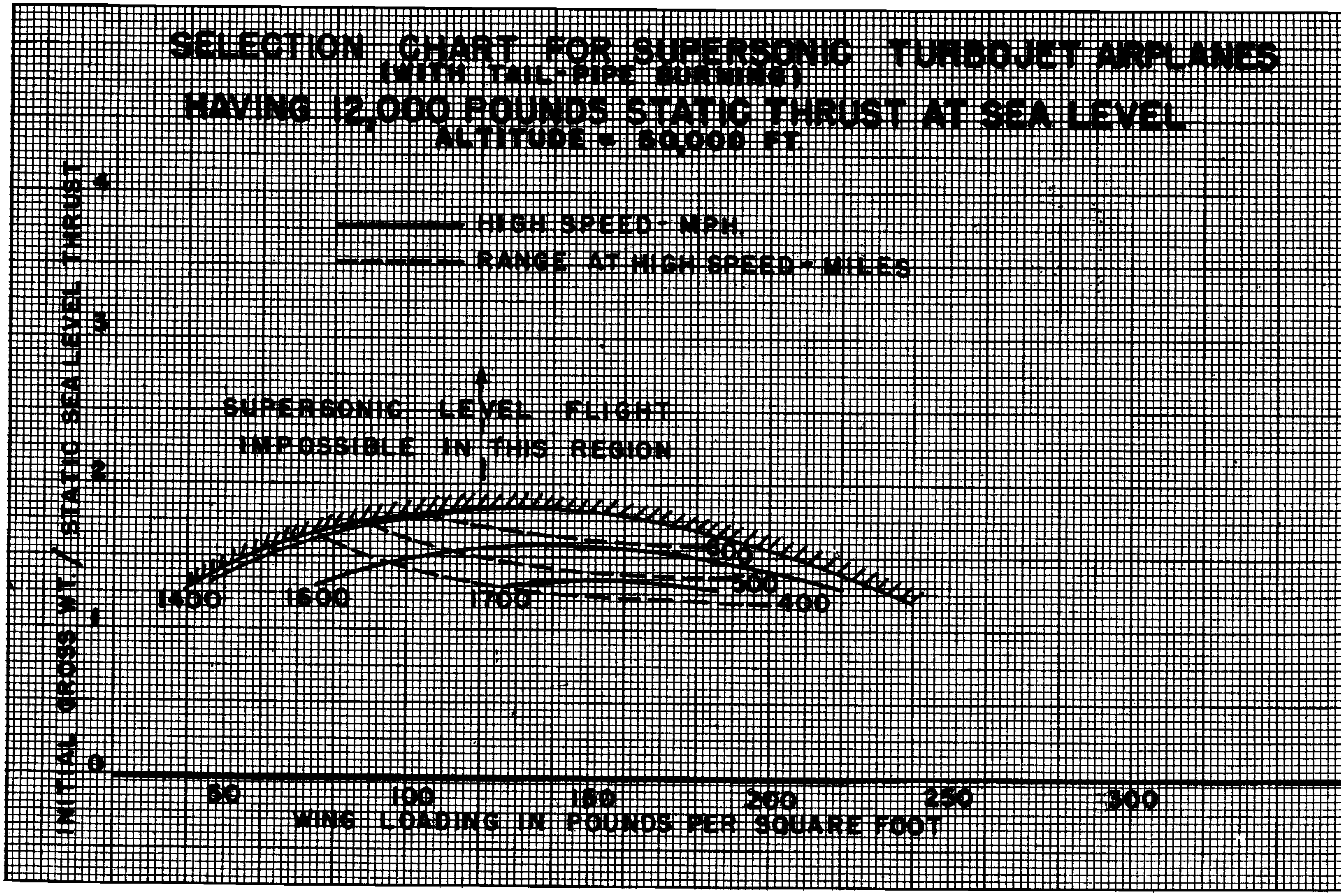

SELECTION CHART FOR SUPERSONIC TURBOJET AIRPLANES
(WITH TAIL-PIPE BURNING)
HAVING 12,000 POUNDS STATIC THRUST AT SEA LEVEL
ALTITUDE = 50,000 FT.
HIGH SPEED - MPH.
RANGE AT HIGH SPEED - MILES
SUPERSONIC LEVEL FLIGHT
IMPOSSIBLE IN THIS REGION
1400
1600
1700
600
500
400
INITIAL GROSS WT. / STATIC SEA LEVEL THRUST
0
1
2
3
4
50
100
150
200
250
300
WING LOADING IN POUNDS PER SQUARE FOOT

Figure 10

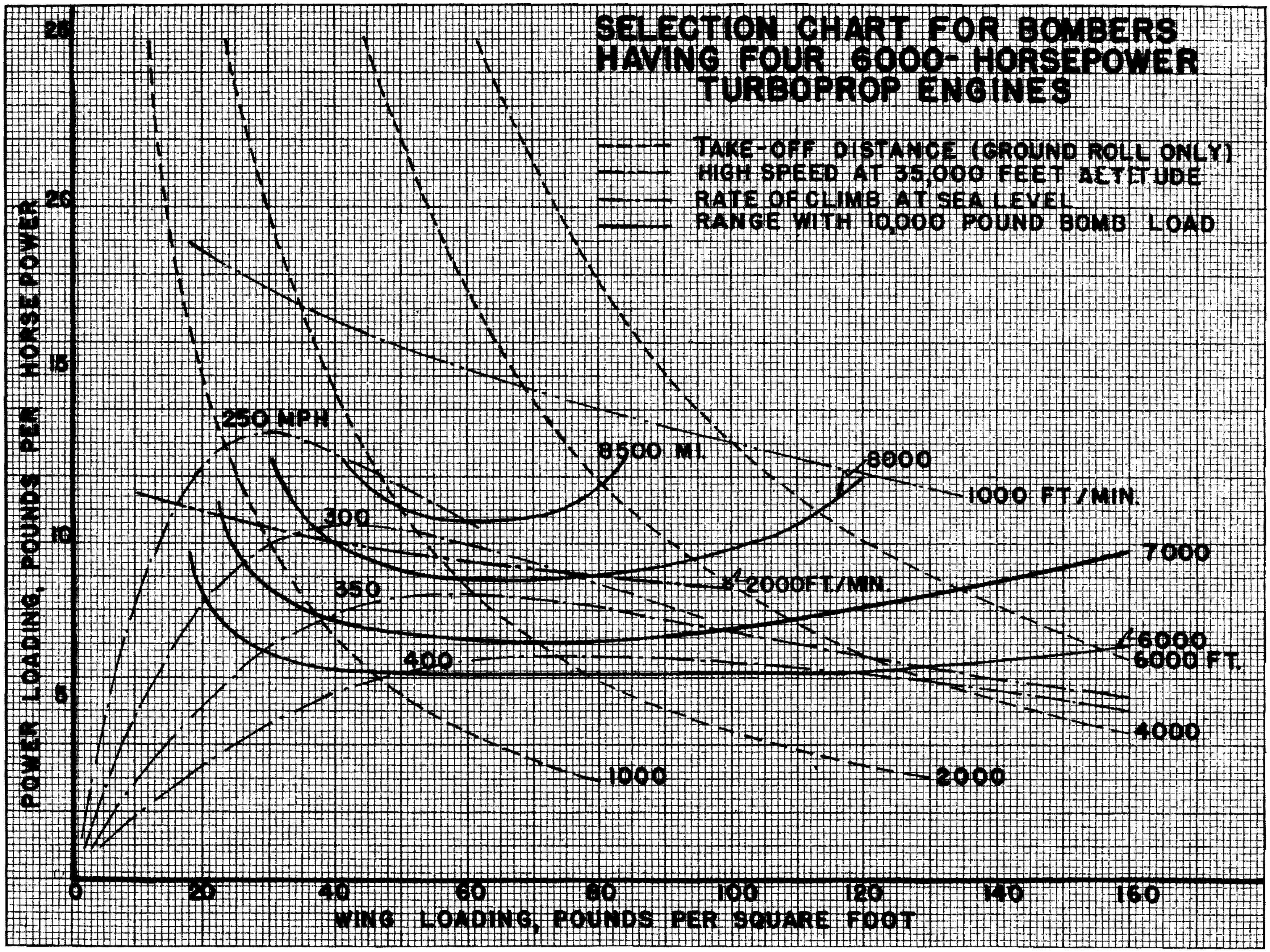

SELECTION CHART FOR BOMBERS HAVING FOUR 6000- HORSEPOWER TURBOPROP ENGINES
TAKE-OFF DISTANCE (GROUND ROLL ONLY)
HIGH SPEED AT 35,000 FEET ALTITUDE
RATE OF CLIMB AT SEA LEVEL
RANGE WITH 10,000 POUND BOMB LOAD
250 MPH
300
350
400
8500 MI.
8000
7000
6000
1000 FT./MIN.
2000 FT./MIN.
6000 FT.
4000
1000
2000
POWER LOADING, POUNDS PER HORSE POWER
WING LOADING, POUNDS PER SQUARE FOOT
0
20
40
60
80
100
120
140
160
5
10
15
20
25

