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An Investigation into the Suitability of Proposed Aircraft Design Memoranda Tests for Deck-landing Aircraft

Part I. Seafire IIc and Barracuda II

By

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with an Appendix by P. A. HUFTON, M.Sc.

Part II. Hellcat I and Avenger I

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Part I.

Seafire IIc and Barracuda II

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with an Appendix by P. A. HUFTON, M.Sc.

Summary—A series of requirements for deck-landing aircraft has been proposed and the suggested programme of tests has been carried out on two Naval aircraft. The results of these tests are given in this report, and their significance has been discussed in the light of the accepted deck-landing qualities of these two aircraft.

1. *Introduction*.—One of the major requirements for naval aircraft is that the manoeuvre of landing on the carrier's deck should make the least possible demands upon the skill and concentration of the pilot. It was therefore considered advisable to lay down some standard to which all future deck-landing aircraft should comply. In order that such a standard might be accepted as reasonable and sufficient, it was necessary to determine how present-day naval types compared with this proposed standard, in view of the accepted qualities of these aircraft for deck landing.

This proposed Aircraft Design Memoranda (A.D.M.) (published in January, 1944, and given in full in Appendix I) lays down a series of tests which are designed to ensure that the aircraft shall be under adequate control during the approach and shall not undergo violent changes in trim with engine power.

This note describes these tests as carried out on a Seafire IIc and a Barracuda II, during the summer of 1944, to determine to what extent these two aircraft conform to the proposed standard.

2. *Description of Aircraft and Instruments*.—The Seafire IIc (Fig. 1 (S)) is a single-seat, single-engine low-wing monoplane fighter aircraft developed from the Spitfire Vb. It is powered by a Merlin 46 engine driving a 10 ft. 3 in. diameter three-blade variable pitch propeller. This aircraft is fully equipped for deck operation (except for wing folding) and was taken as a typical high-performance fleet fighter. It was tested at a mean weight of 6,600 lb. with the C.G. undercarriage down at 0·366 aerodynamic mean chord (a.m.c.), the total flying time being about 12 hours.

The Barracuda II (Fig. 1 (B)) is a multi-seat single-engine monoplane torpedo-bomber-reconnaissance aircraft, designed specifically for carrier operation. It is powered by a Merlin 32 engine driving a 11 ft. 9 in. diameter four-blade variable pitch propeller. It was fully navalized

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and carried the usual array of external bomb racks and radio antennæ. The tests on this aircraft were made at a mean weight of 12,200 lb. with the C.G. undercarriage down at $0 \cdot 338$ a.m.c., and occupied about 10 hours flying time.

The relevant aerodynamic data for the two aircraft are given in Tables 5 and 6.

The instrumentation was practically the same for the two aircraft, the following instruments being fitted to each:—

(1) A.S.I. operating on the aircraft system.

(2) A.S.I. operating from a venturi pitot on a strut under the wing, and a suspended static head.

(3) Engine r.p.m. indicator.

(4) Engine boost pressure gauge.

(5) Pitch indicator (a standard Mk. I gyro horizon modified by the addition of a vernier scale).

(6) Remote reading desynn transmitters and indicators for ailerons, rudder and elevator.

(7) Rate-of-roll meter, with desynn indicator.

(8) Rate-of-yaw meter, with desynn indicator.

(9) Stick force transmitter, with desynn indicator.

(10) Rudder force transmitters with desynn indicators (one on each pedal).

The rate-of-roll and rate-of-yaw instruments consisted of electrically-driven spring-constrained gyros. The rotation to be measured caused precession of the gyro against the spring, the resulting deflection being measured by a micro-desynn transmitter and standard indicator. The stick force transmitter could be used to measure either aileron force or elevator force separately. This instrument, like the rudder force transmitters, measured the deflection of a spring, under the action of the pilot's effort, by means of a micro-desynn transmitter and standard indicator. Not all these instruments were in use at any one time.

The indicators were grouped together on a panel and photographed by means of a Bell & Howell Type A-4 clockwork driven ciné camera, using 35 mm. film, the camera being solenoid operated. In the Seafire, this equipment was carried in the radio compartment behind the pilot's seat, with the operating switches in the cockpit, near the pilot's left hand. In the Barracuda, the equipment was installed in the navigator's compartment and was operated by an observer.

Some difficulty was experienced in operating the suspended static from the Seafire. It had to be carried on the pilot's knees before use, and then paid out over the side of the cockpit when required. The upper end was anchored by means of a quick-release hook outside the cockpit in reach of the pilot, so that the instrument could be jettisoned after use. It was then lowered by a parachute stowed in a bag fastened to the outside of the rear part of the fuselage. Some inconsistencies in the readings of the A.S.I. operating from this static may be attributable to kinks or other defects in the rubber tubing between the head and the aircraft. No such difficulties were encountered in the case of the Barracuda.

3. *Scope of Tests, and Presentation of Results*.—The programme of tests on each aircraft was essentially that proposed in the draft A.D.M., fuller investigations of the various effects being made where necessary. Usually, tests were carried out over a range of speeds and engine powers, covering the normal range of approach conditions for these aircraft, instead of at the single condition suggested in section 4.2 of the A.D.M. This A.D.M. landing condition involves an engine setting which gives a stalling speed of $0 \cdot 87$ times the engine-off stalling speed, and an indicated airspeed equal to the engine-off stalling speed. The condition is referred to repeatedly throughout this report as the " standard " condition. The A.D.M. states that in this " standard " condition the angle of glide should at least be 5 deg.

The results are given in the same order as that in which the tests are described in the proposed A.D.M. (*see* Appendix I), and the same section numbers and headings are employed. Section 4, below, describes the tests, and, where applicable, gives quantitative results for comparison with the provisional figures of the A.D.M. The complete results are given in the figures at the end of this report. Where tests fuller than those called for in the A.D.M. were carried out, these are described under the corresponding A.D.M. headings.

4. *Results of A.D.M. Tests.*—Analysis of the A.D.M. tests necessitated a knowledge of the position error of the standard A.S.I. systems. The position error curves given in Fig. 5 (S) for the Seafire and Fig. 5 (B) for the Barracuda were obtained with the aircraft in the landing condition, using the suspended static head and venturi pitot method.

4.1. *Effect of Engine on Stalling Speed.*—Stalling speeds were measured at a range of engine powers, with the aircraft in the landing condition (*i.e.* with wheels and flaps down, hood open, propeller speed control set for maximum r.p.m.) using the suspended static head and venturi pitot. Figs. 2 (S) and 2 (B) show, for the Seafire and Barracuda respectively, (*a*) the variation in stalling speed and $C_{L\max}$ with engine power as represented by boost pressures, and (*b*) the variation in stalling speed and $C_{L\max}$ with thrust coefficient T_c. Values of T_c for each experimental point were obtained from the charts published by Biermann and Hartman[1] (1938) and by MacDougall[2] (1945). The rather large scatter of some of the points on the curves for the Seafire may be due to some defect in the suspended static system, as suggested in section 2.

The stalling speeds, engine off, with the aircraft in the landing condition, were, for the Seafire 68 knots (78 m.p.h.), falling to 61 knots (70 m.p.h.) when the engine was opened up to zero boost; the corresponding figures for the Barracuda being 64·5 knots (74 m.p.h.) and 54 knots (62 m.p.h.). Owing to the low power/drag ratio of the Barracuda, a considerable amount of engine is used in the approach to land, and in covering the range used, the propeller cannot be kept in fully fine pitch at the higher powers. The Seafire propeller remained in fully fine pitch during this and the remaining tests, since the r.p.m. never approached the constant speeding figure of 3,000 r.p.m.

4.2. *Effect on Engine on Gliding Angle.*—The angle of glide was measured over a range of speeds and engine powers, covering the normal approach conditions for these aircraft, instead of in the proposed " standard " condition described in section 3. The speed range covered with the Seafire was limited at its lower end by the poor slow-flying qualities of this particular aircraft. The lowest speed at which a steady glide could be maintained was about 1·18 times engine-off stalling speed with the C.G. at 0·336 a.m.c. Results are plotted directly as angle of glide against E.A.S. for various amounts of engine, in Figs. 3 (S) and 3 (B). In Figs. 4 (S) and 4 (B), the results are presented in a form which shows the variation in angle of glide with the pilot's safety margin in speed above his stalling speed (engine on). Angle of glide is plotted against the ratio V/V_{ES}, for various constant values of r.p.m. for the Seafire and for constant-boost pressures for the Barracuda, where

V = aircraft speed, E.A.S., m.p.h.

V_{ES} = stalling speed, m.p.h., with engine on, at the same boost pressure as used during the glide.

Also shown are lines of constant airspeed expressed as the ratio V/V_S, where V_S is the engine-off stalling speed, m.p.h.

These diagrams show to what extent the " standard " condition (*see* section 3), can be obtained with these aircraft. If a certain margin in speed above engine-on stalling speed during the glide is specified, *i.e.* a definite value is assigned to V/V_{ES}, and if, in addition the gliding speed is to be a certain fraction of the engine-off stalling speed, *i.e.* if V/V_S is also specified, then the resulting angle of glide may be read off the diagram at the point of intersection of the appropriate curve for V/V_S with the required ordinate for V/V_{ES}. This point will also give the engine setting necessary to produce this condition. In general, if any two of the four quantities— V/V_{ES}, V/V_S, angle of glide, and engine setting, are specified, these diagrams enable the remaining two quantities to be found.

The " standard " condition of section 3 requires an engine setting which will make V_{ES} less than 0·87 times V_S, and a speed V equal to 1·00 times V_S. Hence it is required to make $V/V_S = 1 \cdot 00$ and $V/V_{ES} = 1/0 \cdot 87 = 1 \cdot 15$. For the Barracuda, this results in an angle of glide of 6 deg., which is better than the proposed standard (*see* Point " A ", Fig. 4 (B)). In the case of the Seafire, however, this " standard " condition would probably give an angle of *climb* of about 3 deg. (*see* Point " A," Fig. 4 (S)). It must be emphasised that the low-speed portion of Fig. 4 (S) is extrapolated. The poor slow-flying qualities of this Seafire made it impossible to obtain experimental points in the region below a value of $V/V_S = 1 \cdot 18$.

4.3. *View During the Approach.*—The method of assessing view suggested in the proposed A.D.M. was investigated, but was dropped in favour of the photographic method described in Appendix II. Photographs were obtained with a pinhole camera (Figs. 6 (S) and 6 (B)), and from these the view could be assessed in terms of the angle of yaw (*i.e.* the angle between the line of sight and the plane of symmetry) necessary in order to see the port corner of the carrier's round-down when 300 yards astern. The technique and method of analysis of the photographs are described in Appendix II.

With the Barracuda, measured values of datum attitude were obtained during the glides described in section 4.2, and the wing incidence corresponding to the " standard " condition (*see* section 3) was found to be 14·8 deg. However, analysis of actual landings on H.M.S. *Pretoria Castle* showed that the average landing condition involves an airspeed of about 1·025 V_S and a gliding angle (relative to the air) of 3·2 deg. This condition is represented by Point " B " Fig. 4 (B), and the mean wing incidence was found to be 13·4 deg. (± 1 deg.). Accordingly, the view has been assessed for the " standard " condition and for the average deck landing condition.

For the Seafire no satisfactory measurements of the variation in datum attitude with speed and engine power were available, and therefore the wing incidence has been estimated for the various approach conditions using results published by Lyons and Bisgood[3] (1945), Young and Hufton[4] (1941) and Smelt and Davies[5] (1937) to obtain the lift due to the wing, flaps and slipstream respectively. The closest approximation to the " standard " condition (viz. an angle of glide of 5 deg. at an airspeed of 1·00 V_S), results in the very low value of $V/V_{ES} = 1 \cdot 02$ (*see* Fig. 4 (S)), and the estimated wing incidence for this condition is 13·8 deg. Analysis of actual Seafire landings on H.M.S. *Pretoria Castle* shows that the average deck landing approach condition for this aircraft is represented by a value of $V/V_S = 1 \cdot 10$, with a glide angle of 3 deg. and a corresponding value of $V/V_{ES} = 1 \cdot 15$ (*see* Point " B ", Fig. 4 (S)) giving an incidence of 11 deg. As on the Barracuda the view has been assessed for both these incidences, and the results, for both aircraft, are given in Tables 1 and 2 below. The wind speed over the carrier deck has been taken to be 25 knots. It will be noted that two positions of the pilot's head have been considered, one with the pilot's head central, the other with his head moved the maximum comfortable distance across the cockpit. In this off-centre position, the view through the side windscreen panels on the Barracuda is satisfactory, but on the Seafire distortion in the perspex panels makes it necessary to view the carrier around the rear edge of the windscreen, *i.e.* by direct vision.

In all cases, the width of the carrier round-down has been assumed to be 90 ft.

TABLE 1

View from Barracuda Cockpit

Approach condition	Mean wing incidence	Downward inclination of line of sight to wing chord	Angle of yaw needed to see carrier	
			Head central	Head moved $3\frac{1}{2}$ in. sideways
" Standard " landing condition (section 3) $V/V_{ES} = 1{\cdot}15$ $V/V_S = 1{\cdot}00$ Angle of glide $= 6°$	14·8° (Measured)	18·6°	7°	4°
Average condition for actual deck landings $V/V_{ES} = 1{\cdot}26$ $V/V_S = 1{\cdot}025$ Angle of glide $= 3{\cdot}2°$	13·4° (Measured)	15·3°	5°	2°

TABLE 2

View from Seafire Cockpit

Approach condition	Mean wing incidence	Downward inclination of line of sight to wing chord	Angle of yaw needed to see carrier	
			Head central	Head moved 5 in. sideways
Closest approximation to " Standard " landing condition (section 3) $V/V_{ES} = 1{\cdot}02$ $V/V_S = 1{\cdot}00$ Angle of glide $= 5°$	13·8° (estimated)	16·7°	22°	12°
Average condition for actual deck landings $V/V_{ES} = 1{\cdot}15$ $V/V_S = 1{\cdot}10$ Angle of glide $= 3°$	11·0° (estimated)	12·5°	15°	9°

4.4. *Alternative Requirements.*—(A) *Direct measurement of rate of " sidestep ".*—This test was not carried out as no simple method of measurement has yet been evolved. The sidestep distance could be calculated from records of two accelerometers and an angle of bank indicator in the aircraft, but this extra complication was not considered worthwhile, since measurements of maximum rate of roll and time to bank 10 deg. are easier to make and are more informative.

(B) (i) *Measurement of time to reverse 30 deg. bank.*—In the "standard" approach condition (section 3), the time to change from a steady angle of bank of 30 deg. in one direction to a steady 30 deg. bank in the opposite direction was measured on the Barracuda. The time was 2·5 sec. and the force which the pilot has to apply peaked initially to 45 lb., then fell off to about 25 lb. The result was approximately the same whether the test was done by banking to 30 deg. and then reversing this bank on one continuous manœuvre, or by flying steadily in a 30 deg. banked turn and changing as rapidly as possible to a steady 30 deg. bank in the opposite direction.

Neither this test, nor the following one, were carried out with the Seafire, as the aircraft was damaged before the work was completed.

(B) (ii) *Measurement of rate of flat turn.*—This test could not be done with the Barracuda, due to the danger of rudder overbalance.

4.5. *Control During the Final Stage.*—(i) *Measurement of time to apply* 10 *deg. Bank.*—For this test, the Barracuda was trimmed to fly in the "standard" condition ($V/V_S = 1 \cdot 00$, $V/V_{ES} = 1 \cdot 15$), and full aileron was applied as rapidly as possible. The time for the bank to reach 10 deg. was obtained from integration of the resulting rate-of-roll versus time record. This minimum time was 0·9 secs., and the force required was 40 lb. With the Seafire, where the "standard" condition was unobtainable, the test was done at a speed of $1 \cdot 25\ V_S$. The result, reduced to the average approach speed (for normal Seafires) of $V/V_S = 1 \cdot 10$ gives the time to bank 10 deg. as 0·75 secs., and the force required as 15 lb.

A fuller investigation of the lateral control on both aircraft was made by recording stick force and rate of roll during steady rolls with various aileron angles. The required amount of aileron was applied as rapidly as possible, and then held steady, the rudder being held fixed throughout. Both aircraft were in the landing condition (wheels and flaps down, hood open, etc.) and the speed range covered was from 80 to 115 m.p.h. with the Barracuda, and 90 to 105 m.p.h. with the Seafire. Full results are given in Figs. 7 (B), 8 (B) for the Barracuda and Figs. 7 (S), 8 (S) for the Seafire. Rate of roll is represented by the helix angle $\tan^{-1} pb/2V$, and the stick forces have been reduced to a speed of $1 \cdot 10\ V_S$ for each aircraft.

The maximum values of $pb/2V$ obtainable were 0·09 for the Barracuda (17·5 deg. aileron) and 0·10 (mean of port and starboard) for the Seafire (24 deg. aileron). Some asymmetry between the rolling performance to port and starboard was found with the Seafire. The stick forces given in Figs. 8 (B) and 8 (S) are maximum values, occurring early in the roll.

(ii) *Change in trim tests.*—Two alternative methods of measuring the change in longitudinal and direction trim on closing or opening the throttle are proposed in the A.D.M. Both methods were fully investigated and the results are given in Tables 3 and 4 below. The tests were done at a small range of initial speeds and engine powers, but no significant differences were observed, and the results have been quoted for the following mean initial conditions:—for the Barracuda, mean speed 80 m.p.h. ($1 \cdot 07 V_S$) at $-$ 1 lb./sq. in. boost, and for the Seafire, mean speed 97 m.p.h. ($1 \cdot 24 V_S$) at $-$ 5 lb./sq. in. boost.

Continuous automatic observer records were taken during what may be termed the "dynamic" tests—in which the pilot attempted to keep the vertical acceleration between certain limits. For the "static" tests, records were taken of the steady conditions before and after closing (or opening) the throttle.

There was no tendency, on either aircraft, to drop a wing on closing the throttle, the amount of aileron used was usually about 2–3 deg., and never exceeded 5 deg.

The starboard foot-force indicator failed during the tests on the Barracuda, and measurements of the port foot-force only were obtained. However, from measurements of foot-force during the engine-cut tests, it appears that the force required on opening the throttle would be of the order of 15lb. only.

TABLE 3

Change in Trim Tests on Barracuda

Type of Test	Change in :—					
	Elevator Force	Elevator Angle	Rudder Force	Rudder Angle	Datum Attitude	A.S.I. Reading
(ii) (*a*) Closing throttle, keeping vertical acceleration between 0·8 and 1·0*g*.	2 lb. (push)	5° (down)	25 lb. (port)	15° (port)	1·7°/sec. (nose down)	2 m.p.h./sec. (deceleration)
(ii) (*b*) Closing throttle, speed increased by approx. 10 m.p.h.	2 lb. (push)	5° (down)	28 lb. (port)	15° (port)	8° (nose down)	11 m.p.h. increase
(iii) (*a*) Opening throttle, keeping vertical acceleration between 1·0 and 1·2*g*.	3 lb. (push)	1° (down)	—	5° (st'b'd)	1·5°/sec. (nose down)	4 m.p.h./sec. (acceleration)
(iii) (*b*) Opening throttle, flying at same A.S.I. as initially.	1 lb. (push)	1° (up)	—	6° (st'b'd)	7° (nose up)	0

TABLE 4

Change in Trim Tests on Seafire

Type of Test	Change in :—					
	Elevator Force	Elevator Angle	Rudder Force	Rudder Angle	Datum Attitude	A.S.I. Reading
(ii) (*a*) Closing throttle, keeping vertical acceleration between 0·8 and 1·0*g*.	<1 lb. (push)	1° (down)	10 lb. (port)	4° (port)	1·3°/sec. (nose down)	2 m.p.h./sec. (deceleration)
(ii) (*b*) Closing throttle, speed increased by approx. 10 m.p.h.	<1 lb. (pull)	1° (down)	7 lb. (port)	3° (port)	5° (nose down)	9 m.p.h. (increase)
(iii) (*a*) Opening throttle, keeping vertical acceleration between 1·0 and 1·2*g*.	3 lb. (push)	unsteady	21 lb. (st'b'd)	5° (st'b'd)	1·0°/sec. (nose down)	5 m.p.h./sec. (acceleration)
(iii) (*b*) Opening throttle, flying at same A.S.I. as initially.	5 lb. (pull)	1° (up)	33 lb. (st'b'd)	7° (st'b'd)	12° (nose up)	0

5. *Discussion of Results.*—The tests have been described in the order in which they were carried out. This was the order most convenient for flight testing, but the order of relative importance of the various factors is probably as follows:—

(i) View during the approach.
(ii) Control over the rate of descent.
(iii) Lateral control at low speeds.
(iv) Change in trim with engine power.

In the following section, the significance of the results of these provisional A.D.M. tests is discussed in the light of the accepted deck landing qualities of the two aircraft.

5.1. *View During the Approach.* The tests with the pinhole camera in the Seafire cockpit give results which amply substantiate pilot's opinions as to the inadequacy of the view. An angle of yaw of 22 deg. would be required for an approach from dead astern in a 5-deg. glide at a speed equal to the engine-off stalling speed (approximately, the " standard " condition) with the pilot seated centrally in the cockpit. A large improvement is possible if the pilot moves his head across the cockpit and views the carrier round the rear edge of the windscreen, the required angle of yaw then falls to 12 deg. This is the normal expedient by which the view from the cockpit of the Seafire is improved. Other considerations require a departure from the proposed " standard " approach condition, and this also helps to improve the view. It is found that actual landings with this type of aircraft are made at an approach speed in the region $1 \cdot 05$ to $1 \cdot 15$ times engine-off stalling speed, with an angle of glide (relative to the air) of about 3 deg. The reason for this departure from the " standard " condition is explained in section 5.2 below, but the net result is that more engine is used causing a reduction in incidence, which, together with the reduction in gliding angle, reduces the downward inclination of the line of sight relative to the wing chord and so further improves the view. In this typical approach condition which may be specified by values of $V/V_S = 1 \cdot 10$, glide angle $= 3$ deg., giving $V/V_{ES} = 1 \cdot 15$ (as at Point " B ", Fig. 4 (S)), the required angle of yaw with head central is 15 deg , falling to 9 deg. if the pilot moves his head 5 in. across the cockpit. This is still rather an excessive amount of yaw for low-speed flying at sea level, so a curved approach path is used, instead of a straight approach from dead astern. This enables the required angle between line of sight and the centre line of the aircraft to be obtained with little or no skid, depending on the radius of turn. For example, a turn of 4,000 ft. radius relative to the carrier would provide the required angle of 9 deg. between line of sight and centre-line of the aircraft when 900 ft. astern, the width of the round-down being taken as 90 ft. (*see* Fig. 9). For a turn of larger radius, a certain amount of skid is necessary, and the average Seafire approach does employ a combination of turn and skid, as borne out by analysis of actual landings on H.M.S. *Pretoria Castle*, published by Lean, Duddy and Stott[6] (1944). The measured radius of turn varied between 2,000 ft. and 5,000 ft. relative to the carrier. In the case of the Barracuda, the view from the cockpit is generally considered to be fairly good, and this is borne out by analysis of the pinhole-camera photographs. The angle of yaw required in order to be able to see the corner of the round-down from 900 ft. in an approach in the " standard " condition of section 3 is only about 7 deg. This can be improved to an angle of 4 deg. by a $3\frac{1}{2}$-in. sideways movement of the pilot's head, which the harness will allow comfortably.

Analysis of actual landings on H.M.S. *Pretoria Castle* shows that the average approach speed is rather higher than that proposed for the standard condition, and the amount of engine used is such as to reduce the angle of glide to about 3 deg. As in the case of the Seafire, this has the effect of further improving the view, and, with the pilot's head in the off-centre position, the required angle of yaw falls to 2 deg.

Thus, although the Barracuda does not attain the " ideal " proposed in the A.D.M., the above considerations show how easily the required view can be obtained, by these three slight deviations from the " standard " approach condition (sideways movement of pilot's head, increase in amount of engine used, and a small angle of skid). The view for deck landing is therefore considered to be tolerable.

5.2. *Control of Rate of Descent.*—A fairly complete investigation was made of the pilot's control over the angle of glide on both aircraft. The normal procedure is for corrections to rate of descent during the approach to be made with the throttle, and corrections to speed with the elevator. Full results are shown graphically in Figs. 3 (S), 3 (B), 4 (S), and 4 (B).

The Seafire, which is aerodynamically " clean ", even in the landing condition, requires only a small fraction of the total power available in order to fly level at the approach speed. This means that the increment of lift due to slipstream required for the " standard " approach condition is not available with the amount of engine required for an angle of glide of 5 deg. at the approach speed. Fig. 4 (S) shows that, at a speed equal to $1 \cdot 0\ V_S$, the amount of engine used in a 5 deg. glide is such as to reduce the stalling speed only 2 per cent. Alternatively, if a 15 per cent. margin above engine-on stalling speed is to be available during a 5 deg. glide, there will still be a margin of about 13 per cent. above engine-off stalling speed remaining when the throttle is closed. For this reason the average Seafire approach is made with sufficient engine to decrease the gliding angle to about 3 deg., in which condition the stalling speed is reduced by about 5 per cent. Even so, at the average approach speed of $1 \cdot 15\ V_{ES}$, a margin of 10 per cent. above engine-off stalling speed remains when the throttle is closed, producing a tendency towards aerodynamic bounce, if a full " check " is attempted, as discussed in a report by Hufton[7] (1945).

The lower limit to the angle of glide which can be used is fixed by safety considerations, and is about 5 deg. relative to the carrier, or 3 deg. relative to the air (for average aircraft and carrier speeds). Below this angular boundary lies the turbulent wake of the carrier itself. The effect of this disturbed airflow on the controllability of the approaching aircraft is uncertain, but the region is one which it is better to avoid if possible. Apart from this, an approach from a very shallow glide has several disadvantages. It makes it more difficult for the pilot to appreciate errors in height, more difficult to make the necessary correction for height error, particularly if the aircraft is too low, and the result of such an uncorrected height error is certainly more dramatic. Finally, the pitching of the stern of the ship in a heavy sea makes it necessary to use as steep an approach as possible, to avoid hitting the round-down or touching down forward of the arrester wires.

