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DEPARTMENT OF COMMERCE

CIVIL AERONAUTICS ADMINISTRATION

WASHINGTON, D. C.

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FLIGHT ENGINEERING

REPORT NO. 13

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JUNE 13, 1944

ALTITUDE AND ITS EFFECT UPON AIRPLANE PERFORMANCE

FLIGHT ENGINEERING AND FACTORY INSPECTION DIVISION

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DEPARTMENT OF COMMERCE
CIVIL AERONAUTICS ADMINISTRATION
Washington

July 10, 1944

SAFETY REGULATION RELEASE NO. 164

SUBJECT: Flight Engineering Report No. 13, entitled, "Altitude and Its Effect Upon Airplane Performance"

PREPARED BY: Flight Engineering and Factory Inspection Division.

The brief and rather elementary discussion of the subject which is contained in this report was undertaken primarily to furnish information for inclusion in a publication addressed to private pilots. Upon its completion however it was concluded that considerable abridgment would be necessary to serve the originally intended purpose, but that the entire paper was of sufficient value in that it sheds considerable light upon a very confused subject and also in that the Tables contained herein represent the only general study which is known of the effect of wing and power loading upon the items of performance treated, to warrant its separate publication.

F. M. Lanter
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Director, Safety Regulation

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24B and 40

Rec'd 2-17-45

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PREFACE

The brief and rather elementary discussion of the subject which is contained in this report was undertaken primarily to furnish information for inclusion in a publication addressed to private pilots. Upon its completion however it was concluded that considerable abridgment would be necessary to serve the originally intended purpose, but that the entire paper was of sufficient value in that it sheds considerable light upon a very confused subject and also in that the Tables contained herein represent the only general study which is known of the effect of wing and power loading upon the items of performance treated, to warrant its separate publication. A part of the calculations involved in the preparation of the Tables was done by Mr. M. E. Gaydos and the report has been prepared by Omer Welling, Chief, Flight Analysis Section.

APPROVED BY: Charles F. [signature]

Chief, Flight Engineering and
Factory Inspection Division

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INTRODUCTION

Those who fly airplanes are necessarily concerned with altitude - the altitude of objects or points of terrain on the surface of the earth as well as the altitude of the airplane in order that proper clearance above the former of these may be maintained and also because the performance of the airplane is apparently affected by altitude, being generally reduced with increasing altitude. Altitude is, of course, a vertical dimension - usually that from sea level to the point involved - and the altitude of points or objects on the surface of the earth is ordinarily measured by means of surveying instruments. The use of these instruments to measure the altitude of an airplane in flight is obviously impracticable if not impossible and it has therefore, been necessary to resort to some other means for this purpose. The instrument almost universally so used is the altimeter but in order to understand what this instrument really measures, the relation of the measurement to altitude, and the nature of the effect of altitude upon the performance of an airplane it is necessary to devote some attention to the nature of the air in which the airplane operates.

MECHANICAL PROPERTIES OF THE ATMOSPHERE

The atmosphere surrounds and rests upon the surface of the earth. Each square foot of the earths surface supports approximately 2,100 pounds of atmosphere. The atmosphere is made up of air (which is a mixture of gases) water vapor, smoke and dust particles and other suspended material, but mechanically it may be considered to be air and the air a gas. All gases are compressible and the mechanical properties are such that the weight of a given volume of the gas is directly proportional to the pressure to which it is subjected and inversely proportional to the absolute temperature (degrees, Fahrenheit plus 459.4). Thus for example, a cubic foot of air at the normal sea level pressure of 2,118 pounds per square foot and a temperature of 59°F weighs 0.0765 pounds. If the temperature remains constant but the pressure is doubled, the weight of a cubic foot of air is doubled. If the pressure remains constant and the temperature is elevated to 120°F, the weight will be reduced to:

$$0.0765 \times \frac{59 + 459.4}{120 + 459.4} = 0.0685 \text{ pounds}$$

or 89.6% of the original weight.

The pressure exerted by the atmosphere upon the surface of the earth or upon objects immersed in the atmosphere is due to the weight of the air above the level at which the pressure is measured. If the weight of a cubic foot of air remained constant at any altitude as the weight of a cubic foot of water does remain substantially constant no matter what the depth below the surface, then the atmospheric pressure would decrease 0.0765 pounds per square foot for each foot of altitude gained and would become zero at an altitude of approximately 27,670 feet, indicating that altitude to be the extent of the atmosphere above sea level. As the preceding paragraph has indicated, however, the weight of a cubic foot of air does not remain constant with altitude it rather tends to decrease with increasing altitude due to the corresponding reduction of pressure.

Due, however, to the fact that the temperature ordinarily decreases with increasing altitude, the weight of a cubic foot of air tends to increase with increasing altitude. The effect of the decreasing pressure is much greater than that of the decreasing temperature however, and their combination produces a net decrease in weight with increasing altitude such that under normal conditions it is only half its sea level value at an altitude of 21,850 feet and the pressure at an altitude of 27,670 feet is not zero but rather still approximately 35% of the sea level value. Finally, the upward extent of the atmosphere is theroetically infinite and is known to be several hundred miles.

The foregoing discussion indicates that if the temperature and atmospheric pressure at one level may be measured and if the temperature at all intervening altitudes is known, then the pressure at any other altitude may be determined by calculating, as has been illustrated above, the weight of each intervening cubic foot of air and adding (if the new altitude is less than the original) or subtracting (if greater) the sum of these weights to or from the originally measured. pressure. That is, the pressures at all altitudes and the corresponding unit weights or densities are interrelated by the temperatures which exist at all altitudes, or, the pressure and corresponding density at any altitude are determined by the pressure and temperature at sea level and the temperatures at all intervening altitudes.

THE STANDARD ATMOSPHERE

It is a matter of common knowledge that both atmospheric pressure and temperature at sea level vary from time to time at a given station as well as from station to station at a given time. Thus for example, at almost any station in the United States the pressure may vary over a range of 70 or more pounds per square foot and the temperature over 100 or more degrees Fahrenheit during the course of a year. Also, the rate at which temperature changes with increasing altitude may vary from an increase of 1° or 2°F to a decrease of 5° or 6°F per thousand foot of altitude at a given altitude and further, may vary from altitude to altitude above a given station at a given time. Finally, it is as yet a practical impossibility to predict or forecast what these values of pressure and temperature will be more than a few hours in advance of the time of making the forecast and for this reason, in order practically to deal with the atmosphere for certain purposes, it is necessary to resort to an approximation which may be considered invariable.

The approximation by common agreement, almost universally used, is called, "The Standard Atmosphere" and it is based upon the following assumptions:

a. The atmospheric pressure at sea level is invariably 2,118 pounds per square foot or 14.7 pounds per square inch, or 29.92 inches of mercury.

b. The temperature at sea level is invariably 59°F.

c. The temperature invariably decreases uniformly form 59° at sea level to -67°F at an altitude of 35,300 feet, i.e. the temperature decreases 3.566°F per thousand foot of altitude from sea level to 35,300 feet and not at all above that level.