Figure 11

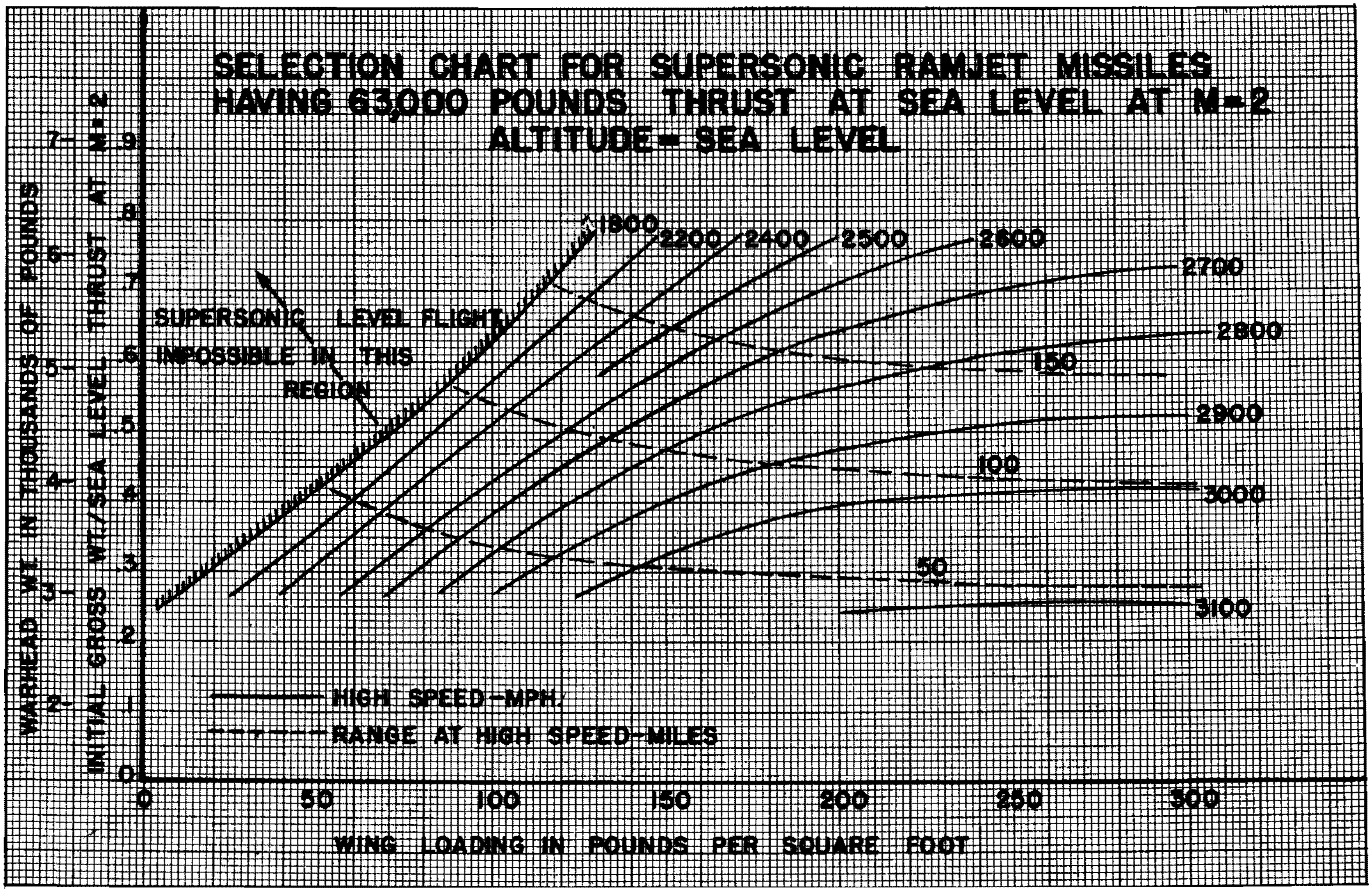

SELECTION CHART FOR SUPERSONIC RAMJET MISSILES
HAVING 63,000 POUNDS THRUST AT SEA LEVEL AT M=2
ALTITUDE = SEA LEVEL
WARHEAD WT. IN THOUSANDS OF POUNDS
INITIAL GROSS WT./SEA LEVEL THRUST AT M=2
SUPERSONIC LEVEL FLIGHT IMPOSSIBLE IN THIS REGION
1800
2200
2400
2500
2600
2700
2800
2900
3000
3100
150
100
50
HIGH SPEED–MPH
RANGE AT HIGH SPEED–MILES
WING LOADING IN POUNDS PER SQUARE FOOT

Figure 12

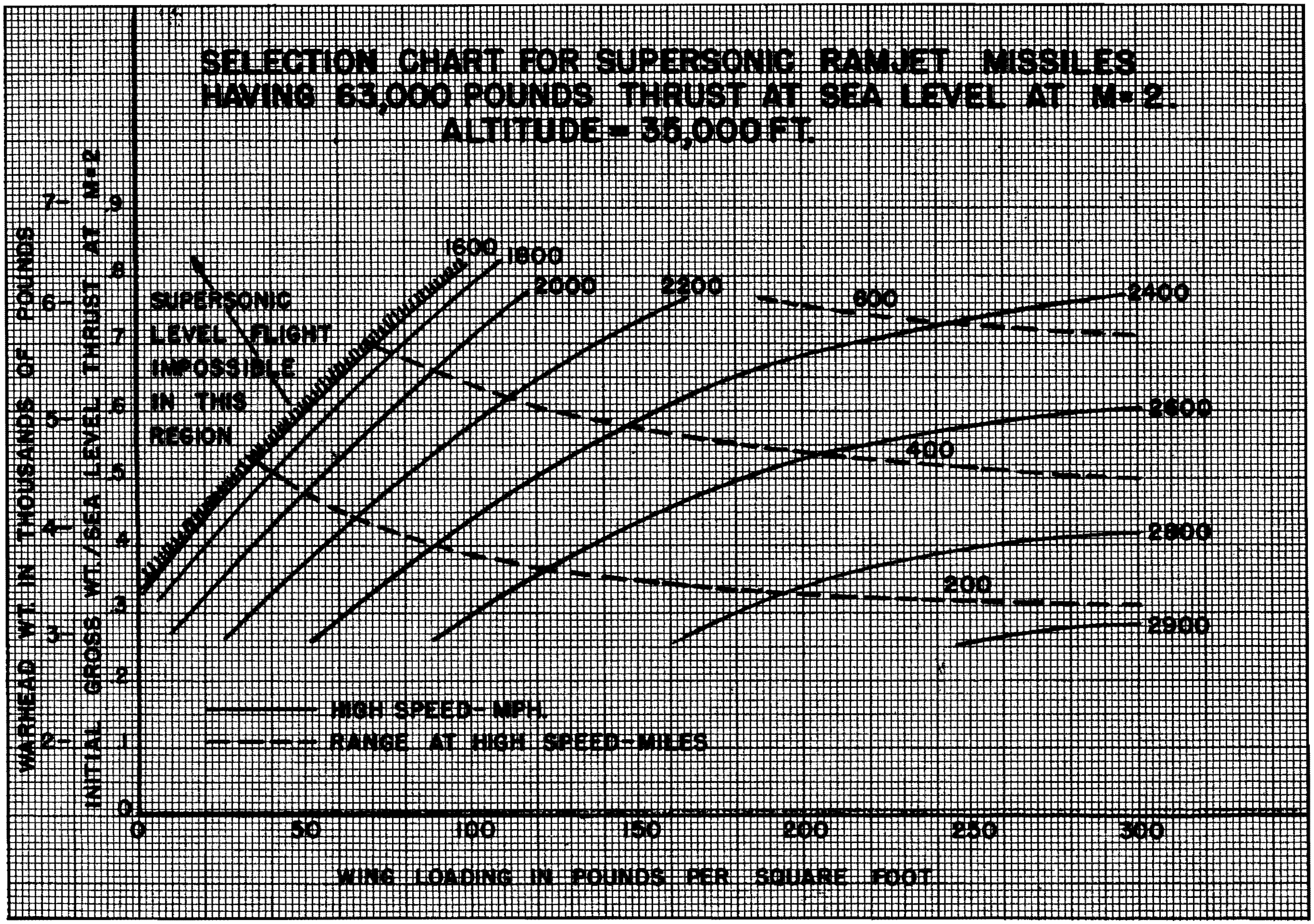
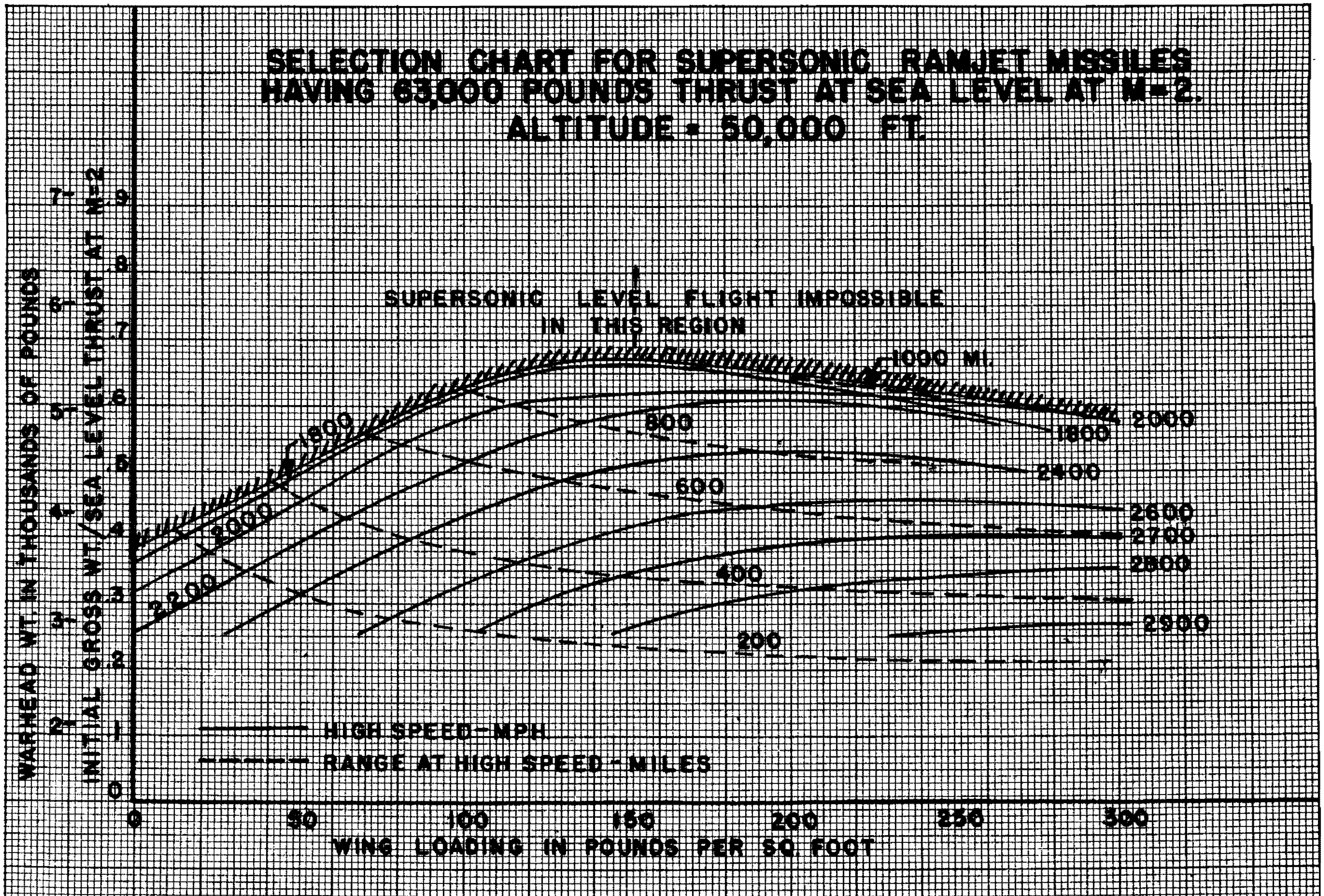