A further disadvantage of this lack of drag is that the small amount of engine employed means that the throttle is only a very short distance from the closed position. Hence only a very small proportion of the throttle travel is available for fine adjustment of the angle of glide, making it difficult to avoid over-correction, and so causing accidents due to " undershooting " or " overshooting ".

The Barracuda, having ample drag to increase the gliding angle, is easily able to satisfy the " standard " condition of section 3. The required angle of glide of 5 deg. at a speed equal to $1 \cdot 0\ V_S$, can be obtained with a margin of about 24 per cent. above engine-on stalling speed, *i.e.* with sufficient engine to reduce the stalling speed by nearly 20 per cent. Analysis of landings on H.M.S. *Pretoria Castle* shows that the average approach is made at a speed equal to about $1 \cdot 025$ times engine-off stalling speed, with a mean angle of glide of $3 \cdot 2$ deg. relative to the air. The amount of engine used is such as to reduce the stalling speed by just over 20 per cent. (*see* Fig. 4 (B), Point " B "). An advantage of this higher speed and lower gliding angle is the improvement in view resulting from the decrease in angle between line of sight and wing chord.

The large amount of engine used during the approach permits fine adjustment of the angle of glide with quite coarse movements of the throttle lever.

5.3. *Lateral Control at Low Speeds.*—Even if the approach path of an aircraft is perfectly satisfactory, considerable piloting skill may be required to touch down without either float or bounce which can easily result in a barrier crash. Float can occur if the pilot over-checks his rate of descent, and bounce, or damage to the undercarriage is likely to occur if he does not check enough; this problem has been discussed by Hufton[7] (1945) and it is shown that the cure is to keep the speed on the approach as low as possible. In the ideal case the approach speed would be determined simply by the proximity to the stalling speed, but in practice the speed necessary to give adequate forward view or satisfactory lateral control may be considerably higher than that required to avoid danger of stalling.

It is therefore important that the lateral control should be adequate for correcting wing dropping due to gusts, and for making corrections to line in conjunction with the rudder at the A.D.M. approach speed of $1 \cdot 15\ V_{ES}$.

Normal rate-of-roll tests were carried out on the Seafire, and integration of the rate-of-roll versus time curves shows that it meets the A.D.M. requirement with 10 deg. bank in $0 \cdot 75$ sec. with full aileron at the average approach speed of $1 \cdot 10\ V_S$. The stick force involved was about 15 lb., instead of the stipulated 5 lb. The variation of stick force during the initiation and development of a roll shows a definite response effect, the peak stick force occurring after the aileron has become steady but before the rate of roll has reached its maximum value. It is considered that during manoeuvres of short duration, such as picking up a wing or making small alterations in alignment, it is these peak values of stick force which are most important, and they have been plotted against aileron angle in Fig. 8 (S). The rate of application of aileron was not necessarily the pilot's maximum.

In the case of the Barracuda, a separate test was made in which the pilot attempted to obtain 10 deg. bank in the shortest possible time, with correspondingly high inertia forces. Part, at least, of the high value of 40 lb. for the force required may be attributed to aileron inertia. The time to bank 10 deg. (0·9 sec.) is higher than that proposed in the A.D.M. The limitation appears to be the large rolling inertia of the aircraft, since the stick force employed was not the pilot's maximum, and the steady rolling velocity, represented by $pb/2V$, is satisfactory.

For various reasons, little was done in measuring the effectiveness of the ailerons and rudder in making corrections to line. On the Barracuda, the test for aileron effectiveness in the "sidestep" manoeuvre was performed, and the results fall short of the A.D.M. requirements, both in time to reverse 30 deg. bank, and in the maximum force required. There is, however, some inconsistency between the aileron requirements for minimum time to obtain 10 deg. bank and those for reversing 30 deg. bank. To roll through a total angle of 60 deg. in 1·5 secs. at a speed of $1 \cdot 0\ V_S$ with the Barracuda requires a mean $pb/2V$ of 0·12, while, allowing the specified 0·75 sec. for the first 10 deg. bank, the steady value of $pb/2V$ would need to be about 0·20. It is clear, therefore, that some modification is necessary to the A.D.M. requirement for minimum time in this manoeuvre. These tests were not performed with the Seafire, but the required aileron power would be equally unattainable.

No tests were done on either aircraft to measure the maximum rate of flat turn, and the direct measurement of the rate of "sidestep" was not attempted, for lack of a suitable safe technique. Rough calculations, based on the time to reverse 30 deg. bank in the case of the Barracuda, show that this aircraft should easily satisfy the proposed A.D.M. requirements with ailerons alone. These calculations have been extended, for both aircraft, as explained in Appendix III. The results are discussed in section 6.2.

5.4. *Change in Trim with Engine Power.*—Two alternative methods of measuring change in trim on closing or opening the throttle were investigated, one essentially dynamic in principle, the other static.

Considering first the "engine-cut" case, the dynamic test required the pilot to maintain a vertical acceleration of between 0·8 and 1·0g with an initial speed of $1 \cdot 0\ V_S$. It is clearly impossible to approach 1·0g from this initial condition without stalling the aircraft. Instead, the sudden loss in slipstream lift, plus the longitudinal deceleration following the loss in thrust, will cause a rapid increase in the rate of sink. This effect will be present, though to a lesser extent even when the initial speed is higher than $1 \cdot 0\ V_S$, as in these tests.

Analysis of control angle records during Seafire landings on H.M.S. *Pretoria Castle* shows that, during the interval between cutting the throttle and picking up an arrester wire (0 to 1·5 seconds), there is a definite flattening out of the approach path by a backward movement of the stick. This is only possible on account of the margin above engine-off stalling speed which exists during a typical approach with this aircraft. The closer the initial speed approaches the engine-off stalling speed, the greater will be the "sinking" effect, and this test shows that the pilot's reaction is to drop the nose of the aircraft to regain a safe flying speed, rather than to attempt to check the sink. It may be desirable to stall the aircraft at the instant of touch-down, but the effect is difficult to reproduce for test purposes.

The static test for change in trim on closing the throttle required the pilot to increase speed from $1 \cdot 0\ V_S$ to $1 \cdot 15\ V_S$. This also involves dropping the nose of the aircraft, and it is therefore to be expected that the longitudinal change in trim would be roughly the same for the two methods of test. Both stick and rudder forces are within the limits proposed in the A.D.M., for both aircraft. The stick forces in the case of the Seafire are consistent with the aircraft being longitudinally unstable, stick free.

It is therefore suggested that the static test gives results very similar to the dynamic test, while the latter becomes increasingly difficult as the initial speed approaches engine-off stalling speed.

On opening up to full throttle following a " wave-off ", the pilot has, in effect, two extreme courses open to him. He may either hold the aircraft down, and allow it to accelerate to its best climbing speed, or he may pull it into the steepest possible climb, at the same speed as initially. The dynamic method of measuring change in trim corresponds to the former, and the static method to the latter technique. The average path taken, following the opening of the throttle to go around again, results from a compromise between these two techniques.

The change in trim on opening the throttle, is, therefore, somewhat different for the two methods of test. On the Seafire, the control forces and movements are all about 50 per cent. greater for the static method of test, the stick forces being reversed as well as increased. On the Barracuda, the stick forces are less for the static test. With either method of test, on both aircraft, the stick and rudder forces are within the limits suggested in the A.D.M.

Here again, the static test is much easier to perform and the results are easier to record and analyse. In carrying out the dynamic test, it was found that conditions are very unsteady following the sudden opening of the throttle, particularly with the Seafire, making it very difficult to keep the vertical acceleration between the prescribed limits.

6. *Suggested Modifications to the Proposed A.D.M.*—6.1. *Requirement for View.*—The standard set for view from the cockpit is such that probably even the best of present-day conventional single-engine naval types would fail to satisfy it. On the other hand, the twin-engine aircraft, or a jet-propelled aircraft, would probably provide a view better even than the A.D.M. standard. Provided, however, that the pilot can clearly see the " batsman " and the round-down, nothing further will be gained by increasing his angle of vision beyond this minimum standard, as far as the difficulty of landing is concerned. It is not possible to say, on the basis of tests on two aircraft, whether the A.D.M. requirement is more stringent than is necessary.

Accepting the proposed standard, it is suggested that the method of assessing view should employ the pinhole-camera technique. It is simpler, quicker and more accurate than the post method, and it provides a permanent record. A further advantage is that the pinhole-camera method gives a simple quantitative assessment of the view, in terms of the angle of yaw which the pilot has to introduce in order to see the " batsman ".

6.2. *Tests for Aileron Effectiveness during the Approach.*—As pointed out in section 5.3, the aileron power required to reverse 30 deg. of bank in 1·5 seconds is far greater than that needed to produce 10 deg. of bank in 0·75 seconds, the value of $pb/2V$ necessary to satisfy the former requirement being quite unattainable with a normal aircraft. It therefore appears necessary either to increase the specified time to, say, 3·0 secs., or to reduce the angle of bank involved. There appears to be ample aileron power to produce lateral displacements of the flight path by means of correctly banked " S "-turns, well within the limits of horizontal distance suggested in the A.D.M. Calculations have been made, with various simplifying assumptions, of the sidestep distance obtainable with various maximum angles of bank in the "S" turn, and the results are shown graphically in Figs. 11 (S) and 11 (B). The method of estimation, and the assumptions made, are described briefly in Appendix III. As an example, a sidestep distance of 25 ft. can be obtained in a forward distance of about 500 ft. with the Seafire, and in roughly the same distance with the Barracuda. The corresponding maximum angles of bank are about 23 deg. and 18 deg. respectively, and the times taken are 4 secs., and 4·7 secs. The proposed overall time limit of 3 secs. would limit the maximum sidestep distances to 9 ft. for the Seafire and 6 ft. for the Barracuda. The conclusion is, therefore, that an ample rate of sidestep is assured if the time to bank 10 deg. from rest is specified to be 0·75 sec., and if an average value of $pb/2V$ (say, 0·08) is available. The time to reverse 30 deg. of bank, and the maximum rate of flat turn do not enter directly into the estimation of sidestep distance in Appendix III, but it may be mentioned that the measured time to reverse 30 deg. of bank agrees fairly well with that estimated from the measured time to obtain 10 deg. of bank and the measured value of $pb/2V$.

7. *Conclusions*.—The Barracuda appears to satisfy most of the requirements of the proposed A.D.M., except for the stick forces during reversal of 30 deg. of bank, and during the rapid application of 10 deg. of bank. The view from the cockpit is also slightly below the proposed standard. The Seafire fails badly to satisfy the important requirements for the effect of engine on gliding angle and view from the cockpit during the approach.

Certain of the suggested figures in the proposed A.D.M. appear to require modification, particularly the time to reverse 30 deg. of bank.

It is suggested that the pinhole camera method of assessing view from the cockpit should be adopted, as giving a quantitative assessment of view. It is also suggested that the " static " method of measuring change in trim with engine power (*i.e.* by comparing the initial and final trim conditions) is considerably simpler, and gives results very similar in magnitude, though not necessarily in direction, to those obtained from the " dynamic " tests.

8. *Further Developments*.—These tests have now been carried out on both Avenger and Hellcat aircraft, and the results will be reported in due course. In addition, deck landing trials have been made with all four aircraft and it is hoped eventually to use the results of these tests to determine what modifications are required in the proposed A.D.M.

The requirements will probably need further modification to cover aircraft with tricycle undercarriages and jet propulsion. The tricycle undercarriage reduces the tendency to bounce after touch down and may allow higher approach speeds, and with jet propulsion the absence of slipstream will necessitate higher approach speeds and may increase the difficulty of correcting height errors.

LIST OF SYMBOLS

b	total wing span
C_L	aircraft lift coefficient $= W/\frac{1}{2}\rho V^2 S$
D	propeller diameter
p	rate of roll
S	gross wing area
T	propeller thrust
T_c	thrust coefficient $= T/\rho V^2 D^2$
t_A	time to obtain 10 deg. of bank starting from zero rate of roll
t_S	time for complete " S " turn
V	true airspeed
V_S	true airspeed at the stall with flaps and undercarriage down and engine throttled back
V_{ES}	true airspeed at the stall with flaps and undercarriage down and engine on
W	all-up weight of aircraft
β	azimuth angle between line of sight and aircraft centre line
θ	change in direction of flight from the original path
λ	value of ψ with aircraft at rest on level ground
ψ	vertical angle between aircraft datum and horizontal
ρ	air density
ϕ	angle of bank
ϕ_M	maximum angle of bank in an " S " turn
A.M.C.	aerodynamic mean chord

APPENDIX I

Proposals for an A.D.M. for Deck Landing Characteristics

By

P. A. HUFTON

1. *Introduction.*—The landing technique used for carrier-borne aircraft differs markedly from standard aerodrome landing technique and demands different characteristics from the aircraft. At the same time finer judgment is needed by the pilot. While it would be agreed that pilots can acquire toleration to bad handling characteristics, these do imply more concentration by the pilot and leave him with closer margins of error. Any slight failure by the pilot, due possibly to strain after a long reconnaissance, would have serious consequences.

It seems desirable, therefore, that a standard should be set up to which deck-landing aircraft should conform. In the present note the form that any requirements should take is deduced from a study of the present deck-landing technique. Where possible an exact quantitative expression for the standard has been given, but it will be seen that these do not cover the more important aspects of the problem. It is clear that more flight experiments are wanted before any final standards can be laid down, and these tests may modify the form in which the requirements are cast.

2. *Technique of Deck Landing.*—A brief description of deck-landing technique is necessary so that the way the various requirements arise may be seen. Starting with the carrier deck, the area on which the aircraft must touch down is that covered by the arrester wires: *i.e.* not more than 400 ft. long with about 6 wires about 50–60 ft. apart, and about 70 ft. wide. It is usual to have two or more crash barriers behind the wires, but these are for emergency use only. The aim of the pilot is to take either the first or second wires; there is some risk of striking the crash barriers if the last wire is used. The aircraft should not land more than about 20 ft. off centre; there is the obvious danger of running over the side if this is exceeded, but there is also the danger of damaging the aircraft due to the side forces set up in the arrester hook. An important feature of carriers is the existence of a disturbed wake behind the stern. This extends upwards from the round-down at an angle of about 4–5 deg. and is characterised by both a reduction of wind velocity and by a downwash. It is, therefore, an area to be avoided in landing-on.

A " batsman " controlling the approaching aircraft is stationed at the port side of the stern.

In Stage 1 of the approach the carrier is circled by the aircraft at a distance up to half a mile away and at a height between 500 and 1,000 ft. The undercarriage is lowered at this stage. When signalled to approach the aircraft turns in on the port beam when astern of the carrier and the flaps and hook are lowered.

With this turn in, Stage 2 of the approach starts. The aircraft turns into line with the ship, with a certain average amount of engine. The approach speed is about 10–15 per cent. above the stalling speed with the amount of engine used, but of course is much nearer the stalling speed engine-off. This low approach speed is desirable for two reasons: to prevent float after flattening out, and to leave as long a time as possible for manœuvring into position. An angle of glide (with respect to the ship) of about $6\frac{1}{2}$ deg. is aimed at and the batsman gives corrections to the pilot, who adjusts his height on the throttle. Corrections to line are made as required by the combined use of aileron and rudder. The approach is continued until the aircraft is over the edge of the deck and at a reasonable height above the deck.

For Stage 3 the batsman orders the pilot to " cut ". The ideal place for touching down is just aft of the first wire. The pilot cuts the throttle and the aircraft should sink firmly on to the wires without swinging, dropping a wing or changing trim violently. Control of the rate of descent on to the deck should be obtainable by adjustments on the throttle. If the pilot or batsman is dissatisfied with the position of the aircraft, the pilot gives full throttle and goes round again. At the end of Stage 2 and during Stage 3, intense gusts due to the flow over the deck may be encountered. This will be specially so if the pilot has approached rather low and runs through the wake of the carrier.

An alternative method in use by the American Navy combines Stages 2 and 3 in a curved approach from the circling stage. After practice most pilots tend to use this method. It is thought that this requires similar characteristics to the technique described above, so that the discussion of the requirements is based on the simple straight approach.

3. *Summary of Above Landing Conditions.*—(i) The gliding angle is adjusted with engine to be not less than about 5 deg. (with reference to atmosphere).

(i) The gliding speed is such that it is, if possible, below the stalling speed without engine, but about $1 \cdot 10$–$1 \cdot 20$ times the stalling speed with the amount of engine defined in (i).

(iii) Under the conditions laid down above, the carrier should be visible from at least 1,000 ft. (the aircraft is assumed to be in a steady glide aimed at the round-down).

(iv) The gliding speed should not be more than 75 knots.

(v) Corrections to the line of the approach by aileron and rudder should be possible.

(vi) Good lateral control during the final stages is essential because of the gustiness.

(vii) If the engine is cut from the condition given in (i) and (ii) the aircraft should sink without marked change of trim.

(viii) If the engine is given full throttle from the condition given in (i) and (ii) there should be no uncontrollable change of trim of the aircraft.

In this list (i) and (ii) ensure that the aircraft is gliding at a sufficiently steep angle, that enough engine is given to reduce the stalling speed so far as to make shallow turns possible at the gliding speed, and that the gliding speed is below the engine-off stalling speed so that the aircraft sinks without float when the engine is throttled back. The rest are obvious control requirements.

4. *Suggested A.D.M. for Flight Tests of Handling Qualities during Deck Landing.*—4.1. *Effect of Engine on Stalling Speed.*—The stalling speed with flaps and undercarriage down shall be measured by suspended static and swivelling pitot with varying amounts of engine boost: the propeller shall be in fine pitch. Sufficient points must be obtained to allow a reasonable curve to be drawn.

The stalling speed engine-off is to be less than 75 knots.

4.2. *Effect of Engine on Gliding Angle.*—The rate of descent of the aircraft with flaps and undercarriage down shall be determined for the least engine setting in 4.1 which gives a stalling speed less than $0 \cdot 87$ times engine-off stalling speed (*see* section 5). The indicated air speed at which the test is done shall be equal to the engine-off stalling speed. The angle of glide shall be greater than 5 deg.

4.3. *View during Approach.*—The attitude ψ deg. of the datum line of the aircraft to the horizontal shall be measured during test section 4.2. The aircraft shall be placed on level horizontal ground with the datum line of the aircraft at an angle of $\psi + 7 \cdot 5$ deg. to the horizontal* on the line which bisects at right angles the line joining two small markers 1 ft. high placed 90 ft. apart. The aircraft is to be towed backwards along its axis until the pilot sitting fully equipped and strapped in, and with the cockpit hood closed can no longer see either of these sighting marks. The port mark should not disappear before the starboard mark and the minimum acceptable distance is 300 yards.†

4.4. *Alternative Requirements.*—*Either* (A): The aircraft is to be trimmed to fly in the condition of 4.2 above. The flight path of the aircraft is to be displaced laterally, the aircraft finishing in a steady glide parallel to the initial path and at the same speed. The distance required to complete

* The angle of $7 \cdot 5$ deg. to be added to the attitude ψ deg. is the angle of glide of the aircraft with respect to the *carrier*. This value is based on a free-air gliding angle of 5 deg., an aircraft approach speed of 75 knots, and a carrier and wind speed of 25 knots. These are average values.

† *See* section 5.

the manœuvre from the start is to be measured, for various lateral displacements. These distances are not to exceed

For lateral displacement	5ft.	15 ft.	25 ft.	*Highly tentative.
Horizontal distance	≯1,000 ft.	1,500 ft.	2,000 ft.	

Or (B): (i) The aircraft is to be trimmed under the conditions of 4.2. 30 deg. of bank is to be applied by aileron, nose kept straight. The minimum time required to reverse the bank to 30 deg. in the opposite way, and the maximum forces, are to be measured. This is to be done in both directions.

The time must be less than 1·5 secs.* and the force less than 10 lbs.*

(ii) The aircraft is to be trimmed to fly in the condition of 4.2. The rate of turn with full rudder, aircraft held level laterally, and the rudder forces are to be measured. This is to be done both to port and starboard.

The rate of turn must be more than 180 deg. minute and the force less than 100 lb.*

4.5. *Control during the Final Stage.*—(i) The aircraft is to be trimmed to fly in the condition of 4.2. above, 10 deg. of bank is to be applied, nose kept straight. The minimum time required to obtain this bank, and the maximum stick forces, are to be measured.

The time is to be less than 0·75 sec.* and the force less than 5 lb.*

(ii) *Either* (*a*) with the aircraft trimmed in the condition of 4.2 above, the throttle should be suddenly closed, the pilot using the elevator to maintain the vertical acceleration between 0·8 and 1·0*g*. There should be no tendency to drop a wing and the change of trim, both longitudinal and directional, should be small and easily controlled.†

Change of elevator force	<10 lb.*
Change of rudder force	<25 lb.*

Or (*b*) the aircraft is to be trimmed to fly in the condition of 4.2 above. Without retrimming, the pilot should then glide steadily engine-off at 1·15 times the engine-off stalling speed. The elevator and rudder forces are to be measured. They are not to exceed

Elevator force	10 lb.*
Rudder force	25 lb.*

(iii) *Either* (*a*) with the aircraft trimmed in the condition of 4.2 above, the throttle should be suddenly opened, the pilot using the elevator to maintain the vertical acceleration between 1·0 and 1·2*g*. The changes of trim should be easly controlled.†

Change of elevator force	< 20 lb.*
Change of rudder force	< 50 lb.*

Or (*b*) the aircraft is to be trimmed to fly in the condition of 4.2 above. Without adjusting the trimmers, the pilot should fly steadily at the same airspeed but at full throttle. The elevator and rudder forces to be measured. They are not to exceed

Elevator force	20 lb.*
Rudder force	50 lb.*

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* *See* section 5.

† It is fairly obvious that this manœuvre is essentially of short duration, and an extra phrase limiting the time to say 3 secs. may be required.

5. *Notes on Requirements and Suggestions for Future Work.*—Throughout the requirements the suggested figures have been starred, and it should be thoroughly understood that these are in no sense final but are merely indications of the order of the quantities. It is expected that these figures will be altered, in some cases profoundly, as the result of flight experiments.

It is worth noting, too, that the order of the tests is that most convenient for flight testing. It cannot be regarded as indicating the relative importance of the various items: in fact most Fleet Air Arm pilots would agree that the requirement for view is the most important.

Turning now to the details of the requirements, 4.1 and 4.2 determine the gliding speed, gliding angle and amount of engine, and are fairly straightforward. There may be practical difficulties in measurement of engine-on stalling speed with a suspended static and this will be examined.

4.3 is also simple on paper, but it may be necessary to revise the requirements because of practical difficulties. It may, for instance, be impossible to obtain an angle of $\psi + 7 \cdot 5$ deg. on the ground. Again alternative methods of doing this will be examined.

The alternative requirements of 4.4 are intended to ensure that corrections to line are possible. The first alternative asks for a direct measurement of the sidestep, leaving the method of obtaining this to the pilot. It is not considered that this requirement is one which can be recommended; if a pilot were asked to obtain the required figures by any means, he might easily overstep the limits of safety; and the method of recording the motion at a safe height presents serious difficulties. The second alternative scheme assumes that the sidestep is produced by a combination of banking and sideslipping, and that if minimum times to bank and rates of flat turn are specified, then a given sidestep distance can be obtained. It is less direct, but very much simpler and safer to perform. Flight tests of both alternative requirements are needed, and a comparison of what the pilot actually does in approaching a carrier with what he does when attempting the first alternative requirement would show whether in fact such a requirement is desirable.

In requirements 4.5 there are again alternative methods of determining the change of trim during the engine cut and baulked landing manoeuvre. One of these methods is dynamic and is fairly close to the real manoeuvre; the second method compares the final steady change of trim. The second method is simpler, and if it is generally agreed that it does in fact give results comparable with the first method, under the conditions laid down, viz. small changes of vertical acceleration, it would be adopted.

Thus a good deal more research flight testing is required before even the form of the tests can be settled, and to put the numbers in the tests will obviously require routine tests on most Naval types. These two types of tests are quite distinct and our proposals for them are as follows.

(A) *Research tests.*—These are intended to clarify the obscure parts of the landing approach, and to test the practicability and satisfactory nature of the proposed requirements. The aircraft used for these trials will carry enough instruments to follow the details of the deck landing manoeuvre. The results of the deck tests will be correlated with the results of the tests of the requirements and these revised as required. Two aircraft have already been obtained for this work, viz. Seafire and Barracuda, and these are being fitted with instruments.

(B) *Routine tests.*—The tests indicated above should be followed by applications of the revised A.D.M. to as many aircraft of Naval type as possible, and the correlation of the results of these tests with pilots' opinions of the aircraft as a deck landing aircraft. It is particularly important that the tests should be applied to those aircraft which are not considered to have good deck handling qualities, so that limiting qualities shall be found as accurately as possible. It is suggested that some non-Naval aircraft might be tested as well. The following aircraft should be tested in this way:—Martlet, Fulmar, Firefly, Firebrand (original condition of flaps), Firebrand (final condition of flaps), Hurricane, Mosquito, Typhoon and Airacobra.

APPENDIX II

View during the Approach

The fundamental requirement proposed for view is that the pilot should be able to see the port corner of the carrier's round-down from 300 yards astern, when the aircraft is in a steady glide aimed at the round-down.. Most present-day Naval types fail to satisfy this requirement, and in order to see the carrier, the pilot improves his view by a combination of these expedients :—

(i) Moving his head sideways.

(ii) Applying yaw.

(iii) Approaching the carrier on a turn, instead of in a straight approach from dead astern.

As a basis of comparison between different aircraft, it is proposed to use the angle of yaw which the pilot has to apply to enable him to see the carrier when approaching in a steady glide from 300 yards dead astern. When a large reduction in this angle can be obtained by the pilot moving his head sideways, this also is to be measured.

The test proposed in section 4.3 of the A.D.M. is difficult to carry out in practice; for most aircraft the incidence required would be greater than the ground incidence, necessitating jacking up the main wheels. Therefore, the following alternative methods of assessing the view have been investigated.

