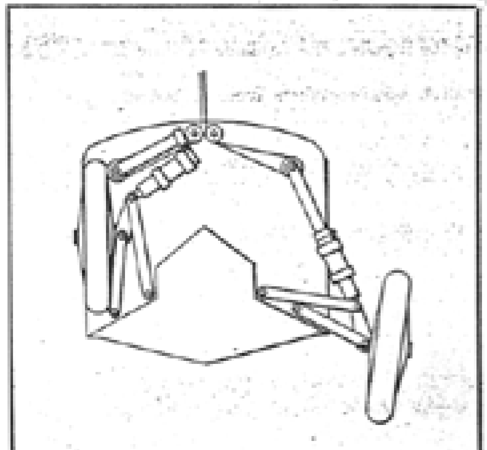
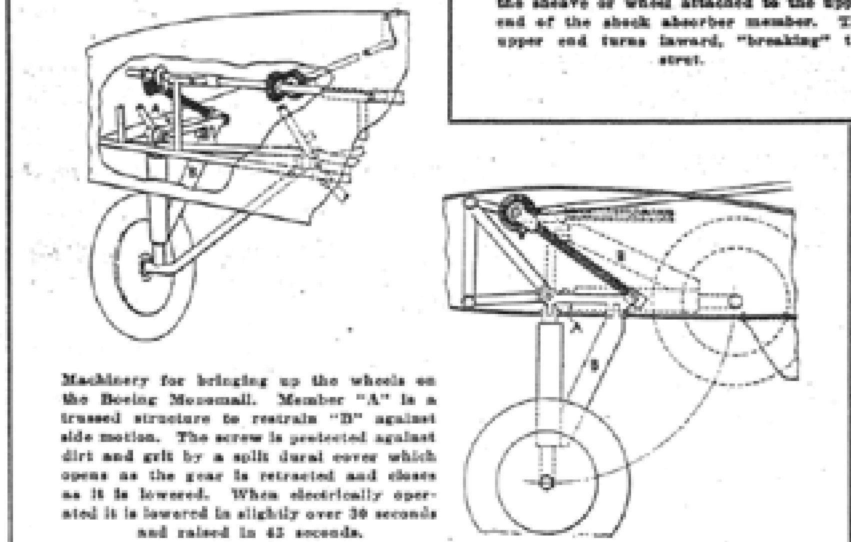


Drawing from Grumman's retractable landing gear patent. The shock absorber strut has the hinge effect, so that under compression the eccentricity tends to keep strut from buckling. In retracting, cables turn the sheave or wheel attached to the upper end of the shock absorber member. The upper end turns inward, "breaking" the strut.



Machinery for bringing up the wheels on the Boeing Monomail. Member "A" is a trussed structure to restrain "B" against side motion. The screw is protected against dirt and grit by a split dural cover which opens as the gear is retracted and closes as it is lowered. When electrically operated it is lowered in slightly over 30 seconds and raised in 43 seconds.

wheel or to carry the landing loads. The Bellanca "Roma," a sesquiplane built in 1928 had one portion of the wing thickened slightly to house the wheel. The landing loads were carried as in the "Airbus" but with the wheels fully retracting into the bottom of the inner bay of the lower wing. In a biplane the wheels might fold inward into the bottom of the fuselage. It is understood that the new Curtiss YO-40 Army observation plane is of this general type, with thick filleted roots on the lower wing.

INTO THE NACELLE

The problem becomes easiest when there are outboard engine nacelles, for the nacelle usually provides ample space for housing the gear and gives the proper tread with the wheels located under the concenetrated loads. On a high wing monoplane design with engines in the leading edge, the landing gear must be very high. Because of the great height it is almost impossible to take the side loads as a cantilever, and therefore the design must be complicated by side struts to the fuselage as on the General YO-27 (see illustration) where they remain out in the air-stream when the wheel is retracted. In the new Dutch built Fokker F-XX a compromise apparently has been reached by retaining the high wing and lowering the engine nacelles below the wing so that a completely retractable gear can be obtained with no side struts to the fuselage, the nacelles carrying all the load. Fokker seems to have decided that the slightly increased drag of the lowered nacelles did not offset the advantages of the high wing, with its inherently less drag and greater lift as well as its absence of buffeting and better passenger vision. (see page 59 and references 3, 4 and 5). However, some designers have reached a compromise in another fashion, locating the nacelle in the best position with reference to the landing gear and then lowering the wing to have the engine in the leading edge. Examples of this are the new Martin bomber, the Boeing bomber and the new Boeing transport, and the new Douglas monoplane transport.

The problem of retraction in these designs is quite similar to that in folding the wheel backward into the low wing of a single-engined plane. The simplest method seems to be to have the upper end of the rear brace strut move, while the forward struts pivot about their upper ends. On the Martin bomber, this is accomplished by having the rear member slide backward on a track when the wheel is hoisted by a cable attached to the axle. The gear is lowered by a cable pulling the rear strut forward to the landing position, where it is locked. In the Boeing Monomail a rotating screw and nut draw the upper end of the rear member forward and upward from the landing position. A number of equivalent systems might have been substituted such as a slide tube, rack and gear, piston and cylinder, etc.

HINGED STRUTS

A very similar plan is to provide the rear strut with a hinge near its center and "break" the strut. The Eaglerock Bullet built in 1929, the Fokker 1930 XO-27 (forerunner of the General YO-27), the Keith-Ryder San Franciscan racer of 1931 and 1932, and the new Clark-General single engined transport furnish examples. On the General transport an hydraulic piston and cylinder bearing against the landing gear strut at its "breaking" point supply the actuating force. The hydraulic system automatically locks so that if a landing is made before the wheels are fully lowered the gear will remain rigid. It is stated that the gear can be raised in eighteen seconds and lowered in eight.

The most familiar design in which the wheels swing up into the wing transversely, is the Lockheed which folds them inward. As in the Verville 1922 Pulitzer

Racer, and the 1928 Bellanca "Roma," the shock absorber is hinged to the front spar. A rear brace member is hinged to the rear spar, but further outboard than the shock absorber hinge so that when the gear is retracted about an axis passing through these hinges, the wheels move backward until they are between the spars. A similar system, but with the wheels moving outward is used on the French Bleriot transport monoplane.

As on the Lockheed the wheel on the Ballanca was raised by a cable, but instead of going to a piston and cylinder it went directly to a winch. The Bellanca had no side bracing and side loads were taken by bending in the front member, which was attached to the top and bottom of the front spar. The inner lower brace to the spar was drawn back out of the way to allow the wheel to be raised. The wheel was lowered by allowing it first to fall of its own weight and then drawing it home by a cable on the outboard side. The wheels could be manually raised in 17 to 21 seconds and lowered in seven to nine seconds.

Perhaps the best example of reducing the drag by leaving the wheels largely in the air stream and merely changing their position is the new Martin bomber. The wheels are retracted behind the outboard nacelles, but not enclosed. With the gear extended, there is a loss of approximately 12 per cent of the top speed with the wheels retracted. It was found that fairing over the wheels did not help the top speed, but with the wheels extended the fairing caused considerable disturbance.

In discussing the problem of retraction, some landing gear linkages and actuating mechanisms have been described. Let us consider the principal advantages and disadvantages of some of the various types. The most desirable seems to be the purely mechanical system, having only levers and hinges. It is usually simple, positive, quick operating, and cheap to build and maintain. A mechanical cable system usually is light, simple, and cheap, but has the problem of fouling of cables and pulleys, frayed cables, failure of synchronization due to the stretching of cables, etc. A slide tube or a track is usually light, simple, cheap, and quick operating, but it may jam. Screw and hydraulic mechanisms, though they are positive in action and can produce great forces, are usually slow unless power actuated. They have a tendency to be heavy, and though the hydraulic systems have the advantage of being very flexible they are confronted with the problems of pumps, leaks, and congealing oil in cold weather.

Though it is not difficult to make a landing gear that may be retracted or lowered manually without great exertion, often considerable time is necessary. The Department of Commerce requires that a gear be lowered in 60 seconds or less. This can usually be accomplished by a clutch which disengages the slow acting retracting mechanism of high mechanical advantage allowing the gear to drop of its own weight. Auxiliary means, such as rubber shock absorber chord, are sometimes used to hasten the lowering of the wheels.

On commercial transports and military planes it is felt that unless the landing gear naturally comes down very quickly the addition of power operated mechanism to an emergency hand actuated mechanism is desirable. The pilot's duties are ever increasing, and especially during landing in bad weather his attention should not be distracted by lowering the landing gear. On military types the power operated type is desirable to avoid distracting the pilot during formation take offs and landings. Power actuated gears are usually electrically driven, though it should be feasible to utilize the air pressure to raise or lower the gear by a flap.

Mention was made of warning devices to insure that the pilot has not forgotten to lower the wheels before landing. Most of these are electrical and are actuated if the throttle is retarded when the wheels are not locked in the down position. Originally lights were used but they proved inadequate and were replaced by a horn near the pilot's ear. The latest device, on the Clark General transport, is an electric motor with an eccentric weight which causes the control stick to vibrate (see page 59, this issue). In addition most planes have indicators, usually colored lights, showing when each wheel is locked in the up or down position. The Department of Commerce requires and indicator "to show the position of the wheels at all times."

Closing, I wish to acknowledge my appreciation to the Army Air Corps, and to those manufacturers and engineers who have so kindly supplied me with data and illustrations. Unfortunately, space did not permit credit being given in each instance nor the publication of all of the material desired.

Document 3-15(e), Roy G. Miller, letter to editor, **Aviation**, *"Retractable Landing Gears,"* **Aviation** *32 (April 1933), with response from Richard M. Mock: 130-131.*

MR. MOCK is to be congratulated on his very interesting article on Retractable Landing Gears, which was published in the February issue of AVIATION. He has invited our attention to practically every phase of this question. Some of his thoughts are, I am sure, new to many of us. There are several points, however, on which he seems to be rather too optimistic as regards the virtues of landing gear retraction.