SELECTION CHART FOR SUPERSONIC RAMJET MISSILES
HAVING 63,000 POUNDS THRUST AT SEA LEVEL AT M=2.
ALTITUDE = 35,000 FT.
SUPERSONIC LEVEL FLIGHT IMPOSSIBLE IN THIS REGION
1600
1800
2000
2200
2400
2600
2800
2900
600
400
200
HIGH SPEED- MPH.
RANGE AT HIGH SPEED-MILES
WARHEAD WT. IN THOUSANDS OF POUNDS
INITIAL GROSS WT./SEA LEVEL THRUST AT M=2
WING LOADING IN POUNDS PER SQUARE FOOT
0
50
100
150
200
250
300
1
2
3
4
5
6
7
8
9

Figure 13



SELECTION CHART FOR SUPERSONIC RAMJET MISSILES
HAVING 63,000 POUNDS THRUST AT SEA LEVEL AT M=2.
ALTITUDE = 50,000 FT.
SUPERSONIC LEVEL FLIGHT IMPOSSIBLE
IN THIS REGION
1000 MI.
800
600
400
200
1800
2000
2200
2400
2600
2700
2800
2900
HIGH SPEED-MPH
RANGE AT HIGH SPEED-MILES
WARHEAD WT. IN THOUSANDS OF POUNDS
INITIAL GROSS WT./SEA LEVEL THRUST AT M=2
WING LOADING IN POUNDS PER SQ. FOOT
0
50
100
150
200
250
300

Figure 14

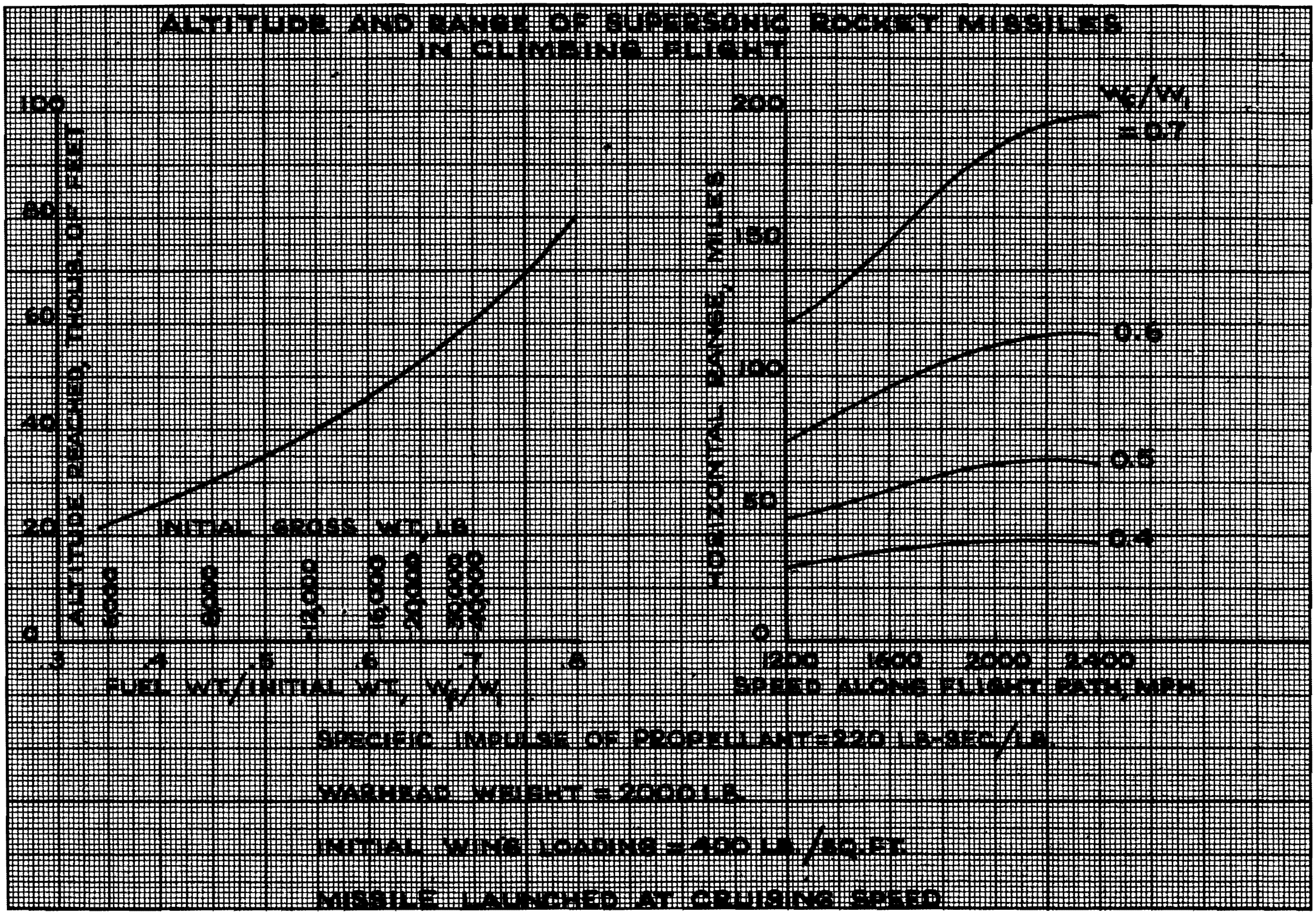

ALTITUDE AND RANGE OF SUPERSONIC ROCKET MISSILES
IN CLIMBING FLIGHT
ALTITUDE REACHED, THOUS. OF FEET
100
80
60
40
20
0
INITIAL GROSS WT, LB.
5000
8000
12,000
16,000
20,000
24,000
30,000
40,000
.3
.4
.5
.6
.7
.8
FUEL WT/INITIAL WT, W_F/W_I
HORIZONTAL RANGE, MILES
200
150
100
50
0
W_F/W_I
= 0.7
0.6
0.5
0.4
1200
1600
2000
2400
SPEED ALONG FLIGHT PATH, MPH.
SPECIFIC IMPULSE OF PROPELLANT = 220 LB-SEC/LB.
WARHEAD WEIGHT = 2000 LB.
INITIAL WING LOADING = 400 LB/SQ.FT.
MISSILE LAUNCHED AT CRUISING SPEED