I. *Post Method.*—The attitude of the aircraft datum to the horizontal is measured for the " standard " approach condition (*see* section 3) during the rate of descent tests of section 4.2. Let this angle be ψ deg. Then, assuming a free air gliding angle of 3 deg. (5 deg. was suggested originally, but measurements during landings on H.M.S. *Pretoria Castle* show that a flatter approach than this is normally used), an approach speed of 75 knots, and a combined carrier speed plus wind speed of 25 knots, the angle of glide relative to the carrier is 5 deg. The pilot's line of sight is thus inclined downwards at an angle to equal $(\psi + 5)$ deg. relative to the datum. To determine the angle of view (in the horizontal plane) that the pilot has when looking in this direction, the aircraft is stood on level ground, and a post of the correct height is held vertically directly in front of the aircraft. The height of the post, and its distance from the aircraft, are chosen so that the angle between the line joining the top of the post to the pilot's eye, and the aircraft datum, is the required $(\psi + 5)$ deg. The post is then moved sideways until its top is just visible to a pilot seated in the cockpit, on his parachute, with the seat in the fully raised position. The arrangement is shown diagrammatically in Fig. 10. If, in order for its top to be visible from the cockpit, the post has to be moved through a sideways distance subtending an angle β at the pilot's eye, then β is the angle of yaw which the pilot would have to apply in order to be able to see in the direction of flight in this condition. When landing on, the carrier round-down is assumed to subtend a total angle of 6 deg. at the pilot's eye when 300 yards away, so, in order to be able to see one corner of the round-down when approaching from directly astern, the pilot has to apply an angle of yaw of $(\beta - 3)$ deg.

II. *Pinhole-camera Method.—Description of Camera.*—For taking photographs from inside the cockpit, a pinhole camera offers two advantages over a normal lens camera :—

(i) A very wide angle of view.

(ii) An infinite depth of focus.

The camera used for these tests was provided with a graticule, consisting of horizontal and vertical lines crossing in the centre of the plate, and with concentric circles giving angular bearings from the axis of the pinhole.

Method of Use.—With the aircraft standing on level ground, the camera is set up in the cockpit with the pinhole in the position of the pilot's left eye when seated on his parachute on the fully raised seat. The camera plate is arranged so that the graticule lines are truly horizontal and vertical, with the horizontal line parallel to the transverse axis of the aircraft. The axis of the pinhole is then horizontal, and parallel to the plane of symmetry of the aircraft. Then, if the attitude of the aircraft datum to the horizontal (on the ground) is λ deg., the intersection of the graticule lines gives the point in his field of view on which the pilot would be focusing if looking straight ahead in a direction inclined down by λ deg. to the datum. If this point on the print falls within the nose of the aircraft, the view ahead is blocked in a direction inclined downwards at an angle λ deg. to the datum, and the pilot could only see in the direction of flight by applying yaw. The magnitude of the angle of yaw to be applied can easily be measured from the print by interpolation between the rings of the graticule at the point where the horizontal line cuts the nose of the aircraft.

If the angle λ deg. above is not the same as the angle $(\psi + 5)$ deg. required during an actual approach (it is usually a few degrees less), then the blockage of view for the required angle can be obtained simply by shifting the horizontal line of the graticule through a vertical distance corresponding to the change in angle required. The distance through which it has to be moved is obtained by interpolation between the rings of the graticule. The angle of yaw to be applied is then measured from the intersection of this new horizontal line with the nose of the aircraft. The angle of yaw necessary in order to see the corner of the round-down is 3 deg. less than the angle of yaw required to see straight ahead in the direction of flight.

Typical photographs obtained with the pinhole camera are given in Figs. 6 (S) and 6 (B). On the Seafire photographs, the horizontal line of the graticule corresponds to a direction of sight inclined downwards at 14 deg. to the root wing chord. The corresponding angle for the Barracuda is 15 deg.

Comparison of Methods.—In interpreting the pinhole camera photographs, the pillars of the windscreen should be neglected, as they are not apparent with binocular vision. Provided this is done, the camera method offers the following advantages over the post method:—

(i) The angle of yaw can be measured from the photographs with an accuracy of $\pm$ 0·2 deg., whereas the post method is only accurate to $\pm$ 0·5 deg.

(ii) Having taken photographs for about three eye positions, results can easily be estimated for any other eye position, and for a range of angles between line of sight and aircraft datum. The post method involves measurements with a different post height for each angle required.

APPENDIX III

Simplified Method of Estimating Sidestep Distance

Lateral displacement of the flight path was assumed to be obtained by means of a continuous, correctly banked " S " turn with no sideslip. The variation of angle of bank with time has been assumed to be linear as shown by the full line in Fig. 12. The rate of roll is assumed to have constant magnitude throughout, the value being obtained from flight measurements of the maximum steady rate of roll, and the time taken for the angle of bank to reach 10 deg. from zero rate of roll.

If it is assumed that the maximum rate of roll is achieved by the time the angle of bank reaches 10 deg.,

and if ϕ_M = maximum angle of bank involved in the " S " turn

t_A sec = time to obtain 10 deg. of bank starting from zero rate of roll

p deg./sec. = maximum measured value of the rate of roll

t_S sec = time for the complete " S " turn,

then, if the time taken to arrest a rate of roll is neglected, it is easily shown that

$$t_S = 3t_A + (4\phi_M - 30)/p ,$$

and the assumed rate of roll $p' = 4\phi_M/t_S$.

Since it is assumed that there is no sideslipping, the radius of turn will be inversely proportional to the tangent of the angle of bank. Suppose that after t secs. the radius of the turn is R ft., the angle of bank is ϕ degrees and the change in direction of flight from the original path is θ degrees.

Then $$R = \frac{V^2}{g \tan \phi}, \quad \text{where } V = \text{airspeed in ft./sec.,}$$

$$V = R\frac{d\theta}{dt},$$

or $$\theta = V\int\frac{dt}{R},$$

and, if S/2 is the sidestep in feet obtained in the first half of the " S " turn,

$$S/2 = V\int_0^{t_{S/2}} \sin\theta dt.$$

This integration has to be performed in two stages due to the discontinuity of ϕ when $t = t_S/4$. In practice such discontinuities cannot occur, and a check has been made by estimating the sidestep distance, assuming that the ϕ vs t curve is built up from sine functions as shown by the dotted line of Fig. 12. This method resulted in a 10 per cent. increase in the estimated sidestep distance, but since the true ϕ vs t variation probably lies between these two cases the much simpler linear variation has been used in the following calculations and the results are therefore probably slightly pessimistic.

The results of the calculations for the Seafire and Barracuda are given in Figs. 11 (S) and 13 (B) for various values of ϕ_M between 10 deg. and 30 deg. The sidestep distance is plotted against forward distance moved, given by Vt_S, for values of t_S from 0 to 7 seconds. The curves of minimum forward distance, shown chain-dotted, are obtained from equation (1) of this Appendix, and show the maximum sidestep distance obtainable for a given forward distance, and the angle of bank employed, assuming that all the available aileron power is used. For a smaller sidestep distance, or a greater forward distance, full aileron power is not required.

TABLE 5

Aerodynamic Data—Seafire IIC

General

Mean weight during trials, lb.	6,600
S (gross wing area) sq. ft.	242
Engine	Merlin 46
Rated H.P. at 3,000 r.p.m. at 21,500 ft. and at + 9 lb./sq. in. boost pressure	1190
Power loading, lb./b.h.p.	5·55
Wing loading, lb./sq. ft.	27·3
Span loading, lb./sq. ft.	4·8
C.G. h (mean chord = S/span)	0·336 c
Airscrew diameter, ft.	10·25
Airscrew pitch, fine	30°
Airscrew pitch, coarse	65°
Gear ratio	0·477 : 1

Wings

Area (gross), S, sq. ft.	242
Span, $2s$, ft.	36·87
Mean chord, c, ft.	6·53
Aspect ratio	5·67
Dihedral	6°
Sweepback of $\frac{1}{4}c$ line	0°
Chord, ft., root	8·33
Chord, ft., tip	4·2
Section, root	NACA 2213
Section, tip	NACA 2203
Wing twist, root-tip	2·5°

Flaps

Type	Split
Maximum angle	85°
Flap area/S	0·065
Flap chord/local wing chord	0·12
Flap span/$2s$	0·445

Longitudinal Control

Tail surface area (gross), S' sq. ft.	32·9
Elevator area/S'	0·403
e'/c (e' = distance, C.G. to $\frac{1}{3}$ T.P. chord)	2·76
S'/S	0·136
Tail volume coeff., $S'e'/Sc$	0·375
Elevator angles (max.)	±28°
Type of balance	Horn
Percentage balance	9·1
Stick gearing, $d\eta/dx$, deg./in.	3·9
Trimming tab area (total), sq. ft.	0·86
Trimming tab angles (max.), down	10°
Trimming tab angles (max.), up	15°

Directional Control

Fin and rudder area, S'', sq. ft.	12·75
Rudder area/S''	0·607
e''/s (e'' = distance, C.G. to centroid of S'')	1·07
Fin and rudder volume coeff., $S''e''/Ss$	0·0563
Rudder angles (max.), port	28°
Rudder angles (max.), starboard	29·5°
Type of balance	Horn
Percentage balance	4·5
Pedal gearing, $d\xi/dx$, deg./in.	7·45
Trimming tab area, sq. ft.	0·38
Trimming tab angles (max.), port	13°
Trimming tab angles (max.), starboard	6°

Lateral Control

Type of aileron	Frise
Aileron area (total) sq. ft.	18·9
Aileron area/S	0·078
Aileron chord/local chord	0·235
Aileron span/$2s$	0·37
Aileron angles (max.), down	22°
Aileron angles (max.), up	25°
Percentage balance	27·5
Stick gearing, $d\xi/dx$, deg./in.	2·75

TABLE 6

Aerodynamic Data—Barracuda II

General

Mean weight during trials, lb.	12,200
S (gross wing area), sq. ft.	402
Engine—rated H.P. at 3,000 r.p.m. at 2,500 ft. and + 18 lb. per sq. in. boost pressure	1,645
Power loading, lb./b.h.p.	7·3
Wing loading, lb./sq. ft.	30·3
Span loading, lb./sq. ft.	4·96
C.G. h (mean chord = S/span)	0·338 c
Airscrew diameter, ft.	11·75
Airscrew pitch: fine	15°
Airscrew pitch: coarse	50°
Gear ratio	0·477 : 1

Wings

Area (gross), S, sq. ft.	402
Span, $2s$, ft.	49·2
Mean chord, c, ft.	8·17
Aspect ratio	6·0
Dihedral	1° 21′
Chord, ft.: root	10·08
Chord, ft.: tip	6·29
Section: Root	NACA 23021 at 4·5°
Section: Tip	NACA 23010 at 1·5°
Wing twist, root-tip	3·0°

Flaps

Type	Youngman
Maximum angle: diving	−32° up
Maximum angle: landing	+47° down
Flap area/S	0·13
Flap chord/local wing chord	0·29
Flap span/$2s$	0·53

Longitudinal Control

Tail surface area (gross), S', sq. ft.	69
Elevator area/S'	0·44
e'/c (e' = distance, C.G. to $\frac{1}{4}$ T.P. chord)	2·89
S'/S	0·17
Tail volume coeff., $S'e'/Sc$	0·50
Elevator angles (max.): up	30°
Elevator angles (max.): down	24°
Type of balance	Geared tab & horn
Percentage balance	28·3
Stick gearing, $d\eta/dx$, deg./in.	3·0
Trimming tab area (total), sq. ft.	1·49
Trimming tab angles (max.): down	11·5°
Trimming tab angles (max.): up	13°
Balance tab angles (max.): down	14°
Balance tab angles (max.): up	9·5°

Directional Control

Fin and rudder area, S'', sq. ft.	34·9
Rudder area/S''	0·435
e''/s (e'' = distance, C.G. to centroid of S'')	1·01
Fin and rudder volume coeff., $S''e''/Ss$	0·088
Rudder angles (max.): port	30°
Rudder angles (max.): starboard	30°
Type of balance	Geared tab & horn
Percentage balance	16·9
Pedal gearing, $d\xi/dx$, deg./in.	7·7
Tab area, sq. ft.: trimmer	0·833
Tab area, sq. ft.: balance	0·833
Trimming tab angles (max.): port	17°
Trimming tab angles (max.): starboard	8°
Balance tab angles	±13·5°

Lateral Control

Type of aileron	Frise
Aileron area (total), sq. ft.	37·8
Aileron area/S	0·094
Aileron chord/local chord	0·235
Aileron span/$2s$	0·39
Aileron angles (max.): down	15°
Aileron angles (max.): up	19°
Percentage balance: inboard	26·4
Percentage balance: outboard	35·0
Stick gearing, $d\xi/dx$, deg./in.	3·0 at centre

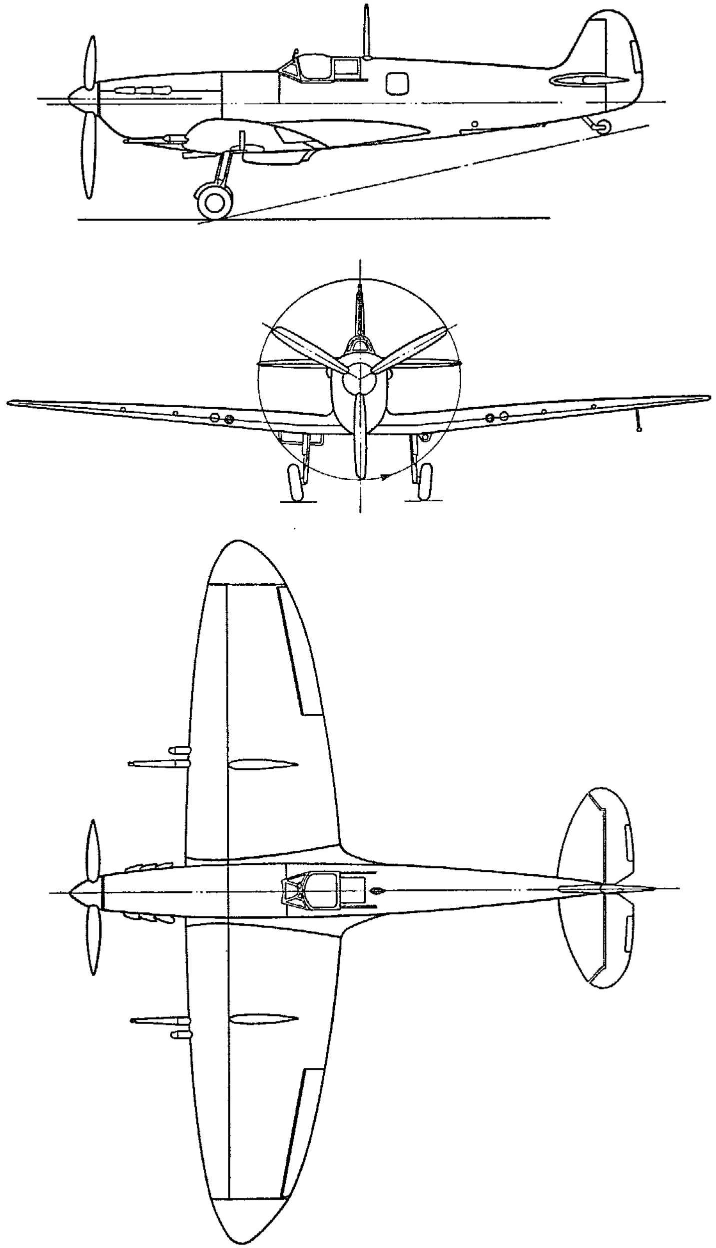


FIG. 1 (S). General Arrangement of Seafire II.

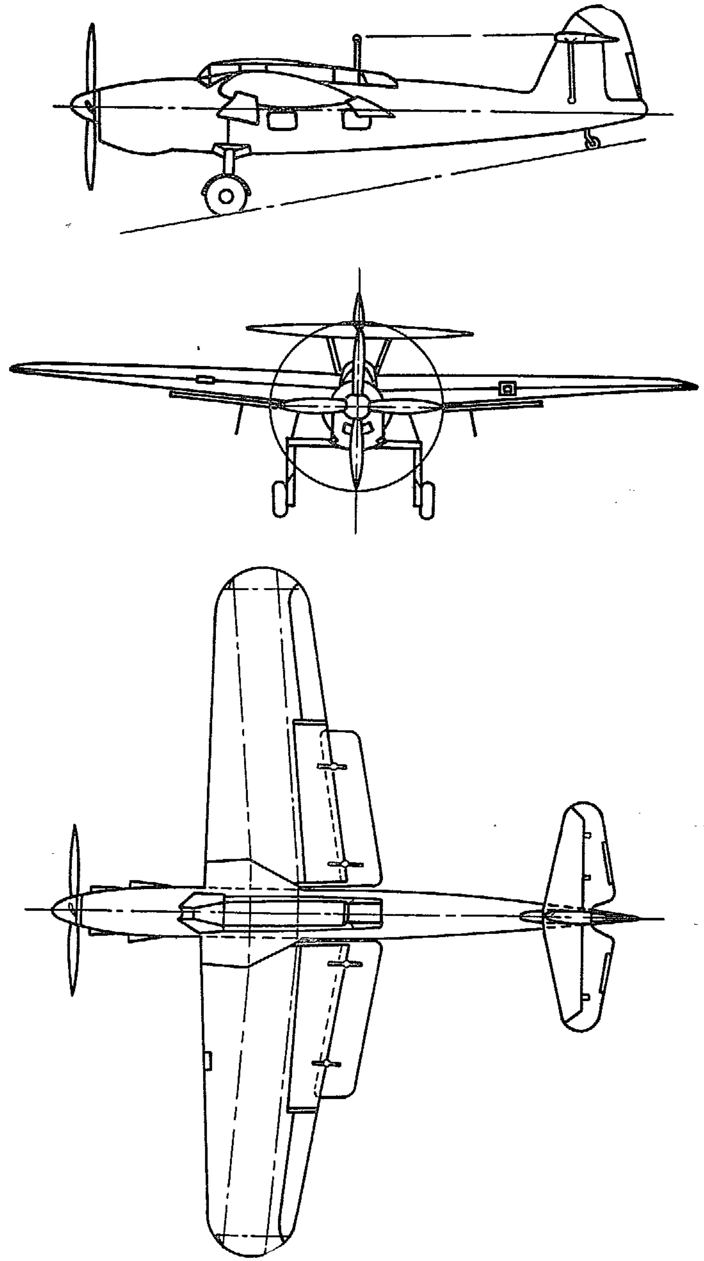


FIG. 1 (B). General Arrangement of Barracuda.

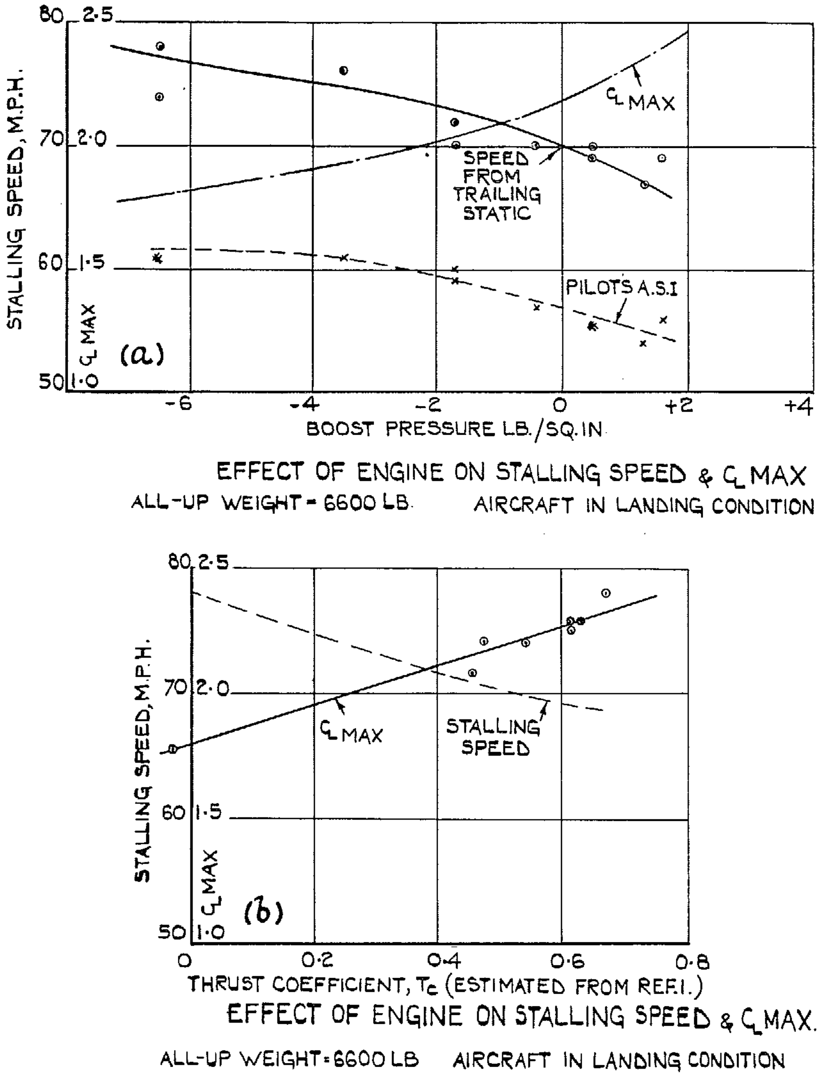


FIG. 2 (S). Stalling Speed and $C_{L\ max}$—Seafire MB.125.

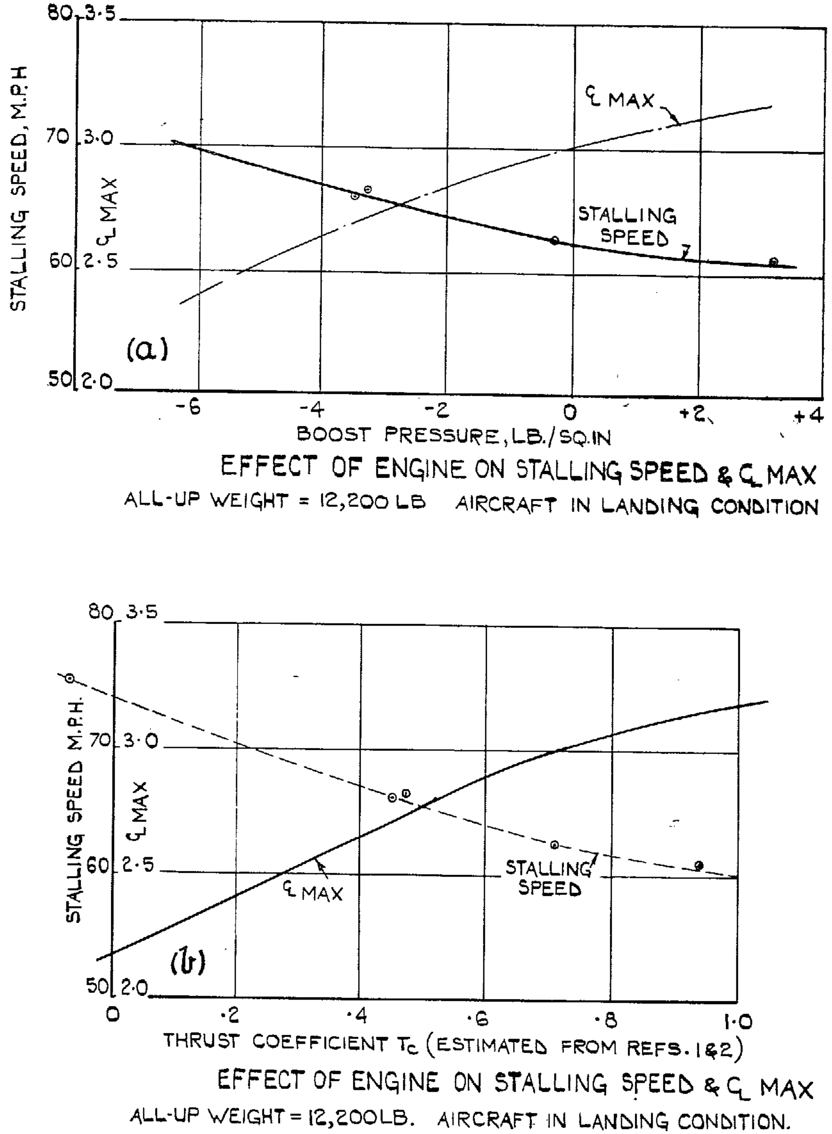


FIG. 2 (B). Stalling Speed and $C_{L\ max}$—Barracuda LS.540.