It is stated that an airplane having a top speed of 175 m.p.h. with a faired gear of fixed design showed an improvement of 25 m.p.h. when fitted with a retractable gear. This amounts to an increase of 14 percent in speed and corresponds to a reduction in drag of about 33 percent. The drag of the landing gear, according to this, would be 50 percent, as much as the drag of the airplane with landing gear retracted.

The frontal area of a faired landing gear usually amounts to less than 10 percent of the total frontal area of the airplane, including landing gear, fuselage, pilot's compartment, wing, tail, etc. It is hard to believe that 10 percent of the frontal area would have half as much resistance as the other 90 percent. This would mean that the resistance coefficient for the landing gear was 4.4 times as high as the average for the balance of the airplane, as based on frontal area. It is true that landing gear fairing cannot be complete or perfect, but neither can the shape of other parts be such as to produce a minimum drag for a given frontal area. The fuselage, in order to offer sufficient length for tail surfaces, must usually have a fineness ratio of at least 2.5 times the ideal fineness ratio. Landing gear fairing suffers from no such limitation. Furthermore, the fuselage lines must enclose the engine, cool it and dispose of the exhaust gases. The pilot's enclosure and a number of minor items (such as door hinges, door handles, cowl fasteners and the like) add to the fuselage drag. The wings and the tail surfaces have about 2.5 times the fineness ratio of the ideal streamline strut section, and their shape is chosen for considerations other than minimum drag, as based on frontal area. When we consider that an increase of 2.5 times the ideal fineness ratio corresponds to an increase of from 50 to 100 percent in drag, it appears that the airplane we are discussing would have been about eight times as good a job of streamlining as its landing gear. The resistance coefficient of a well-faired landing gear may be equal to the average for other parts of the airplane, but it certainly is not 4.4 times as great.

In the more or less haphazard performance testing employed by the average commercial airplane organization weather conditions and other important variables may be neglected. Change of propeller setting, changing propellers, admitting more air to a starved carburetor, changing to high compression pistons or charging supercharged gears are tricks of the trade which may be employed by an enthusiast to prove his point where he is out from under the watchful eye of an Army or Navy trial board. He is apt to minimize the importance of such changes even in his own mind and consider that they are not worth mentioning.

The estimate of 60 lb. for the difference in weight between a fixed and a retractable gear for a transport of 8-12 seats seems to be rather optimistic even 60 lb. may eliminate one passenger where the weight allowance is on the ragged edge.

One feature of the retractable landing gear which has received very little attention is the effect which its drag has on stability and control. In preparing to land, a pilot ordinarily needs all of his faculties to judge the approach and to handle the controls. If the flying qualities of the airplane and the "fell" of the controls suddenly change when the landing gear is extended the result may be a bad landing or, worse, a stall while turning into the field. Roy G. Miller, Hartford, Conn.

Mr. Miller's interesting comment is very much appreciated. In the general statement I made regarding the possible improvement in an airplane with an efficient retractable gear compared with a fixed external gear, I indicated the increase in speed rather than the reduction in total drag, because I felt that a definite figure for gain in speed was more comprehensible than an abstract drag figure. I made the assumption that faster airplanes would have somewhat less drag and not achieve the improved speed through excess of power alone. Therefore, I assumed that, because of the fixed size of struts, wheels, and tires with materials and loadings now used, as the general cleanness of the airplane was improved, it was not possible to reduce the drag of the landing gear in direct proportion and that the ratio of landing gear drag to total drag would increase. This has been observed on a number of actual designs which were tested with streamlined fixed gears as well as retractable gears. As I mentioned in the article, the possible gain depends much upon the ingenuity of the designer and the purpose, type and size of the airplane.

In the general statement I mentioned that for a conventional airplane with 165-170 m.p.h. with a fixed external undercarriage a gain of 10 percent was possible, corresponding to a reduction in drag of about 24 percent, allowing 1 percent for improved propeller efficiency due to change in V/ND.

The airplane which I used as an example, as having its speed actually increased 14 percent by fully retracting the landing gear, was the low wing Lockheed Sirius. This model with external fixed gear had an exceptionally clean fuselage with filleted cantilever wing and tail. Recent tests show that the cockpit opening did not greatly affect the drag. However, the landing gear drag was far from being consistent with the low drag of the rest of the airplane. The wheel was braced to the wing by a vertical shock strut and a "Vee" of two struts on the inboard side, one to the front spar and the other to the rear spar. The space between the front and rear members of the Vee was completely filled by a flat surface meeting the wing at a fairly acute angle. I believe that this surface and the interference between the surface and the wing accounted for a large part of the drag. The intersection between the shock strut and the wheel fairing might have been improved slightly, while the wheel fairing in top view was considerably longer than the most efficient shape. In addition, the opening in the bottom and the projecting tire also contributed to the drag.

In the caption of the picture illustrating the Lockheed gear I mentioned a flap that was added to the later models completely covering the wheel when retracted. Because of the higher speed this made a difference of 6 m.p.h., according to Mr. Lloyd Stearman, president of the Lockheed Aircraft Corporation. This 6 m.p.h. corresponds to a reduction in drag of 8 percent neglecting any change in propeller efficiency, which would give some 18 percent rather than 14 percent.

I believe the figures for the increase in speed on this plane to be accurate, because a number of planes of this type, both with retractable and fixed gears, were built and all seemed to indicate about the same increase in speed.

Substantiating the above, Mr. George W. DeBell of the General Aviation Manufacturing Corporation states that their Clark GA43, low wing transport, increased its speed over 15 percent by retracting the landing gear. The plane was tested, both with a single strut cantilever gear with struts and wheel streamlined, and with the retractable gear now used. As both the speed of this plane and the Lockheed referred to are believed to be in the same range these values of 14 percent and 15 percent seem to be in agreement.

Mr. T.P. Wright, vice president of the Curtiss Aeroplane & Motor Company, states that in checking over results obtained with several of their models in which retractable landing gears have been installed and on which other data pertaining to no retractable landing gear is available he finds that the speed increase over the tripod type fixed gear is 10 percent. He draws attention to the fact that it is incorrect to compare these speeds with such gear retracted and extended, as in the extended position they have more drag than a fixed external gear initially designed for the plane. His estimate of 10 percent is on the basis of the true comparison, though he does not mention at what speed this is attained.

From the figures I gave in my article, I believe it is clear that if 10 percent, or even somewhat less were gained, it is advantageous to retract the gear.

Regarding the weight of 60 lb. for the addition of a retractable gear to an 8,000 lb. transport, I draw attention to the fact that the complete retraction mechanism on the new Martin bomber weighs only 79 lb. and this is a much larger airplane. Mr. DeBell states that the increase in weight of their plane, which is about this size, was 100 to 110 lb., considering each gear complete. He says this is relatively high, as the majority of the parts are not affected by the weight of the plane, such as the pump, and the valve mechanism, the oil lines, and the cowling to cover the landing gear after retraction. In this plane, this portion of the weight, unaffected by the size of the plane, is 38 lb. I believe that a hydraulic system is inherently heavy.

Mr. Wright agrees that if compared with a tripod gear this weight of 60 lb. appears to be correct. He mentions that on several of their recent planes they have used a single strut cantilever gear which is considerably heavier than the tripod type. With this type of gear he states that their experience is that the speed would only be increased 5 m.p.h. when going to the fully retractable gear, but that the weight would actually be lessened by some 50 or 60 lb.

In considering the additional weight of the retractable landing gear, one should not neglect the saving in fuel due to the improved speed and the weight of the streamlining on the fixed external gear. On the Lockheed Sirius the weight of the wheel fairing is 50 lb.

Regarding the fourth paragraph of Mr. Miller's letter I can only say that I attempted to be conservative in any of the general statements I made and that most were based upon data known to be accurate. I doubt if the gains indicated to me by some of the commercial manufacturers were obtained by supercharging or the use of higher compression ratios. However, I call attention to the fact that the higher speeds, due to the retractable gear, may slightly increase the dynamic pressure on the carburetor intake, slightly supercharging the engine and increasing its power. This I believe is another point supporting the retractable gear, as is the fact that at the higher speeds the opening for taking in air to cool the engine can be reduced, reducing the drag.

I do not believe that very many detailed data exist on the effect of the drag of the landing gear, when down, on the stability or flying characteristics of the airplane.

Richard M. Mock

Document 3-16(a-h)

(a) Fred E. Weick of Hampton Virginia, U.S. Patent 2,110,516. Filed 18 Jan. 1934; issued 8 March 1938. Fred E. Weick Collection, LHA.

(b) Fred E. Weick & Associates, Inc., to Waco Aircraft Co., Troy, OH, 8 Apr. 1938, Weick Collection, LHA.

(c) Weick & Associates, Inc., to Waco Aircraft Co., Troy, OH, 2 Feb. 1939, Weick Collection, LHA.

(d) C. J. Brukner, President, The Waco Aircraft Company, Troy, OH, to Fred E. Weick and Associates, Inc., Evelyn Place, College Park, MD, 16 Feb. 1939, Weick Collection, LHA.

(e) Weick & Associates, Inc., to The Waco Aircraft Company, Troy, OH, Attn: Mr. C. J. Brukner, President, 22 March 1939, Weick Collection, LHA.

(f) Weick & Associates, to Timm Aircraft Corporation, Grand Central Air Terminal, 1020 Airway, Glendale, CA, Attn: Mr. R. A. Powell, Vice-President, 22 March 1939, Weick Collection, LHA.

(g) Alpheus Barnes, The Wright Company, 11 Pine Street, New York, to Mr. Grover C. Loening, c/o U.S.S. *Mississippi,* Pensacola, FL, 16 Jan. 1915, reprinted in Grover Loening, *Our Wings Grow Faster* (Garden City, NY: Doubleday, Doran & Co., Inc., 1935), p. 45.

(h) Grover Loening, "What Might Have Been," in *Our Wings Grow Faster* (1935), pp. 43-46.