The numbers involved in these assumptions are approximate mean or representative values of a great many actual measurements of pressure and temperature at various points on and above the surface of the earth at various intervals of time and these measurements indicate that the probability of departure of a particular measurement from the "standard" value is a maximum at or near the surface of the earth and generally decreases with increasing altitude. As has been indicated earlier, these assumed values of a sea level pressure and temperature and of temperatures at all altitudes define a pressure and a density corresponding with each altitude and the standard atmosphere is ordinarily presented in the form of a table showing for each of suitable altitudes the corresponding pressure, temperature and density.

THE ALTIMETER

The altimeter is basically an instrument which measures atmospheric pressure. It does not measure altitude directly. In spite of this, however, the dial is graduated in feet of altitude and the instrument is so designed that when it is subjected to a given pressure, the pointer on the dial indicates the altitude at which that pressure occurs in the standard atmosphere. The reading of the instrument may or may not be the actual altitude depending upon how closely the actual conditions of pressure and temperature correspond with those assumed by the standard atmosphere. For example, the Civil Aeronautics Administration assumes, for the purpose of investigation of the power plant cooling characteristics, a very hot day such that the temperature from sea level to an altitude of 5,000 feet is 110°F. If, on such a day the atmospheric pressure at sea level is that assumed by the standard atmosphere, namely 2,118 pounds per square foot or 29.92 inches of mercury the pressure at an altitude of 5,000 feet will be such that the altimeter will read 4,500 feet. If, however, and this is quite possible, the barometer at sea level reads 30.42 inches of mercury, then at 5,000 feet the altimeter will read approximately 4,000 feet. It may thus be seen that the altimeter reading is only an approximation to the altitude and that the maximum error likely to be encountered is of the order 1,000 feet.

When the altimeter is used to provide clearance between an airplane in flight and points or objects upon the surface of the earth, allowance is ordinarily made for the possibility of error by providing a margin great enough to permit some clearance in spite of the error. When used to indicate vertical clearance between two or more airplanes in flight, the altimeter is ordinarily a more precise instrument because at the same altitude all mechanically correct instruments will show the same reading no matter what the relation of the reading to the actual altitude. Also, certain altimeters are provided with an adjustment such that if the atmospheric pressure at sea level directly below the airplane is known, the instrument may be set to that pressure and thus practically eliminate the error due to departure of the pressure from the "standard" value which has been pointed out in the preceding paragraph.

THE EFFECT OF ALTITUDE UPON AIRPLANE PERFORMANCE

In the light of the foregoing discussion, it is now possible to consider the performance of the airplane and the apparent effect of altitude upon this. Performance includes such items as take-off distance, rate of climb, stalling speed, endurance, ceiling, etc. but no attempt will here be made to deal comprehensively with all items of performance. Instead, take-off distance, rate of climb, and stalling speed are selected as being of greatest interest and sufficiently representative of all performance to illustrate generally the apparent effect of altitude upon performance and the following discussion of these is confined to consideration of an airplane equipped with an unsupercharged or "sea level" engine and a fixed pitch propeller.

a. Take-Off Ground Run

The purpose of the take-off ground run is to accelerate the airplane from a standing start (zero ground speed) to the minimum airspeed at which flight is possible or safe. The force producing the acceleration is the difference between the thrust exerted by the propeller and the resistance to moving offered by the airplane. The thrust results from the power drawn from the engine and is, at a given airspeed and density, very nearly proportional to the power. The resistance to motion is made up of the rolling resistance of the landing gear wheels and the air drag of the airplane. The minimum airspeed which must be attained is that at which, for the airplane attitude involved, the lift, equals the weight of the airplane. The distance, speed, and accelerating force are so interrelated that, for a given force, the distance is approximately proportional to the square of the speed, i.e., if the speed is increased 20%, the distance increases approximately 44%, etc.; or, for a given speed, the distance is inversely proportional to the force; i.e., if the force is doubled, the distance is halved.

The lift, and drag forces acting upon an airplane are at a given airplane attitude, directly proportional to the square of the airspeed and to the density or weight of a cubic foot of air; i.e., at a given density, if the speed is doubled the forces are increased to four times their original value or, at a given speed, if the density is halved the forces are halved. At the end of the take-off ground run the lift load on the airplane must equal its weight no matter what the combination of airspeed and density involved and for this reason, if at one time the density or weight of the air is less than at another, the above relation indicates that the speed must be greater in order that density times the square of airspeed, and therefore lift, remain the same. Under these conditions, as indicated by the preceding paragraph and

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due to the necessary increase in airspeed, the take-off distance will increase and, in general, tends to increase in proportion as density or weight of the air decreases. Also, since both lift and drag are proportional to the same product of density and the square of airspeed and further, since the lift must always equal the weight of the airplane, therefore the drag remains constant no matter what the density and necessary airspeed for take-off.

The power developed by an engine is proportional to the weight of the mixture of air and fuel burned in the cylinders in a unit of time. The volume of the mixture handled by the engine in a unit of time is approximately proportional to the RPM at which it operates, since the dimensions of the cylinder, that is, its bore and stroke are fixed. For this reason, if the engine operates at a given RPM, as the density or weight of the air entering the cylinders is decreased so is the power. As has been indicated by the previous discussion, the thrust developed during the take-off run is very nearly proportional to the power and therefore, the thrust is also approximately inversely proportional to the density or weight of a cubic foot of the air in which the airplane operates during take-off; that is, if the density is reduced to 80 percent of its original value, the thrust will also be reduced to approximately 80 percent of its original value. This means that the accelerating force during the take-off run, which is equal to the difference between the thrust and the resistance of the airplane to motion, will also be reduced by an even greater amount. If, for example, under the original conditions the resistance is 25 percent of the thrust, leaving 75 percent of the thrust as an accelerated force, then a reduction of the thrust to 80 percent of its original value will reduce the accelerating force to 55 percent of the original value at thrust. It has been stated above that the take-off distance is inversely proportional to the accelerating force. It follows therefore, that reduction in density, which reduces the accelerating force, necessarily increases the take-off distance and does so at a rate which is greater than proportional to the density. Combining the two effects of density upon the take-off distance which have been discussed above, it must be concluded that the take-off distance increases with reducing density at a rate which is somewhat greater than inversely proportional to the square of the density.