Figure 15

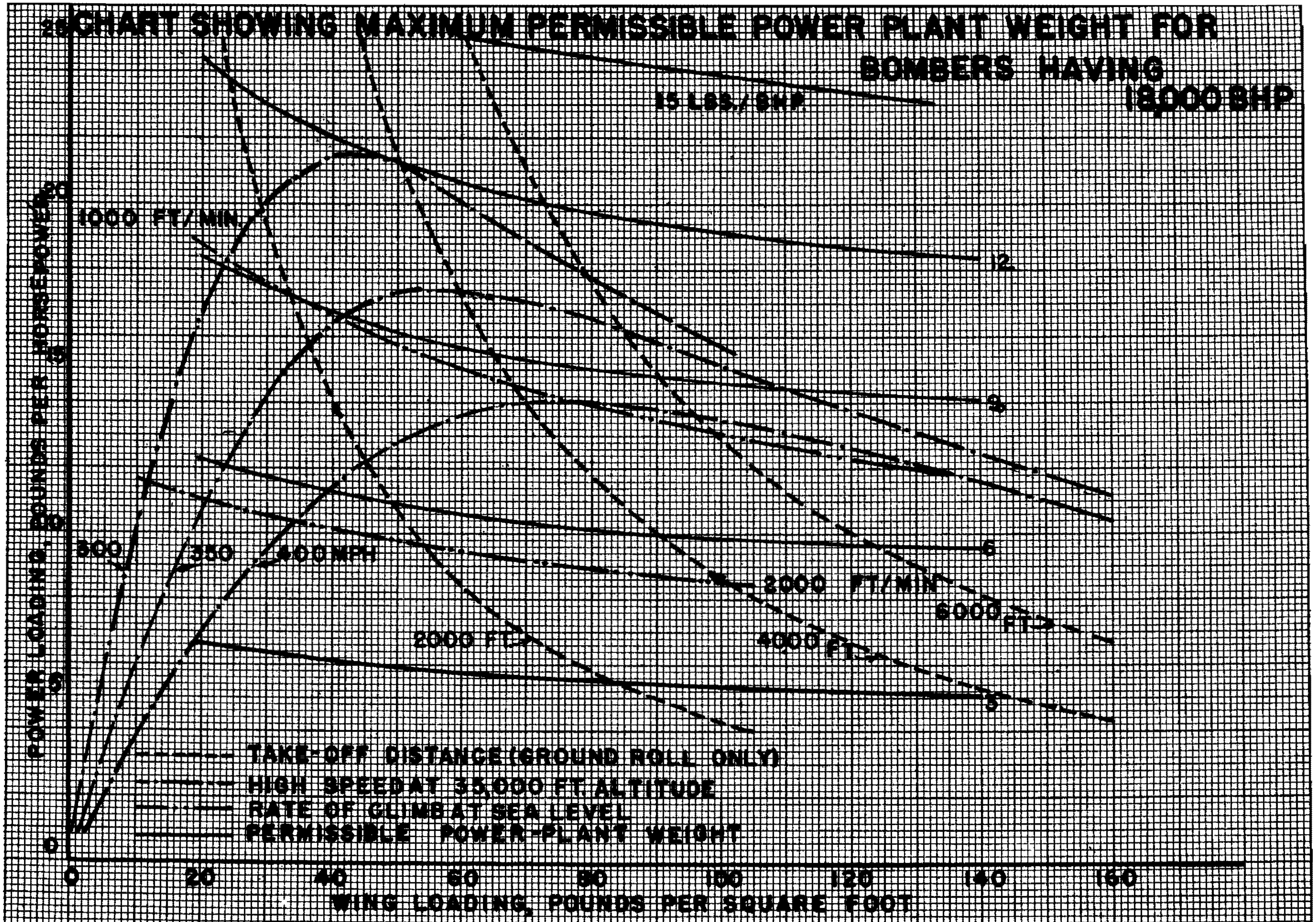


CHART SHOWING MAXIMUM PERMISSIBLE POWER PLANT WEIGHT FOR
BOMBERS HAVING
18,000 SHP
15 LBS./SHP.
12
9
6
5
1000 FT/MIN.
2000 FT/MIN.
300
350
400 MPH
2000 FT.
4000 FT.
6000 FT.
TAKE-OFF DISTANCE (GROUND ROLL ONLY)
HIGH SPEED AT 35,000 FT ALTITUDE
RATE OF CLIMB AT SEA LEVEL
PERMISSIBLE POWER-PLANT WEIGHT
POWER LOADING, POUNDS PER HORSEPOWER
0
5
10
15
20
25
WING LOADING, POUNDS PER SQUARE FOOT
0
20
40
60
80
100
120
140
160