Fred E. Weick received two different patents covering different aspects of the steerable landing gear incorporated into the W-1 and W-1A. (His patent attorney was Allen E. Peck of Washington, DC, who had been a pilot during World War I and thus was familiar with aeronautics.) The first was U.S. Patent 1,848,037, filed on 16 February 1931 and issued on 1 March 1932; it mostly concerned his airplane's limited elevator control arrangement. The second (presented below) was U.S. Patent 2,110,516, filed on 18 January 1934 and was not issued until 8 March 1938; it covered some of the airplane's other features, including combinations of the tricycle gear with two-control operations. In addition, this second patent made claims involving the use of a flap to aid takeoff when a tricycle gear was used. Weick's patent application stayed in limbo for over four years because its claims interfered with an application by Joseph M. Gwinn, Jr., of Buffalo, New York. Gwinn had been working on a little airplane called the Gwinn Aircar, which had aims somewhat similar to those of the W-1 and W-1A. The patent court eventually decided that the date of Weick's conception of the W-1 was about a year earlier than Gwinn's design, and Weick received the patent.

The first six documents in the string to follow concern Weick's second patent and his efforts to protect it against infringements. As readers will see, companies building aircraft with some version of steerable tricycle landing gear in the late 1930s, such as the Waco Aircraft Company of Troy, Ohio, argued that their particular design differed from Weick's and therefore did not infringe his patent. Once the basic advantages of this sort of gear were well known within the aeronautical community, it was inevitable that other designers would apply it for their own purposes. This obviously did not work to Weick's financial benefit, but the widespread adoption of the new landing gear arrangement served the needs of American aviation well. Weick consoled himself with the fact that what he had contributed was not a product for sale but a gift to the aviation world.

To the end of his life, Weick felt that he had never seen a tricycle gear on a production airplane that was as effective in making a wide variety of landings, both easily and safely, as that of the W-1 and W-1A. His little plane's nose wheel was placed far out ahead; the gear had 12 inches of shock absorber and was designed much stronger than it needed to be. He and his associates could afford to go overboard in this direction because with the W-1's pusher arrangement, the plane's balance was helped by a little extra weight in the nose. And with the 18 inches of shock-absorber travel for the rear wheels, safe landings on smooth surfaces could be made, as long as the vertical velocity did not exceed about 20 feet per second, almost regardless of the manner in which the airplane was brought to the ground. With the wing-tip clearance made possible by the airplane's high-wing arrangement and five degrees of dihedral, pilots made satisfactory landings while still turning with a 30° bank. The airplane virtually straightened out by itself. With a tractor arrangement, however, it was almost impossible to get the full potential out of the gear. To achieve the effective wheelbase, one would have had to put the nose wheel out in front of the propeller, and this could not be done structurally.

Readers should find the last two documents in this string curious. Both relate to the U.S. Circuit Court Court of Appeals ruling in January 1914 that Glenn Curtiss and his associates had in fact infringed the Wright brothers' patent. The next-to-last document communicates a buoyant reaction of the Wright Company to that ruling. The last provides an excerpt from the autobiography of Grover Loening in which the pioneering aircraft builder gave voice to the potential that seemed to exist in 1914 for an out-and-out aircraft monopoly dominated by Wright.

The whole matter of patents in U.S. aeronautical history deserves a much closer look than historians have given it. Patent policy certainly factored into the reinvention of the airplane, but whether it enhanced or stifled innovation is not altogether clear. Scholars have paid considerable attention to the famous patent suit involving the Wrights and Glenn Curtiss, most recently in Tom D. Crouch, "Blaming Wilbur and Orville: The Wright Patent Suit and the Growth of American Aeronautics," in *Atmospheric Flight in the Twentieth Century*, eds. Peter Galison and Alex Roland (Dordrecht: Kluwer Academic Publishers, 2000), pp. 287-300. In this essay, Crouch argues that the Wright patent did not retard U.S. aviation in the way many observers over the years have claimed. What choked it far more was the failure of the U.S. government to recognize the crucial importance of the new technology.

In the same volume that includes Crouch's essay, there is another essay on patents that offers an even more significant set of important new insights about U.S. aeronautical patent policy history. In "Pools of Invention: The Role of Patents in the Development of American Aircraft, 1917-1997" (pp. 323-345), Alex Roland analyzes the infamous cross-licensing agreement of 1917 that set up a patent "pool" by which U.S. aircraft manufacturers could get around the problems of the existing Wright and Curtiss patents plus avoid the future bottlenecks and high cost that otherwise would have been sure to come with patent infringement suits. Roland's argument is too detailed to do more with here than briefly summarize. Contrary to what one might think, the patent pool was not a factor stifling innovation in the U.S. aircraft industry, because the industry chose not to protect itself in that way. "An invention published in a patent could often be worked around faster by the competition than if it were kept secret until incorporated in an actual aircraft," Roland explains, "whence the competition would have to reverse-engineer it, master its production, and redesign an existing plane to install it as a modification" (p. 339). Keeping things proprietary inside a company for as long as possible was much more vital to the industry than patenting.

Roland shows how this distinguished aircraft manufacturing from many other types of U.S. industries, in which patents played a greater role. The U.S. aircraft industry came to be dominated by very large corporations in which teams produced most innovations rather than lone inventors. Improved design and greater efficiency in manufacture was much more important to the aircraft industry than outright invention. Design process and manufacturing practice was the key to a competitive advantage, not patentable ideas per se. Roland also underscores how the federal government would finance 85 percent of U.S. aerospace research over the years. In this case, patents meant little. Industry did not have much incentive to patent inventions when the government contracts that sustained them (primarily contracts with the military) typically required free licensing. With the government sharing this technology freely with its other contractors, the value of patents diminished even further. This may be the reason why aviation historians have paid so little attention to patents, Roland speculates. Patents simply have not meant much to the U.S. aircraft industry.

This provocative conclusion suggests that the aircraft manufacturers' patent pool—which was later ruled a violation of the Sherman Anti-Trust Act—was not a factor stifling innovation in U.S. aircraft manufacturing. But at least one legal scholar and antitrust expert seems to disagree. In a May 1997 address in San Antonio before the American Intellectual Property Law Association, Joel I. Klein, Assistant U.S. Attorney General for the Clinton Administration's Antitrust Division, Department of Justice, and the main driving force behind the government's antitrust suit against Microsoft, used the early history of aeronautical patents in the United States as a case study of how cross-licensing and patent pools can in fact result in a blunting of competition.

Klein reviewed the history in the following way. On the eve of the country's formal entry into World War I in April 1917, U.S. aircraft manufacturing was in quite a mess. The government needed hundreds of new aircraft, but patents held by the Wright-Martin Aircraft Corp. and by the Curtiss Aeroplane & Motor Corp.—and those patents contested between the two firms—blocked production. Both companies demanded high royalties, which for government procurement meant excessive airplane costs. In response to numerous complaints from the army and navy, in January 1917, the National Advisory Committee for Aeronautics recommended creating an aircraft manufacturers association, one whose first task was to affect the cross-licensing of aeronautical patents. Seven months later, the Manufacturers Aircraft Association (MAA) came to life, with an endorsement from a special advisory committee convened by Assistant Secretary of the Navy, Franklin D. Roosevelt. The MAA quickly reached an agreement, which eventually turned into the Amended Patent Cross-License Agreement of 31 December 1928. From the time of the original agreement in 1917 on (until a consent decree of November 1975 brought on by the Justice Department finally abolished the MAA and terminated the cross-licensing agreement as a Sherman Anti-Trust violation), member companies could enjoy full and inexpensive access to all patents held by the other members. This agreement signaled one of the most crucial milestones in American aviation development, for it prevented a virtual deadlock in aircraft construction that could have come, with countless patent infringement suits.

The patent pool encompassed practically all U.S. airplane manufacturers; the charter members of the MAA involved several of the top companies then in airplane manufacture, including Aeromarine Plane and Motor Company, Burgess Company, Curtiss Aeroplane and Motor Corporation, L.W.F. Engineering, Standard Aero Corporation, Sturtevant Aeroplane Company, Thomas-Morse Aircraft, and Wright-Martin Aircraft Corporation. Many others soon signed on, to everyone's advantage. As Klein explained in his 1997 address, the MAA pool "resolved all pending infringement claims and bound the members to give each other nonexclusive licenses to 'all airplane patents of the United States now or hereafter owned or controlled by them.' " (Klein, "Cross Licensing and Antitrust Law," Address before the American Intellectual Property Association, San Antonio, TX, 2 May 1997, copy available at www.apeccp.org.tw/doc/USA/Policy/speech/1123.htm, p. 5). So tidy was the arrangement that MAA members agreed not to put any relevant patent out of the pool's reach. The cost to members was a flat $200-per-aircraft royalty. One hundred thirty-five dollars of that amount (67.5 percent) went to Wright-Martin and $40 (20 percent) to Curtiss until their patents ran out or until each of the two companies received a maximum of $2 million each, whichever came first; the other $25 per-aircraft-royalty (12.5 percent) stayed in the pool for MAA administrative costs. An MAA arbitration panel decided which patents merited royalties and how much those royalties were to be in dollars. In sum, the MAA worked to manage the entire field of inventions for airplanes, including patent research of all types, the prosecution of patent applications in the U.S. Patent Office, the granting of patent licenses, as well as the arbitration of disputes and royalty awards. This tightly knit arrangement laid the basis for U.S. aeronautical patent policy for the next 58 years, until the consent decree ended it in 1975.

It should be apparent from this summary why Attorney General Klein, one of the moving forces in the U.S. government's antitrust case against Microsoft, would be interested in the history of the MAA and how its management of a patent pool affected subsequent aeronautical innovation in the United States. The formation of the pool in the summer of 1917 provoked a mild uproar, with charges of an "aircraft trust." Under duress, Secretary of War Newton D. Baker, personally delighted with what the pool meant for aircraft procurement, asked U.S. Attorney General Thomas W. Gregory for an antitrust advisory opinion. It took Gregory only a few weeks to answer that the anticompetitive effects of the pool came nowhere close to outweighing "the very real procompetitive benefits resulting from assembling those patents into an affordable package available to all comers" (Klein, p. 6). The MAA pool solved what attorneys today would call the "bilateral monopoly problem;" in 1917, that sort of monopoly was on the verge of being exercised by Wright-Martin and Curtiss. These two companies had the entire embryonic U.S. aircraft industry over a barrel—not to mention each other as well. In September 1917, Attorney

General Gregory essentially concluded that combining the Wright and Curtiss patents into a wider pool would not only be procompetitive (and work out for both companies), but prove also to be the best solution for everyone involved, including the U.S. military then engaged in world war.