In order to indicate the magnitude of this effect Table I has been prepared. Since the relative magnitude of the effect depends also upon the particular combination of wing loading and power loading involved in a given airplane, the table considers various wing loadings and, for each, several take-off power loadings over the range likely to be encountered. The power loading is the weight of the airplane divided by the rated take-off power of the engine (or their sum if more than one engine) at sea level.

b. Rate of Climb

An airplane climbs because the thrust horsepower available from the engine is in excess of that required for level flight at the same airspeed. The power is by definition a force times a velocity. The thrust horsepower is therefore, thrust times the speed of flight and the power required for level flight is the drag of the airplane times the speed of flight. It has been seen in the discussion of the take-off distance that, so long as the lift force on the airplane remains equal to the weight, the drag will remain constant. In any straight and steady flight condition the lift load on the wing is approximately equal to the weight of the airplane. Since, as has been pointed out above, the drag is proportional to the product of density times the square of airspeed and power is equal to drag times airspeed it follows that power required is very nearly proportional to the cube of the airspeed times the density. The thrust horsepower available is equal to the brake horsepower of engine times the propulsive efficiency and generally tends to increase with increasing airspeed. Since the rate of climb results from the difference between thrust horsepower available and power required, it is usually convenient to plot these two quantities against airspeed as has been done in Figure 1. As has been pointed out in the discussion of the take-off distance, the brake horsepower and therefore, the thrust horsepower are approximately proportional to the density. This is indicated by several curves of thrust horsepower available for each of several densities in Figure 1. As the density decreases the speed required for level flight increases in order that the product of density times the square of velocity and therefore the lift remains constant. Since the velocity must increase so therefore, must the horsepower required for level flight. This is also indicated by several curves on Figure 1. The combination of these two effects of density upon horsepower required and thrust horsepower available is such that the difference between these two powers and therefore, the rate of climb reduces with decreasing density. The magnitude of this effect is illustrated by the quantity shown in Table II, which also considers wing and power loading.

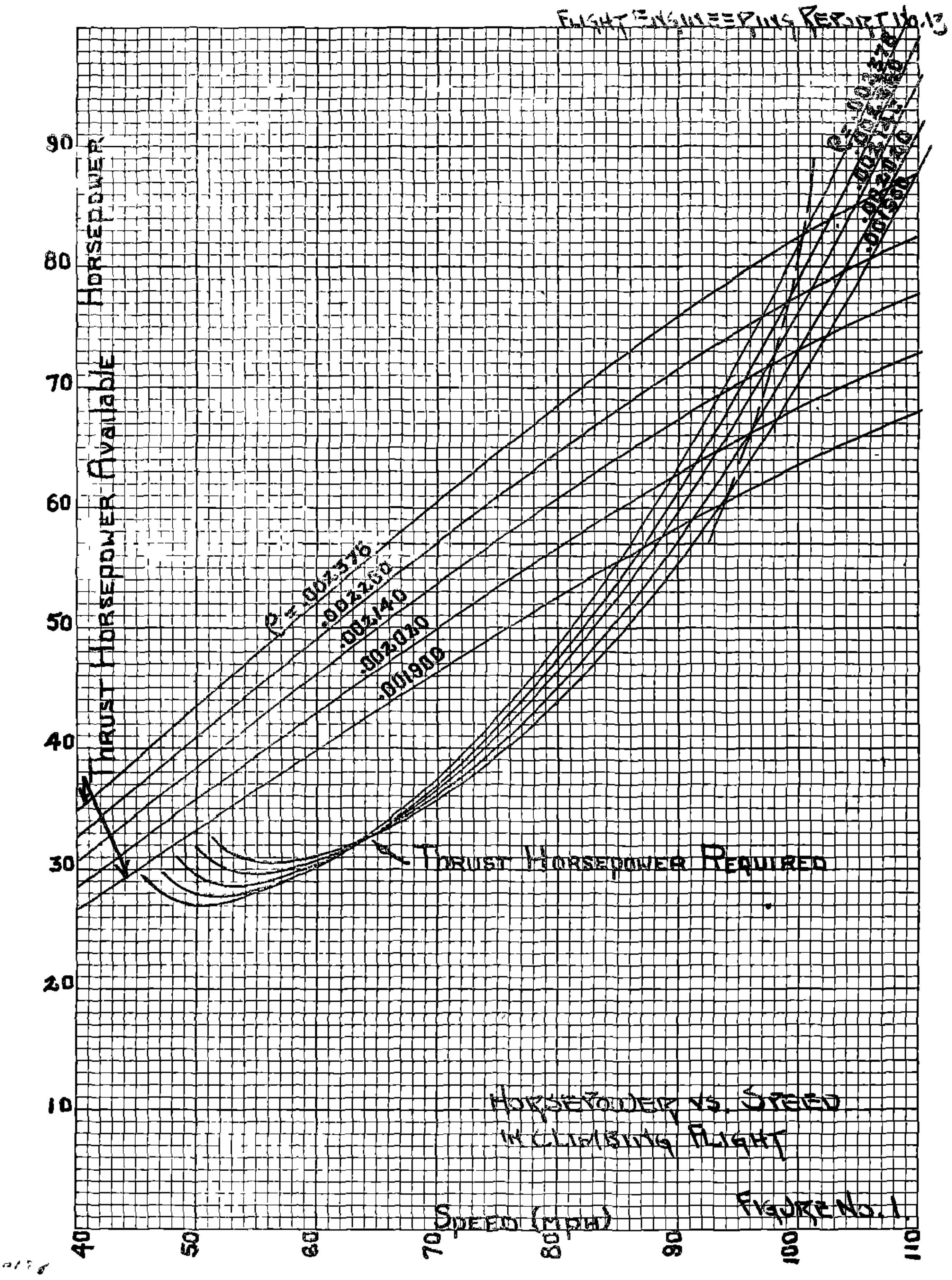
c. Stalling Speed

The stalling speed of an airplane is the lowest speed at which the lift load on the wing may be equal to the weight of the airplane. For the reason that the lift load is proportional to the product of density and the square of speed, it follows that if the density be reduced the speed must increase in order that this product remain unchanged. Table III has been prepared to indicate the magnitude of this variation. The stalling speeds shown in the Table are representative of those which would be obtained for a plain wing without any high lift devices. They will not necessarily correspond with the stalling speed for any particular airplane, but such comparisons as it has been possible to make between these speeds and the results of actual tests indicate that the values shown in the Table are representative of the results of tests for airplanes of good design. It should be borne in mind that the stalling speed here discussed is the actual speed of the airplane with respect to the air through which it moves. It is not the speed with respect to the ground nor yet the "speed" read from the airspeed indicator except under a very special set of circumstances almost never encountered in actual flight in an actual airplane. The airspeed indicator is however a pressure instrument and is so calibrated that when corrected for all known errors associated with its installation its reading is independent of density, i. e., for an airplane of a given wing loading and at a given outside air temperature the indicator will read the same at the stall at any altitude.

CONCLUDING REMARKS

It may be noted that each of the items of performance considered above has been related to the density or weight of a cubic foot of the air in which the airplane operates and further that as this weight of a cubic foot of air decreases the take-off distance is increased, the rate of climb is reduced, and the stalling speed is increased. Nothing has been said about altitude. The apparent effect of altitude upon these, or for that matter, any other items of performance, is due precisely to the fact, as has been indicated earlier, that the density of the air tends to decrease with increasing altitude. This reduction in density with increasing altitude is however, due not to the altitude but to the pressure and temperature which may exist at an altitude. In other words, altitude as such has no effect upon performance at all but pressure and temperature do effect it critically and there happens to be an approximate relation among altitude, pressure, and temperature such that altitude appears to affect performance directly. This illusion is further heightened by the fact that the atmospheric pressure gage (altimeter) in the airplane is calibrated to read feet of altitude. Since the altimeter measures nothing but pressure, it yields no information concerning the weight of the air and this can only be learned by considering also the temperature. Tables I, II, and III have therefore, been prepared in terms not of altitude or density, neither of which may be measured directly by means of any instrument installed in an airplane, but rather in terms of "altimeter reading" (i.e. pressure) and outside air temperature. With a sensitive altimeter, this reading should be taken with the instrument set at 29.92 inches of mercury.