Figure 16

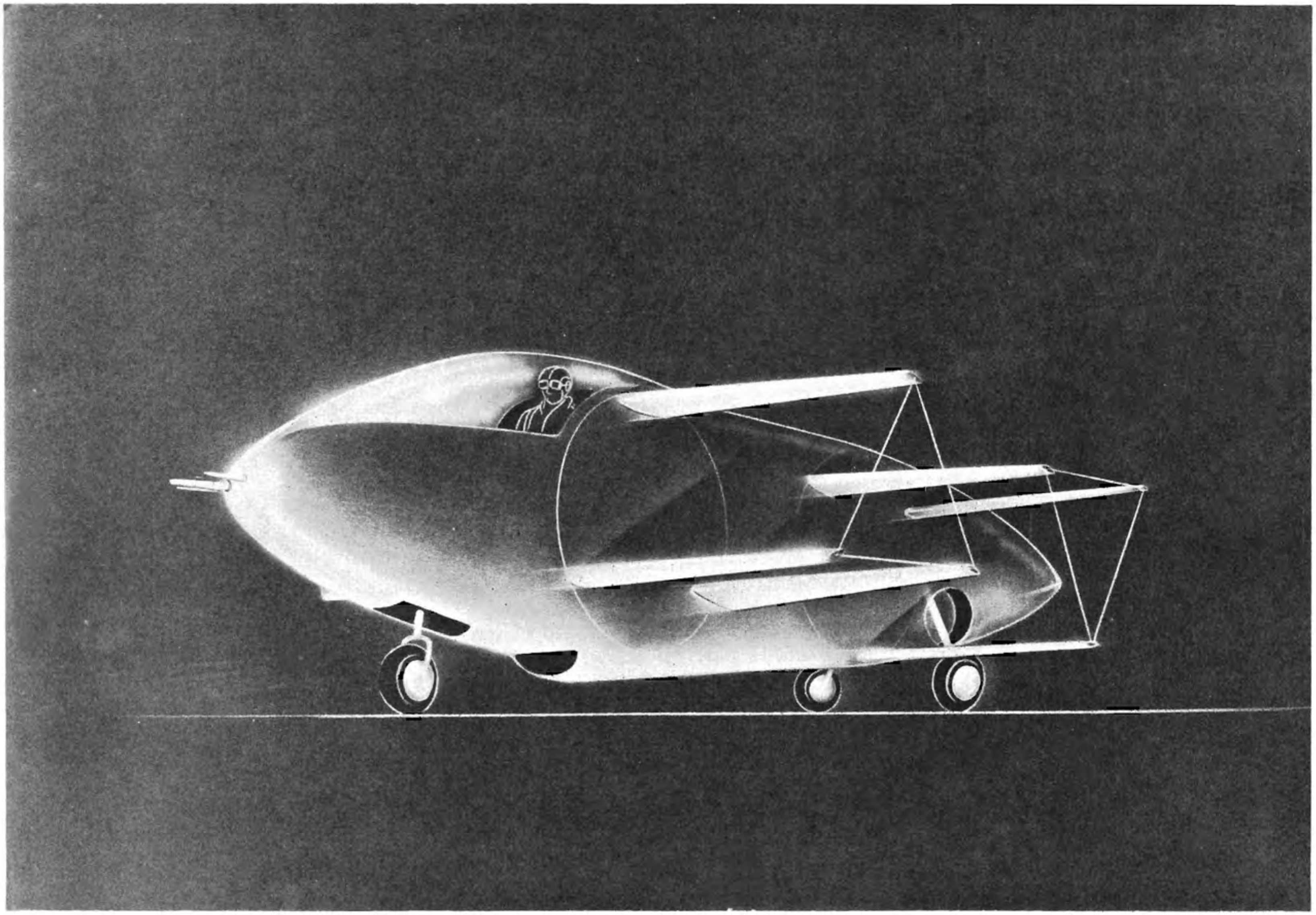


Figure 17 — Proposed cyclogyro designed by AMC

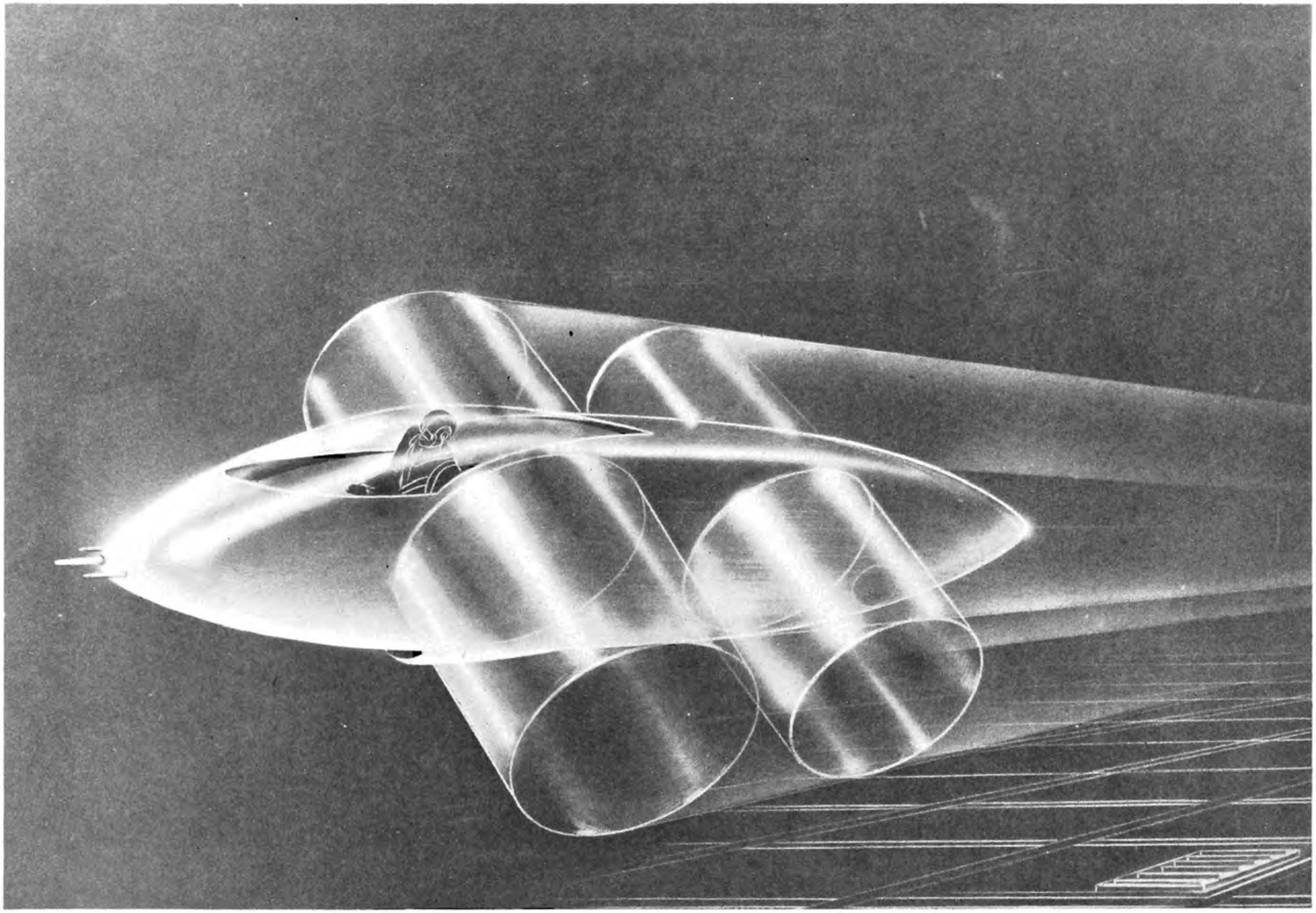
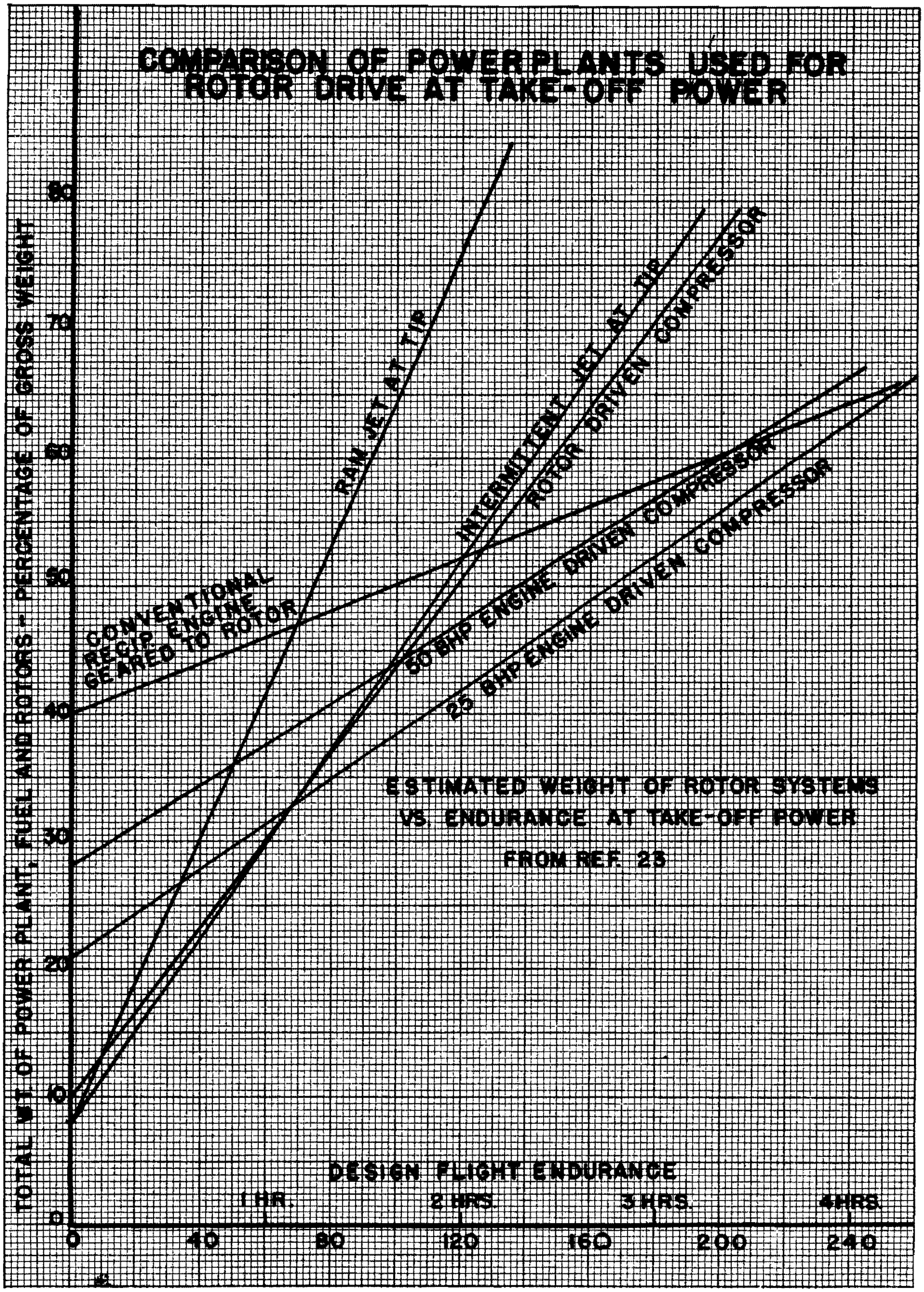
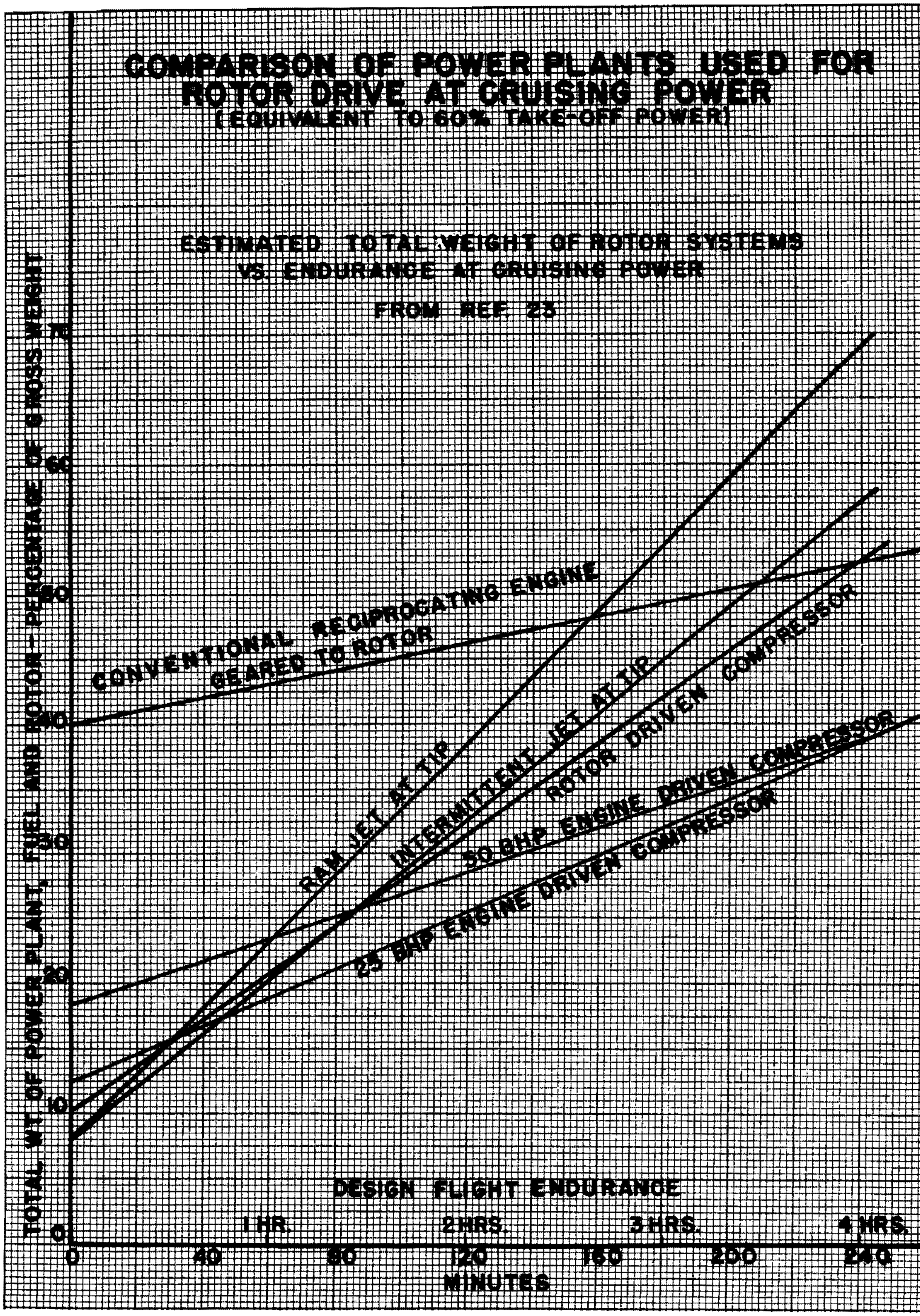


Figure 18 — Artist's conception of cyclogyro in flight



COMPARISON OF POWERPLANTS USED FOR
ROTOR DRIVE AT TAKE-OFF POWER
TOTAL WT. OF POWER PLANT, FUEL AND ROTORS - PERCENTAGE OF GROSS WEIGHT
RAM JET AT TIP
INTERMITTENT JET AT TIP
ROTOR DRIVEN COMPRESSOR
CONVENTIONAL
RECIP. ENGINE
GEARED TO ROTOR
50 BHP ENGINE DRIVEN COMPRESSOR
25 BHP ENGINE DRIVEN COMPRESSOR
ESTIMATED WEIGHT OF ROTOR SYSTEMS
VS. ENDURANCE AT TAKE-OFF POWER
FROM REF. 23
DESIGN FLIGHT ENDURANCE
1 HR.
2 HRS.
3 HRS.
4HRS.
0
10
20
30
40
50
60
70
80
0
40
80
120
160
200
240

Figure 19



COMPARISON OF POWER PLANTS USED FOR ROTOR DRIVE AT CRUISING POWER
(EQUIVALENT TO 60% TAKE-OFF POWER)
ESTIMATED TOTAL WEIGHT OF ROTOR SYSTEMS VS. ENDURANCE AT CRUISING POWER
FROM REF. 23
CONVENTIONAL RECIPROCATING ENGINE GEARED TO ROTOR
RAM JET AT TIP
INTERMITTENT JET AT TIP
ROTOR DRIVEN COMPRESSOR
50 BHP ENGINE DRIVEN COMPRESSOR
25 BHP ENGINE DRIVEN COMPRESSOR
TOTAL WT. OF POWER PLANT, FUEL AND ROTOR – PERCENTAGE OF GROSS WEIGHT
0
10
20
30
40
50
60
70
DESIGN FLIGHT ENDURANCE
1 HR.
2 HRS.
3 HRS.
4 HRS.
0
40
80
120
160
200
240
MINUTES