Much more so than historian Roland, attorney Klein questioned the effects of the patent pool on long-term innovation. Without actually evaluating American progress in aircraft manufacturing from 1917 on, he emphasized the possible anticompetitive effects of such arrangements when patent pooling deters or discourages participants from engaging in ambitious R & D, thereby retarding innovation. Granting licenses to one another for current and future technology at minimal cost encourages "free-riding" and reduces incentives to compete in R&D efforts, Klein emphasized. In 1917, the urgencies of World War I undermined this concern. The U.S. government needed airplanes, a lot of them, and fast. It was not a good time for fretting about stifling some innovation that might or might not bear fruit 20 or 30 years down the road. In this context, the ruling in favor of the patent pool made pragmatic sense.

Still, in Klein's view, the establishment of the MAA's historic cross-licensing agreement raised significant questions about just how competitive—or noncompetitive—the system of aeronautical patent management in the United States actually proved to be. How innovative could the system be if the patent pool turned out to be the only route by which new technology could travel to the marketplace and into the design of airplanes? How much incentive could there have been for new invention if the pool agreement provided for only a "reasonable" royalty, an amount typically set much lower by the industry's own arbitration jury than it would otherwise often have been?

For American aeronautical development, the matter of patent policy seems to be fundamentally important. Historians need to address the competing conclusions of Klein and Roland. Such research may prove especially important to an understanding of the airplane design revolution of the interwar period, for the young American aircraft manufacturing industry simply could not have afforded to "reinvent" the airplane if every single airplane part, including major systems such as wing design, airfoils, engine arrangement, and landing gear, were all patent-protected. What it needed was an approach that severely limited what a patent protected and that made available at low cost the "generic knowledge" of how to design effective aircraft. This seems to be what the MAA system did, however much in restraint of trade.

U.S. aeronautical patent policy in the 1920s and 1930s seems to be vitally important to the history of the NACA as well. Not only did the NACA promote the original creation of the MAA in order for it to affect the cross-licensing of aeronautical patents (NACA chairman Joseph S. Ames was one of the MAA's founding trustees), the NACA also served through this period as the country's major source of aeronautical innovation, through its R&D establishment. One might argue that by fostering the concept of the patent pool, the NACA ensured that its own organization would serve in that capacity rather than U.S. industry, thereby cementing its role for decades to come. This helped engender the very result that historian Roland cites in his study, that is, that the government would come to finance 85 percent of aerospace research in the United States. In other words, the NACA may have served as a substitute for whatever spirit of competition was lost through the pooling arrangement. Still one can wonder, as Klein does, what new inventions and innovations may have been lost from potential R&D by the U.S. aircraft manufacturing due to the coziness of the overall patent pooling arrangement.

Regarding patents in the special context of our study, it seems clear that very few advances in aerodynamics were patent-protected. With the exception of specific forms or devices such as the "Harlan flap," few aerodynamic improvements took patentable form. As readers will see in the documents below, even when an inventor received a patent, as Weick did for his landing gear, manufacturers managed to integrate the refinement without paying much, if anything, for it.

Document 3-16(a), Fred E. Weick of Hampton Virginia, U.S. Patent 2,110,516. Filed 18 Jan. 1934; issued 8 March 1938.

UNITED STATES PATENT OFFICE

AIRPLANE

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A corporation of Virginia

Application January 18, 1938, Serial No. 185,634

This invention relates to certain new and useful improvements in airplanes; and the nature and objects of the invention will be apparent to and readily understood by those skilled in the art in the light of the following explanation and detailed description of the accompanying drawings illustrating what I believe to be the preferred embodiments or aerodynamical and mechanical expressions of my invention, from among various other forms, embodiments, designs, combinations and constructions of which the invention is capable within the spirit and the scope thereof.

This application is a substitute for and continuation in part of the pending application filed by me July 5, 1934, Serial No. 733,893, for improvements in airplanes.

Fundamentally it is a general aim and a primary object of my present invention to reduce or substantially eliminate the basic hazards and dangers that result from certain features and characteristics inherent in the prevailing designs and types of conventional heavier-than-air craft, or "airplanes" as such craft are generically termed herein. It is generally recognized and established that the conventional airplane, because of such inherent dangers and hazards, can only be practically piloted with any degree of safety under the varying conditions encountered in flight and in taking off and landing, by highly trained and skillful pilots; and that the general use of the conventional airplane is therefore restricted to those having the time and finances for the training and the ability to successfully acquire from such training the necessary and essential piloting technique and skill for safety in flight.

One of the major hazards of the conventional airplane is the landing operation which requires delicate and skillful handling of the airplane and excellent vision and a high degree of depth perception from the pilot to successfully carry perception from the pilot to successfully carry out. This hazard is a direct result of the inability of the conventional airplane to land steeply because of a flat glide and of high landing speed with a long landing run, and further due to the difficulty of contacting the ground accurately at a desired point because of the limited range of gliding angles available to the pilot, even by side slipping the airplane.

A feature of an airplane of my present invention resides in a design providing an airplane that cannot be stalled and has a wide range of gliding angles available to the pilot, and also has a low landing speed and a short landing run on the ground, which together with internal control and stability at all speeds and angles, results in the substantial elimination of the landing hazards because practically no skill is required to accurately maneuver and land the airplane even on small landing areas and over the usual landing area border obstructions.

Another feature of the invention which contributes to the safety of landing and to the ease of pilot handling on the ground, is presented by my design and arrangement of the landing gear in which provision is made for absorbing the maximum vertical landing velocities and for preventing nosing over or "ground looping" under any conditions of landing that may be encountered; and further in which the landing surface engaging elements are arranged so that the landing gear is directionally stable and upon landing surface contact always causes the airplane to tend to follow the direction of landing instead of a path defined by the fore and aft axis or direction in which the airplane is headed, unless the direction of landing is along such axis.

A further feature of the invention is presented by the relative arrangement of the directionally stable landing gear, the body of the airplane and the lifting surface therefore by which the airplane in normal position supported on the ground by the landing gear is in substantially normal cruising flight attitude with the body in substantial horizontal position and the lifting surface is at a relatively small angle of incidence, that is, substantially the angle of incidence for cruising flight.

Another feature resides in the combination with the above arrangement of landing gear, body and lifting surface, of means for increasing the lift coefficient of the lifting surface for take off of the airplane without substantially changing the normal cruising flight attitude of the airplane with the body maintained in substantially horizontal position and the lifting surface at the relatively small angle of incidence, during take off of the airplane.

The conventional airplane essentially has three controls, that is, directional (rudder), lateral (ailerons), and longitudinal (elevator). Such three controls are required in order that the conventional airplane may operably meet all of the conditions encountered in flight operations, as for example, in landing in a crosswind, or in making a landing under conditions that necessitate side slipping the airplane. Generally, the directional or rudder control is operated by the pilot's feet, while the lateral and longitudinal controls are operated manually by the pilot, so that, the pilot must acquire the essential skill and ability to coordinate and synchronize the operation of these controls through his feet and hands. Experience has established the fact that such coordination is difficult to acquire and that crossing or improper coordination of the controls, particularly when flying close to the ground, is a frequent cause of serious accidents with the conventional airplane.

An important feature and a characteristic of this investigation is the provision of a basic design of airplane in which but two controls are required for flight operations, both in the air and in landing and taking off, under all conditions encountered, with the resulting elimination of the difficulties of coordinating and synchronizing three controls as in the conventional airplane, and if desired the elimination of a foot operated control to thereby avoid the necessity of coordinating not only a plurality of controls but also in coordinating and synchronizing the feet and hands in operating such controls.

An airplane design of my invention is further featured by lateral stability and lateral control for and throughout the entire range of speed and angles of attack, to thereby eliminate the dangerous characteristics encountered in the conventional airplane due to lateral instability at low speeds and angles of attack at and approaching the stall, and the insufficient lateral control for the conventional airplane under such flight conditions.

Another feature embodied in a design of airplane of this invention in combination and aerodynamic cooperation with the other features thereof, that contributes to the safety and reduces the piloting skill required, resides in eliminating that characteristic generally found in the conventional airplane of balance at higher angles of attack with power-on than with power-off to other by avoiding the possibility of losing altitude when it is desired to climb, while always insuring climb when full power is applied with the airplane below the maximum level flight speed.

A further general object and a feature of my invention is the provision for maximum range of vision for the pilot, especially in a forward and downward direction, in a design of airplane having the foregoing characteristics for safety by reducing the piloting skill required; and further in the provisions for comfort and reduction of fire hazard, and all of the foregoing in a design that is adapted for relatively low cost production and that is capable of minimum upkeep and operating costs in use.

With the above general features, characteristics and objects in view, as well as certain other features and characteristics that will appear and be readily recognized in the following description, by invention consists in certain novel features in design and in combination and arrangements of aerodynamic and structural elements and parts, all as will be more fully and particularly referred to and specified hereinafter.

As an example of one aerodynamic and structural design expression of the principles, features and characteristics of my invention, I have illustrated in the accompanying drawings, a small, light-weight and relatively low cost airplane of the two-place, high monoplane wing type, that is particularly adapted for general private owner and novice pilot use because of the inherent safety characteristics and low degree of piloting skill required to operate it with safety as compared to the conventional airplane. I have selected the illustrated design of airplane embodying the invention, primarily because an airplane of this general design and type embodying certain of the basic features hereof has been constructed and flight tested and the safety characteristics and the low degree of piloting skill required satisfactorily indicated.

However, there is no desire or intention to limit the invention in all and its various features and characteristics to embodiment in a design and type of airplane of the example hereof. It is recognized and intended that the invention can be embodied in and expressed by various other designs and types of airplanes, as will be apparent to those skilled in the art, and my present invention includes all of such embodiments and adaptations within the broad spirit and scope of the invention.