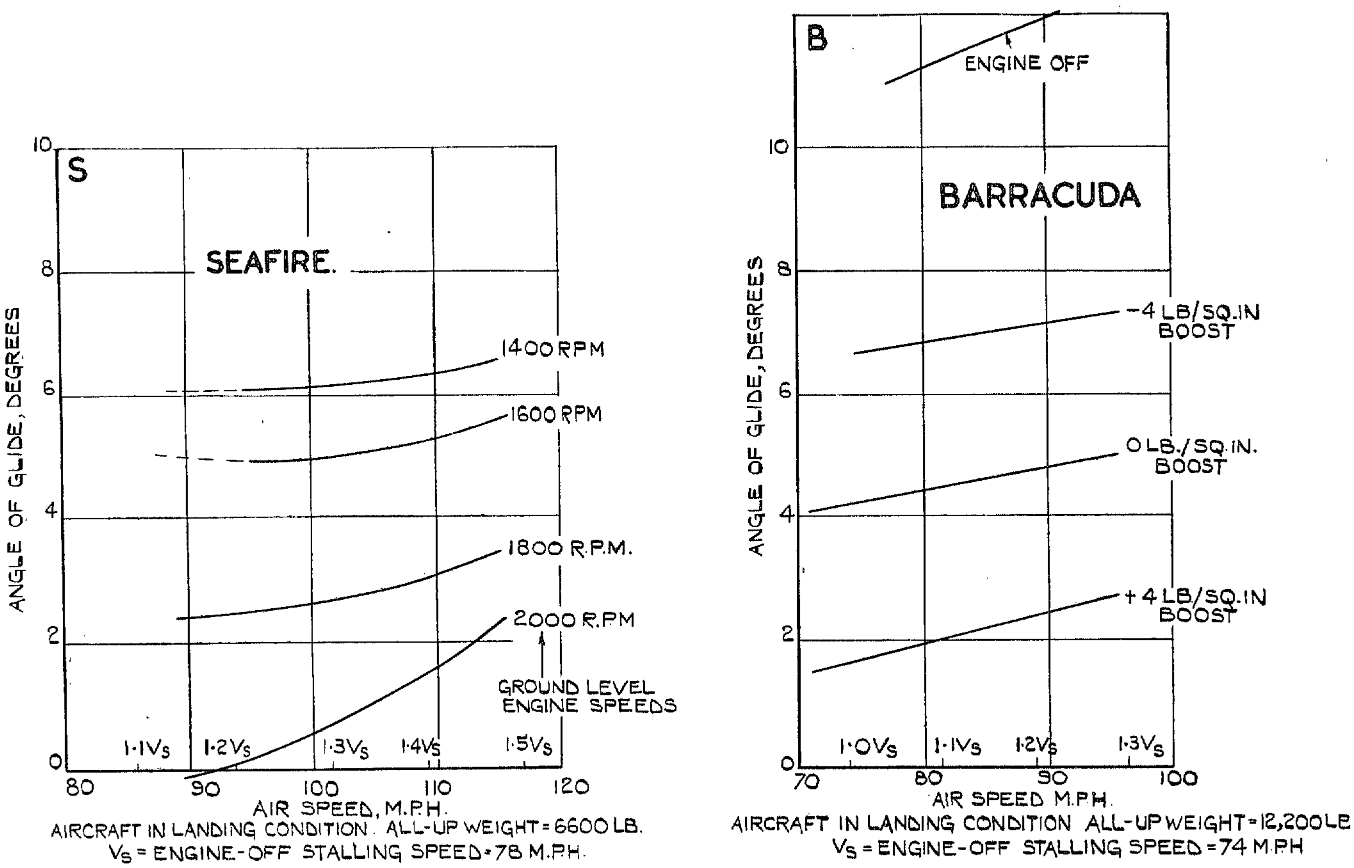
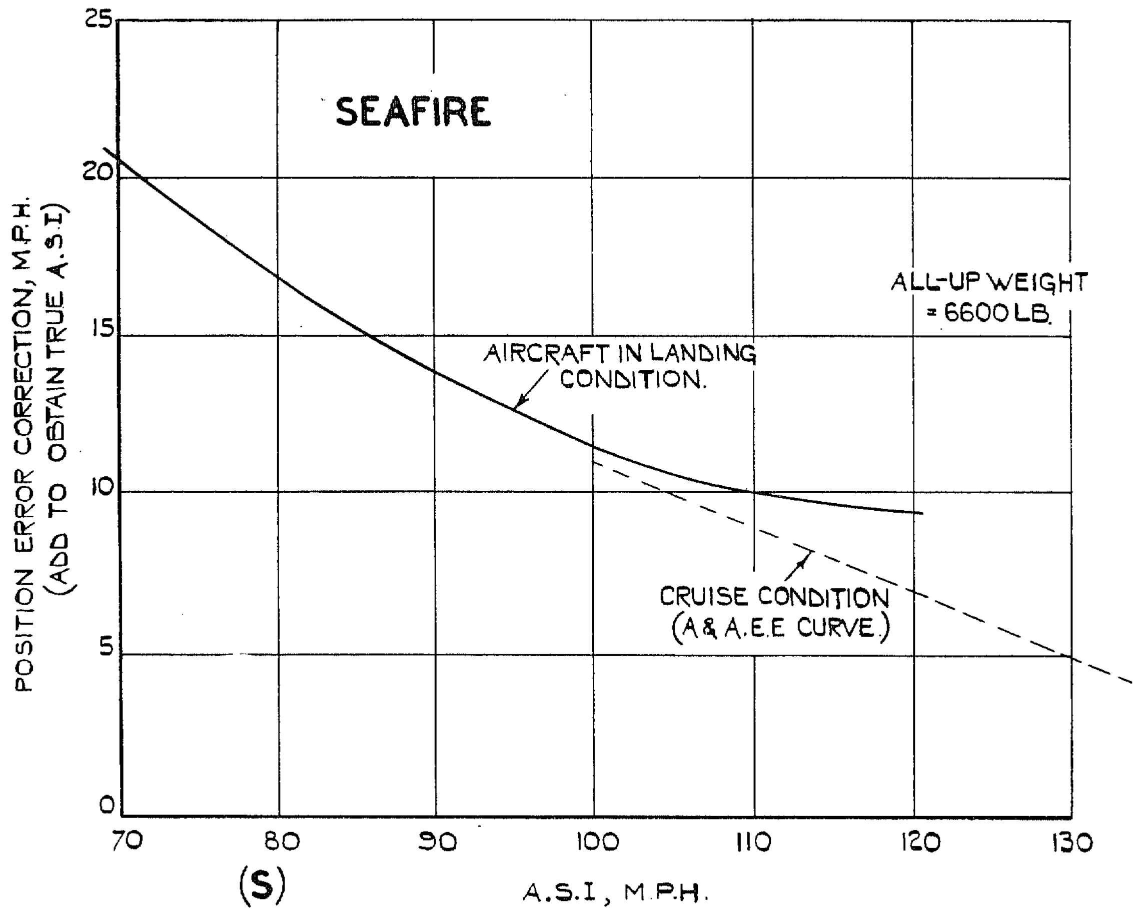


FIG. 3. Effect of Engine on Gliding Angle.



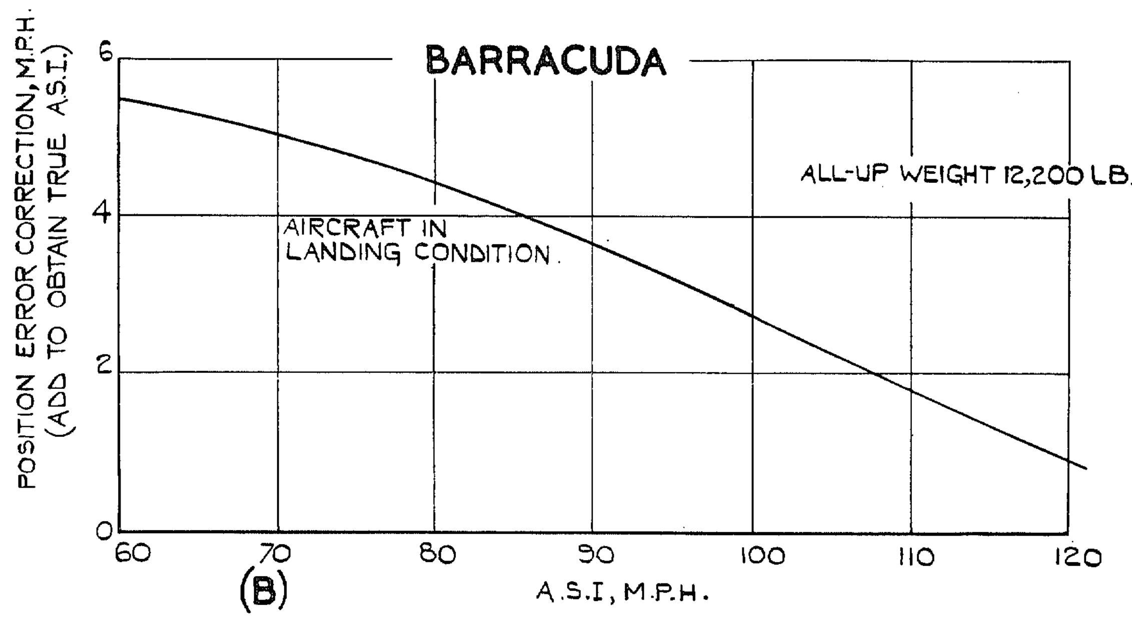


FIG. 5. Position Error Curves.

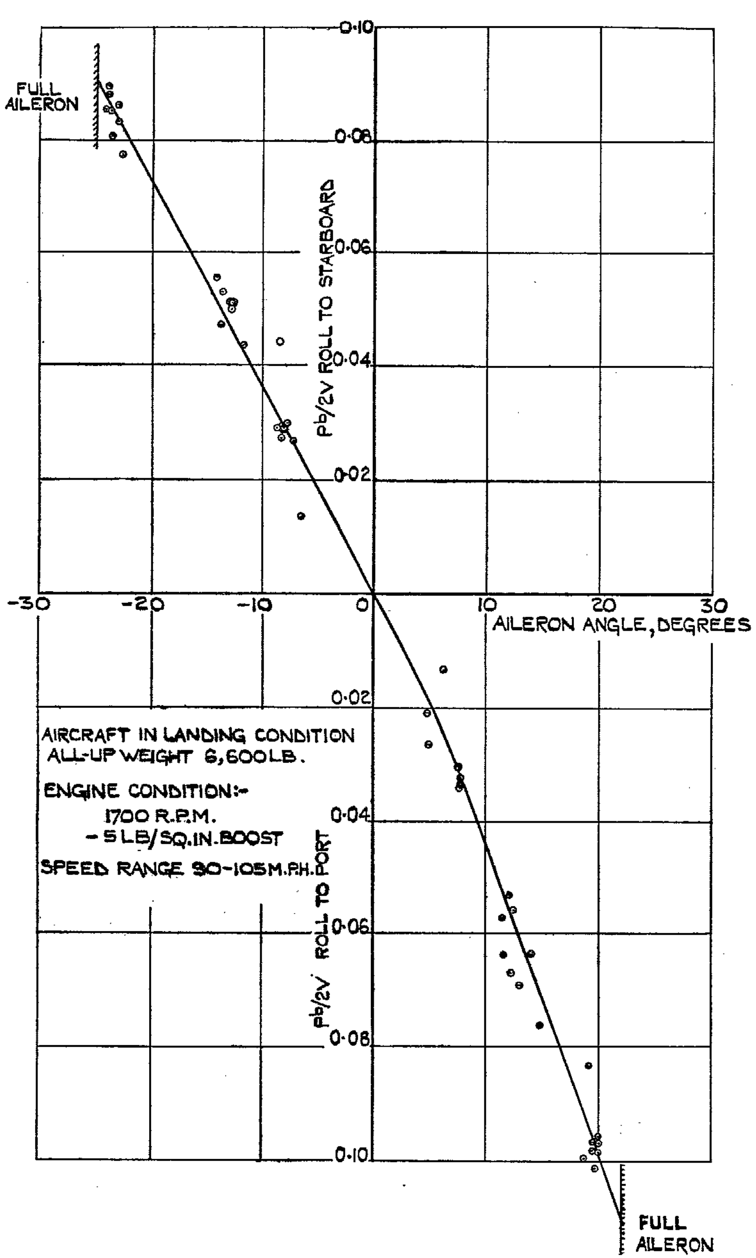


FIG. 7 (S). Variation of Rate of Roll with Aileron Angle—Seafire.

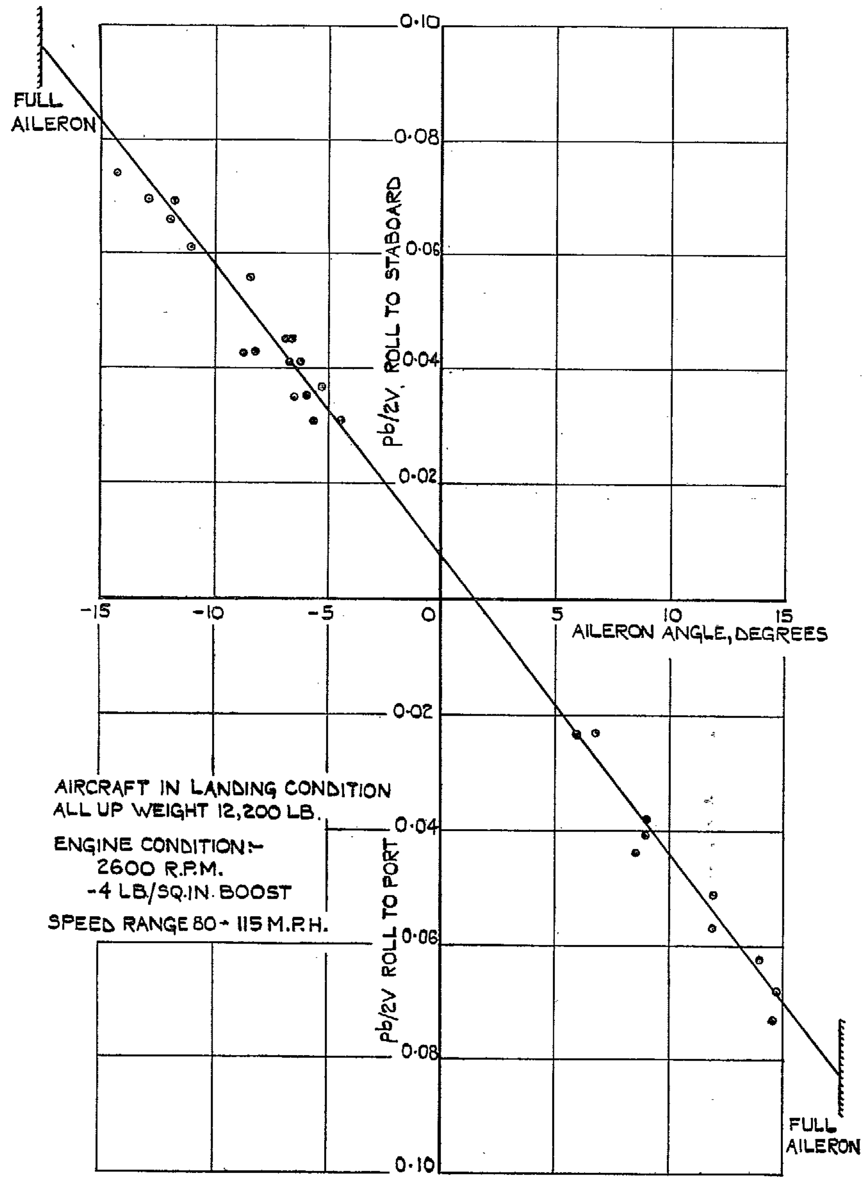


FIG. 7 (B). Variation of Rate of Roll with Aileron Angle—Barracuda.

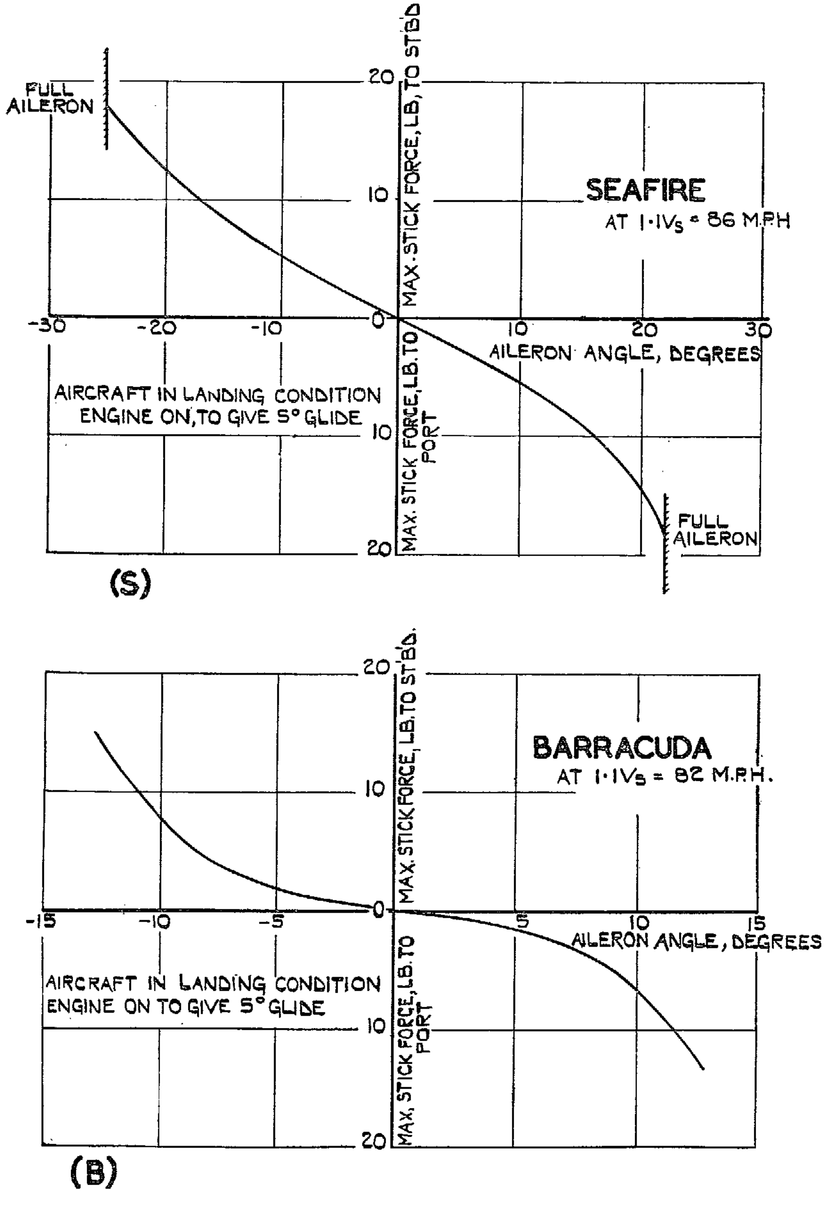


FIG. 8. Variation of Stick Force with Aileron Angle, Landing Condition, at $1 \cdot 10V_S$.

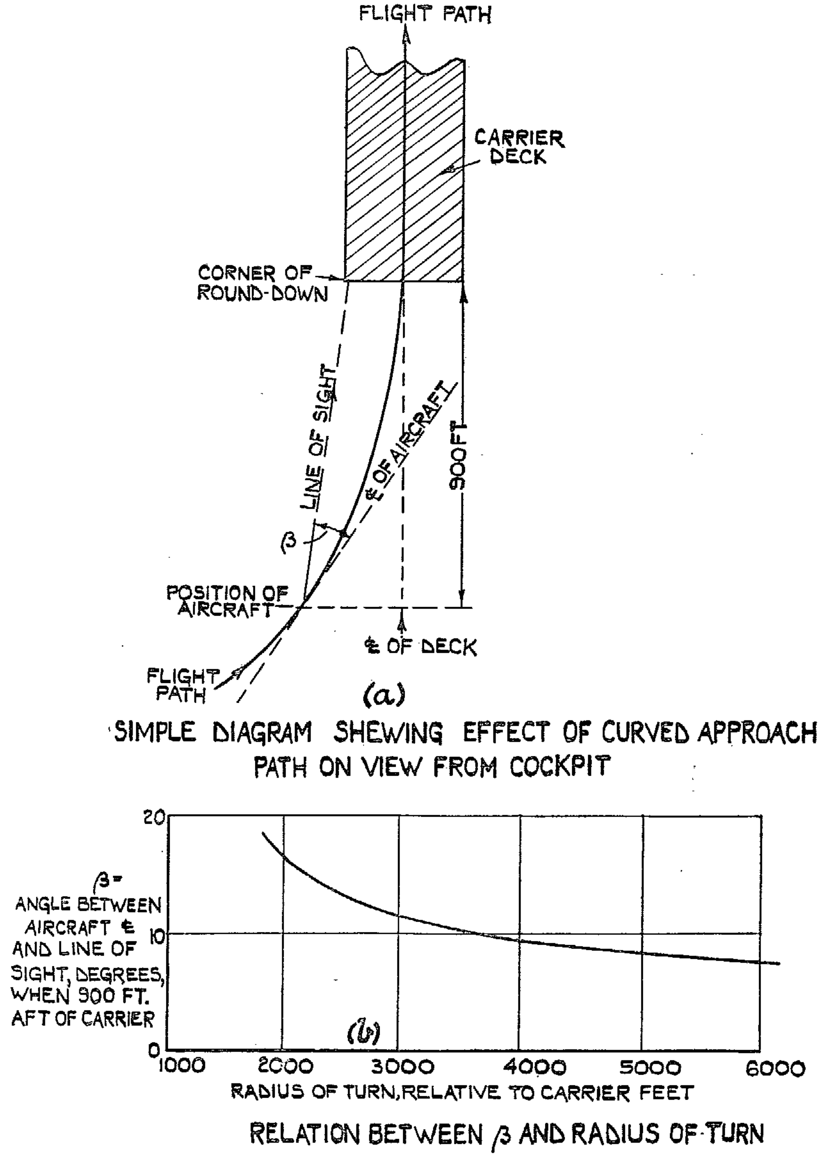


FIG. 9. Improvement in View Due to Use of Curved Approach Path.

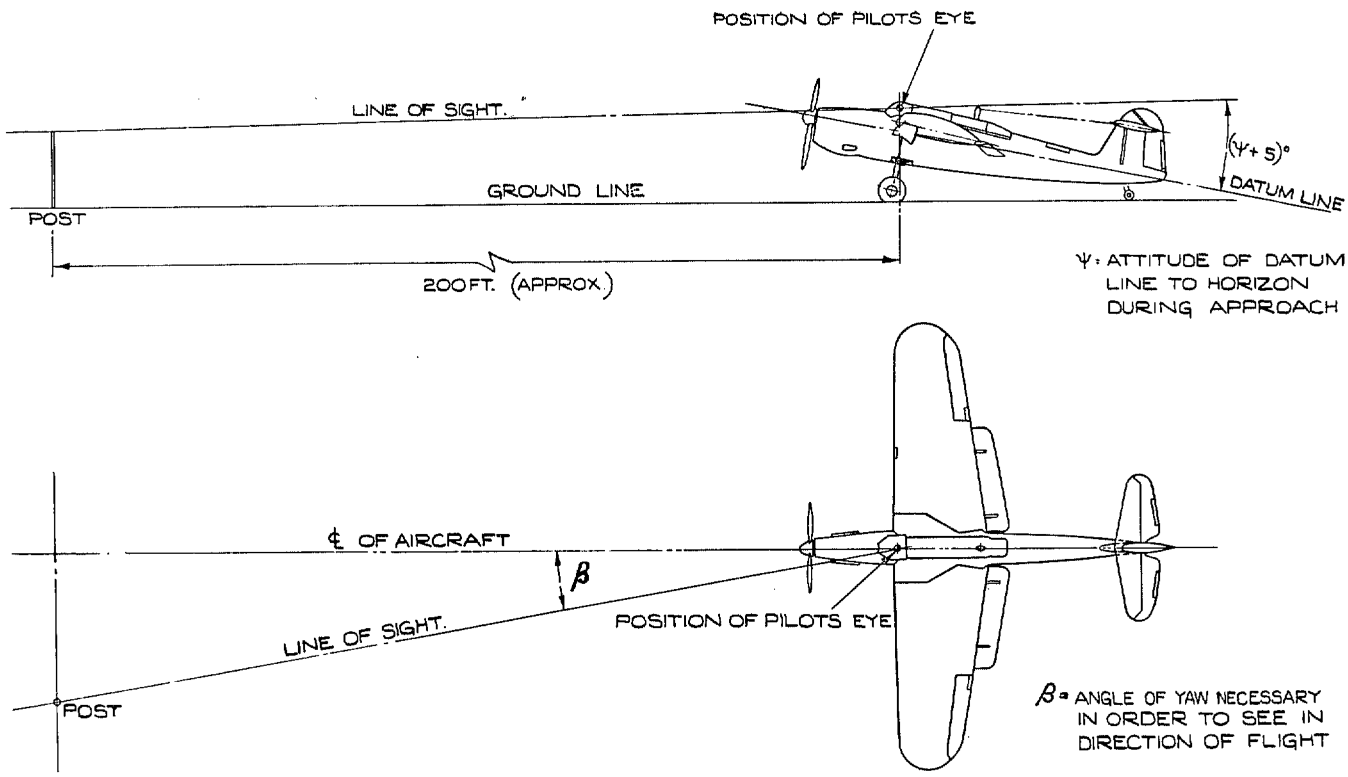


FIG. 10. Post Method for Assessing View from Cockpit.

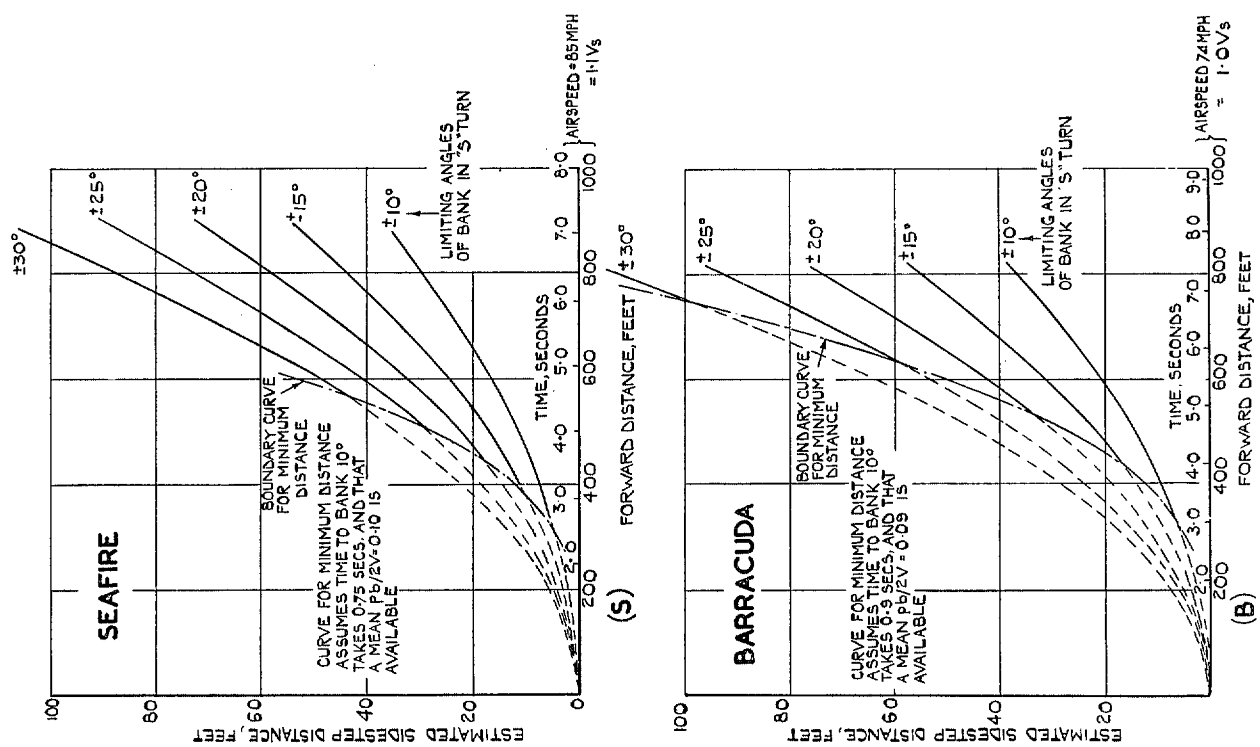


FIG. 11. Estimated Sidestep Distances for Complete "S" Turn.