The calculations involved in the preparation of the tables are outlined in Appendix I.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HORSEPOWER VS. SPEED IN CLIMBING FLIGHT
FIGURE No. 1
THRUST HORSEPOWER AVAILABLE
THRUST HORSEPOWER REQUIRED
SPEED (MPH)
ρ=.002376
.002260
.002140
.002030
.001900
40
50
60
70
80
90
100
110
10
20
30

TABLE I - TAKE-OFF GROUND RUN ($V_{T-O} = 1.1 V_S$)

Altimeter Reading	Wing Loading	5				10				15				20			25			30		40		50		60	
	Power Loading	15	20	25	30	10	15	20	25	5	10	15	20	5	10	15	5	10	15	5	10	5	10	5	10	5	10
	Temperature °F.																										
Sea Level	0	173	246	330	429	296	484	719	1 000	294	518	872	1 321	336	754	1 273	439	1,000	1 736	546	1,290	744	1,731	926	2,180	1,116	2,720
	20	189	270	366	476	322	530	787	1 110	254	566	957	1,470	366	826	1 430	477	1,088	1 911	594	1,370	811	1,905	1,010	2,475	1,260	2 880
	40	207	298	405	530	354	589	877	1,250	277	620	1,058	1 685	420	907	1 577	520	1 197	2,104	646	1,522	881	2,100	1,100	2 625	1,365	3,035
	60	226	326	446	589	384	642	965	1 390	301	677	1,160	1 808	442	994	1 742	576	1 310	2 340	702	1,669	960	2,300	1,210	2,895	1,447	3,485
	80	252	352	493	655	419	702	1,062	1 510	328	741	1 282	2,018	474	1,071	1 930	619	1,441	2,600	767	1,818	1,050	2,530	1,309	3 150	1 590	3 710
	100	270	392	541	720	457	768	1 170	1 710	354	802	1 400	2 215	512	1 182	2,120	666	1 564	2 845	830	1,978	1,140	2 765	1,418	3,490	1 720	4,090
1 000	0	193	276	374	489	327	540	904	1,441	256	570	967	1,487	370	839	1,449	498	1,100	1,930	600	1,190	815	1,980	1,080	2,435	1,190	2,900
	20	211	304	413	542	360	600	899	1 379	282	633	1,079	1 661	404	922	1 602	504	1 215	2,150	659	1,540	895	2,135	1,120	2 695	1,320	3,220
	40	232	342	459	605	391	656	990	1 426	306	690	1 187	1,853	442	1 013	1 782	576	1 310	2 430	720	1 760	960	2,340	1,220	2 965	1,405	3,550
	60	253	368	503	670	429	720	1,092	1 595	334	758	1 312	2,065	482	1,148	1 979	640	1 490	2 720	786	1 869	1 040	2,600	1,330	3,265	1,510	3,870
	80	276	401	554	740	466	790	1 210	1 770	361	825	1 442	2 300	521	1 217	2 176	700	1,610	3 040	854	2,045	1 150	2,860	1,450	3 600	1 735	4 250
	100	299	439	610	821	506	863	1,332	2,380	391	863	1 579	2 550	564	1 325	2 188	750	1 760	3,370	920	2 222	1,265	3 150	1 572	3,965	1,885	4,650
2 000	0	212	304	415	545	370	600	898	1 290	282	631	1,080	1 670	405	930	1,613	540	1 210	2 215	660	1,550						
	20	232	335	460	606	393	660	950	1,440	305	690	1 190	1 860	444	1 023	1,810	595	1 350	2,490	725	1 711						
	40	252	370	510	677	430	725	1,111	1 664	333	755	1 310	2 080	480	1 108	1 978	650	1 500	2 750	790	1 825						
	60	278	407	563	755	472	802	1 231	1 820	363	830	1 455	2 330	528	1 230	2,208	700	1,650	3,080	880	2,085						
	80	306	449	625	844	516	882	1,370	2,042	397	914	1 620	2 620	576	1 353	2,440	760	1 800	3,420	935	2,280						
	100	330	487	684	930	558	958	1,472	2,260	426	990	1 760	2 890	620	1,467	2 698	810	1,930	3 880	1,020	2,480						
3,000	0	234	340	465	615	396	666	1 010	1,460	309	700	1 210	1 900	446	1 027	1 825	600	1 365	2 510	784	1 711	985	2 375	1,238	3,082	1,465	3,610
	20	258	378	520	695	432	742	1 140	1,660	339	771	1 350	2 150	491	1 140	2,050	650	1,510	2 820	796	1 880	1,080	2,605	1,366	3 335	1,620	4,015
	40	285	414	575	771	481	820	1,260	1,865	369	847	1,490	2 380	528	1,251	2 260	710	1,685	3,160	867	2,090	1,185	2,390	1,482	3,670	1,790	4,430
	60	310	455	635	860	523	896	1 390	2 080	400	926	1,640	2 675	581	1 370	2,490	765	1 850	3,400	946	2,305	1,285	3,190	1,630	4,055	1,965	4,880