The design of the illustrated embodiment of my invention is of the high monoplane wing type having an outrigged tail and a pusher propeller and includes the body B, high monoplane wing W and the outrigged tail or empennage E carried from the wing W. The body B is of the closed cabin or nacelle type in which the occupants are enclosed in an upwardly extended, light-weight cabin portion C with a forward windshield and side window arrangement that affords particularly good vision out of the forward side windows for the pilot. The body B and its cabin portion C enclose two seats, in this instance, the forward one S of which is shown as providing the preferable pilot's seat from which the airplane is flown.

The wing W is mounted as and provides a high monoplane wing that extends across the upper or top side of the cabin structure C and above the main body B as defined by the portion of such body that extends forwardly of cabin structure C. The wing W terminates forwardly with it leading edge structure or portion spaced rearwardly from the forward end of body B, and also preferably as here shown, terminating at or short of the forward side of cabin structure C. The trailing edge portion of wing W is disposed preferably to the rear of or approximately at and above the rear or tail end of body B although the design is not essentially limited to such relative positions of body and wing. Preferably, the wing W has a decided dihedral angle for a purpose to be hereinafter explained.

Outrigger girders or spars 10 are mounted and supported from the wing structure W spaced from opposite sides, respectively, of body B, and extend rearwardly from the wing in substantially parallel relation. The rear ends of the outrigger girders 10 mount and carry thereon an empennage E, that consists in the present example of the spaced vertical fins or stabilizers 11 mounted on the spars 10 and extending thereabove and therebelow, the horizontal stabilizer 12 extending across and between spars 10, and the vertically swingable elevator 14 pivotally mounted along the trailing edge of stabilizer 12 and between girders 10 and 11.

A motor M is mounted on the rear portion of body B, preferably the upper portion thereof and along the fore and aft axis of the body, and drives a pusher propeller P which is disposed and positioned between the outrigger girders or spars 10, the central section of the wing W between the spars 10. The design thus presents an initial safety feature in locating the propeller P as surrounded and guarded by the wing and body at the forward side, the girders 10 at the opposite lateral sides and the empennage E at the rear side, so that injury from inadvertent contact with the propeller when the airplane is on the ground is practically eliminated.

The landing gear for the airplane, which in this example happens to be of the land type, consists of the spaced rear wheels 15 having a very wide tread and disposed at opposite sides of the rear portion of body B beneath wing W and aft of the center of gravity of the airplane, and the forward wheel 16 mounted at the forward end of body B along the longitudinal axis of the airplane and forward of its center of gravity. These wheels are each of the so-called "air wheel" type familiar in the art and capable of withstanding considerable side loads without failure. The arrangement of the wheels 15 and 16 of the landing gear relative to the body B and the lifting surface W is such that in normal position supported on the ground by the landing gear, the airplane is in substantially normal cruising flight attitude with the body B in substantially horizontal position with its longitudinal axis approximately parallel to the ground and with the lifting surface W at the relatively small angle of incidence for cruising flight. The mounting and relative arrangement and operation of this landing gear and the wheels 15 and 16 thereof form important features of my invention and will be referred to and explained in detail hereinafter.

Basically, according to an airplane design of my present invention, the landing and take off difficulties and hazards of the conventional airplane, are materially reduced by providing for a wide range of glide angles and a steep angle of climb. I attain the desired result by designing the wing W of a so-called "high lift" type having a high drag and by which the airplane can attain a steep angle of glide with a relatively low rate of vertical decent. The type of high lift wing W here selected embodies an auxiliary airfoil A fixed in a certain spaced relation forwardly of and along the leading edge of wing W, and by which as familiar to those skilled in the art, a high drag and lift can be obtained. Attention is specifically directed at this point to the fact that my invention is not limited or restricted to any particular form and type of "high-lift" wing, or wing to give the required increase in drag and lift, as wings of the "flap" or "slotted" or other suitable types may be employed if desired or found expedient.

For example, I have in the present instance, included trailing edge flaps F on the wing W of the manually operated and controlled type, and such flaps can be efficiently used in conjunction with the type of high lift wing W, or a wing of the automatically operating flap type can be substituted for wing W. In connection with the flaps F, I have purely diagrammatically and without regard to efficient location, illustrated a manual control for operating flaps F, which flaps are suitable connected together for simultaneous raising and lowering. The manual control may include the operating hand lever 17 in the body B accessible to the pilot, connected with and operating bell crank 18 by the push-pull tube 19, which bell crank is connected to the flaps F by the tubes 20 operatively connected to and coupled by a suitable bell crank in the wing. Primarily, in the design of this example, the manually controlled flaps F are provided for selective use by a pilot if he finds it is difficult to accustom himself to the normal action for the design of this example with its wing W, of pulling the control stick back to increase the angle of glide in the landing maneuver. By using the flaps F, the angle of glide can be satisfactorily adjusted in landing, through operation of the control level 17, as will be readily understood by one skilled in this art.

As lateral instability at high angles and low speeds is a primary danger in conventional airplanes, I have by my present invention insured lateral stability and control throughout the entire range of speeds and angles of attack which can be maintained in flight. I have accomplished this by providing sufficient longitudinal stability in the airplane, and by limiting the upward travel of the longitudinal control of elevator 14 to a point where the airplane not only cannot be maintained in a stall but also cannot be forced into a spin. Such longitudinal control limitation is fully discussed in United States Letters Patent No. 1,848,037 issued to me March 1, 1932, and it will suffice to here state that the upward travel of elevator 14 in the present design is limited in any suitable manner, such for example as disclosed in my aforesaid patent, to a point where the airplane cannot be maintained in a stall or forced into a spin. In the present design with the high lift wing W, the stall occurs at a high angle of attack, approximately 25°, and even though the upward travel of elevator 14 is insufficient to enable flying at the stall, ample elevator control is found present and available throughout the flight range.

The problem of lateral control at and beyond the stall, inherent in the conventional airplane, is eliminated from an airplane designed in accordance with this invention, because of the longitudinal control and stability relationship and the inability of the airplane to be flown or maintained at the stall. Therefore, conventional or other lateral control means may be employed for the design and satisfactory lateral control is insured therefrom at all speeds and angles of attack at which the airplane may be flown.

By the design and arrangement, in cooperation, of the high lift wing W and the limitation of upward travel of elevator 14, together with lateral stability and control throughout the range of speed and angles of attack, the airplane requires no particular skill to land it, other than maneuvering to contract the ground at the desired point. The wind range of gliding angles including a steep angle of glide make the landing approach an easy maneuver calling for no particular or special degree of

skill. The airplane will itself practically take care of contact with the ground without particular attention on the part of the pilot, and if landed with its wings approximately level laterally at any speed within approximately 30 miles per hour of the minimum speed, the landing will be safe whether the airplane is leveled off before ground contact or continued in the glide straight to the ground, with the landing gear of the present example, that embodies certain important features of the present invention.

The landing gear as hereinbefore referred to includes the rear wheels 15 behind, and the forward wheel 16 ahead, of the center of gravity of the airplane and so arranged and mounted as to make it practically impossible for the airplane to nose over. The rear wheels 15 are preferably provided with the usual or any suitable brakes (not shown) but even with a full application of the brakes the arrangement of the landing gear is such as to prevent the airplane nosing over. The rear wheels 15 are disposed spaced a wide distance apart at opposite sides of and spaced from the body B toward the rear thereof and behind the center of gravity of the airplane. Each rear wheel 15 is mounted on a truss 21 extended laterally from the adjacent side of body B and mounted for swinging thereon to permit vertical movement of the wheel, and a long travel shock absorbing strut 22 extending between truss 21 and the wing W there above.

In accordance with the invention, the forward landing wheel 16 of the landing gear is mounted and arranged so as to be normally freely laterally swingable or castering for cooperation with the directionally fixed rear landing wheels 15 to provide the directionally stable landing gear for the airplane. I have disclosed herein one possible form of mounting and arrangement to attain the lateral swinging or castering operation of the forward landing wheel 16, in which example, the forward wheel 16 is carried by a long travel shock absorbing strut that includes the upper section 23 mounted in the nose or forward end of the structural frame of body B and the lower section 24 rotatable and also vertically movable in the upper section. The lower end of the strut section 24 is provided with the fork 24a in which the forward landing wheel 16 is mounted and by which the wheel is vertically movable and laterally swingable or rotatable to carry out its castering function. The shock absorbing strut 23—24 that mounts and carries forward wheel 16 is mounted on the body B in fixed position with its vertical axis inclined rearwardly, for example, a rearward inclination of approximately twenty degrees (20°) may be used. The forward wheel 16 is thus mounted and arranged so that the area of landing surface contact of the wheel is to the rear of the point at which the projection of the rearwardly inclined vertically disposed axis about which the wheel rotates meets the landing surface, and, as a result, this forward landing wheel will function to caster or rotate into the direction of travel. The normally freely castering forward landing wheel 16 cooperates with the directionally fixed real wheels 15 that are located aft of the center of gravity of the airplane, in such a manner that upon ground contact of the landing gear the castering front wheel 16 will caster or rotate into the direction of travel of the airplane. For example, in landing the airplane with such directionally stable landing gear, if the airplane is landing with side drift, then immediately upon ground contact of the gear, the front wheel 16 will caster or rotate into the direction of travel of the airplane and in cooperation with the directionally fixed rear wheels, automatically turn or head the side drifting airplane into the direction of travel.

Such a directionally stable landing gear also enables accurate handling of the airplane in taxiing on the ground and substantially eliminates any tendency of the airplane to ground loop.