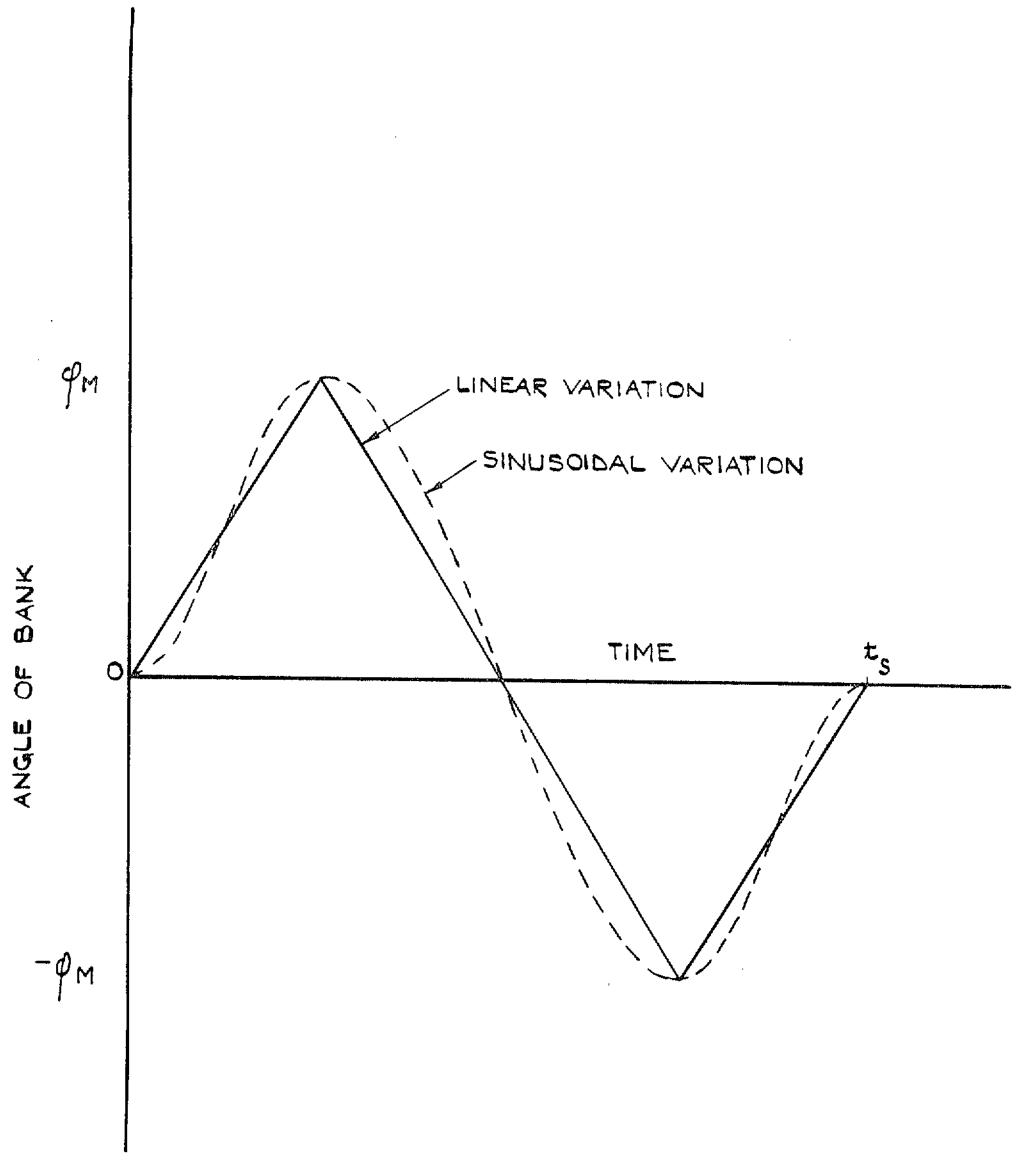


FIG. 12. Variation of Angle of Bank with Time during Sidestep Manoeuvre.

Part II.

Hellcat I and Avenger I

By

D. Lean, B.Sc., D. Johnson and J. R. Stott, B.Sc.

Summary.—The programme of tests described in Part I has been carried out on two further Naval aircraft—a Hellcat I and an Avenger I. The method of test and the results obtained are presented and discussed, and it is concluded that these two aircraft are subject to the same criticisms as the Seafire IIc and Barracuda II, though to a lesser degree. The recommendations made in Part I regarding modifications to the method of test and to some of the quantitative requirements are substantiated by the results obtained with these two aircraft.

1. *Introduction.*—This report is a record of a series of tests which have been carried out on two Naval aircraft—a Hellcat I and an Avenger I—to determine to what extent these aircraft conform to a proposed specification for deck-landing aircraft. Part I dealt with the results of the same tests performed on a Seafire IIc and a Barracuda II aircraft, and gave a full account of this proposed Aircraft Design Memoranda (A.D.M.), and the necessity for it, in an Appendix. The present report will, therefore, be confined to the results of the tests on the Hellcat and the Avenger, with a brief discussion of the more important results and their effect on the proposed standards.

2. *Description of Aircraft and Instrumentation.*—The Hellcat I is a single-engine single-seat monoplane fleet fighter built by the Grumman Aircraft Engineering Corporation, Long Island, New York. It is powered by a Double Wasp R–2800–10W engine driving a 13·08-ft. diameter three-bladed Hamilton Standard Hydromatic propeller. This aircraft was fully equipped for deck operation, and was flown at a mean weight of 11,750 lb. with the C.G. at 0·254 of the standard mean chord. The test flying occupied a total of about 15 hours.

The Avenger I is a single-engine three-seat monoplane torpedo-bomber built by the same company as the Hellcat. It is powered by a Cyclone R–2600–8 engine driving a 13·0-ft. diameter three-bladed Hamilton Standard Hydromatic propeller. This aircraft was also fully navalized, and was flown at a mean weight of 14,600 lb. with the C.G. at 0·27 standard mean chord. The test flying occupied a total of about 17 hours.

Three-view general arrangement drawings of both aircraft are shown in Figs. 1 (H) and 1 (A), and the relevant aerodynamic data given in Tables 9 and 10. Fig. 2 shows the location of the standard mean chord, C.G. position, etc. for the two aircraft.

The instrumentation was very similar to that already described for the Seafire and Barracuda. It will therefore suffice to give a list of instruments as fitted to both aircraft.

(1) A.S.I., connected to the standard aircraft system.

(2) A.S.I., connected to a venturi-pitot and a suspended static head.

(3) Altimeter, connected to the aircraft static system.

(4) Engine r.p.m. indicator } in parallel with the pilot's instruments.
(5) Manifold pressure gauge }

(6) Pitch gyro—a modified Mk. I artificial horizon with a vernier scale fitted.

(7) Desynn-operated indicators for measurement of

- (*a*) aileron angle,
- (*b*) rudder angle,
- (*c*) elevator angle,
- (*d*) elevator force *or* aileron force,
- (*e*) rudder force (port and starboard pedals),
- (*f*) throttle position,
- (*g*) rate of roll,
- (*h*) rate of yaw,
- (*i*) flap angle (Hellcat only).

These indicators were mounted on a single panel and photographed by a Bell and Howell Type A-4 35 mm. ciné camera, at a normal speed of 8 frames per second, though a speed of 16 frames per second was used for some tests.

Operation of the suspended static head was simple matter with the Avenger, but in the case of the Hellcat it was necessary, in order to avoid earlier difficulties, to make the operation of this instrument by the pilot as simple as possible. The static head, with 100 ft. of tubing, was stowed in a streamlined box carried under the starboard wing. It could be released to trail below the aircraft by pulling on a toggle in the cockpit, and, at the conclusion of the test, it could be jettisoned on a parachute by pulling a second toggle. This method worked well as regards release and jettisoning, but the head was usually damaged on hitting the ground, through using an insufficiently large parachute (54 in. diameter).

The provision of a Desynn indicator for the flaps on the Hellcat was made necessary by the fact that these flaps are spring loaded so that the maximum flap angle can only be maintained at a speed less than 86 knots. Above this speed the flaps start to retract automatically and progressively as the speed increased until an angle of 15 deg. is reached at 150 knots. The variation in flap angle with speed is shown in Fig. 3 (H). The resulting loss in C_L, also shown in this diagram, is estimated from the charts published by Young and Hufton[4] (1941).

3. *Test Procedure, and Method of Presentation of Results.*—The order in which the tests were carried out was essentially that proposed in the A.D.M. In some cases a more extensive investigation was made than was actually called for in the A.D.M., and the results are given under the appropriate heading.

In all cases, unless otherwise stated, the tests were done with the aircraft in the landing condition, *i.e.*, wheels and flaps down, hood open, oil cooler open, cowling gills shut and (for Hellcat only) inter-cooler shut. The propeller speed control lever was always set for maximum r.p.m., hence the propeller was normally in fully fine pitch.

The "standard" approach condition proposed in the A.D.M. (*i.e.* at 1·0 times engine-off stalling speed, with sufficient engine to make this speed equal to 1·15 times engine-on stalling speed, the resulting angle of glide being at least 5 deg. relative to the air) was adhered to as closely as possible. Where applicable, tests were done on the Hellcat at about 1·03 times engine-off stalling speed in a 5 deg. glide; with the Avenger, the initial speed was usually about 1·10 times engine-off stalling speed in a 5 deg. glide, the behaviour of this aircraft being somewhat uncertain at lower speeds, except in perfectly calm air, due to a reversal of its wing-dropping tendency as the stall is approached (*see* section 4.1). Where appropriate, the results have been reduced to a speed of 1·00 times engine-off stalling speed.

4. *Results of A.D.M. Tests.*—4.1. *Effect of Engine on Stalling Speed.*—Stalling speeds were measured with the aircraft in the landing condition, using the venturi pitot and suspended static-head method, at a wide range of engine powers. Results are shown plotted in Figs. 4 (H) and 4 (A). These figures show (*a*) the variation of stalling speed with engine power as represented by manifold pressure and/or throttle position and (*b*) the variation in maximum lift coefficient with thrust coefficient.

Considering the Hellcat first, the stalling speed with the throttle fully closed is 74 knots (85 m.p.h.), whereas the stalling speed at zero thrust is 71 knots (81·6 m.p.h.). This latter figure is taken as the engine-off stalling speed at the mean weight during the tests, viz.: 11,750 lb., corresponding to a maximum lift coefficient of 2·06. Variation of stalling speed with throttle setting and manifold pressure is shown in Fig. 4 (H), from which it is seen that the stalling speed can be depressed by about 7 m.p.h. per inch of throttle movement, *i.e.* by about 12 m.p.h. at 30 in. mercury manifold pressure. The increase in maximum lift coefficient is almost exactly equal to the thrust coefficient.

It is interesting to note that, with wheels and flaps up, a maximum lift coefficient of about 1·64 was obtained with engine off. This is an exceptionally high value for the maximum lift coefficient without flaps, although it agrees well with model results on the plain wing, published by Jacobs, Pinkerton and Greenberg[8] (1936), at a Reynolds number of about 8 million compared with the full scale Reynolds number of about 6 million. This section, and that on the Avenger, has 1·8 per cent. camber at 0·15*c*, the thickness varying from 15 per cent. to 9 per cent. between root and tip.

The stall with flaps down, engine on, is described as gentle, but without warning. The right wing drops.

In the case of the Avenger, the stalling speed, engine off (*i.e.* at zero thrust), is 66 knots (76 m.p.h.) at 14,600 lb., giving a maximum lift coefficient of 2·02. The variation of stalling speed with manifold pressure is shown in Fig. 4 (A), the rate of variation being approximately the same as for the Hellcat. The stalling speed is reduced by about 11 m.p.h. when the engine is opened up to 30 in. of mercury manifold pressure. The increase in maximum lift coefficient is equal to 0·65 times the thrust coefficient.

With wheels and flaps up, a maximum lift coefficient of about 1·63 is obtainable, the increment in maximum lift due to flaps being 0·39. Here again, the measured value of $C_{L\max}$ is in agreement with the model tests referred to above.

With flaps down, engine on, the stall is preceded by a slight drop of the starboard wing, but at the stall the aircraft flicks over to port. This reversal makes lateral control rather difficult at speeds very close to the stall.

Values of thrust coefficient have been estimated from the charts contained in a report by McHugh and Pepper[9] (1944).

The proposed A.D.M. requirement is that the stalling speed, engine-off, should be less than 75 knots.

4.2. *Effect of Engine on Gliding Angle.*—As in the case of the tests on the Seafire and Barracuda, a comprehensive set of rate-of-descent measurements was made over a range of speeds and engine conditions, instead of at single measurement at the " standard " condition (*see* section 3).

Angles of glide in the range 0 deg.–12 deg. were obtained at speeds from the stall up to 1·3 times engine-off stalling speed with the Hellcat and up to 1·5 times engine-off stalling speed with the Avenger. Fig. 5 (H) shows the variation of angle of glide with airspeed at various constant values of engine r.p.m. in the case of the Hellcat. Fig. 5 (A) gives the variation of angle of glide with speed at constant manifold pressures for the Avenger. Due to the comparatively high drag and low fine-pitch setting of the propeller on this aircraft, constant speeding begins at fairly steep angles of glide, so that lines of constant r.p.m. would not cover the required range with the propeller in fine pitch.

A clearer picture of the pilot's control over the angle of glide and the safety margin above the engine-on stalling speed is obtained by plotting the gliding angle against the ratio V/V_{ES}, where V is the airspeed and V_{ES} is the engine-on stalling speed at the same manifold pressure as used during the glide. Diagrams of this type for the Hellcat and the Avenger are given in Figs. 6 (H) and 6 (A) respectively. In Fig. 6 (H) the angle of glide is given (*a*) at constant engine r.p.m. (ground level equivalent), (*b*) at constant airspeed expressed as the ratio V/V_S, where V_S is the engine-off stalling speed, and (*c*) at constant throttle setting, measured in inches travel from the closed position. The diagram for the Avenger is similar, except that the lines of constant manifold pressure are omitted to avoid confusion with the lines of constant throttle setting.

If the angle of glide, and the safety margin above engine-on stalling speed are specified, then these diagrams show (*a*) the actual airspeed as a fraction of the engine-off stalling speed, and (*b*) the required engine condition in terms of throttle setting and engine r.p.m. In general, if any two of these quantities are known, then these diagrams enable the remaining quantities to be determined.

The " standard " approach condition proposed in the A.D.M. (*see* section 3), is represented by Point " A " on both diagrams. In the case of the Hellcat, it can be seen that a safety margin represented by $V/V_{ES} = 1 \cdot 15$ at a speed of $1 \cdot 0\ V_S$ would result in an angle of glide of about $1 \cdot 5$ deg., instead of the specified 5 deg. An angle of glide of 5 deg. at a speed of $1 \cdot 0\ V_S$ gives a safety margin of only $6 \cdot 5$ per cent. above engine-on stalling speed. Preliminary analysis of records obtained with this aircraft during landings on H.M.S. *Pretoria Castle* suggests that an average approach speed is in the region of $1 \cdot 05\ V_S$ with an angle of glide of about $3 \cdot 7$ deg. relative to the air as a point " B ", Fig. 6 (H). This has an important bearing on view from the cockpit (*see* section 4.3 below).

In the case of the Avenger, the " standard " approach condition (point " A ", Fig. 6 (A)), results in an angle of glide of about $4 \cdot 5$ deg., in other words, the A.D.M. requirement is very nearly satisfied. Point " B " on Fig. 6 (A) represents the average approach condition employed during landings on H.M.S. *Pretoria Castle, i.e.* an approach speed of $0 \cdot 98\ V_S$ at an angle of glide of $3 \cdot 8$ deg.

From diagrams 6 (H) and 6 (A) it is possible to extract more direct information on the way in which the pilot can adjust his angle of glide by means of the throttle. This is illustrated in Figs. 7 (H) and 7 (A) which show the variation in gliding angle with throttle setting at constant speeds up to $1 \cdot 2\ V_S$. The throttle controllability factors, in inches of throttle movement per degree change in angle of glide, are $0 \cdot 134$ and $0 \cdot 102$ for the Hellcat and Avenger respectively, roughly independent of speed, at a mean angle of glide of 4 deg. in the approach.

Position error curves, necessary for the analysis of the above tests, are given in Figs. 8 (H) and 8 (A) for Hellcat and Avenger respectively. A bad feature with this Hellcat is the variation of position error with engine power, particularly at low speeds. This is probably due to the use of a static vent on the thrust line, not far from the trailing edge of the wing (*see* Fig. 2 (H)). Later aircraft have reverted to the conventional pitot-static head.

4.3. *View from the Cockpit.*—The view from the cockpit has been assessed from a set of pinhole-camera photographs, taken by the method described in Appendix II of Part I. Typical photographs are reproduced in Figs. 9 (H) and 9 (A).

For the Hellcat, the angle between line of sight and wing chord has been obtained for two approach conditions. The first is the closest practicable approximation to the " standard " condition of the A.D.M. (*see* section 3) specified by an angle of glide, relative to the air, of 5 deg. at a speed of $1 \cdot 0\ V_S$. From the rate of descent measurements, the attitude of the datum to the horizontal in this condition is found to be $10 \cdot 0$ deg. Assuming a wind speed over the carrier deck of 25 knots, the angle of the approach path relative to the carrier is $7 \cdot 7$ deg., so that the angle between the line of sight and the wing chord is $17 \cdot 7$ deg. The corresponding angle for the second (*i.e.* the average) approach condition (a speed of $1 \cdot 05\ V_S$ at an angle of glide of $3 \cdot 7$ deg. relative to the air) is 13 deg. Two positions of the pilot's left eye were investigated, (*a*) with head central, and (*b*) with head moved $4 \cdot 7$ in. across to port, which is the maximum comfortable distance allowed by the straps.

In the first of these approach conditions (approximating to the " standard " condition) the angle of yaw necessary in order to be able to see the port corner of the carrier round-down from dead astern at a range of 300 yards is $14 \cdot 9$ deg. when the pilot's head is central, falling to $7 \cdot 3$ deg. when he moves his head the maximum comfortable distance across the cockpit. In the second condition, the required angle of yaw is 7 deg. with the pilot's head central, or 3 deg. when he moves his head fully across the cockpit.

The chosen positions of the pilot's eye are those corresponding to a pilot of average height (about 5 ft. 11 in.) with the seat in the fully raised position. The view will naturally be critically dependent upon this height.

For the Avenger, two approach conditions have been investigated. The first is the " standard " A.D.M. condition (specified by $V/V_S = 1 \cdot 00$ with an angle of glide of 5 deg. relative to the air). Assuming a 25-knot wind speed over the carrier deck, as before, the angle between the line of sight and the datum is 17 deg., and in this condition, the angle of yaw required in order to be able to see the port corner of the carrier round-down from dead astern at a range of 300 yards is 16 deg. when the pilot's head is central, or 7 deg. when he moves his head $5\frac{1}{4}$ in. across the cockpit, which is the maximum comfortable displacement permitted by the straps. The second condition is the average of about 20 landings on H.M.S. *Pretoria Castle*, and is given by $V/V_S = 0 \cdot 98$ at an angle of glide of 3·8 deg., the angle between the line of sight and the datum being 15·5 deg. In this condition, 12 deg. of yaw would be required in order to be able to see the port corner of the carrier round-down at 300 yards range when the pilot's head is central. This angle is reduced to 5 deg. if he moves his head 5 in. across the cockpit—the maximum allowed by his straps. The pilot is again assumed to be of average height. It should be noted that the proposed A.D.M. requires the pilot to be able to see the port corner of the round-down from dead astern at a range of 300 yards when seated with head central, the aircraft being headed straight towards the carrier.

4.4. *Alternative Requirements*.—A. *Direct measurement of sidestep distance*.—The direct measurement of the rate at which the flight path can be displaced laterally was not attempted on either aircraft, for lack of a suitable simple, yet safe, technique. However, calculations have been made of the lateral displacement produced by a correctly banked continuous S-turn, based on the measured time to produce 10 deg. of bank and the measured maximum steady rate of roll available. Then, if a maximum forward distance (*i.e.* a maximum time) is specified, the maximum allowable angle of bank in the complete S-turn is fixed by the rolling power of the ailerons, and hence the maximum lateral displacement can be estimated. The method of calculation is described in Appendix III of Part I.

These calculations have been made for both aircraft for an approach speed of $1 \cdot 0\ V_S$, with maximum angles of bank of from 10 deg. to 30 deg. Figs. 10 (H) and 10 (A) show the relation between forward distance moved and sidestep distance, assuming full use is made of the available aileron power. At any higher forward speed the attainable sidestep distance will be less, since it can be shown that for a given aircraft the maximum angle of bank that can be used for an S-turn in a given forward distance is independent of the speed of flight, but the time taken is inversely proportional to the speed, and the sidestep distance is approximately proportional to the square of the time taken to complete the S-turn.

The A.D.M. requirement (25-ft. sidestep in 2,000 ft. forward distance) is easily met, in fact, a sidestep distance of 25 ft. can be achieved in a forward distance of between 500–600 ft. for either aircraft at $1 \cdot 0\ V_S$. For a higher speed, the sidestep distance is reduced by dividing by a factor $(V/V_S)^2$.

B. (i) *Time to reverse* 30 *deg. of bank*.—This test, and the succeeding one (B. (ii)) were originally proposed to provide data for the calculation of rate of sidestep (*see* section 4.4A above) but it has been found that the time to reverse 30 deg. of bank can be estimated very closely from the measured time to attain 10 deg. of bank from the start of the aileron movement and the measured maximum steady rate of roll (*see* section 4.5 (i) below). The estimate for the Hellcat is slightly (about 10 per cent.) less than the measured time, since the estimate takes no account of the time taken to arrest the roll.

It has been shown in section 4.4 (A) above that the sidestep distance can be calculated simply from the measured time to apply 10 deg. bank and the maximum steady rate of roll. Tests to measure such quantities as time to reverse 30 deg. bank, rate of flat turn, etc., are not necessary for this calculation and have only been included here for the sake of completeness.

The time to reverse 30 deg. of bank was, however, measured in two ways. Firstly, the aircraft was trimmed to fly straight with wings level in the condition most nearly approaching the " standard " condition (section 3). It was then banked rapidly to 30 deg. in one direction;

this bank was then reversed to 30 deg. in the opposite direction as rapidly as possible, and then the aircraft was brought back to straight flight once more. The time from maximum bank in one direction to maximum bank in the other direction, and the stick force which the pilot has to employ during the reversal were both measured. In the second method, the aircraft was first trimmed to fly in a steady 30 deg. banked gliding turn in the " standard " condition, and this bank was reversed as rapidly as possible to a steady 30 deg. in the opposite direction. The time, from the start of the ailron movement till a steady 30 deg. opposite bank had been attained, and the maximum force employed by the pilot, were again measured. These times have been corrected to a speed of $1 \cdot 0\ V_S$, using full aileron and applying exactly 30 deg. bank, by simple proportion.

In the case of the Hellcat, the times obtained by the two methods agree quite well, the first giving $3 \cdot 2$ sec., the second giving $3 \cdot 5$ sec. for the manoeuvre. The second value ($3 \cdot 5$ sec.) includes the time taken to apply the aileron. The stick forces have been corrected to $1 \cdot 0\ V_S$ at full aileron, assuming a square law relation for speed and aileron angle. This method of correction is justifiable only because the speed differences are small, about $0 \cdot 1\ V_S$. The first method gives the maximum force required as 21 lb. The second method requires a force of 31 lb. A large part of these forces is probably due to inertia forces, or lack of favourable response effect, the steady stick force for full aileron at this speed being about 5 lb.

For the Avenger, only the first of the two methods was investigated, *i.e.* with the bank varying from 0 deg. to 30 deg. in one direction then reversing to 30 deg. in the opposite direction as rapidly as possible. The measured time for reversal from 30 deg. in one direction to 30 deg. in the other direction, reduced to a speed of $1 \cdot 0\ V_S$, with full aileron, is $3 \cdot 4$ sec., the force required being 20 lb. This time is rather less than the estimated time, based on the time to bank 10 deg. and the steady rate of roll, since the speed tends to increase during the manoeuvre, thereby increasing the rate of roll, and, in addition, the measured time does not include the time to apply the aileron. The increase in speed in the case of the Hellcat was less pronounced than in the case of the Avenger, so that the estimated and measured times agree more closely.

The A.D.M. requirement is that the manoeuvre should take less than $1 \cdot 5$ sec., the stick force being less than 10 lb.

B. (ii) *Maximum rate of flat turn.*—The maximum steady rate of flat turn was measured on both aircraft, in both directions, using full rudder, the aircraft being trimmed to fly straight in the condition most nearly approaching the standard condition (section 3). The pedal forces were also measured.

With the Hellcat, flat turns to starboard could not be maintained, the turn rapidly deteriorating into a straight sideslip. The maximum rate of turn to starboard reduced to a speed of $1 \cdot 0\ V_S$ was only 80 deg. per minute, the corresponding pedal force being 130 lb. The maximum rate of flat turn to port at $1 \cdot 0\ V_S$ was found to be 230 deg. per minute, the corresponding rudder force being 140 lb. The reason for this asymmetric turning performance is that 13 deg. of starboard rudder are required to maintain straight flight in this condition. This leaves only 17 deg. extra rudder travel for turns to starboard, whereas for turns to port, 43 deg. of travel are available. Even with full starboard rudder trim, a pedal force of over 30 lb. is required to maintain straight flight.

For the Avenger, rates of flat turn (reduced to $1 \cdot 0\ V_S$) of 160 and 280 deg. per minute were obtained to starboard and port respectively, the corresponding pedal forces being 109 and 96 lb. Here again, the rate of turn decreased after the initial yawing motion following displacement of the rudder, though it did not show a progressive decrease as with the Hellcat. The difference in the rates of turn in the two directions is probably due to the fact that 9 deg. of starboard rudder are required for straight flight in this initial condition.

The A.D.M. requirement is that the rate of turn shall be greater than 180 deg. per minute, and the rudder force less than 100 lb.