	80	338	499	700	953	569	980	1 538	2 340	435	1,010	1 808	2 982	646	1,533	2 830	830	2,030	3 970	1,025	2 520	1,400	3,485	1 760	4,425	2,130	
	100	368	544	772	1,063	615	1 071	1,695	2,680	470	1,100	1 990	3 320	682	1 635	3 060	900	2 185	4,450	1 109	2 740	1,515	3 800	1,908	4 850	2,320	
4,000	0	266	390	538	722	452	765	1 180	1 739	349	765	1,400	2 845	505	1,177	2 118	665	1 530	2 900	800	1,945						
	20	294	431	599	797	497	849	1 320	1 965	387	880	1 560	2 575	553	1 302	2,400	720	1,610	3 220	885	2,150						
	40	322	477	666	890	545	939	1,471	2 225	417	968	1 730	2 860	603	1,431	2,642	795	1,910	3,600	965	2 325						
	60	351	521	735	1 010	591	1 027	1,623	2 490	452	1 070	1 902	3 180	652	1 560	2 900	870	2,105	4 035	1,050	2,610						
	80	386	575	815	1 140	647	1 135	1 810	2 925	492	1 160	2 120	3 600	711	1 721	3 260	930	2 300	4 510	1,145	2,970						
	100	419	630	903	1,268	706	1 250	2 020	3 210	537	1 262	2 340	4 040	772	1 885	3 620	1,000	2,470	5 125	1 240	3,125						
5,000	0	293	430	601	811	495	848	1 320	1,980	380	876	1 550	2 520	550	1,292	2 355	730	1,720	3 300	897	2,180	1,220	3,080	1 545	3,910	1,820	4,680
	20	324	477	670	915	546	943	1,480	2 250	417	932	1,735	2 860	604	1,432	2 642	800	1 910	3,690	990	2,425	1,350	3,410	1 700	4,360	2,030	5,190
	40	354	525	747	1,022	595	1,040	1,646	2 558	454	1,060	1 997	3 200	654	1 562	2 930	875	2 090	4,040	1,070	2,660	1,470	3,750	1,850	4 900	2,272	5,750
	60	387	581	834	1,151	650	1,142	1 843	2 920	491	1,160	2 110	3,605	711	1,725	3,260	985	2,300	4,500	1,161	2,980	1,590	4,150	2,000	5,300	2,405	6,300
	80	421	639	920	1,283	710	1 252	2,050	3 290	536	1 270	2 350	4,090	773	1 885	3 620	1,025	2 510	5 000	1 270	3 220	1 720	4,590	2,182	5,870	2,610	6,940
	100	456	694	1,010	1,420	768	1,363	2 260	3 682	576	1,380	2 565	4 550	835	2 050	3 985	1,102	2 750	5,600	1 370	3,520	1,870	5,080	2,375	6,450	2,850	7,690
6,000	0	326	484	681	930	550	950	1,502	2 280	421	989	1 833	2 880	610	1 450	2,680	810	1 930	3 725	1,000	2,430						
	20	359	539	758	1 048	600	1 052	1,682	2 590	458	1,073	1 953	3 310	663	1 590	2 995	900	2 180	4 130	1,130	2 840						
	40	397	595	850	1 183	647	1 173	1 900	2 990	502	1,189	2 184	3 760	730	1 770	3 385	990	2 415	4,605	1,200	3,070						
	60	431	655	944	1,327	727	1 286	2 100	3,400	546	1,308	2 380	4 250	795	1 948	3 760	1,075	2,680	5 225	1 310	3,370						
	80	471	722	1 048	1,500	787	1,415	2 360	3 860	590	1,420	2 675	4 790	857	2 120	4 160	1 135	2 905	5 830	1,425	3 710						
	100	513	789	1,122	1,680	859	1,542	2 620	4 430	641	1 559	2 982	5 450	929	2 325	4 650	1,220	3 140	6 600	1 525	4,080						
7,000	0	366	545	776	1,075	615	1,077	1 771	2 685	467	1,100	2 000	3,400	678	1 635	3 090	980	2,170	4,700	1 105	2 775	1,510	3,900	1 910	5,040	2 190	6,060
	20	407	609	872	1,228	684	1 210	1,940	3 080	535	1,250	2,360	4 105	749	1,830	3 524	1,080	2 500	4 650	1 174	3,255	1,700	4 530	2,165	5,840	2,550	6,960
	40	451	685	990	1,412	755	1 347	2 205	3 635	569	1 368	2 560	4 540	821	2,025	3,960	1,110	2 750	5 270	1 330	3,480	1,850	5,000	2,315	6,400	2,790	7,650
	60	490	772	1,092	1,580	815	1,470	2 442	4 100	613	1,486	2 820	5 140	890	2 220	4 440	1,190	3 000	5,900	1,442	3 820	2,000	5,470	2 540	7,050	3,050	8,440
	80	541	825	1,220	1 790	893	1,623	2 740	4 750	664	1 568	3,120	5 835	956	2,420	4 900	1,280	3 300	6,600	1,581	4,200	2 180	6,080	2,750	7,840	3,330	
	100	585	910	1,361	2 025	973	1 731	3 181	5 450	720	1,780	3 495	6 750	1 048	2 680	5,560	1,380	3 600	7 400	1,722	4,650	2,350	6 770	3,000	8,700	3,640	[illegible]