Figure 20

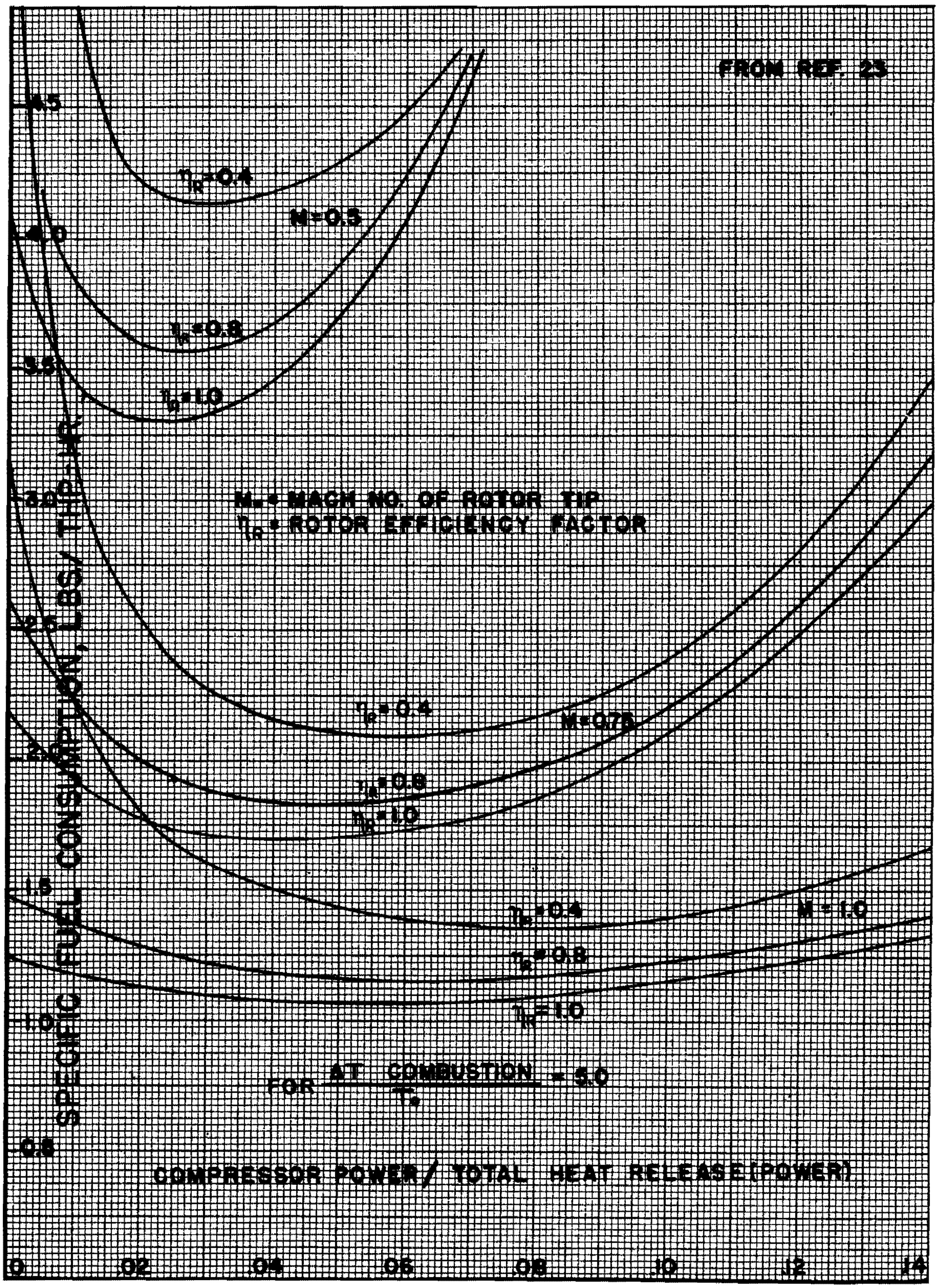


Figure 21 — Specific Fuel Consumption for a Compressor-Driven Rotor as a Function of Compression, for Several Tip Mach Numbers

PART III

AIRCRAFT MATERIALS AND STRUCTURES

By

N. M. NEWMARK, CONSULTANT

PART III

AIRCRAFT MATERIALS AND STRUCTURES

1 NOVEMBER 1945

SUMMARY

To meet the requirements of transonic and supersonic flight without increase in relative structural weight of military aircraft will require improvement in properties of materials, discovery of new alloys, and development of methods of fabrication and of production, as well as development of methods of design, construction, and analysis adapted to take advantage of the materials' properties. In all of these fields research is necessary; research directed toward the discovery of funadmental relations and concepts, as well as research directed toward the refinement of production processes. Fundamental research as distinct from applied research needs support more than ever because of a relative lack of support in the past.

This report summarizes very briefly the present state of knowledge and proposes problems for investigation relating to structural materials, sandwich construction, fastenings, general structural problems in structural mechanics and methods of analysis, determination of external loads, and the effect of the external loading on structures composed of various materials as a function of the properties of the material.

INTRODUCTION

The structure of an airplane is designed by taking into account the external loads and service conditions combined with consideration of the allowable stresses or deformations of the materials composing the structure. It is clear that the process of structural design cannot be considered separately from the properties of the material. Although under present conditions there is not a great deal of difference in weight or quality of performance of structures designed for the most efficient utilization of their material by first-rate designers, whether the material be aluminum, magnesium, wood, or steel; for certain special applications one material may offer considerable advantage over another. There is a reasonable prospect of considerable savings in weight or of improvement in other characteristics if materials stronger, stiffer or lighter than those now available can be developed. However, it is generally such considerations as production techniques, ease of fabrication, and service and maintenance characteristics that make some one material much more suitable than others for particular applications.

It is interesting to note that in spite of improvements in materials in the past 25 years, the ratio of weight of structure plus landing gear to gross weight of military aircraft, although fluctuating slightly, has remained substantially constant in the range from about 30 to 35 per cent, practically independent of size or type of airplane. It should not be inferred that this represents lack of progress. It has required considerable improvement in design and fabrication techniques and in materials to keep the weight of the airplane structure from increasing in the face of the more severe requirements imposed on the design.

However, the requirements of the future will be even more severe. To maintain contours in transonic and supersonic flight, and to prevent flutter, may require sacrifice of weight for the sake of stiffness. Certainly it is necessary that development of new materials with improved properties, and development of methods of design and production using the improved materials, must proceed as rapidly in the future as in the past, if not at a more accelerated rate.

MATERIALS

From the materials standpoint, research is necessary in production, forming, and treating processes as well as in the development of materials having the most favorable properties for specific structural uses.

Based on experience in the past decade of the major airplane fabricators, it appears likely that the predominant material for use in structural components will continue to be aluminum, although experience with the use of magnesium is constantly increasing. Development of a magnesium alloy with a compressive yield strength of over 40,000 lb/sq in., might change the picture materially if fabrication procedures for handling the material were also developed. In spite of the experimental work that has so far been done with laminated plastics, unless new developments in the future show a considerable improvement, especially in ductility, the use of such materials for major structural components cannot be important. For particular applications, especially in landing gear and other parts where extremely high strength is necessary, steel will continue to be the most important material.

Notable developments in aluminum alloys in recent years are the production of high strengths by artificial aging, and the increased resistance to corrosion and stress corrosion cracking by alloy coatings or "cladding" on the sheet material. The newest alloys, of the aluminum, zinc, and magnesium type, 75S-T and R303, have yield strengths of well over 60,000 lb/sq in., and tensile ultimate strengths as high as 72,000 lb/sq in., for sheet, going up to almost 90,000 lb/sq in., for extrusions. The values for sheet are exceeded only by the cold-worked and age-hardened alloy, 24S-T86, which has a considerably lower ductility.

It has long been a maxim that high ductility and a relatively large difference between yield strength and ultimate strength are necessary for metals to be used in structures. However, recent experience with cold-worked and age-hardened aluminum alloys has indicated that, except for the problems of fabrication and forming involved, the numerical values of ultimate elongation can be cut down materially without impairing the performance of a well-designed structure.

With the new developments in materials the problem of notch sensitivity becomes of increasing importance. Although there seems to be at least a rough correlation between notch sensitivity, or sensitivity to stress concentrations, for static loads compared with impact loads, there seems to be no correlation at all with resistance to repeated applications of load, or to so-called fatigue. The ordinary measure of ductility, permanent elongation in a standard tension test coupon, depends on the gage length over which the measurement is made, and may have no relation at all to the sensitivity of the material to a stress concentration of the type encountered in a structure, either under static loads or under repeated loads. The factors affecting the failure of a material are still not clearly understood. Much fundamental work remains to be done in developing a better measure of the suitability of a material for a given application.

Some further increase in strength of aluminum alloys may come from refinement of techniques and methods already established, but no major increase seems likely without some radical change in alloying elements or processes of treatment. Increase of strength of magnesium alloys (particularly compressive yield strength) would have to be accomplished by means that would not involve too serious a decrease in ductility. The problems of corrosion resistance and of stress corrosion are important also.

It is important to encourage fundamental research into the factors affecting the physical and chemical properties of alloys of the light metals. Such research may eventually point the way to alloys far superior to those now available. Systematic empirical search for better combinations of alloying elements should also be encouraged. The search certainly should not be limited to alloys of magnesium and aluminum, but because of the relative abundance of the former, it seems desirable to concentrate efforts on magnesium to a much greater extent than in the past. Developments of methods of producing aluminum from more plentiful ores than bauxite would be an essential if future military demands are likely to require more aluminum than can be produced from known deposits.

The almost fantastic possibilities of beryllium clamor for investigation. Although beryllium is as light as magnesium, it has a modulus of elasticity greater than that of steel; but, unfortunately, the poor ductility of beryllium makes it almost unworkable. A most important drawback to its use is its scarcity.