4.5. *Control during the Final Stage.*—(i) *Time to bank* 10 *deg.*—With the aircraft trimmed to fly straight in the usual initial condition (*see* section 3), full aileron was applied at the maximum possible rate. The variation of angle of bank with time was obtained either from the inclination of the vertical bar of the pitch gyro or from integration of the rate-of-roll record. The time to bank 10 deg. was measured from the start of the aileron movement.

With the Hellcat, 10 deg. of bank can be applied in 0·85 sec. at 1·0 V_S, the force required being 24 lb. In the case of the Avenger, the time is 1·20 sec., and the stick force 18 lb. In both cases, a large proportion of the stick force required is due to aileron inertia. The corresponding A.D.M. figures are 0·75 sec. and 5 lb. stick force.

In addition to these tests, a full set of rate-of-roll measurements was made on both aircraft. The steady rate of roll was measured, following the application of various amounts of aileron. Tests were done at several speeds, both in the landing condition and in normal cruising conditions. Results are given in Figs. 11 (H) and 11 (A) for the Hellcat and Avenger respectively. Rate of roll is given in terms of the ratio $pb/2V$. Maximum values of $pb/V2$ obtainable with the Hellcat are 0·08 to port and 0·06 to starboard. Corresponding figures for the Avenger are 0·07 to port and 0·084 to starboard. Maximum values of stick force, reduced to 1·0 V_S, are plotted against aileron angle in Figs. 12 (H) and 12 (A). These maximum stick forces do not include the effect of aileron inertia, nor any lightening of force due to response.

4.5. (ii) *Change in trim on Closing the throttle.*—The change in trim on closing the throttle was investigated in two ways, one method being dynamic in character, the other static. For the first method, the aircraft was trimmed to fly straight in the " standard " condition (*see* section 3). The throttle was then closed and the pilot attempted to maintain the vertical acceleration between 0·8 and 1·0*g*, this acceleration being indicated on a direct-reading accelerometer hanging in the cockpit. Automatic observer records were obtained during the 3–4 sec. period following the closing of the throttle, and from these records the change in control angles and forces, and the change in attitude and speed could be obtained.

The second method of test involved the same initial conditions as for the dynamic method. After closing the throttle, the pilot increased his speed by about 10 per cent. of the stalling speed and when a steady state had been attained, without re-trimming, a record was taken of control angles and forces, etc. A record of the initial conditions was obtained before closing the throttle, and hence the change in the various quantities could be obtained.

The results are given in Tables 7 and 8 below (section 4·5 (iii)) for the Hellcat and Avenger respectively. The figures given are the mean of several tests by each method.

4.5. (iii) *Change in trim on opening the throttle.*—The change in trim on opening the throttle for a baulked landing was also investigated in two ways, dynamic and static. Starting from the usual initial condition, the dynamic method required the pilot to keep the vertical acceleration between 1·0 and 1·2*g* after opening the throttle to full take-off power. From the automatic observer record, the change in control angles and forces, and the change in attitude and speed could be obtained.

The static method of test required that, after opening the throttle, the pilot should fly the aircraft steadily at the same airspeed as initially, without re-trimming. From automatic observer records of the initial and final steady conditions, the changes in the relevant quantities could be obtained. Collected results, for both aircraft are given in Tables 7 and 8 below, the figures being, as before, the mean of several tests by each method.

No figures are given for change in lateral trim, but the change in aileron angle never exceeded 3 deg.

TABLE 7

Change in Trim Tests, Hellcat JX.815

Description of Test	Initial Speed	Initial Engine Condition (ground level)	Change in :—					
			Elevator Angle	Elevator Force	Rudder Angle	Rudder Force	Datum Attitude	Speed
Closing throttle, keeping vertical acceleration between 0·8 and 1·0 g.	73 knots ($1·03\ V_s$)	1870 r.p.m. 18 inches Hg Boost	2·0° up	4·5 lb. pull	6·1° port	19 lb. port	2·0°/sec. nose down	—1·0 knots/sec. (deceleration)
Closing throttle, increasing speed by approx. $0·1\ V_s$.	73 knots ($1·03\ V_s$)	1870 r.p.m. 17·5 inches Hg Boost	2·1° up	7·3 lb. pull	10·3 port	30 lb. port	7·7° nose down	+7·4 knots ($0·11\ V_s$)
Opening throttle, keeping vertical acceleration between 1·0 and 1·2g.	73 knots ($1·03\ V_s$)	1930 r.p.m. 19 inches Hg Boost	4·3° down	9·5 lb. push	2·4° starboard	72 lb. starboard	0	+4·7 knots/sec (acceleration)
Opening throttle, flying at same speed as initially	73 knots ($1·03\ V_s$)	1930 r.p.m. 19 inches Hg Boost	3·9° down	9·5 lb. push	13·4° starboard	55 lb. starboard	9·2° nose up	0

TABLE 8

Change in Trim Tests, Avenger JZ.640

Description of Test	Initial Speed	Initial Engine Condition (ground level)	Change in :—					
			Elevator Angle	Elevator Force	Rudder Angle	Rudder Force	Datum Attitude	Speed
Closing throttle, keeping vertical acceleration between 0·8 and 1·0g.	69 knots ($1 \cdot 05\ V_s$)	2550 r.p.m. 17·5 inches Hg Boost	2·3° up	2 lb. pull	1·7° port	17 lb. port	1·5°/sec. nose down	—1·3 knots/sec. (deceleration)
Closing throttle, increasing speed by approx. $0 \cdot 1\ V_s$.	69 knots ($1 \cdot 05 V_s$)	2570 r.p.m. 17 inches Hg Boost	4·5° up	No record	4·2° port	27 lb. port	4·1° nose down	+ 11 knots ($0 \cdot 17\ V_s$)
Opening throttle, keeping vertical acceleration between 1·0 and 1·2g.	69 knots ($1 \cdot 05\ V_s$)	2600 r.p.m. 18 inches Hg Boost	1·9° down	3 lb. push	3·5° starboard	13 lb. starboard	1·7°/sec. nose down	+ 1 knot/sec. (acceleration)
Opening throttle, flying at same speed as initially.	69 knots ($1 \cdot 05\ V_s$)	2580 r.p.m. 17 inches Hg Boost	1·6° up	No record	3·0° starboard	21 lb. starboard	2·0° nose up	0

The proposed A.D.M. requirement is that, on closing the throttle, the stick force should be less than 10 lb. and the rudder force less than 25 lb. On opening the throttle, the stick force should be less than 20 lb. and the rudder force less than 50 lb.

5. *Discussion of Results.*—It was suggested in Part I that, although the order in which the tests have been performed and described was that most convenient for flight testing, it was not the order of highest importance of the various factors under investigation. The order of relative importance of the four most important factors is probably as follows:—

(1) View from the cockpit.

(2) Control over the angle of glide.

(3) Lateral control during the approach.

(4) Change in trim with engine power.

Discussion of the results will therefore be given under these four main headings.

5.1. *View from the Cockpit.*—The view from the cockpit of the Hellcat during an approach in the " standard " condition (*i.e.* at speed of $1 \cdot 0\ V_S$, in a 5 deg. glide relative to the air) falls short of the A.D.M. requirement—in fact, with the pilot's head central, an angle of yaw of nearly 15 deg. would be required in order to see the port corner of the round-down from dead astern at a range of 300 yards. This angle of yaw can immediately be reduced by the pilot to about 7 deg. by the simple expedient of moving his head across the cockpit.

The typical approach condition for this aircraft differs from this " standard " condition in two ways. The approach speed is in the region of $1 \cdot 05\ V_S$ and extra engine power is used to reduce the angle of glide to about $3 \cdot 7$ deg. relative to the air. The reason for this choice of speed and angle of glide is discussed more fully in the next section (section 5.2), but, other considerations apart, it has an important effect on the view from the cockpit. The increased speed and decreased angle of glide have the effect of reducing the angle between the line of sight and the wing chord by about $4 \cdot 5$ deg., and in this condition only 3 deg. of yaw would be required in order to see the carrier, if the pilot moves his head fully across the cockpit. However, since any slight increase in incidence (as caused by adjustments to the angle of glide) would immediately block the view of the carrier, the pilot must have a little in hand as regards view. The usual technique is therefore for the pilot to use a gentle curved approach with his head moved across the cockpit. The effect of a curved path in eliminating the necessity for approaching with a larger amount of skid was discussed in Part I.

The view from the Avenger cockpit during an approach at $1 \cdot 0\ V_S$ in a 5 deg. glide is worse than that of the Hellcat, but is improved slightly in the average approach condition. The configuration of the instrument panel is such that once the line of sight intersects the top of this panel, a large angle of yaw is necessary in order to see around it. However, view over the top of this panel is obtained with a fairly small decrease in angle between line of sight and wingchord or a small increase in the height of the pilot. Although the average approach condition ($0 \cdot 98\ V_S$ at $3 \cdot 8$ deg. angle of glide) requires 5 deg. of yaw in order to be able to see the carrier, this particular pilot was above average height, and could therefore see ahead without using yaw. The approach with this aircraft could therefore have been made from dead astern, but, in fact, a gentle curved approach was used, so that the view should be adequate to cope with possible increases in incidence.

It should be emphasised, however, that the preceding discussion is based on the assumption that the pilot is of average height (about 5 ft. 11 in.). A shorter pilot is naturally at a disadvantage, in fact, a decrease in height of the pilot's eye of 1 in. results in a decrease in the downward angle of view available of about 2 deg. This decrease may have a considerable effect on the angle of yaw required, particularly with a broad flat top to the instrument panel as in the case of the Avenger.

5.2. *Control over the Angle of Glide.*—The Hellcat fails to satisfy the A.D.M. requirement as regards angle of glide. At a speed of $1 \cdot 0$ V_S with sufficient engine on to depress the stalling speed to 87 per cent. of the engine-off stalling speed, *i.e.* $V/V_{ES} = 1 \cdot 15$, the angle of glide relative to the air is only $1 \cdot 4$ deg., as at Point " A ", Fig. 6 (H). In order to obtain the required 5 deg. glide at a speed of $1 \cdot 0$ V_S, the amount of engine employed is such that the margin above engine-on stalling speed is only $6 \cdot 5$ per cent. It is clear, therefore, that a close approximation to the " standard " approach condition is not possible with this aircraft. A higher speed and more engine are employed, at a smaller angle of glide. Preliminary analysis of tests on H.M.S. *Pretoria Castle* suggests that the approach speed is about $1 \cdot 05$ V_S, at an angle of glide of $3 \cdot 7$ deg. relative to the air. This results in a margin of 13 per cent. above engine-on stalling speed. The view from the cockpit is also considerably improved, as explained in section 5.1.

An important feature, not covered by the proposed A.D.M., is the way in which the angle of glide can be adjusted by movement of the throttle. Information on this point is obtained by cross-plotting along lines of constant airspeed (*i.e.* constant V/V_S) in Fig. 6 (H) to obtain the relation between angle of glide and throttle position. Fig. 7 (H) shows that the change in throttle setting per degree change in angle of glide is fairly independent of speed and has a value of $0 \cdot 134$ in. per degree at a mean angle of glide of 4 deg. This figure compares well with figures estimated by Duddy[10] (1945) for other deck-landing aircraft.

Seafire II	$0 \cdot 053$ in. per degree.
Barracuda II ..	$0 \cdot 132$ in. per degree.
Corsair I	$0 \cdot 105$ in. per degree.

In view of the importance of being able to control the angle of glide accurately without resorting to microscopic movements of the throttle knob, it may be worth while to lay down some minimum standard for this factor. A simple way of improving the control is simply by lengthening the throttle lever.

The Avenger more nearly satisfies the proposed A.D.M. requirement; the angle of glide in the " standard " condition is $4 \cdot 5$ deg. The average approach condition recorded on H.M.S. *Pretoria Castle* gives a speed of $0 \cdot 98$ V_S at an angle of glide of $3 \cdot 8$ deg.; the resultant margin over engine-on stalling speed being 15 per cent., which is considered satisfactory.

The variation of angle of glide with throttle setting (Fig. 7 (A)) indicates a throttle controllability factor of $0 \cdot 102$ in. per degree at a mean angle of glide of 4 deg. This figure seems to be rather low for an aircraft of this type.

5.3. *Lateral Control during the Approach.*—Tests for aileron effectiveness during the approach include measurements of the time to bank 10 deg. and the time to reverse 30 deg. of bank. Neither of these two aircraft satisfy the proposed A.D.M. requirement of 10 deg. of bank in $0 \cdot 75$ sec. with a stick force less than 5 lb. The differences in the times to bank 10 deg. on the two aircraft, viz. $0 \cdot 85$ sec. for the Hellcat, $1 \cdot 20$ sec. for the Avenger, can be accounted for by the difference in their respective maximum steady rates of roll. The ratio $pb/2V$ is about the same for both, but the Avenger has the greater span and lower stalling speed, so that its rate of roll is less. The value of $pb/2V$ of $0 \cdot 07$ for the Hellcat is considered to be rather low for a naval fighter. The time to apply full aileron is about $0 \cdot 3$ sec. in each case, which is as low as can be expected. In both cases, the stick forces are much higher than the proposed figure of 5 lb. (24 lb. for Hellcat, 18 lb. for Avenger). This is inevitable, since the stick forces for steady application of full aileron (*i.e.* excluding inertia forces) are greater than 5 lb., for either aircraft.

With regard to the test for the time to reverse 30 deg. of bank, it was explained in Part I that the proposed time limit of $1 \cdot 5$ secs. would require a phenomenal value of $pb/2V$, and it was suggested that this time limit might have to be extended to $3 \cdot 0$ sec. Both these aircraft (Hellcat and Avenger) require about $3 \cdot 5$ sec. in order to reverse 30 deg. of bank, but it should be noted that this manoeuvre of reversing 30 deg. of bank is only one part of the side-step manoeuvre. It has been shown (section 4.4 (*a*) above) that the measured time to reverse 10 deg. of bank and the measured maximum steady rate of roll will result in an adequate rate of sidestep, so that, in itself, strict limitation of the time to reverse 30 deg. of bank is unnecessary. The time taken for the sidestep manoeuvre, based on a complete S-turn, will, however, exceed the proposed time limit of 3 sec. unless the angle of bank in the turn is less than about 10 deg. which severely limits the sidestep distance attainable. It is considered, however, that the important factor is not the time taken, but the forward distance travelled, and to this extent, both these aircraft easily satisfy the proposed requirements. The tests to measure the time to reverse 30 deg. bank, and the maximum rate of flat turn could, therefore, be omitted from the A.D.M.

The maximum rate of flat turn does not enter into the calculations on rate of sidestep, since it is assumed that the turn is made with ailerons, the rate of turn always being appropriate to the angle of bank and the rudder being used simply to prevent sideslip.

5.4. *Change in Trim with Engine Power*.—The results of the change in trim tests, given in Tables 7 and 8 in section 4.5, show that the Avenger more or less satisfies the A.D.M. requirements as regards stick and rudder forces, while the Hellcat is satisfactory except as regards rudder forces on opening the throttle.

This general result is roughly the same whether the tests are done by the dynamic or by the static method, although in general the static method results in slightly higher control forces and movements. The chief exception is the rudder force on the Hellcat when the throttle is opened.

The relative merits, or otherwise, of the two methods of measuring change in trim were fully discussed in Part I. The conclusion was reached that, while the dynamic method of test more nearly corresponded to the actual manoeuvre under investigation, the static method would give results in sufficiently close agreement with the other. It was therefore suggested that the static method should be employed since, in addition, it is much simpler to perform. This conclusion is borne out by the above results.

6. *Conclusions*.—The Hellcat fails to satisfy the proposed requirement for view and for angle of glide in the standard approach condition. It is also generally slightly below standard as regards lateral control during the approach. The Avenger is below standard for view, and for the time to apply 10 deg. of bank.

It is suggested that it might be necessary to fix a minimum figure for the control over angle of glide with the throttle; a suggested figure is $0 \cdot 15$ in. per degree change in angle of glide.

It is also suggested that the rate of sidestep will be adequate provided that the A.D.M. requirements for lateral control are met. There appears to be no necessity to measure directly the time to reverse 30 deg. of bank, or the maximum rate of flat turn.

The criticisms of the Hellcat and the Avenger are very similar to those already made regarding the Seafire and Barracuda , in fact, it seems probable that any conventional single-engined fighter will suffer from the same defects as the Seafire and Hellcat—chiefly due to lack of drag during the approach and restriction of view. The proposed general requirements could probably best be met in the case of a twin-engined propeller-driven aircraft with very large flaps and ailerons designed to be very effective right down to the stall.

7. *Further Developments*.—It is hoped eventually to combine the results of the tests on these four aircraft in a review of the proposed requirements, and to make recommendations regarding the form of the tests and suggest possible modifications to the quantitative requirements.

TABLE 9

Aerodynamic Data, Hellcat I JX.815

General	
Mean weight during trials, lb.	11,750
S (gross wing area), sq. ft.	334
Engine	Double Wasp R–2800–10W
Rated H.P. at 2,700 r.p.m. at ground level and 54 in. Hg. r.p.m.	2000
Power loading, lb./b.h.p.	5·88
Wing loading, lb./sq. ft.	35·2
Span loading, lb./sq. ft.	6·40
C.G. h (mean chord $= S$/span)	0·254 c
Airscrew diameter, ft.	13·08
Airscrew blade angle at $0{\cdot}7R$ fine	19°
Airscrew blade angle at $0{\cdot}7R$ coarse	58°
Gear ratio	0·500 : 1

Wings	
Area (gross), S, sq. ft.	334
Span, $2s$, ft.	42·83
Mean chord, c, ft.	7·80
Aspect ratio	5·50
Dihedral (outer panels)	7·5°
Sweepback of $\frac{1}{4}c$ line	1·3°
Chord, ft. root	9·89
Chord, ft. tip	5·30
Section root	NACA 23015·6
Section tip	NACA 23009
Wing twist, root-tip	0

Flaps	
Type	NACA Slotted
Maximum angle	48°
Flap area/S	0·119
Flap chord/local wing chord	0·188
Nett. flap span/$2s$	0·552
Speed at which automatic retraction commences, E.A.S.	86 knots.

Longitudinal Control	
Tail surface area (gross), S', sq. ft.	77·84
Elevator area/S'	0·331
e'/c (e' = distance C.G. to $\frac{1}{3}$ T.P. chord)	2·40
S'/S	0·233
Tail volume coeff., $S'e'/Sc$	0·559
Elevator angles (max.) up	27°
Elevator angles (max.) down	13°
Type of balance	Nose
Percentage balance	27·5
Stick gearing, $d\eta/dx$, deg./in.	2·15
Trimming tab area (total), sq. ft.	2·0
Trimming tab angles (max.) down	18°
Trimming tab angles (max.) up	4°

Directional Control	
Fin and rudder area, S'', sq. ft.	23·4
Rudder area/S''	0·385
e''/s (e'' = distance C.G. to centroid of S'')	0·98
Fin and rudder volume coeff., $S''e''/Ss$	0·0687
Rudder angles, max. port	30°
Rudder angles, max. starboard	30°
Type of balance	Horn
Percentage balance	21·1
Pedal gearing, $d\xi/dx$, deg./in.	8·33
Trimming tab area, sq. ft.	0·62
Trimming tab angles port	30°
Trimming tab angles starboard	8°

Lateral Control	
Type of aileron	Frise, + spring tab
Aileron area (total), sq. ft.	15·7
Aileron area/S	0·047
Aileron chord/local chord	0·200
Total aileron span/$2s$	0·298
Aileron angles (max.) down	13°
Aileron angles (max.) up	17°
Percentage balance	34·0
Stick gearing, $d\xi/dx$, deg./in.	1·78
Trimming tab area, sq. ft.	0·40
Trimming tab angles up	8°
Trimming tab angles down	8°
Spring tab span/aileron span	0·22
Spring tab chord/aileron chord	0·25

Note.—Port spring tab is used for trimming.

TABLE 10

Aerodynamic Data, Avenger JZ.640

General

Mean weight during trials	14,610
S (gross wing area), sq. ft.	490
Engine	Cyclone R-2600-8
B.H.P. 2,600 r.p.m. S.L. (take off)	1,700
B.H.P. 2,400 r.p.m. S.L. 1–6,700 ft. low ratio	1,500
B.H.P. 2,400 r.p.m. 9,500–14,800 ft. high ratio	1,350
Power loading (T.O.), lb./B.H.P.	8·6
Wing loading, lb./sq. ft.	29·8
Span loading, lb./sq. ft.	4·96
Airscrew diameter, ft.	13
C.G. h (mean chord = S/span)	0·27 c
Airscrew blade angle at $0 \cdot 7R$ fine	22°
Airscrew blade angle at $0 \cdot 7R$ coarse	45°
Gear ratio	0·572 : 1

Wings

Area (gross), S, sq. ft.	490
Span, $2s$, ft.	54′ 2″
Mean chord, c, ft.	9·02
Aspect ratio	6
Dihedral (outer panels)	6°
Sweepback of $\frac{1}{4}c$ line	2·4
Chord, ft. root	11′ 11″
Chord, ft. tip	4′ 7″
Section root	NACA 23015
Section tip	NACA 23009
Wing twist, root-tip	0
Span of leading edge slats (total)	7′ 6″

Flaps

Type	Split
Maximum angle (deg.)	45
Flap area/S	0·1275
Flap chord/local wing chord	0·215
Nett flap span/$2s$	0·68

Longitudinal Control

Tail surface area (gross), S', sq. ft.	110·84
Elevator area/S'	0·423
e'/c (e' = distance C.G. to $\frac{1}{4}$ T.P. chord)	2·39
S'/s	0·226
Tail volume coeff., $S'e'/Sc$	0·54
Elevator angles (max.) up	20°
Elevator angles (max.) down	10°
Type of balance	Horn
Percentage balance	41·2
Stick gearing, $d\eta/dx$, deg./in.	1·725
Trim tab area (total), sq. ft.	0·92
Trim tab angles (max.) up	12·5°
Trim tab angles (max.) down	7·5°

Directional Control

Fin and rudder area, S'', sq. ft.	42·7
Rudder area/S''	0·38
e''/s (e'' = C.G. to centroid of S'')	0·83
Fin and rudder volume coeff., $s''e''/Ss$	0·0725
Rudder angles (max.) port	24°
Rudder angles (max.) starboard	24°
Type of balance	Horn
Percentage balance	30·3
Pedal gearing, $d\xi/dx$, deg./in.	8
Trimming tab area, sq. ft.	2·12
Trimming tab angles port	24°
Trimming tab angles starboard	16°

Lateral Control

Type of aileron	Frise
Aileron area (total), sq. ft.	19
Aileron area/S	0·039
Aileron chord/local chord	0·268
Total aileron span/$2s$	0·276
Aileron angles (max.) up	21°
Aileron angles (max.) down	19°
Percentage balance	33
Stick gearing, $d\xi/dx$, deg./in.	2·75
Trim tab area, sq. ft.	0·90
Trimming tab angles up	8°
Trimming tab angles down	8°

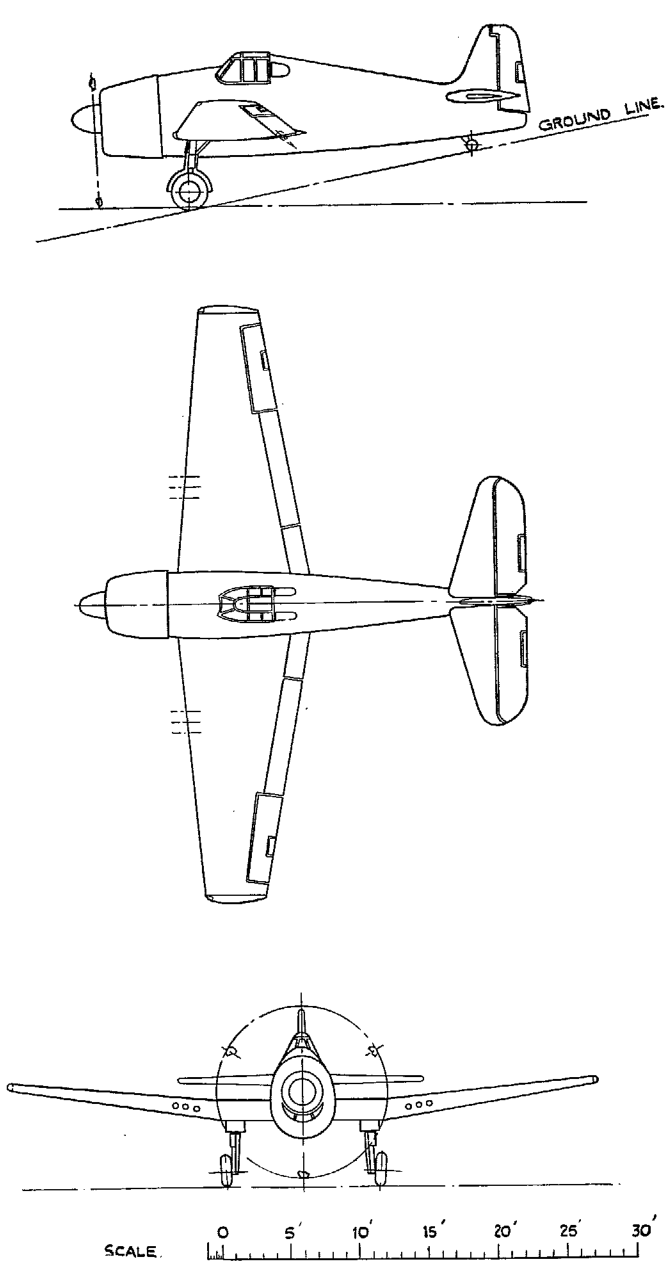


FIG. 1 (H). General Arrangement of Hellcat I.