TABLE II - RATE OF CLIMB AT 1.5V_S

Altimeter Reading	Wing Loading	5				10				15				20			25			30		40		50		60	
	Power Loading	15	20	25	30	10	15	20	25	5	10	15	20	5	10	15	5	10	15	5	20	5	10	5	10	5	10
	Temperature °F.																										
Sea Level	0	1,245	838	593	430	1,925	1,108	703	465	4,265	1,825	1,013	607	4,190	1,755	940	4,130	1,695	880	4,080	1,643	3,990	1,553	3,910	1,475	3,850	1,410
	20	1,203	810	564	405	1,862	1,064	668	427	4,160	1,765	966	570	4,080	1,692	893	4,020	1,632	832	3,970	1,578	3,880	1,488	3,790	1,408	3,730	1,348
	40	1,165	775	538	384	1,805	1,022	635	398	4,055	1,705	923	534	3,975	1,630	848	3,910	1,570	785	3,860	1,513	3,770	1,422	3,680	1,342	3,615	1,270
	60	1,125	745	513	360	1,750	981	600	369	3,950	1,650	880	500	3,875	1,573	805	3,810	1,510	741	3,755	1,455	3,660	1,360	3,570	1,277	3,505	1,203
	80	1,088	715	488	338	1,697	943	567	342	3,850	1,594	840	466	3,770	1,515	760	3,710	1,450	696	3,650	1,393	3,550	1,297	3,465	1,215	3,395	1,138
	100	1,050	685	464	316	1,643	903	538	315	3,750	1,536	800	430	3,670	1,458	720	3,605	1,392	651	3,550	1,335	3,450	1,235	3,360	1,150	3,290	1,075
1,000	0	1,183	791	553	396	1,830	1,043	652	414	4,090	1,734	945	554	4,020	1,660	872	3,960	1,600	812	3,905	1,550	3,810	1,457	3,735	1,376	3,670	1,306
	20	1,143	762	526	372	1,773	1,002	617	365	3,990	1,673	902	518	3,910	1,600	826	3,850	1,538	764	3,795	1,485	3,700	1,390	3,615	1,308	3,550	1,238
	40	1,104	728	500	350	1,717	960	585	339	3,890	1,616	860	483	3,810	1,540	783	3,740	1,476	719	3,690	1,422	3,590	1,327	3,510	1,243	3,435	1,170
	60	1,067	698	475	327	1,660	920	551	324	3,790	1,560	818	448	3,705	1,482	740	3,640	1,418	674	3,585	1,363	3,490	1,265	3,400	1,180	3,325	1,105
	80	1,030	669	450	304	1,611	881	518	300	3,690	1,505	777	415	3,605	1,425	696	3,545	1,360	630	3,485	1,305	3,385	1,204	3,295	1,117	3,220	1,040
	100	994	640	425	283	1,558	842	488	273	3,590	1,450	737	380	3,510	1,369	654	3,445	1,303	588	3,385	1,245	3,285	1,143	3,195	1,053	3,120	978
2,000	0	1,122	743	514	362	1,760	981	602	373	3,920	1,640	881	502	3,845	1,566	805	3,785	1,505	745	3,730	1,455	3,630	1,358	3,555	1,275	3,485	1,205
	20	1,083	714	487	338	1,680	940	567	346	3,820	1,582	838	467	3,735	1,507	759	3,675	1,444	697	3,620	1,390	3,525	1,293	3,440	1,208	3,370	1,137
	40	1,045	682	461	316	1,628	900	535	319	3,720	1,525	796	431	3,635	1,447	717	3,570	1,383	652	3,515	1,328	3,420	1,230	3,330	1,144	3,255	1,070
	60	1,008	652	437	293	1,575	859	502	287	3,620	1,470	754	397	3,535	1,392	673	3,470	1,326	608	3,415	1,270	3,315	1,170	3,225	1,082	3,150	1,005
	80	973	624	413	272	1,525	820	470	259	3,525	1,418	714	364	3,440	1,335	631	3,380	1,268	563	3,320	1,214	3,215	1,110	3,125	1,020	3,050	942
	100	936	594	380	250	1,474	782	440	233	3,430	1,361	673	330	3,345	1,280	592	3,280	1,212	522	3,225	1,155	3,115	1,051	3,025	958	2,950	882
3,000	0	1,060	695	474	328	1,690	918	552	334	3,745	1,548	826	450	3,670	1,472	738	3,610	1,410	677	3,555	1,360	3,455	1,260	3,375	1,176	3,300	1,105
	20	1,023	666	448	304	1,594	876	517	301	3,650	1,490	773	415	3,570	1,415	693	3,500	1,350	628	3,450	1,297	3,350	1,195	3,260	1,110	3,190	1,036
	40	987	636	423	282	1,540	837	485	272	3,550	1,434	731	380	3,470	1,355	650	3,400	1,290	585	3,345	1,235	3,245	1,133	3,155	1,046	3,080	970
	60	950	606	400	259	1,488	798	453	245	3,455	1,380	691	346	3,370	1,300	608	3,300	1,234	541	3,250	1,178	3,145	1,073	3,055	985	2,975	915
	80	915	578	375	239	1,439	759	421	218	3,360	1,323	650	313	3,275	1,245	566	3,210	1,177	498	3,150	1,122	3,045	1,015	2,955	923	2,875	845
	100	880	550	350	217	1,390	723	392	195	3,270	1,278	611	280	3,180	1,190	527	3,120	1,122	455	3,060	1,066	2,950	960	2,855	863	2,780	786
4,000	0	1,000	647	435	293	1,558	854	500	318	3,570	1,456	751	397	3,495	1,378	672	3,430	1,314	608	3,385	1,265	3,275	1,162	3,195	1,076	3,120	1,003
	20	963	618	408	270	1,504	813	466	258	3,475	1,400	707	363	3,395	1,320	625	3,330	1,255	560	3,270	1,203	3,170	1,097	3,080	1,010	3,005	935
	40	927	588	383	248	1,450	774	436	230	3,380	1,343	667	328	3,295	1,264	584	3,220	1,198	517	3,170	1,149	3,070	1,036	2,980	948	2,900	870
	60	892	560	360	226	1,400	735	404	210	3,290	1,292	626	294	3,200	1,210	543	3,130	1,141	473	3,080	1,086	2,970	978	2,880	886	2,800	807
	80	857	532	338	205	1,352	700	372	176	3,195	1,241	587	261	3,110	1,155	502	3,045	1,085	432	2,990	1,030	2,875	921	2,785	827	2,705	747
	100	822	504	313	184	1,304	660	343	152	3,108	1,190	549	230	3,015	1,100	463	2,955	1,038	390	2,900	976	2,785	866	2,690	767	2,610	690
5,000	0	938	600	395	258	1,466	790	449	242	3,400	1,364	686	345	3,320	1,284	605	3,250	1,220	540	3,205	1,170	3,100	1,064	3,015	977	2,935	903
	20	902	570	370	236	1,414	750	416	216	3,305	1,306	643	310	3,220	1,227	559	3,150	1,160	493	3,100	1,108	2,995	1,000	2,905	912	2,825	834
	40	867	542	345	214	1,362	712	385	189	3,210	1,253	603	277	3,125	1,172	518	3,055	1,103	450	3,000	1,050	2,895	940	2,805	850	2,720	772
	60	833	514	322	193	1,313	673	354	162	3,125	1,202	562	243	3,035	1,119	476	2,965	1,048	406	2,910	995	2,800	883	2,710	790	2,625	708
	80	800	486	300	172	1,266	638	323	136	3,030	1,153	523	210	2,940	1,064	437	2,880	995	365	2,820	940	2,705	827	2,610	730	2,530	650
	100	764	460	275	151	1,220	602	293	110	2,945	1,105	485	178	2,850	1,012	400	2,790	943	325	2,735	888	2,620	774	2,520	681	2,440	595
6,000	0	876	552	355	224	1,375	726	400	208	3,230	1,272	620	293	3,145	1,190	538	3,080	1,124	472	3,030	1,075	2,940	966	2,835	880	2,760	803
	20	842	523	330	202	1,324	688	367	174	3,130	1,215	578	259	3,050	1,135	493	2,980	1,065	425	2,930	1,015	2,820	908	2,730	813	2,650	735
	40	807	495	306	180	1,273	648	335	142	1,040	1,164	538	224	2,955	1,080	452	2,885	1,008	383	2,830	957	2,720	843	2,630	752	2,545	672
	60	773	467	283	158	1,226	612	305	121	2,955	1,113	500	192	2,865	1,027	411	2,795	955	340	2,740	903	2,630	786	2,530	694	2,450	612
	80	743	440	262	139	1,180	576	274	95	2,870	1,066	460	160	2,780	975	372	2,710	903	300	2,655	850	2,535	733	2,440	635	2,360	553
	100	707	414	238	118	1,135	541	245	70	2,785	1,020	423	120	2,690	923	335	2,630	852	259	2,575	798	2,450	683	2,350	577	2,275	500
7,000	0	817	505	318	192	1,287	664	349	164	3,060	1,181	556	242	2,975	1,100	472	2,910	1,032	405	2,860	983	2,750	870	2,660	782	2,580	705
	20	782	475	292	168	1,238	623	318	135	2,965	1,125	515	208	2,880	1,045	428	2,810	973	360	2,760	922	2,650	806	2,555	727	2,470	638
	40	748	448	267	147	1,184	576	287	106	2,875	1,074	475	174	2,790	992	388	2,715	916	316	2,665	867	2,550	748	2,460	656	2,370	575
	60	715	422	246	127	1,140	550	258	80	2,795	1,026	437	142	2,700	939	347	2,630	865	275	2,580	813	2,460	693	2,365	600	2,280	525
	80	684	396	225	107	1,095	518	229	56	2,710	980	400	111	2,615	888	310	2,550	815	235	2,495	760	2,370	642	2,275	542	2,190	460
	100	651	370	202	85	1,052	482	198	31	2,625	935	362	79	2,530	837	274	2,470	765	195	2,415	711	2,290	590	2,185	487	2,110	405