A study prepared by the Metals Conservation and Substitution Group, and reported to the War Production Board (Report No. 63, dated 23 September 1942) indicates great difficulties with beryllium alloys on account of low elongation and workability. Alloys with aluminum containing about 30% beryllium have fairly high strength and a modulus in the neighborhood of 20,000,000 lb/sq in., approximately twice that of aluminum. Even these alloys are not attractive in comparison with other light metals. Further investigation may lead to lightweight high-modulus alloys having strengths, ductility, and soundness worth considering for aircraft structures. However, if large quantities of beryllium are involved, an adequate source of supply must be discovered.

Future possibilities in light metal alloys may also include production of materials by atomic disintegration or synthesis.

With regard to heavier metals, alloy steels have considerable importance, but generally only for restricted uses in certain parts of the structure. Nevertheless, much more work is desirable on development of steels having more favorable properties for these applications, particularly insofar as control of properties, especially hardenability, is concerned.

The current trend is away from the use of wood as a major structural element. Such considerations as variability in properties, and uncertainty as to the amount and quality of the supply, seem to rule out wood as an important structural material for military aircraft. However, there are important possibilities of such uses as core material in sandwich construction that make it desirable to consider the possible use of both hard woods and light woods such as balsa in combination with metals or plastics.

Regarding the field of plastics, the most favorable materials for structural applications appear to be the laminated plastics, and particularly glass fiber laminates. In general these materials seem attractive only in sandwich construction and are discussed in the next section. The field of synthetic plastics for lightweight core materials in sandwich construction needs further investigation.

As an indication of comparative strengths, weights, and other properties of some of the strongest materials, typical properties are given in Table I. Although the concept of specific strength, or strength-weight ratio is an important one, comparative figures for different materials are likely to be misleading, and are consequently not given in the table. For actual use the material with the most favorable properties among those cited, except for special applications, is the aluminum alloy 75S-T.

In general, thin sheets of magnesium are to be avoided because of the tendency of this material to become embrittled due to work-hardening. However, where magnesium can be used in adequate thicknesses it offers interesting possibilities in view of its light weight.

It is almost impossible to consider materials without paying attention to production processes. In the field of metals, especially, the ways in which the metals can be furnished to the manufacturer have an important bearing on the use of the material in an airplane. The extrusion of complicated aluminum and magnesium shapes has made both of these materials much more effective in use. Future possibilities that may be of utmost importance and that should be investigated regard the extrusion of all-metal sandwiches for skins and shear webs, the casting of major structural elements, such as wings or tail surfaces of magnesium, and possibly the upsetting of edges of plates or ends of members for greater efficiency of splices and connections.

TABLE I — TYPICAL PROPERTIES OF SELECTED MATERIALS

	MATERIAL	*WEIGHT* *lb/cu in.*	*MODULUS OF ELASTICITY* *106 lb/sq in.*	*TENSION* *Yield Strength* *lb/sq in.*	*TENSION* *Ultimate* *lb/sq in.*	*COMPRESSION* *Yield Strength* *lb/sq in.*	*FATIGUE STRENGTH** *lb/sq in.*	*PERMANENT ELONGATION IN* *2-in. gage length*
Aluminum Alloys	Alclad 14S-T Sheet	0.101	10.6	60,000	68,000	61,000	18,000	10
	24S-T Sheet	0.100	10.6	46,000	68,000	44,000	18,000	19
	24S-RT Sheet	0.100	10.6	57,000	73,000	55,000	—	13
	Alclad 24S-T86 Sheet	0.100	10.6	69,000	73,000	70,000	—	6
	75S-T Extrusions	0.101	10.4	80,000	88,000	80,000	22,500	10
Magnesium Alloys	FS-1h Sheet	0.064	6.5	33,000	43,000	26,000	14,000	11
	J-1h	0.065	6.5	34,000	47,000	27,000	14,000	9
	O-1 Extrusions	0.065	6.5	29,000	46,000	22,000	—	9
	O-1A Extrusions	0.065	6.5	32,000	49,000	27,000	—	7
Stainless Steel	17-7, Type 301, Full Hard, Cold-Rolled	0.287	26	170,000	200,000	165,000	80,000	13
	Stainless "w" type 322 Cold-Worked and Aged	0.287	26	210,000	220,000	200,000	—	3
Beryllium Alloys	Beryllium, 99.5%	0.068	42	17,000	17,000	—	—	1.0
	Be-30%, Al-70%	0.085	18	72,000	75,600	—	—	1.0
	Be-18%, Mg-6%, Al-76%	0.09	15	82,000	89,000	—	—	0.5
Fiberglass Laminates	Parallel Fiber Glass Cloth with Mr-1A Resin	0.071	6.0	105,000	105,000	26,100	—	0
	Crossed Fiber Glass Cloth, Average, Several Resins	0.063	2.2	47,400	47,400	45,000	—	0
	Conolon F-13, Parallel	0.059	4.7	120,000	120,000	54,000	—	0
	Conolon F-13, Crossed	0.058	2.6	48,000	48,000	35,000	—	0

* Based on complete reversal of stress; 500,000,000 cycles.

SANDWICH CONSTRUCTION

In addition to the development of improved materials, consideration must be given to better ways of using materials in aircraft structural designs. Some possibilities have already been mentioned, the most striking of which is the relatively new form of construction commonly called sandwich construction. There are a number of applications that already have been studied to some extent, but much more extensive research is needed.

The sandwich, as its name implies, consists of two skins with a core or separator. Double-skin or sandwich construction with metal cores or fillers made of the same materials as the skin may be made in various ways, although some types of construction offer little, if any, advantage over the sheet and stiffener combinations that are ordinarily used. Formed, stamped, or corrugated metal cores have been used, and in some cases, with a skin on one side only. Other promising all-metal applications are possible extrusions of ribbed sheets, made to be used with or without an inner skin, sheet-rolled integrally with a grid-like system of stiffeners, and possibly combinations of different metals in outer skin, core, and inner skin.

Other types of sandwiches are of equal importance to the all-metal construction. The use of metal sheets with a strong hardwood core has some promise for extremely heavy construction. Metal skin with a light filler has been used successfully with such fillers as balsa wood, foamed rubber, and foamed or cellular plastics. A most promising type of sandwich is one built up of laminations of fiberglas cloth, paper, or other materials, impregnated with a plastic filler and bonded to a lightweight core material. This application brings in entirely new construction and inspection problems, but seems to have many merits.

The Air Materiel Command has pioneered in applications of glass laminates to aircraft structures, having built and successfully demonstrated the merits of a fiberglas skin-balsa core combination (for the tail cone of a training plane) and more recently having constructed a wing of fiberglas skin with a cellular cellulose acetate core. A more extensive research program with consideration of production methods and inspection techniques, as well as of strength and stability, may point the way to major developments in aircraft structures. In evaluating the merits of laminated plastics, due consideration must be given to the effect of temperature changes, moisture absorption, and above all, to the effect of stress concentrations and dynamic loads which assume major importance because of the low ductility of most of the important materials.

Naturally, a thorough study of possible core materials is needed for the most effective development of sandwich construction. Of the materials that have been tried by various investigators to most promising appear to be balsa wood (either end-grain or side-grain), balsa impregnated with plastic filler, cellular cellulose acetate (especially with a small amount of chopped glass fibers), foamed rubber, and a honeycomb grid of glass cloth. Probably the most complete investigation to date of the relative merits of different materials has been carried out at Wright Field by the Air Materiel Command. However, there are many questions remaining to be answered, such as the optimum thickness of the sandwich, the best method of bonding the skin to the core, the best relative thickness of inner and outer skins, and the development of a less brittle foamed plastic core, in addition to the all-important problems of development of simple and dependable means of fabrication, and of simple methods of repair.

A recent application that appears to have interesting possibilities is the use of a light foamed plastic as a filler to support the upper and lower skin of control surfaces, and in trailing edge construction in general. With the usual ribs and formers omitted, the entire piece becomes a sandwich with an unusually large core.

The use of sandwich-type construction seems to offer many advantages, particularly in increased rigidity of skin for monocoque construction, in elimination of wrinkles in skin or shear webs, and in simplification of construction. Important problems that need to be solved concern not only the fastening of the skin to the core, but the fastening of the sandwich itself to spars and other members, and the best means of splicing at joints.

FASTENINGS

The whole subject of fastenings or of methods of joining the different parts of the structure is one that needs further study and development. So far the most effective and dependable way of making joints in the light metal alloys is by means of riveting. However, the development of rivets to keep pace with the strength of the metals to be joined has not progressed rapidly enough. Much more extensive work can be profitably spent on the problem of developing rivets and riveting and dimpling procedures for the joining of the most recent light metal alloys and for possible stronger future alloys.

However, other methods of fastening also need study. For removable joints, bolts and screws must be used. In other places, it may be necessary or desirable to use bolts because of stress requirements or as a construction expedient. The design of the bolts themselves, especially for bolts in tension, needs further study.

The proper design of fittings, brackets, and attachments for connection of major structural parts needs a great deal of study. More efficient ways of connecting the wing to the fuselage, or the tail surfaces to the fuselage can save considerable weight not only in the fittings but in the remainder of the structure which must distribute the loads coming from the members that are joined together. Another part of this same problem is the effect on a structure, made up of dissimilar materials joined together, of temperature changes, shock loads, and the corrosion produced by the different materials in contact. The relatively simple problem of the elastic action of parts of different modulus of elasticity can probably be solved with analytical methods that are available.

For many applications it is desirable or expedient to use brazing or soldering as a means of fastening. Recent developments of self-brazing aluminum alloys offer interesting possibilities, and further progress along this line should be encouraged.