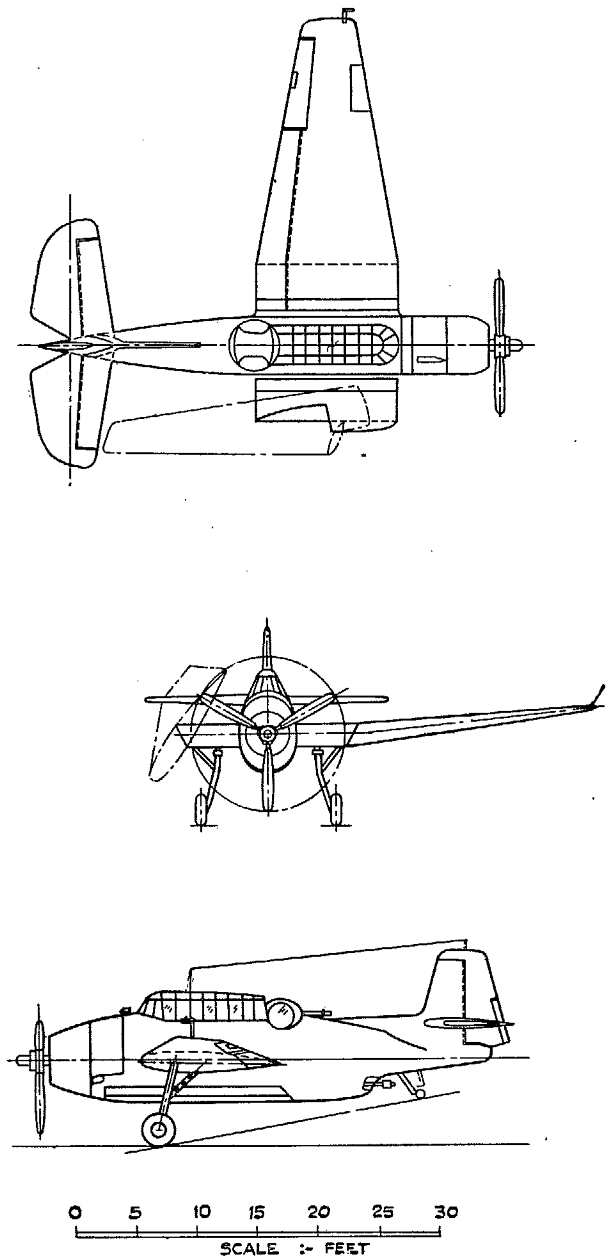


FIG. 1 (A). General Arrangement of Avenger I.

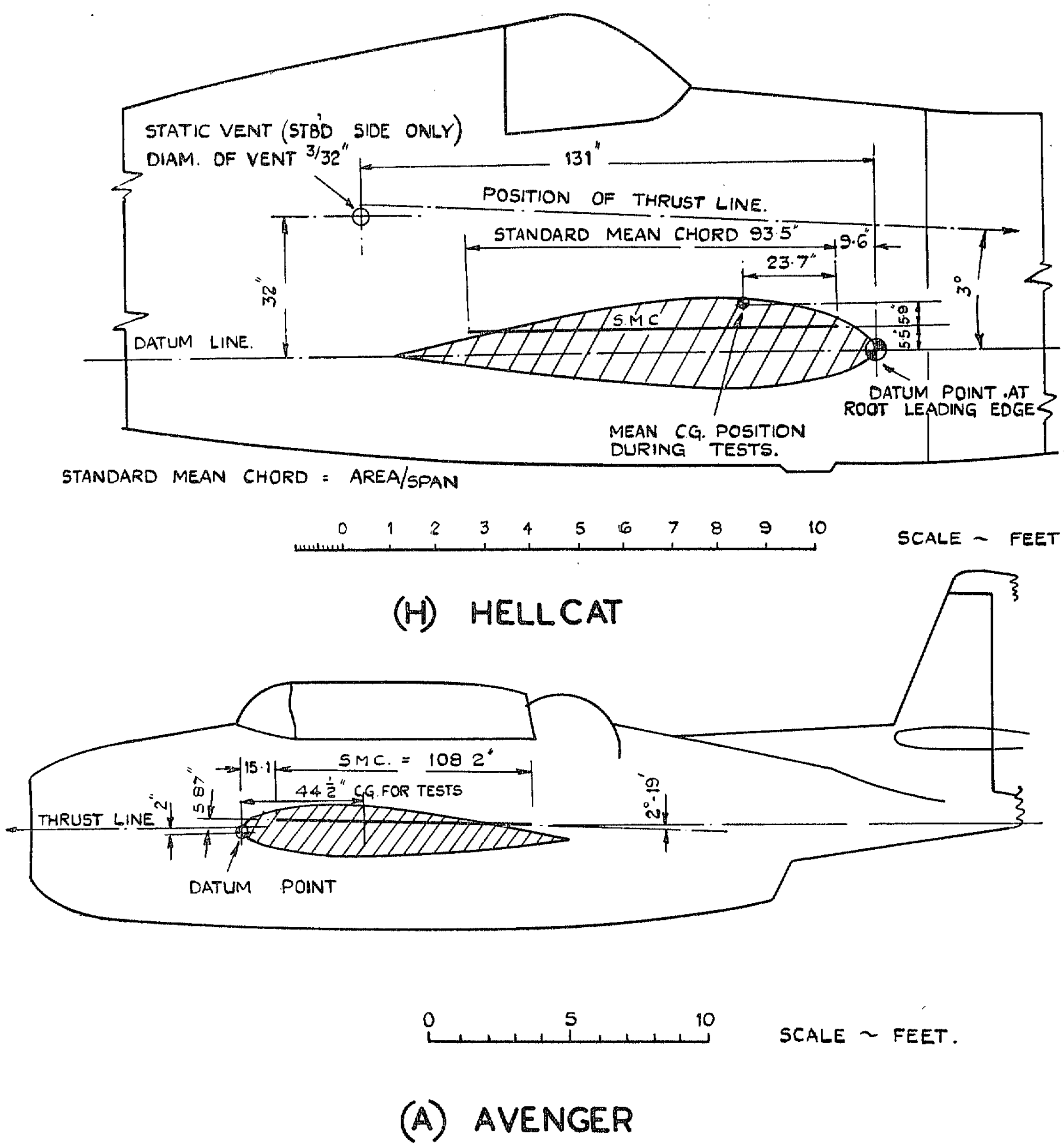


FIG. 2. Location of Standard Mean Chord, Datum Point, C.G., Thrust Line and Static Vent (Hellcat only).

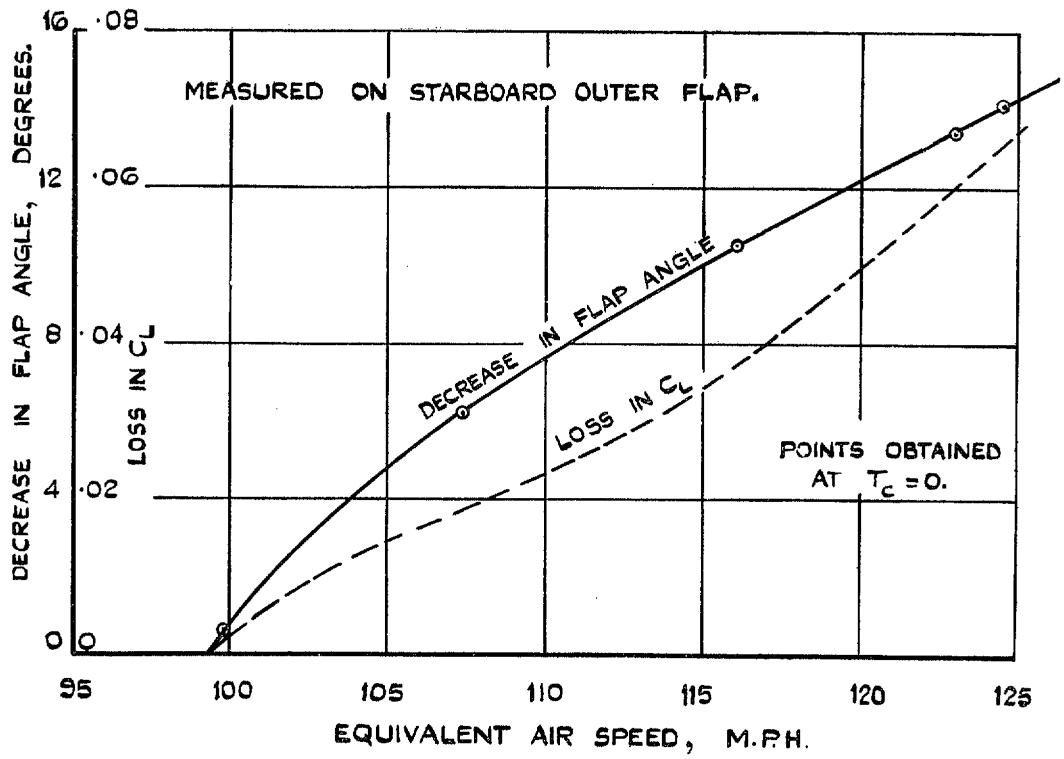


FIG. 3 (H). Variation of Flap Angle with Speed—Hellcat.

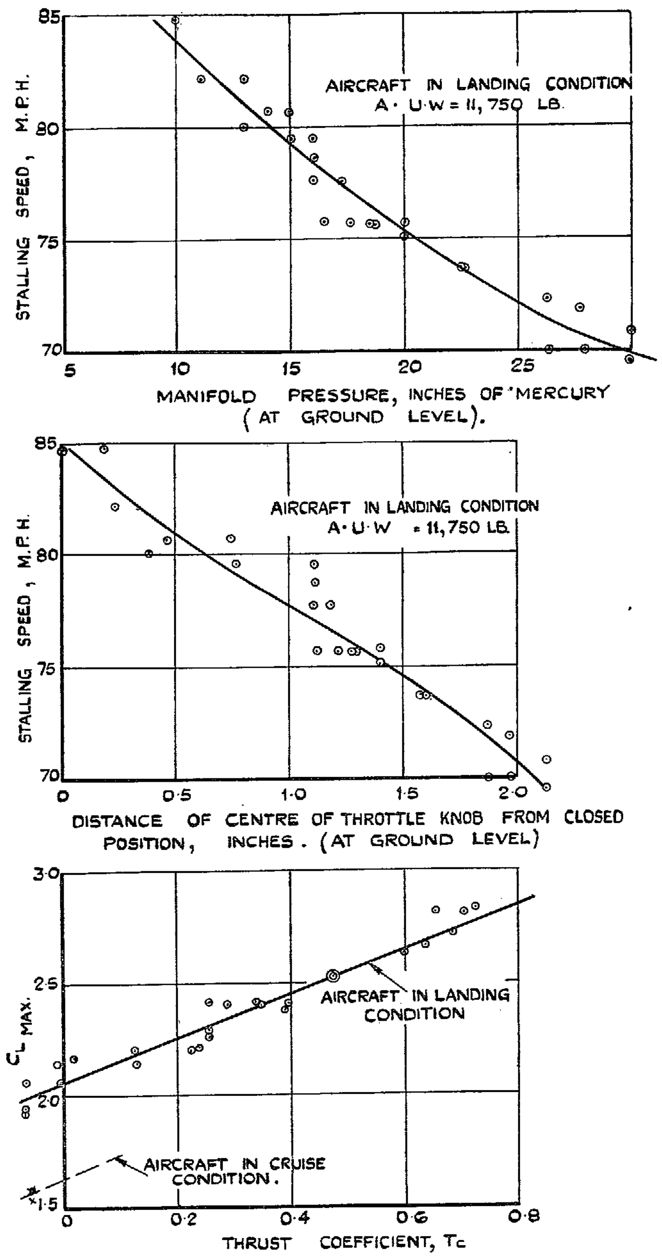


FIG. 4 (H). Effect of Engine on Stalling Speed and $C_{L\ max.}$—Hellcat.

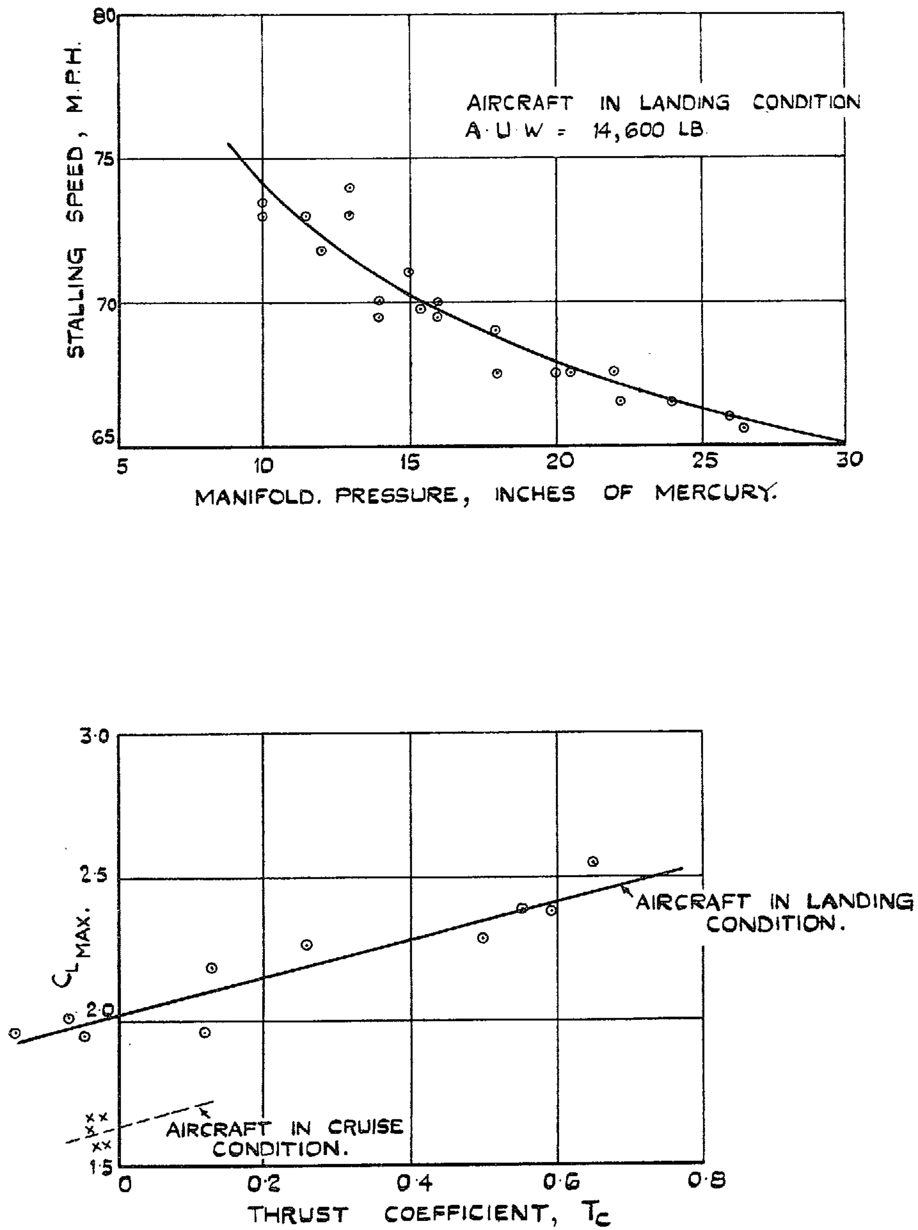


FIG. 4 (A). Effect of Engine on Stalling Speed and $C_{L\ max.}$—Avenger.

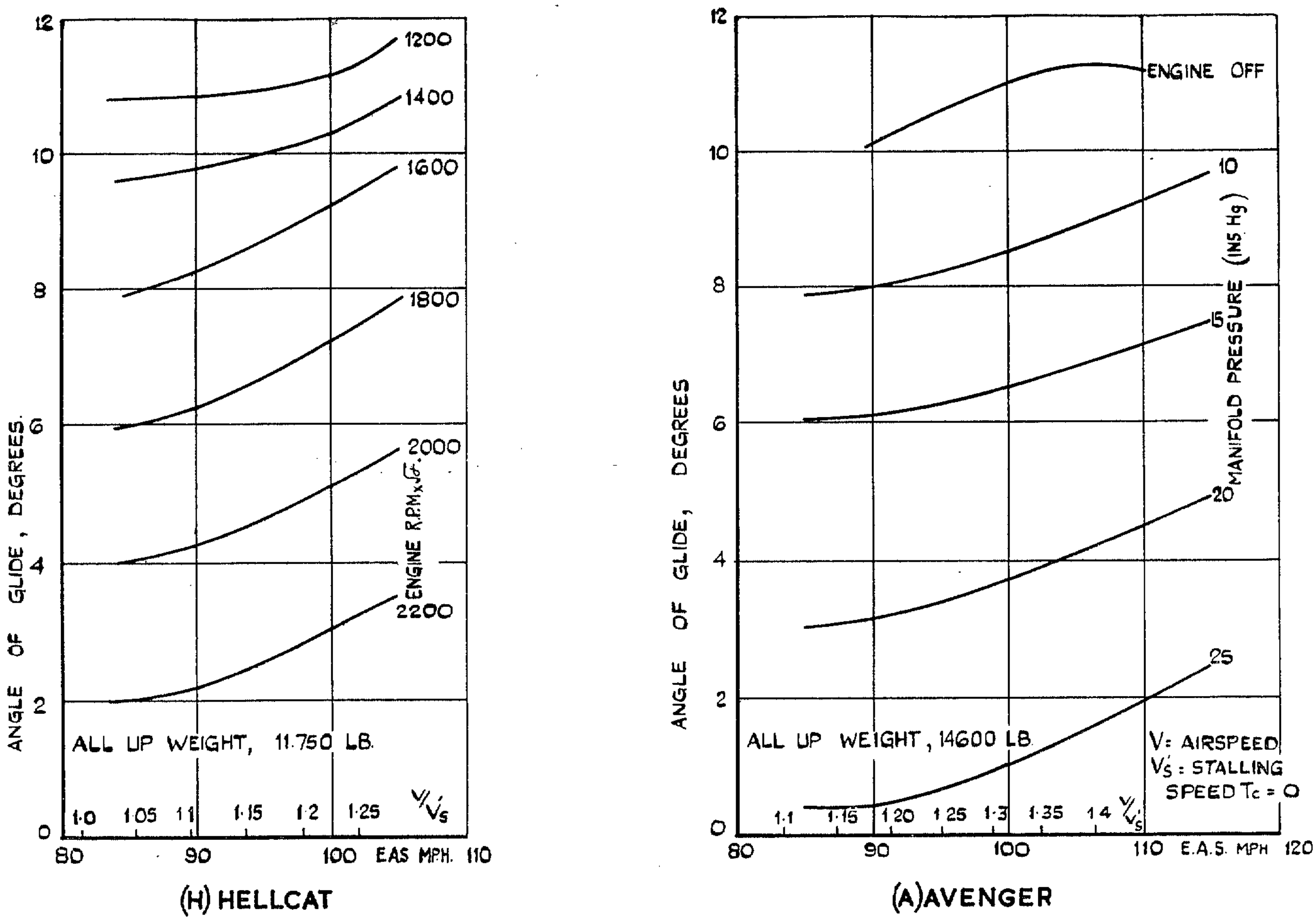


FIG. 5. Effect of Engine on Gliding Angle.

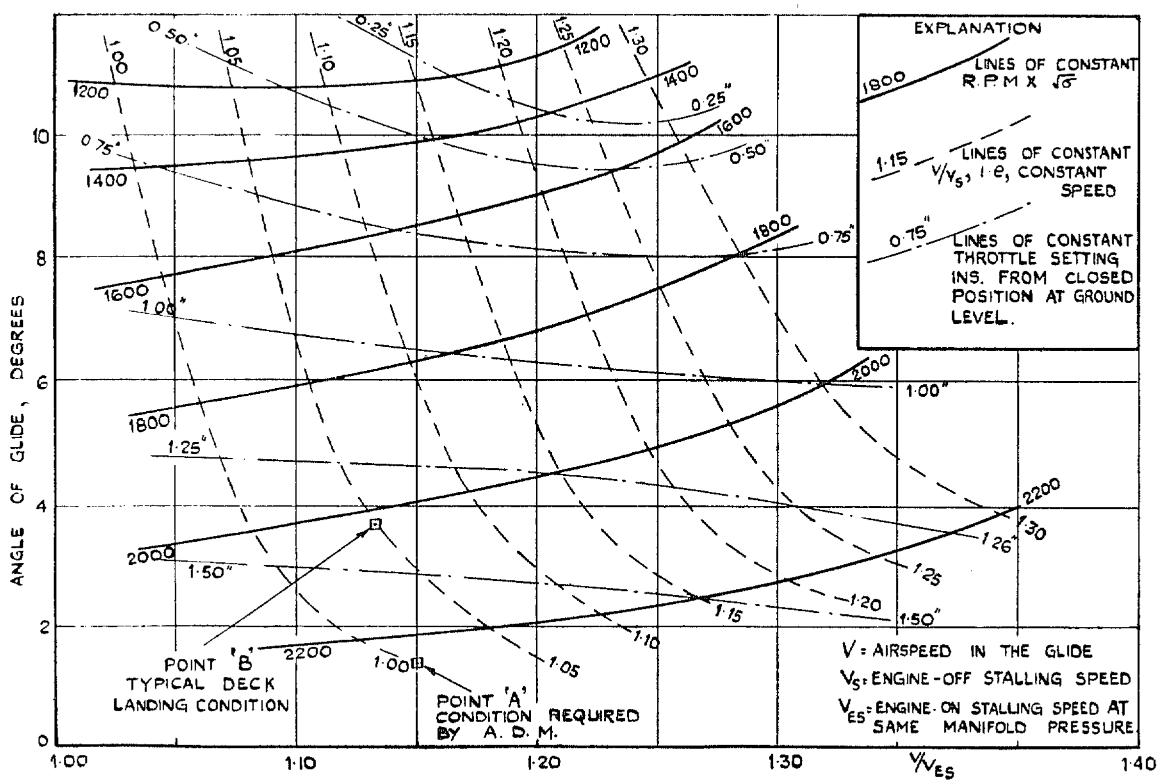


FIG. 6 (H). Effect of Engine on Gliding Angle and Safety Margin—Hellcat.

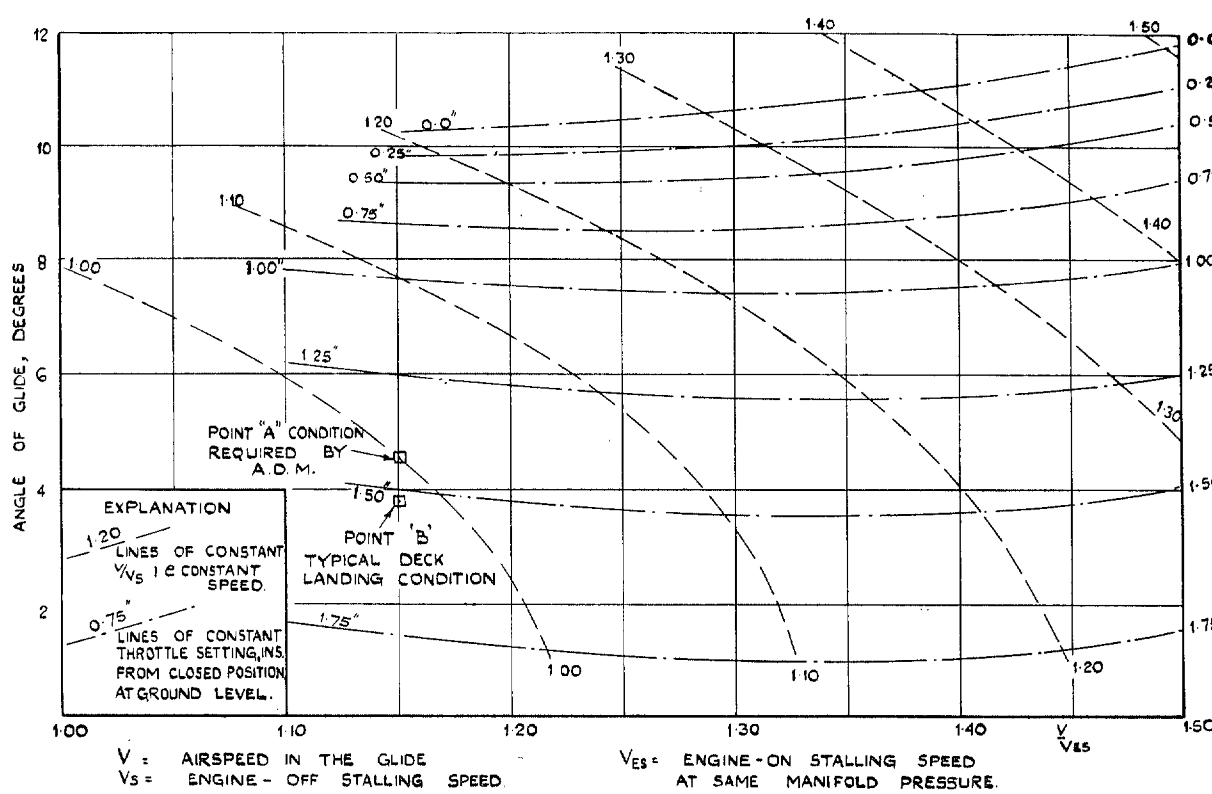


FIG. 6 (A). Effect of Engine on Gliding Angle and Safety Margin—Avenger.