TABLE III - STALLING SPEED

Altimeter Reading	Wing Loading / Temperature °F.	5	10	15	20	25	30	40	50	60
Sea Level	0	36.9	50.6	60.5	68 0	74 1	79.3	87.7	94.1	103.0
	20	37.6	51 6	61.6	69.3	75 5	80 8	89.4	96.0	105.0
	40	38.4	52.7	63 0	70 9	77 1	82 6	91.3	98.0	107 2
	60	39.1	53.7	64.2	72 1	78.6	84.1	93 1	99.9	109 2
	80	40 0	54.9	65 6	73 7	80.4	85 9	95 1	102.0	111.8
	100	40.7	55.9	66 8	75 0	81 8	87 5	96 8	104.0	113.8
1,000	0	37.6	51.5	61.6	69.2	75.5	80 8	89 4	95.9	105.0
	20	38.4	52.6	63.0	70 7	77.1	82.6	91.4	98.0	107.2
	40	39.2	53.7	64 2	72 2	78.7	84 2	93.1	100.0	109.4
	60	40.0	54.7	65 5	73 6	80.2	85.9	95.0	102.0	111.7
	80	40.7	55.8	66.7	75.0	81.8	87.5	96.9	103.9	113.8
	100	41.5	56.8	68.0	76.3	83.2	89.1	98.5	105.7	115.9
2,000	0	38.2	52.5	62.6	70 4	76.7	82.1	90.9	97.5	106.9
	20	39.0	53.6	64.0	71.8	78.4	83 8	92.8	99.5	109.0
	40	39.8	54.7	65.2	73.4	80.0	85.6	94.6	101.7	111.2
	60	40.6	55.9	66 6	74 9	81.6	87.3	96.6	103.8	113.7
	80	41.4	57 0	68.0	76.4	83 2	89 1	98 5	105.8	115.9
	100	42.2	58.0	69 1	77 8	84.7	90.7	100.2	107.8	118.0
3,000	0	38.9	53.3	63.8	71.6	78.1	83.6	92 5	99.2	108.7
	20	39.7	54.4	65 2	73.2	79 8	85.5	94.5	101.2	111.1
	40	40.6	55.6	66.5	74 7	81.5	87.[illegible]	96.5	103.4	113.4
	60	41.4	56.7	67.8	76 1	83.1	88.9	98.4	105.6	115.6
	80	42.2	57.8	69.1	77 5	84.6	90 5	100.1	107.3	117.8
	100	43.0	58.8	70.4	79 0	86 2	92.3	102.0	109 [illegible]	119.9
4,000	0	39.7	54 5	65.1	73.1	79.7	85.3	94.4	101.2	110 9
	20	40.6	55.7	66 5	74 7	81 4	87 2	96.5	103 4	113 3
	40	41.4	56 9	68 0	76.2	83 2	89.0	98 5	105 4	115.7
	60	42.2	57.9	69 2	77.7	84 7	90 6	100.2	107 5	117 9
	80	43.0	59 1	70.5	79.2	86.4	92.4	102.2	109.7	120.0
	100	43.8	60 1	71.9	80.6	88 0	94 0	104.1	111.5	122.2
5,000	0	40.4	55 5	66.2	74 5	81.1	86.8	96 0	103.1	112.9
	20	41.3	56.7	67 6	76.1	82 9	88.7	98 1	105.3	115.3
	40	42.2	57.9	69 1	77 7	84.6	90 5	100.1	107 8	117 9
	60	43.0	59.0	70.5	79.2	86.2	92.3	102.1	109.8	120.0
	80	43.8	60.1	71.8	80.7	88.0	94.1	104.1	112.0	122.3
	100	44.6	61.3	73.1	82.3	89 6	95.9	106.1	114.0	124.9
6,000	0	41.1	56.5	67 5	75.8	82.6	88.4	97.8	105.0	114.9
	20	42.0	57.7	69.0	77 4	84 5	90.3	100.0	107.2	117.4
	40	42.9	59.0	70.4	79.1	86.1	92.1	102.0	109.4	119.9
	60	43.7	60.0	71.8	80.5	87.8	93.9	104.0	111 7	122.0
	80	44.5	61.2	73.1	82.1	89.5	95.8	106.0	113 9	124.5
	100	45.4	62.4	74 5	83.7	91.2	97 5	108.0	116.0	126.9
7,000	0	42.0	57.5	68.8	77.3	84.2	90 1	99.6	107.0	117 0
	20	43.0	58.9	70.4	79.0	86.1	92.1	102.0	109.3	119.9
	40	43.9	60.0	71.8	80.6	87 8	94.0	104.0	111.7	122.0
	60	44.6	61.1	73.2	82 2	89.5	95.9	106.0	113.9	124.4
	80	45.5	62.3	74.5	83.7	91.2	97.6	108.0	116.0	126.8
	100	46.4	63.5	75.9	85.4	93.[illegible]	99.5	110.0	118.1	129.1

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APPENDIX I - CALCULATIONS

The calculations upon which Tables I, II, and III are based assume the airplanes equipped with sea level engines and fixed pitch propellers. They further assume that, for an engine operating at fixed throttle setting and RPM in the standard atmosphere, the variation of power with density is:

$$\frac{BHP}{BHP_o} = 1.132 \; \rho/\rho_o \; -.132 \qquad (1)$$

and, for departures of the temperature from the standard value for a given pressure in the standard atmosphere, the variation of power is:

$$\frac{BHP}{BHP_o} = \sqrt{\frac{T_s}{T}} \qquad (2)$$

An analysis of the 105 airplanes of Appendix IV to Flight Engineering Report No. 5 indicates the maximum lift coefficient to increase with wing loading at a rate approximated very closely by the following equation which has been used throughout.

$$C_{L_{max}} = 1.20 + .0152\, w \qquad (3)$$

Although the actual variation of RPM and propulsive efficiency at constant throttle setting and indicated airspeed with density will actually depend upon the selection of a particular propeller for a particular airplane engine combination, it has been assumed that these remain constant and further, that under these conditions, the power output of the engine is proportional to RPM.

The following have also been assumed as representative and constant values:

Coefficient of Rolling Friction, μ = 0.050

Minimum Parasite Drag Coefficient, C_{D_O} = C.02750

Atmospheric density has been calculated by means of the following expression:

$$\rho = \frac{.0412 \times \text{Pressure ("HG)}}{\text{Absolute Temperature (°F)}} \qquad (4)$$

The calculations and the remainder of the assumptions are outlined below for each of the items of performance considered.

TAKE-OFF GROUND RUN

The take-off ground run may be expressed:

$$S_{t-o} = \frac{V^2_{t-o}}{2\,a}$$

Where: V = the take-off speed in feet/sec.
a = an effective constant acceleration.