The shock absorbing mountings for the front and rear wheels of the landing gear, consisting of the shock absorbing struts 22 and 23—24, have a long vertical travel to sustain and absorb the landing loads at the maximum vertical velocities of landing. As the rear wheels 15 sustain the largest load, their shock absorbing struts have a greater vertical travel than the strut 23—24 for the directionally stable front wheel 16. The maximum vertical positions of the landing wheels with the shock absorbing struts collapsed are shown in dotted lines while the lowered positions of the wheels with the struts extended are shown in full lines. In connection with the travel of the shock absorbing means and landing wheels, I have found that with an airplane of the invention weighing approximately 1150 pounds, the landing gear should be capable of withstanding a vertical velocity of about 25 feet per second, and an 18 inch vertical travel for rear wheel shock absorbers 22 with a 12 inch travel for front wheel shock absorber 23—24, should be satisfactory.

While in the specific example hereof, I have shown a three wheel landing gear with a single forward directionally stable wheel 16, it is to be clearly understood that my invention includes a plurality of forward directionally stable landing wheels, spaced as may be desirable, and with or without the disclosed arrangement of directionally fixed rear wheels. Also, attention is called to the fact that other landing surface engaging elements than wheels may be employed including skis, water landing members such as floats, pontoons and the like, as the invention is in no sense limited to ground engaging landing wheels.

As a further feature of the invention, the forward landing wheel 16 is made steerable for ground handling and taxiing of the airplane, and in the instant example, steering of the directionally stable front wheel 16 is carried out by means of a forwardly extended horizontally disposed arm 25 that is mounted for lateral swinging on a vertical shaft 26 on which it is mounted at its rear end. A brace or truss 25a is preferably mounted extended between the forward end of arm 25 and the lower end of shaft 26, the shaft 26 being of course suitable mounted for rotation around a vertical axis in fixed position in the body B. A rod or link 27 having a bifurcated forward end pivotally connected to the wheel fork 24a, extends rearwardly upwardly and freely slidably through a vertically disposed guide 28 mounted on the forward end laterally swingable arm 25. The guide 28 is pivotally mounted on the arm 25 for free rotation about a vertical axis.

By swinging arm 25 to the right or left the rod 27 is swung to rotate or turn the

front, directionally stable landing wheel 16 to the right or left to steer the airplane when on the ground. The pivotal mounting of rod 27 to wheel fork 24a permits of free vertical travel of the shock absorber section 24, and the pivotal mounting of guide 28 permits free lateral swinging of arm 25 and 37 while operatively coupled. Steering operation or movement of arm 25 is carried out in the present example by and from the pilot's control system for the airplane, as will be described and explained hereafter.

An important feature of the invention made possible by the basic design and directionally stable landing gear, as hereinbefore described, is the use of but two controls by the pilot for complete flight, landing and take-off operation under all the varying conditions encountered in such operations. By this feature, either the rudder or directional control, or the aileron or lateral control of the conventional three-control system may be eliminated. In the preferred design and control arrangement hereo the airplane is provided with only a longitudinal or elevator control and a single control for changing the direction of flight, the conventional rudder control being eliminated. Such two-controls do away with the possibility of crossing controls and materially simplify the process of learning to fly, particularly eliminating the necessity for coordinating foot and manual control operating members.

The airplane is provided with the fixed, preferably adjustable, horizontal stabilizer 12 with the usual vertically swingable elevator 14 for longitudinal control. The usual rudder or directional control is eliminated and fixed vertical fins 11 of sufficient area for directional stability are provided carried at the tail of the airplane on the outrigger 10. Directional and lateral control, that is, control in yaw and roll, is obtained from the wing mounted opposite control surfaces 30, which in this instance are of the so-called spoiler type familiar in the art. Such spoilers 30 are mounted on opposite wings in the upper surface thereof and are differentially vertically swingable to raised position and to lowered position within the wing. The spoilers 20 give a yawing moment similar to that given by a rudder and also at the same time give a rolling moment for lateral control.

A pilot operated control system for operating the two control arrangement described includes a usual pivotally mounted control column 31, rockable or swingable fore and aft of the airplane for longitudinal control, and mounting at its upper end a control wheel 32 rotatable for directional control. The control column 31 is mounted in position for operation by the pilot from forward seat S and is pivotally mounted for fore and aft rocking about the pivot 31a. Control cables 33 and 34 are connected to column 31 and below pivot 31a, respectively, and are extended rearwardly over suitable pulleys to guide them along outrigger girders 10 to the upper and lower ends of the elevator horn 14a, so that, forward movement of the control column will lower the elevator and rearward movement thereof will raise the elevator.

The hand wheel 32 on the upper end of control column 31 rotates a drum 32a, to which the opposite cables 35 and 36 are connected and from which cables 35 and 36 extend and are guided over suitable pulleys to the inner or lower ends of the horns 37 of opposite spoilers 30, respectively, within wing W. The upper ends of the spoiler horns 37 above the wing are connected in the usual manner by a cable 38 guided over suitable pulleys into and through the wing. By rotating wheel 32 to the right and left spoilers 30 are differentially vertically swung to directionally control the airplane to the right or left and at the same time generate a rolling moment acting in the proper direction.

If desired, conventional ailerons can be used in place of the spoilers 30, but due to the improved yaw characteristics from the spoilers the latter are preferred.

With the directionally stable landing gear of the invention, and the other characteristics of the design giving the high drag and lift with steep angle of glide and low landing speed and short landing run together with lateral stability and control throughout the range of speed and angles of attack, the two-control system as described gives full control for flight and for landing and taking off under all conditions and even in cross winds and with side drift.

The directionally stable forward landing wheel is steerably connected into the pilot's control system and this is accomplished by connecting opposite sides of the swingable arm 25 with the opposite spoiler operating cables 35 and 36 by the cables 40 and 41, respectively. Cables 40 and 41 are connected to opposite sides of arm 25 and then extended around opposite pairs of pulleys and connected to cables 35 and 36 leading to and operated by hand wheel 32. Thus, as the hand wheel 32 is turned to the right or left with the airplane on the ground, the directionally stable landing gear 16 is turned to the right or left to steer the airplane in handling or taxiing in movement on the ground. Of course, a separate steering control can be provided for front wheel 16, or this wheel can be unconnected and self-turning or castering if desired, but preferably, the steerable and directionally stable landing gear wheel 16 is operatively coupled with the pilot's control for ground steering.

The two controls made possible by the design of airplane of the invention may, instead of the elevator and the wing mounted directional controls consist of tail mounted rudder or directional controls and the tail mounted elevator or longitudinal control. For instance I have purely diagrammatically illustrated an arrangement of a two-control system for the airplane of the present example, which provides the elevator 14 for longitudinal control and the rudders 45 swingably mounted in the usual manner along the trailing edges of the vertical fins 11 on the outrigger girders 10. The elevator 14 is operatively coupled to control column 31 by cables 33 and 34 while the rudders 45 are connected by cables 46 and 47 with the drum 32a of the control wheel 32. Cables 46 and 47 extend over and are guided by suitable pulleys, from wheel 32 to the spaced rudders 45 where they are connected to the respective rudder horns 45a. A cable or wire 48 connects the two rudders to operatively couple them for swinging by cables 46 and 47, as will be readily understood.

With the two control arrangement, sufficient control directionally and laterally is obtainable in cooperation with the other design features of the invention, if the wing W is given sufficient dihedral as here shown, for handling and maneuvering the airplane in the air, as well as in landing or taking off even in cross winds. As in the two-control system arrangement eliminates the foot operated control and the necessity of acquiring the technique and skill required for coordinating a foot and a manually operated control.

The two controls may consist of the elevator 14 providing the pitch control, and a single control for changing the direction of flight that consists of the opposite ailerons or roll control surfaces R and the vertical rudders 45 connected with the ailerons R through the ailerons operating mechanism or cables so as to be simultaneously operated with the ailerons. In the present example, the ailerons or roll control surfaces R of this single control for changing the direction of flight are of the more or less conventional trailing type mounted on the wing W at opposite sides of the longitudinal axis of the airplane and actuated by a control mechanism for differential operation in the more or less conventional manner.

Such an aileron operating mechanism may, for example, embody a horn or crank 50 fixed on and extending above and below each aileron R together with a cable 51 interconnecting the upper ends of the opposite aileron horns 50, the cable 51 being carried over and around suitable pulleys 51a at opposite sides of the wing. A cable 52 is connected to the lower end of the horn 50 of the right-hand aileron R and this cable 52 extends over suitable pulleys 52a to the drum 32a of the control wheel 32 on the upper end of the control column 31. A cable 53 is connected to the lower end of the horn 50 of the left-hand aileron R and extends around suitable pulleys 53a to the drum 32a of the control wheel 32 on the upper end of the control column 31. Thus, by turning control wheel 32 to the right or to the left, the opposite ailerons R are differentially displaced through the arrangement of control cables 51, 52, and 53, in the usual manner to obtain roll control for the airplane.

The vertical rudders 45 have the hereinbefore described operating cables 46 and 47 extended forwardly along the outrigger spars 10 and in the present form on two control systems of the invention, these rudder operating cables 46 and 47 are connected into and with the control cables 52 and 53, respectively, of the right and left wing ailerons R. For instance, rudder operating cable 47 for the right-hand rudder 45 after passing over suitable pulleys is connected to the cables 53 from the left wing aileron R, while the cable 46 from the left rudder 45, after passing over suitable pulleys is connected to the cable 52 from the right ring aileron R. Thus, the cables in the example hereof, form a means for connecting or coupling the rudders with the ailerons for simultaneous operation as a single control. If desired, any suitable detachable connecting means may be employed for coupling cables 46 and 47 to the cables 52 and 53, respectively.

The rudder operating cables 46 and 47 so connected into and with the aileron operating cables 52 and 53 thus provide for simultaneous operation of the rudders and ailerons for roll and yaw to thereby provide a wing control for changing the direction of flight of the airplane. This single control is, through the cables 52 and

53, operable from and by the steering or control wheel 32 of the pilot operated control unit which includes the control column 31. Rotation of the control wheel 32 to the right will lower the left wing aileron R and simultaneously swing or displace the vertical rudders 45 to the right, so that the direction of flight of the airplane will be changed to the right. When the control wheel 32 is rotated to the left, the reverse movements or displacements of the ailerons R and rudders 45 take place and the direction of flight of the airplane is changed towards the left.