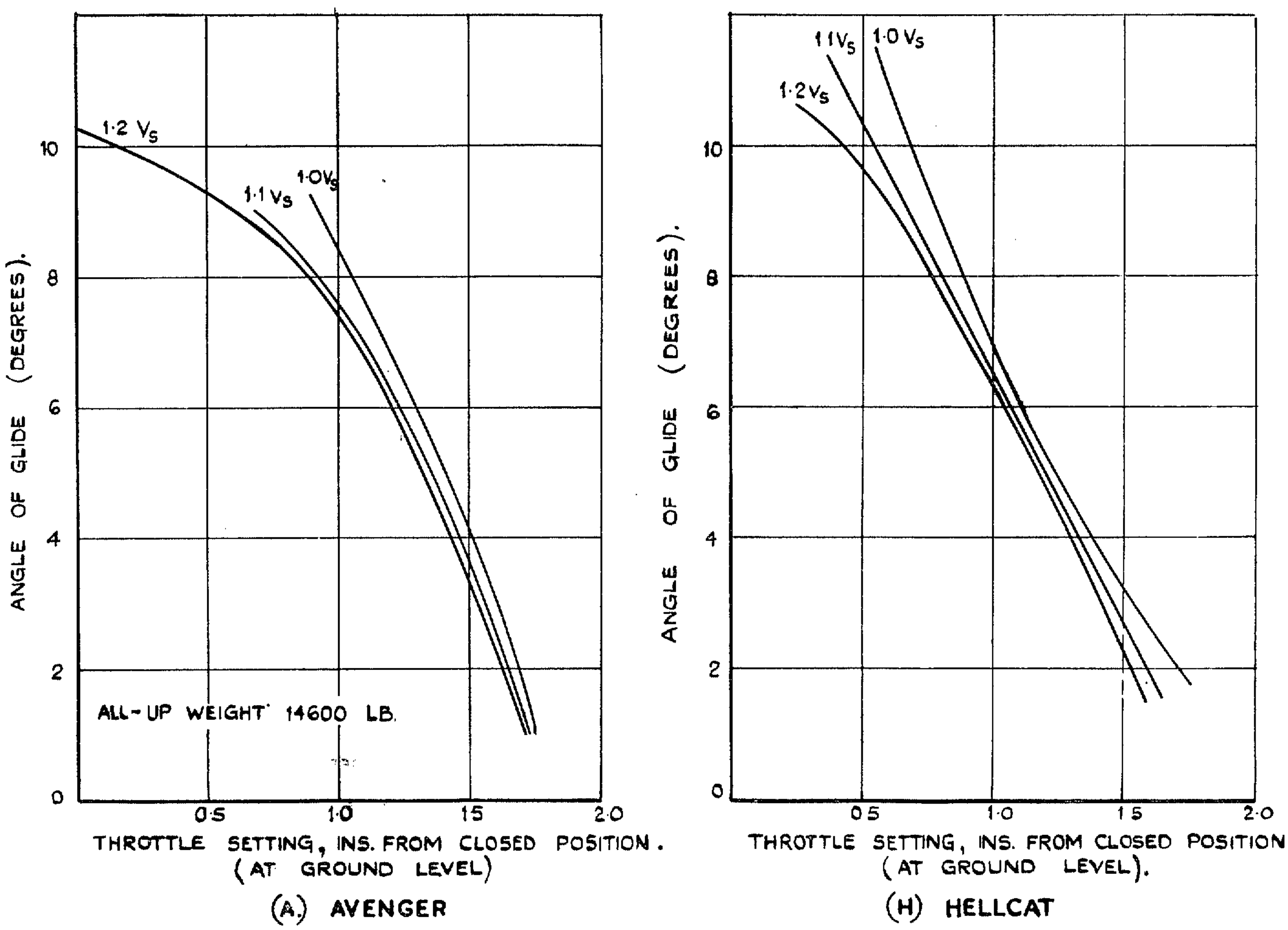


FIG. 7. Effect of Engine on Glide Angle at Constant Speed.

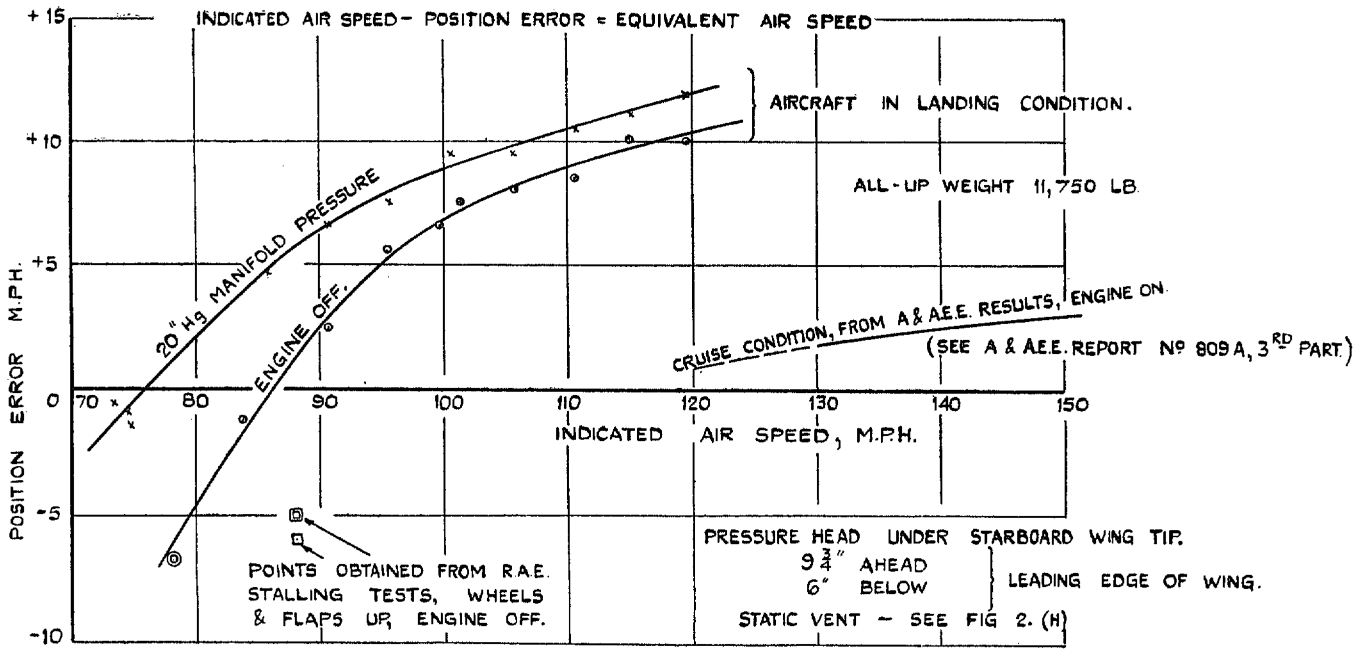


FIG. 8 (H). Position Error Curves—Hellcat.

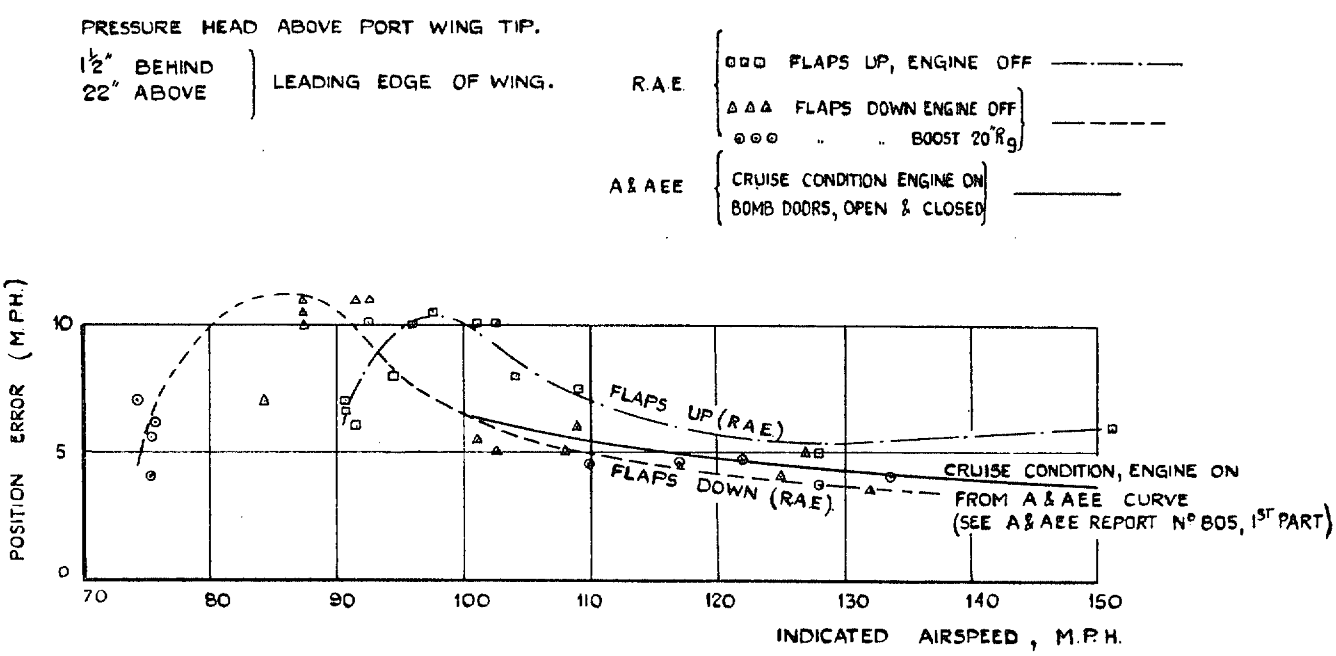
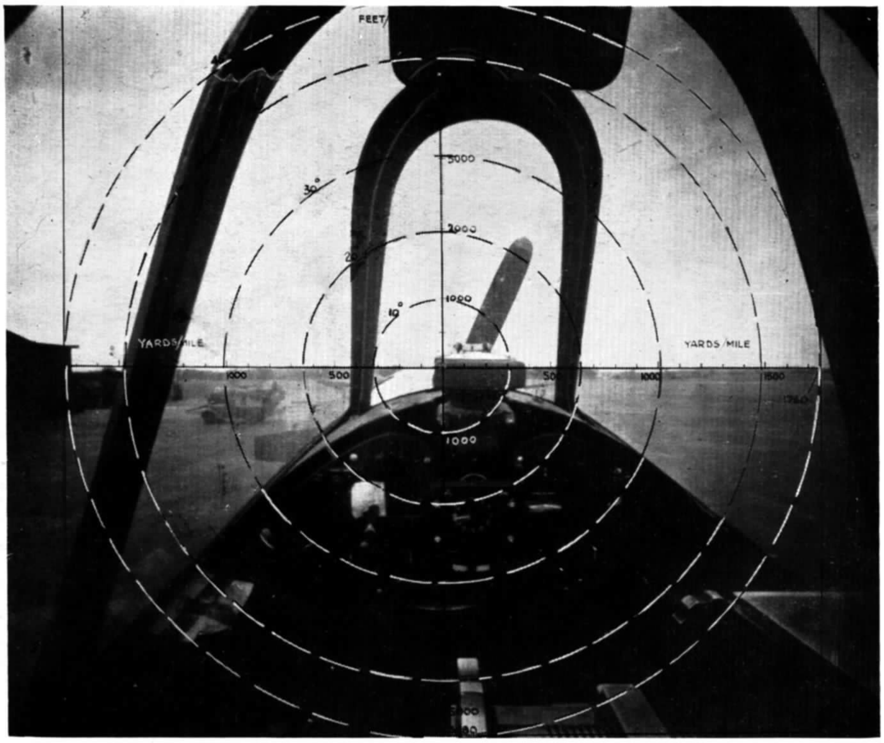


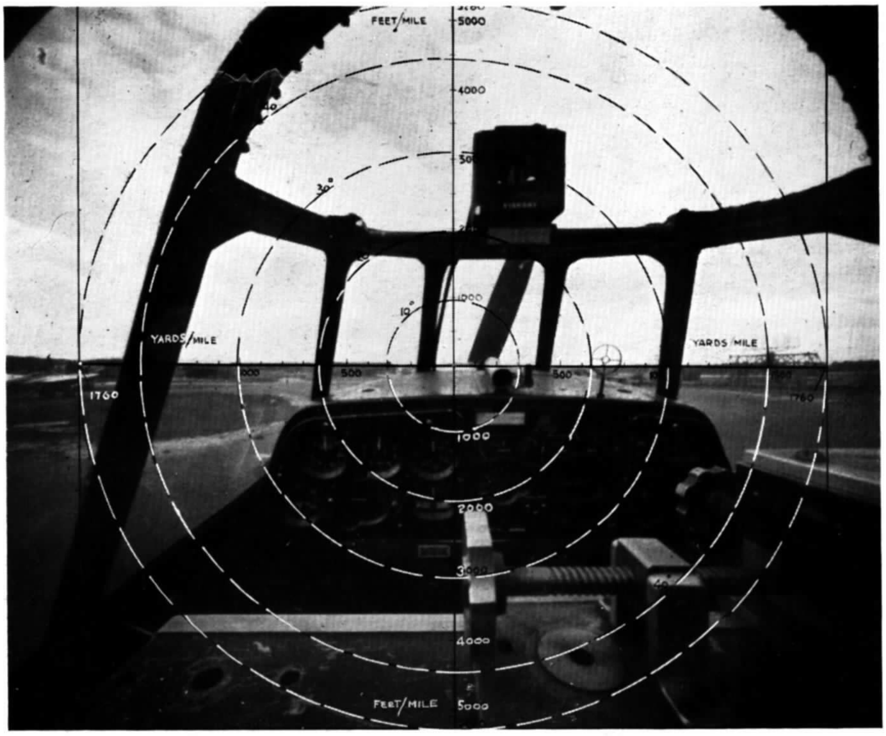
FIG. 8 (A). Position Error Curves.



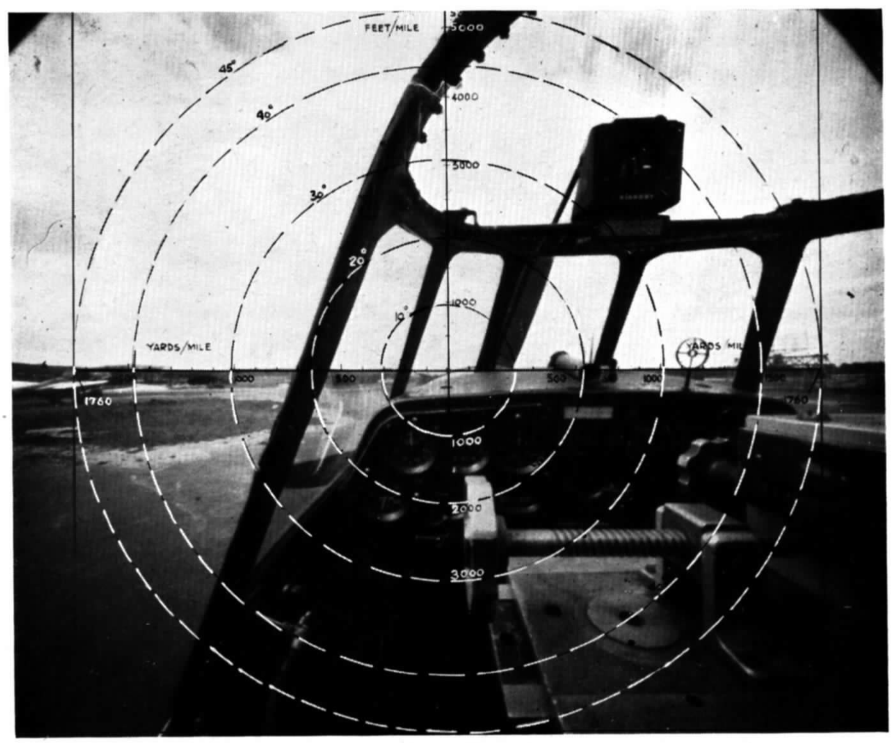
Photograph A. Camera position corresponding to pilot's left eye when seated with head central, seat fully up. Centre of graticule corresponds to a direction of sight inclined down at 10·3° to wing chord.

Photograph B. Camera position corresponding to pilot's left eye when seated with head moved 4·7 inches from central position, seat fully up. Centre of graticule corresponds to a direction of sight inclined down at 10·3° to wing chord.

FIG. 9 (H). View from Cockpit—Hellcat.



Photograph A. Camera position corresponding to pilot's left eye when seated with head central, seat fully up. Centre of graticule corresponds to a direction of sight inclined down at 13·3° to wing chord.



Photograph B. Camera position corresponding to pilot's left eye when seated with head moved $5\frac{1}{4}$ inches from central position, seat fully up. Centre of graticule corresponds to a direction of sight inclined down at 13·3° to wing chord.

FIG. 9 (A). View from Cockpit—Avenger.

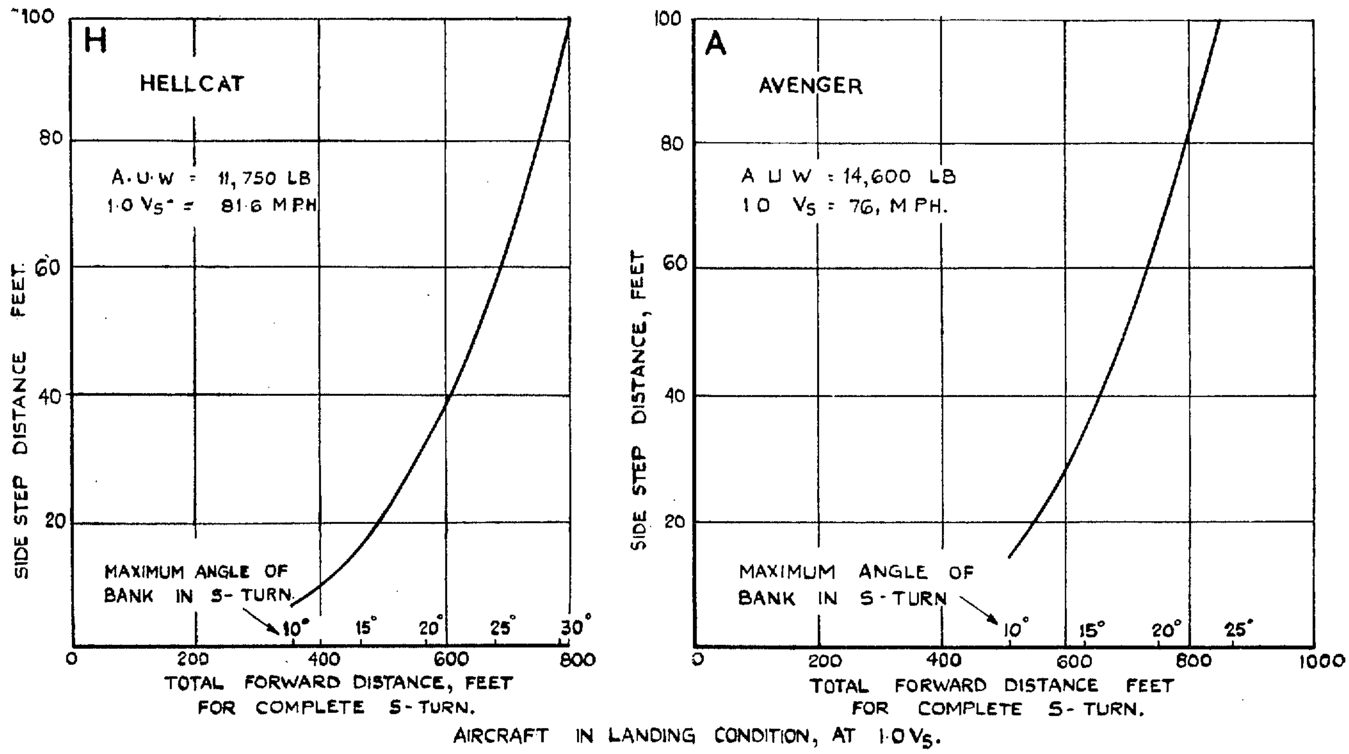


FIG. 10. Estimated Sidestep Distances for Hellcat and Avenger.

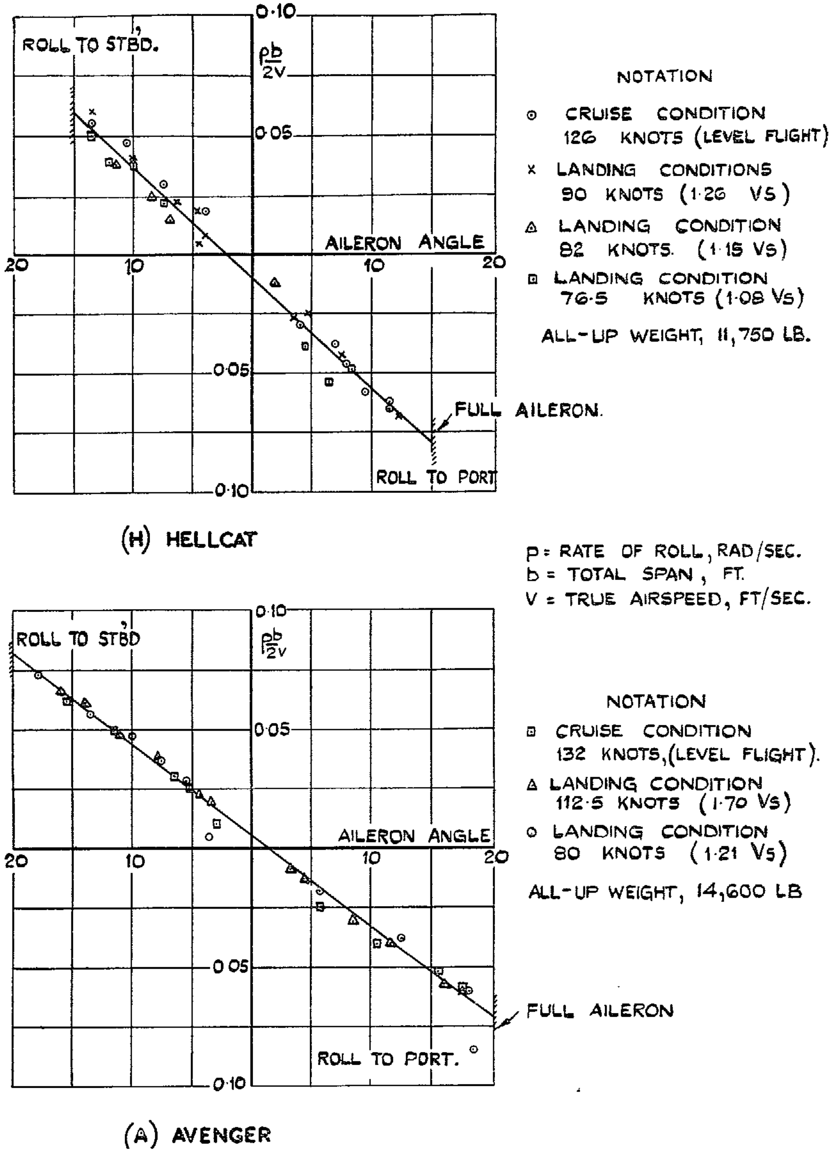


FIG. 11. Variation of Rate of Roll with Aileron Angle.

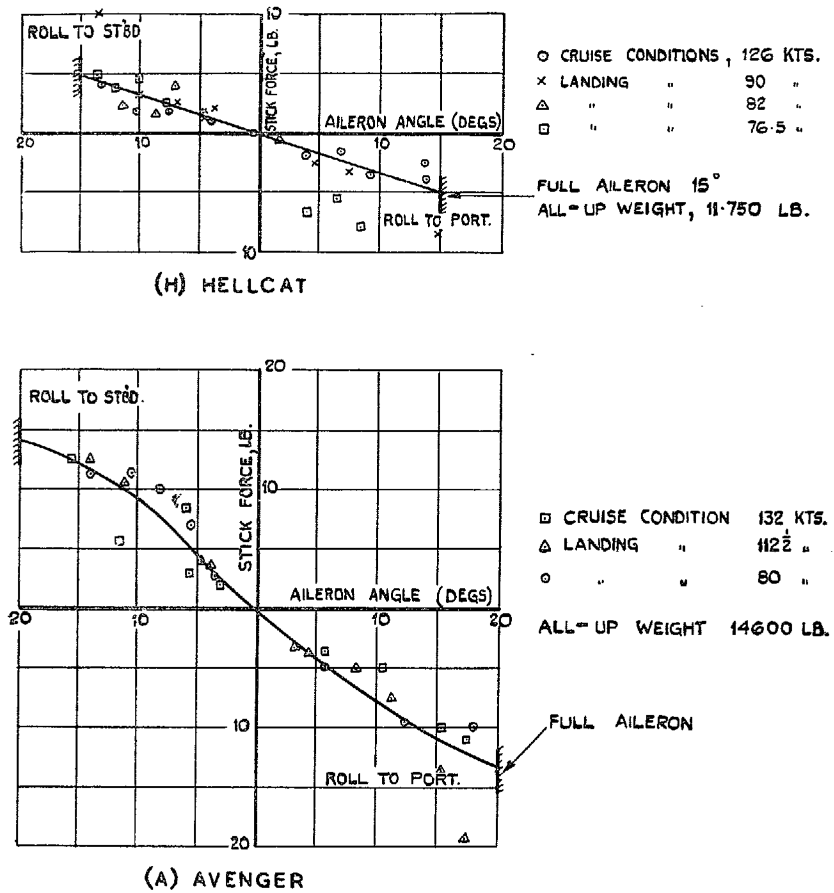


FIG. 12. Variation of Stick Force with Aileron Angle at $1 \cdot 0 V_S$.

REFERENCES

No.	*Author*	*Title, etc.*
1	D. Biermann and E. P. Hartman ..	Tests of Five full-scale Propellers in the Presence of a Radial and a Liquid-cooled Engine Nacelle, including Tests of Two Spinners. NACA Report No. 642. 1938. A.R.C. 3979.
2	A. B. Haines, A. R. C. MacDougall and R. J. Monaghan.	Charts for the Determination of the Performance of a Propeller under Static Take-off, Climbing and Cruising Conditions. R. & M. 2086. March, 1946.
3	D. J. Lyons and P. L. Bisgood	An Analysis of the Lift Slope of Aerofoils of Small Aspect Ratio, including Fins, with Design Charts for Aerofoils and Control Surfaces. R. & M. 2308. January, 1945.
4	A. D. Young and P. A. Hufton	Note on the Lift and Profile drag Effects of Split and Slotted Flaps. R. & M. 2545. September, 1941.
5	R. Smelt and H. Davies	An Estimate of the Increase in Lift due to Slipstream. R. & M. 1788. December, 1936.
6	D. Lean, R. R. Duddy and J. R. Stott ..	Seafire Deck Landings on H.M.S. " Pretoria Castle ". Investigation of Piloting Technique during the Approach. R.A.E. Technical Note No. Aero. 1486. A.R.C. 7961. July, 1944. (Unpublished).
7	P. A. Hufton	The Deck Landing Problem. R.A.E. Technical Note No. Aero. 1689. A.R.C. 8950. September, 1945. (Unpublished).
8	E. N. Jacobs, R. M. Pinkerton and H. Greenberg.	Tests of Related Forward Camber Aerofoils in the Variable Density Wind Tunnel. NACA Report No. 610. December, 1936.
9	J. G. McHugh and E. Pepper	The Propeller and Cooling Airflow Characteristics of a Twin-engine Airplane Model equipped with NACA D_s—type Cowlings and with Propellers of NACA 16 Series Section. NACA Advance Confidential Report No. L.4120. A.R.C. 9328. September, 1944.
10	R. R. Duddy	A Comparison of the Deck-landing Qualities of Three Aircraft. R.A.E. Technical Note No. Aero. 1690. A.R.C. 8951. September, 1945. (Unpublished).

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