It is assumed that:

$$V_{t-o} = 1.1\ V_s = 1.1 \sqrt{\frac{2\ W}{\rho\ C_{L_{max}}}}$$

Now:

$$a = \frac{g}{W}\ (T - R)$$

and:

$$T = \frac{550\ BHP\ \eta}{V_a}$$

Also:

$$R = \mu W + \tfrac{1}{2}\ \rho\ V_a^2\ f, \text{ where } W = \text{ airplane weight;}$$

That is:

a is a function of velocity during the take-off ground run and the expression for R assumes the airplane to operate at zero lift throughout the run. The actual calculation of the variation of a with V_a for a great many cases indicates the actual value at $V_a = 0.70\ V_{t-o}$ is a very close approximation to the effective constant value involved in equation (5). It is therefore assumed that:

$$V_a = 0.70\ V_{t-o} = 0.77 \sqrt{\frac{2\ w}{\rho\ C_{L_{max}}}} = 1.09 \sqrt{\frac{w}{\rho\ C_{L_{max}}}}$$

Then:

$$T = \frac{505\ BHP\ \eta}{\sqrt{\frac{w}{\rho\ C_{L_{max}}}}} = \frac{505\ BHP\ \eta \sqrt{\rho C_{L_{max}}}}{\sqrt{\ }}$$

$$R = \mu W + \frac{1.188\ \rho\ w\ f}{2\ \ C_{L_{max}}} = \mu\ w\ S + \frac{.596\ w\ f}{C_{L_{max}}}$$

Where S = wing area in square feet.
Now: $f = C_{D_o}\ S = 0.0275\ S$

$$\therefore \quad R = \mu\ w\ S + \frac{.0163}{C_{L_{max}}}\ w\ \underline{S} = \frac{w\ S}{C_{L_{max}}}\ (\mu\, C_{L_{max}} + .0163),$$

and: $$a = g \left[\frac{505\ BHP\ \eta \sqrt{\rho C_{L_{max}}}}{\sqrt{\ }}\ \ \frac{w\ S}{C_{L_{max}}}\ (\mu\, C_{L_{max}} + .0163) \right]$$

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$$= g\left[\frac{505\,\eta\sqrt{\rho C_{L_{max}}}}{p\sqrt{w}} - \left(\mu + \frac{.0163}{C_{L_{max}}}\right)\right]$$

$$= g\left[\frac{505\,\eta\sqrt{\rho C_{L_{max}}}}{p\sqrt{w}} - \frac{\mu C_{L_{max}} + .0163}{C_{L_{max}}}\right]$$

$$= \frac{g}{p}\;\frac{505\,\eta\,\rho^{\frac{1}{2}}\,C_{L_{max}}^{3/2} - p\,\mu\,C_{L_{max}}\sqrt{w} - .0163\sqrt{w}\,p}{\sqrt{w}\;C_{L_{max}}}$$

Finally:

$$S_{t-o} = \frac{V_{t-o}^2}{2\,a} = \frac{1.21 \times 2\,w}{2\,a\,\rho\,C_{L_{max}}} = \frac{1.21\,w}{a\,\rho\,C_{L_{max}}}$$

$$= \frac{1.21\,w\,p\sqrt{w}\;C_{L_{max}}}{g\,\rho\,C_{L_{max}}\left[505\,\eta\,\rho^{\frac{1}{2}}\,C_{L_{max}}^{3/2} - p\,\mu\,C_{L_{max}}\sqrt{w} - .0163\,p\sqrt{w}\right]}$$

$$= \frac{1.21\,w\,p}{g\,\rho\left[505\,\eta\;C_{L_{max}}^{3/2}\;\sqrt{\rho/w} - p\left(\mu C_{L_{max}} + .0163\right)\right]} \qquad (5)$$

The solution of this equation contained in Table I is based upon the assumption that the propeller has been so selected that at 70% of the take-off speed the engine turns 90% of the rated RPM and the propulsive efficiency is 0.450. Then:

$$p = \frac{W}{BHP}$$

$$BHP = BHP_o \;\; (1.132\;\rho/\rho_o \; - .132)\sqrt{\frac{T_s}{T}}$$

BHP_o = Rated Take-Off Power

ρ_o = .002378 Slugs per Cubic Foot.

RATE OF CLIMB

The rate of climb of an airplane in feet per minute may be expressed:

$$C = \frac{33{,}000}{W}\left[BHP\,\eta - \frac{\rho\,f\,V^3}{1{,}100} - \frac{2\,W^2}{550\,\pi\,\rho\,e\,b^2\,V}\right]$$

Now:

$f = C_{D_o}\;S = 0.0275\;S$, and: $AR = \frac{b^2}{S}$, whence: $b^2 = ARS$

Also:

$w = \frac{W}{S}$, and $p = \frac{W}{BHP}$

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Making these substitutions:

$$C = 33{,}000 \left(\frac{\eta}{p} - \frac{\rho C_{D_o} V^3}{1{,}100\, w} - \frac{2\, w}{550 \pi \rho\, e\, AR V} \right)$$

For the purposes of Table II, it is assumed that the climbing airspeed is 150 percent of the indicated stalling speed; i.e.

$$V = 1.5 \sqrt{\frac{2\, w}{\rho\, C_{L_{max}}}} = 2.120 \sqrt{\frac{w}{\rho\, C_{L_{max}}}}$$

Then:

$$C = 33{,}000 \left(\frac{\eta}{p} - \frac{9.53 \rho\, C_{D_o}\, w^{3/2}}{1{,}100\, w\, \rho^{3/2}\, C_{L_{max}}^{3/2}} - \frac{2\, w\, \rho^{\frac{1}{2}}\, C_{L_{max}}^{\frac{1}{2}}}{1{,}165 \pi \rho\, e\, AR\, w^{\frac{1}{2}}} \right)$$

$$= 33{,}000 \left(\frac{\eta}{p} - \frac{.00866\, C_{D_o}\, w^{\frac{1}{2}}}{\rho^{\frac{1}{2}}\, C_{L_{max}}^{3/2}} - \frac{.000546\, w^{\frac{1}{2}}\, C_{L_{max}}^{\frac{1}{2}}}{\rho^{\frac{1}{2}}\, e\, AR} \right)$$

$$= 33{,}000 \frac{\eta}{p} - 286 \frac{C_{D_o}\, w^{\frac{1}{2}}}{\rho^{\frac{1}{2}}\, C_{L_{max}}^{3/2}} - 18 \frac{w^{\frac{1}{2}}\, C_{L_{max}}^{\frac{1}{2}}}{\rho^{\frac{1}{2}}\, e\, AR}$$

Calling:

$$\left.\begin{aligned} C_{D_o} &= 0.0275 \\ \eta &= 0.70 \\ e &= .75 \\ AR &= 8 \end{aligned}\right\} \quad \text{Representative Values}$$

Then:

$$C = \frac{23{,}000}{p} - \frac{7.845 \sqrt{\frac{w}{\rho}}}{C_{L_{max}}^{3/2}} - 3.00 \sqrt{\frac{w\, C_{L_{max}}}{\rho}} \qquad \text{———— (6)}$$

STALLING SPEED

$$V_s = 0.6818 \sqrt{\frac{2\, w}{\rho\, C_{L_{max}}}} = 0.965 \sqrt{\frac{w}{\rho\, C_{L_{max}}}} \qquad \text{———— (7)}$$