The foregoing arrangement of two controls consisting of the control for pitch and the single control for roll and yaw comprising the combination of aileron or roll control surfaces and vertical rudders in combination with the directionally stable landing gear of the present invention as hereinbefore described and explained, enables complete control and maneuvering of the airplane for all normal conditions of take off, flight and landing, including the landing of the airplane with side drift, by the operation solely of such two controls.

In accordance with another feature of an airplane design of my invention, the airplane illustrated balances at a slightly lower angle of attack with the power on than with the power off. As the longitudinal control is limited to prevent sustained stalling with either power on or power off, the airplane will always climb if full power is applied with the airplane at speeds below the maximum level flight speed. In this manner, I have eliminated the possibility of the airplane losing altitude in straight flight at low speed with full power on. The general design of the illustrated example which provides the pusher propeller and a high line of thrust, contributes to balancing the airplane at a slightly lower angle of attack with power on than with power off, so that it becomes impossible to maintain a stalled attitude in either case.

In connection with the landing characteristics and the ground handling of the airplane, my design provides the relation between the landing gear and wing of the airplane such that the angle of incidence of the wing is approximately 0° when the airplane is at rest on the ground. In landing, therefore, as soon as the ground is contacted, the wing angle of attack is immediately reduced to 0° so that wing lift is reduced to a negligible amount to prevent any tendency of the airplane to float off the ground if landed above its minimum landing speed. This feature of the design also materially facilitates handling the airplane on the ground in high winds.

The invention provides a relation between the body B, the wing or lifting surface W and the rear wheels 15 and forward wheel 16 of the landing gear, such that with the airplane supported in normal position by said wheels on the ground, the body B is maintained in substantially horizontal attitude with its longitudinal axis approximately parallel to the ground. Thus, the airplane, when supported on the ground in normal position has the body B and the wing or lifting surface W in substantially normal cruising flight attitude with the occupants' seats in their normal attitude for natural seated position. The invention further provides for the maintenance of this normal cruising flight attitude of the airplane during take off, that is, with the body B in its substantially horizontal position and the wing or lifting surface W at its normal cruising flight angle of incidence.

The relation of the wing or lifting surface W to the landing wheels and the body B in normal ground attitude of the airplane is such that the wing or lifting surface W has an angle of incidence, which, for the particular wing used in the present example, will give substantially the lift coefficient used in normal cruising flight. For instance, with the particular wing or lifting surface W of the examples hereof, the angle of incidence of the wing when the airplane is in normal position supported on the ground by the landing wheels, that is, is in normal cruising flight attitude, is approximately zero degrees (0°). Hence, in normal cruising flight attitude of the airplane, the wing also has such approximately zero degree (0°) angle of incidence. Such an approximate zero degree (0°) angle of incidence, where, as in the example hereof, the angle of incidence of the wing with the airplane in normal ground position is the same as the angle of incidence of the wing in normal cruising flight attitude of the airplane is such an angle as will give the wing a lift coefficient of a certain percentage of the maximum lift coefficient for the wing. In the form of the wing W, having the auxiliary airfoil A, the wing, when at its approximately zero degree (0°) angle of incidence, has a lift coefficient approximately or of the order of one-fifth (1/5) of the maximum lift coefficient for such wing. On the other hand, the wing W, without the auxiliary airfoil A, when at the approximately zero degree (0°) angle of incidence, has a lift coefficient approximately or of the order of one-fourth (1/4) to one-third (1/3) of the maximum lift coefficient for such wing.

In terms of speed, such cruising angle of incidence range may be said to be such that in order for the wing to develop a sufficient lift to enable the airplane to take off, the airplane must travel at a rate of speed of at least of the order of fifty percent (50%) in excess of the minimum landing speed for the airplane.

When taking off with the airplane having the landing wheels, body and lifting surface so relatively arranged, the normal horizontal attitude of the body B and the above-referred to angles of incidence for the wing or lifting surface W may be maintained. However, due to the aforesaid relatively low angles of incidence, a relatively long take off run is necessary in order for the airplane to attain a speed necessary for the wing at such angles of incidence to develop a lift sufficient to take off the airplane. Under certain ground conditions, and particularly muddy or sandy conditions, it is difficult, or, at times, even impossible, to attain a sufficient speed for the airplane to develop a lift that will get the airplane off the ground. With the landing gear of the invention having the wheels 15 to the rear of the center of gravity, and particularly where the airplane may, for example, as in the case of an amphibian, have a high thrust line, it is practically impossible under the foregoing ground conditions to get the tail of the airplane down because in such case, the drag on the wheels gives a moment tending to press the nose of the airplane down. Thus, an increase in the angle of attack of the wing W by changing the attitude of the body B and the wing W in order to increase the lift during take off under such conditions can not be accomplished.

In accordance with a feature of my invention, means are provided in combination with the foregoing arrangement of the landing wheels, body and wing of the airplane, through the medium of which the lift of the wing W can be increased during take off without changing the normal cruising flight or ground supported attitude of the airplane and its body B and wing W. For instance, as one example of such a wing lift varying means, I have provided a lift varying wing flap F on the wing W together with the pilot controlled mechanism for operating this flap that includes the control lever 17 as hereinbefore described. Thus, during the take off of the airplane, the lift of the wing W can be arbitrarily selectively increased by the pilot to thereby enable the airplane to take off with a minimum of ground run and without changing the normal horizontal attitude of the body B or the normal cruising flight angle of incidence of the main or fixed portion of the wing or lifting surface W.

The various features of the invention are not necessarily limited in a design of airplanes to the inclusion therewith of the two-control system, as such features may be used to advantage with the conventional three-control system, and similarly the use of a conventional wing is not precluded in a design in which certain of the other features are incorporated, as such other features can still contribute advantages when embodied in a design of airplane having conventional wings. The directionally stable landing gear of my present invention is not restricted to use with an airplane embodying any of the other features and characteristics of the invention, but is of general use on airplanes of various other designs, including the conventional.

As a result of the principles, features and characteristics of my invention, an airplane designed to embody and incorporate them has a high degree of safety and requires very little skill to fly and to land and take off. Due to the wide range of gliding angles for the pilot to select from, together with the characteristics of low minimum gliding speed, lateral stability and control throughout the full speed and angle of attack range, and the inability to remain in the stall, the airplane can be easily and accurately landed on a very small space and practically by merely guiding the airplane to the field and letting it glide into contact with the ground. The combination with the foregoing characteristics of the directionally stable landing gear further simplifies and reduces the landing skill required and enables safe landings in cross winds. The necessity for but two controls made possible by the above features of the design, still further reduces the skill required to operate the airplane and makes learning to fly and operate the airplane a simple and rapid process. The operation of taking off is also rendered easy by the characteristics of the design which result in a short take off run and steep climb, thereby making it possible to easily take off from small fields and clear surrounding obstructions.

It is also evident that various other changes, modifications, variations, substitutions, eliminations and additions might be resorted to without departing from the spirit and the scope of my invention and hence I do not desire to limit my invention in all respects to the exact and specific disclosures hereof.

What I claim is:

1. In aircraft, the combination of a body, a lifting surface, directionally fixed landing wheels to the rear of the center of gravity of the aircraft, a normally freely castering landing wheel forward of the aircraft center of gravity and adapted for cooperation with said directionally fixed landing wheels to provide a landing means for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with side drift and means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and a control for changing the direction of flight.

2. In an airplane, in combination, a system of controls for the airplane consisting solely of a pitch control and a control for roll, and a landing and taxiing gear for the airplane that is normally directionally stable when supporting the airplane on a landing surface with the airplane moving forward, said landing gear embodying directionally fixed landing surface engaging means mounted on the airplane to the rear of the airplane center of gravity, and normally freely castering landing surface engaging means mounted on the airplane forward of the airplane center of gravity and adapted for cooperation with said directionally fixed landing surface engaging means to provide a landing gear for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing gear when the airplane is landing with said drift.

3. In an aircraft, the combination of a body, a lifting surface, directionally fixed landing surface engaging means to the rear of the center of gravity of the aircraft, a normally freely castering landing surface engaging means forward of the aircraft center of gravity and adapted for cooperation with said directionally fixed landing surface engaging means for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with side drift, and means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and a control for changing the direction of flight that includes a swingable vertical rudder surface.

4. In an aircraft, in combination, directionally fixed landing surface engaging means to the rear of the center of gravity of the aircraft, a normally freely castering landing surface engaging means forward of the aircraft center of gravity, and adapted for cooperation with said directionally fixed landing surface engaging means for changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with side drift, and means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and one or more vertical rudder surfaces laterally swingable for changing the direction of flight.

5. In an aircraft, in combination, a body, a lifting surface, a directionally fixed landing surface engaging means to the rear of the center of gravity of the aircraft, a normally freely castering landing surface engaging means forward of the aircraft center of gravity and adapted for cooperation with said directionally fixed landing surface engaging means for changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with

side drift, and means for controlling the aircraft in normal maneuvers in landing and flying comprising a pitch control embodying a horizontally disposed elevator surface vertically swingable about a horizontal axis, and a vertical rudder surface laterally swingable for changing the direction of flight, said vertical rudder surface being located to the rear of the aircraft center of gravity.

6. In an aircraft, in combination, a directionally fixed landing surface engaging means to the rear of the center of gravity of the aircraft, a normally freely castering landing surface engaging means forward of the aircraft center of gravity and adapted for cooperation with said directionally fixed landing surface engaging means for changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with side drift, and means for controlling the aircraft in normal maneuvers in landing and flying comprising a pitch control consisting of a horizontally disposed elevator surface vertically movable about a horizontal axis and a control for changing the direction of flight that includes lateral control surfaces at opposite sides of the longitudinal axis of the aircraft.

7. In an aircraft, a directionally fixed landing surface engaging means to the rear of the center of gravity of the aircraft, castering landing surface engaging means forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing surface engaging means to provide a landing means for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing means when the aircraft is landing with side drift, in combination with, an air control system for the aircraft consisting solely of a pitch control and a single control only for both yawing and rolling control for the aircraft.