The most serious competitor to riveting of joints is the process of welding: fusion welding (gas or arc), spot or seam welding, and for special applications, flash and pressure welding. For joining aluminum or magnesium alloys, fusion welding processes such as the heliarc process, the recently developed multiarc process, and others are available, but so far no procedure has been developed which deposits material that has as high a strength as the parent material that is welded. Also, the available processes inevitably produce changes in the metallic structure of the adjacent material and cause internal stresses near the weld. These conditions can be relieved to some extent by annealing and heat treatment after welding, but the process is not simple and becomes almost impossible for major structural assemblies. In spite of the difficulties, fusion welding has advantages for certain applications, and has been used successfully, both in aluminum and magnesium alloys for simple parts and structural elements. For this reason, further research is desirable.

Both aluminum and magnesium can be successfully joined by spot welding or by seam welding, and a great deal of experimental work has been done on these methods.

In general, the problem seems to be one of sufficient control of the quality of the work and of inspection afterwards. In plants where the procedures have been carefully watched with attention to every detail, spot welding of the light alloys has been very successful as well as convenient. However, there have been also serious difficulties especially where the proper techniques have not been well enough appreciated. Spot or seam welding of stainless steel has been much less of a difficulty; as a matter of fact, this is the most convenient and desirable way to join stainless steel. Both aluminum and magnesium alloys are inherently less suitable for spot welding because of their lower resistance and the necessity for cleaning the surfaces to eliminate surface resistance.

Recent developments of cements and adhesives for joining metals have some favorable characteristics. Such procedures as Cycleweld, Pliobond, Metlbond, and others develop a relatively high shearing resistance in parts that can be subjected to heat and pressure. In general, the adhesives have a relatively low tensile strength and are not suitable for joints which are subject to tension or peeling action. Several of the above-mentioned adhesives have been successfully used in joining stiffeners to sheet in secondary structural parts, and also in such parts as tail fins, and in some cases in control surfaces. Production difficulties are great except where the pressure required can be maintained easily. Problems of fatigue, corrosion resistance, and the means of making repairs have still to be studied adequately.

GENERAL STRUCTURAL PROBLEMS

There are many structural problems which have not been discussed herein, about which further information is necessary for the reduction of weight or the increase in strength of aircraft structural elements. Such problems involve the effectiveness of sheet and stiffener combinations, the action of stiffened shell structures such as are used in the fuselage of large cargo or bomber planes, shear lag, the action of cutouts and lightening or access holes, and the proper reinforcement around cutouts, effective end-fixity of continuous compression members such as the stiffened sheet of a wing, action of shear beams and especially curved shear beams, the proper design of flooring, and similar problems. These problems are in general being studied adequately by such agencies as the National Advisory Committee for Aeronautics, by the Air Materiel Command at Wright Field, and by the aircraft manufacturers themselves.

It is of utmost importance that there be a free exchange of information among the various groups working on these problems and that fundamental research in structural mechanics be encouraged as well as research in applications to particular design problems. The development of suitable methods of analysis of the action of an airplane structure can be hastened by support of research in methods of analysis of plates, shells, stiffened plates or shells, and general aircraft structural frameworks. Methods should be available for plastic action as well as for elastic behavior. The most promising procedures appear to be numerical procedures, step-by-step processes, and methods such as the general procedure recently developed of successive relaxation of constraints.

It is of particular importance to develop adequate methods for taking into account the action of shock leads or rapidly applied loads, either in flight or on landing.

EXTERNAL LOADING

Of fundamental importance in the design of an airplane structure is knowledge of the loading to which the structure may be subjected. This is essentially a statistical problem; it is important to know the relative frequency of loads of different intensities, and their effect on the structure. The problem of collecting information on actual load factors from gusts or from maneuvers is an important and difficult one. The National Advisory Committee for Aeronautics and the Air Materiel Command have already been studying the problem, and their programs of research should be supported and extended.

It is desirable to have measurements of accelerations in flight; however, this information should be supplemented by and correlated with measurements of actual strains in various members of airplanes. These studies should be extended also to take-off and landing loads, and to other influences that affect the structure.

Such things as blast from rockets, recoil of guns, effect of antiaircraft fire and gun fire, and other factors that affect the design of military aircraft need to be studied particularly. With regard to these problems, it is necessary to evaluate the relative merits of providing some measure of protection to the aircraft and the operating personnel or equipment, compared with the disdavantage in weight and loss of speed entailed by armor and defensive armament. This requires a sort of operational analysis. However, a mere study of existing data from recent combat records is not sufficient since conditions have already changed to such an extent that much of the data may be outmoded. Such studies must continually be revised in the light of new developments.

An important feature of load conditions is the time rate of loading and the effect of this rate on the material composing the structure. A start has been made on the latter problem by the National Defense Research Committee and the War Metallurgy Committee, but only a relatively small amount of progress has been achieved.

RELATION OF MATERIALS PROPERTIES TO LOAD CONDITIONS

To assist the designer in providing adequately for the loading conditions on the aircraft structure, a comprehensive survey of the properties of the materials available for use must be made. Such properties as are ordinarily determined as routine (including static strength in tension, compression and shear, and stress-strain curves for these simple loadings) must be supplemented by similar studies for biaxial loading and in some cases for three-dimensional states of stress, by determination of impact strength, notch sensitivity in static and in dynamic loading, and by determination of fatigue strength, or more specifically, by endurance life for such load intensity-frequency curves as are reasonable for the structural use intended. Statistical data on expected values of these properties, as well as some measure of variation, may be more reasonable and useful than a specification of minimum allowable values.

Although for military aircraft, life expectancy is a rather intangible quantity determined in large part by factors beyond the control and often beyond the knowledge of the designer, it is important that a more rational procedure than is at present available be developed for taking into account dynamic or repeated loads. Even in military aircraft there is some evidence for the belief that failures due to a relatively small number of repetitions of high loads may become important. Current trends toward higher working stresses with lower design-load factors, leading to a higher stress under routine conditions (coupled with use of materials having relatively lower fatigue strengths in proportion to yield or ultimate strengths, with the additional factor of greater notch sensitivity in fatigue) indicate difficulties may arise even in planes designed for a short service life unless these conditions are taken into account.

There are several phases of the whole problem to be considered. With respect to loads, determination of frequency, rate of increase, duration, and magnitude are important. In general, these can only be inferred from experimental evidence as embodied in records of accelerations and of strain in flight, landing, and taxiing. Consequently, it is necessary to develop methods of analysis for relating loads to accelerations and stresses in the structure. Otherwise the results of experience with one type or design of an airplane cannot be successfully extended to other designs.

With respect to the materials, it is necessary to determine endurance life, as distinct from endurance limit, for such rates of loading, frequency, and magnitude of stresses as are indicated by the determination of the loads. However, it is important that the emphasis be placed on the action of material in such a physical form as is used in a structure. Most fatigue failures (and, as a matter of fact, most static failures) occur in details, joints, and connections. Consequently, both for static loads and repeated loads, the so-called notch sensitivity of the material is an important element. Improvement of properties of structural elements can sometimes be achieved by such procedures as shot peening or other surface treatment.

Especially for rapidly applied loads, the shape of the structural element considered and its attachment to other parts of the structure may have a much more important bearing on the resistance of the structure than the intrinsic properties of the material determined from more or less routine acceptance tests.

In addition to the factors cited, which have a bearing on design of cargo planes and transports as well as on combat planes, there are combat service conditions which may have an altogether different bearing on the design. Such materials properties as resistance to blast and fragment damage enter into the picture for airplanes to serve combat requirements.

RECOMMENDATIONS

It is recommended that the Army Air Forces take such steps as are required to insure that an adequate program of research in aircraft materials and structures will be carried on. Some programs are already handled effectively by existing facilities, notably the Air Materiel Command, the National Advisory Committee for Aeronautics, various University laboratories, materials manufacturers, and the aircraft manufacturers themselves. In some cases it may be desirable to supplement the work already being done; in other cases, it may be necessary to institute new programs. In general, a greater concentration of effort is needed in the field of fundamental research as distinct from applied research.

Among the problems which must be considered of importance for the future development of military aircraft are the following:

MATERIALS

Development of new materials in general with improved properties.

Development of fabrication and production processes for the new materials.

Fundamental research into the factors affecting the physical and chemical properties of the light metal alloys.

Development of magnesium alloys with greater compressive strength.

Investigation of beryllium alloys.

Development of methods for better control of the properties of alloy steels.

Extrusion of skin-stiffener combinations or all-metal sandwiches for wing covering, shear webs, and flooring.

Casting of major structural parts for monocoque construction.

SANDWICH CONSTRUCTION

Study of the factors affecting the behavior of so-called sandwich construction.

Possible uses of integrally stiffened sheet.

Combinations of different materials in sandwiches.

Further study of fiberglas laminates as sandwich skin.

Production, inspection, and repair techniques with plastic sandwich construction.

Development of core materials for sandwich construction.

Development of a rational method of design of a sandwich element (spacing of skins, relative thickness, etc.).

JOINTS AND FASTENINGS

Development of high-strength rivets for the newer high-strength materials.

Dimpling procedures for relatively brittle materials.

Design of fittings and attachments.

Effect of temperature changes and shock loads on a structure composed of two or more different materials.

Further development of welding procedures for aluminum and magnesium alloys.

Elimination of locked-in stresses due to welding.

Development of adhesives for metal-to-metal, metal-to-wood, or metal-to-plastic connections having high resistance to peeling and to tension.

GENERAL STRUCTURAL PROBLEMS

Fundamental research in structural mechanics.

Methods of design of particular structural elements.

Development of methods of analysis for aircraft structures, particularly methods capable of taking into account plastic behavior of materials.

Development of methods of analysis for shock loads.

LOADING CONDITIONS

Study of actual loading conditions, including frequency of loads of various intensities, and time rate of loading.

Measurements of strains as well as accelerations in flight, and for other conditions.

Correlation of strains with loads.

RELATION OF MATERIALS PROPERTIES TO LOADING CONDITIONS

Fundamental research into the factors determining failure of a material under uniaxial and multiaxial loading.

Notch sensitivity of materials.

Endurance life for actural loading conditions.

Resistance of materials and structures to blast, fragments, sudden high temperatures, etc.