8. In an aircraft in combination, means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and a control for changing the direction of flight, a landing and taxiing gear for the aircraft embodying directionally fixed landing wheels to the rear of the center of gravity of the aircraft, and a castering landing wheel forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing wheels for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing gear the aircraft is landing with side drift, and pilot actuated means operatively connected with said forward castering wheel for arbitrarily swinging said wheel for steering the aircraft on a landing surface.

9. In an aircraft in combination, means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and a control for changing the direction of flight, a landing and taxiing gear for the aircraft embodying directionally fixed landing wheels to the rear of the center of gravity of the aircraft and a castering landing wheel forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing wheels for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing gear when the aircraft is landing with side drift, mechanism for actuating said direction of flight control, and said castering landing wheel being operatively connected with said control actuating mechanism whereby operation of said mechanism to actuate the direction of flight control swings said forward castering wheel to steer the aircraft when on a landing surface.

10. In an aircraft, in combination, means for controlling the aircraft in normal maneuvers in landing and flying comprising solely a pitch control and a control for changing the direction of flight, a single manually actuated control operating unit for operating both the pitch and the direction of flight controls, directionally fixed landing wheels to the rear of the aircraft center of gravity, a castering landing wheel forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing wheels to provide a landing means for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing wheels when the aircraft is landing with side drift, and said castering landing wheel being operatively connected to said single control operating unit so that selective operation of said unit to actuate the directional control swings said castering landing wheel for arbitrarily steering the aircraft on a landing surface.

11. In an aircraft, in combination, a system of air controls for the aircraft consisting solely of a pitch control and a single control only for both yawing and rolling control of the aircraft, a single manually operable control unit for operating both said pitch control and said single control for yaw and roll, a landing and taxiing gear for the aircraft comprising directionally fixed landing wheels to the rear of the aircraft center of gravity and a castering landing wheel forward of the aircraft center of gravity, and means operatively connecting said forward castering landing wheel with the manually operable control unit whereby selective operation of the latter to actuate said single control for yaw and roll swings said castering wheel for arbitrarily steering the aircraft on a landing surface.

In an aircraft, a body, a lifting surface, a landing gear for the aircraft, embodying directionally fixed landing wheels to the rear of the air craft center of gravity and a landing wheel forward of the air craft center of gravity, said wheels being so mounted and arranged that with the air craft supported thereby in normal position on the ground, said body is in substantially horizontally disposed attitude with its longitudinal axis approximately parallel to the ground, said lifting surface being so mounted relative to the body and to the landing wheels that with the aircraft in normal position supported on the ground the lifting surface has an angle of incidence of approximately zero degrees (0°), in combination with means for increasing the lift coefficient of said lifting surface when the aircraft is moving forwardly in normal position supported on the ground by said landing wheels without changing the normal horizontal attitude of said body or the aforesaid normal angle of incidence of said lifting surface.

13. In an aircraft, in combination, a body, a lifting surface, a landing gear for the aircraft consisting of directionally fixed wheels to the rear of the aircraft center of gravity and a castering landing wheel forward of the aircraft center of gravity, said wheels being so mounted and arranged relative to the body and lifting surface

that with the aircraft supported by the wheels in normal position on the ground, said body is in substantially horizontally disposed attitude with its longitudinal axis approximately parallel to the ground, said lifting surface being mounted relative to the body and to the landing wheels so that with the aircraft in normal position supported on the ground the lifting surface had an angle of incidence such that the lift coefficient for the lifting surface at such incidence angle does not exceed approximately one-third (1/3) of the maximum lift coefficient for such lifting surface, means for increasing the lift coefficient of said lifting surface when the aircraft is moving forwardly in normal position supported on the ground by said landing wheels without changing the normal substantially horizontal attitude of said body or the aforesaid normal angle of incidence of said wing, and mechanism under the control of the pilot for arbitrarily selectively operating said lift increasing means.

14. In an aircraft, in combination, a body, a landing gear for the aircraft embodying landing wheels respectively spaced forward of and to the rear of the aircraft center of gravity, a lifting surface for the aircraft, said landing wheels being so mounted and arranged relative to said body and to said lifting surface that when the aircraft is supported by said wheels in normal position on the landing surface, said body is in substantially horizontally disposed normal cruising flight attitude and said lifting surface is at an angle of incidence such that the lift coefficient for the lifting surface at such angle does not exceed approximately one-third (1/3) of the maximum lift coefficient for such lifting surface, a flap member mounted on said lifting surface for movement to positions increasing the lift coefficient of said surface, and pilot actuated mechanism for moving said flap member whereby the lift coefficient of said lifting surface can be increased for the take off run of said aircraft to thereby enable the aircraft to take off without changing the normal horizontal attitude of the aircraft body or the normal angle of incidence of the main portion of said lifting surface.

15. In an aircraft, in combination, a body, a landing gear for the aircraft comprising directionally fixed landing wheels to the rear of the aircraft center of gravity, a normally freely castering landing wheel forward of the aircraft center of gravity adapted for cooperation with the said directionally fixed landing wheels for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing gear when the aircraft is landing with side drift, a lifting surface for the aircraft, said directionally fixed and said castering landing wheels being so mounted and arranged relative to said body and to said lifting surface that when the aircraft is supported by said wheels in normal position on the ground, said body is in substantially horizontally disposed attitude and said lifting surface is at an angle of incidence such that it requires the aircraft to travel at a rate of speed of over approximately fifty percent (50%) in excess of its minimum landing speed in order to take off, a flap member mounted on said lifting surface for movement from normal position to positions increasing the lift of said surface, and pilot actuated mechanism for selectively arbitrarily moving said flap member to a lift increasing position during the take off run of said aircraft without changing the normal horizontal attitude of the aircraft body or the normal angle of incidence of said lifting surface to thereby enable the aircraft to take off at a rate of travel less than that required for the lifting surface at such normal angle of incidence and with the flap in normal position.

16. In an aircraft, a body, a lifting surface, directionally fixed landing wheels to the rear of the center of gravity of the aircraft, normally freely castering landing wheels forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing wheels for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing wheels when the aircraft is landing with side drift, in combination with, air controls comprising a control for pitch embodying an elevator surface vertically displaceable about a horizontal axis, a control for roll embodying control surfaces at opposite sides of the longitudinal axis of the aircraft, and a control for yaw embodying a laterally swingable vertical rudder surface, means for connecting the vertical rudder surface and said roll control surfaces together for simultaneous operation only, as a single combined yaw and roll control, and a manually operable control unit operatively connected with said combined yaw and roll control, whereby the aircraft can be landed with side drift solely by the operation of said pitch control and the said combined yaw and roll control.

17. In an aircraft, a body, a lifting surface, directionally fixed landing wheels to the rear of the aircraft center of gravity, a normally freely castering landing wheel forward of the aircraft center of gravity adapted for cooperation with said directionally fixed landing wheels for automatically changing the aircraft heading to the direction of landing upon ground contact of said landing wheels when the aircraft is landing with side drift, in combination with an air control system comprising solely a control for pitch and a control for changing the direction of flight, said direction of flight control embodying a laterally swingable vertical rudder surface and roll control surfaces at opposite sides of the longitudinal axis of the aircraft, means for connecting said vertical rudder surface and said roll control surfaces for simultaneous operation only, as a single control, and a single manually operated control unit connected with said pitch control and with said control for changing the direction of flight whereby the aircraft can be landed with side drift solely by operation of said pitch control and said direction of flight control.

FRED E. WEICK

Document 3-16(b), Fred E. Weick & Associates, Inc., to Waco Aircraft Co., Troy OH, 8 Apr. 1938.

April 8, 1938.

Timm Aircraft Corporation, Waco Aircraft Company
Glendale, California; Troy, Ohio

Gentlemen:

We see from the aeronautical magazines that your company is manufacturing an airplane incorporating both a tricycle landing gear and a wing flap.

U.S. Patent No. 2,110,516 has just been granted us for this combination, as well as for the combination of a tricycle gear with two air controls. Both of these combinations were developed on our W-1 airplane, described in various publications (Aviation for July, 1934, and January, 1936). A copy of the patent is enclosed for your information.

We plan to issue a license to any manufacturer who desires to use the features incorporated in this patent, and will shortly send you a license agreement for your consideration. We have worked out a royalty rate that is sufficiently low to avoid burdening the industry or retarding its development.

We shall be glad to answer any questions you may have on this subject.

Very truly yours,

FRED E. WEICK & ASSOCIATES, INC.

Document 3-16(c), Weick & Associates, Inc., to Waco Aircraft Co., Troy, OH, 2 Feb. 1939.

February 2, 1939.

The Waco Aircraft Company
Troy, Ohio
Attention: Mr. C. J. Brukner, President

Gentlemen:

Last July we forwarded you two copies of a license agreement in connection with U.S. Patent No. 2,110,516, which agreement we requested you to consider.

Since that time, our President, Mr. Weick, has spoken with you twice, at which time you indicated that you might be willing to sign this agreement, but that you did not consider your present airplane as infringing on the patent under consideration. In view of this stand, will you give us in detail the reasons why you believe your airplane does not infringe of the patent.

We would appreciate a reply at your early convenience.

Very truly yours,

FRED E. WEICK & ASSOCIATES, INC.

Document 3-16(d), C. J. Brukner, President, The Waco Aircraft Company, Troy, OH, to Fred E. Weick and Associates, Inc., Evelyn Place, College Park, MD, 16 Feb. 1939.

Feb. 16, 1939

Fred E. Weick and Associates, Inc
Evelyn Place,
College Park, Maryland

Att. R. Sanders, Exec, Vice-President

Gentlemen:

Your letter of February 2nd indicates a confused impression of the discussions the writer has had with your President, Mr. Fred Weick, on the subject of Patent No. 2,110,516. Since, naturally, there would be no occasion for our signing an agreement with you until we began, or intended to begin, to utilize the patent in our production, we analyzed our tricycle model airplane with each claim of this patent and reached the conclusion that we were not utilizing the same.

Since there are several claims that contain the term, "approximately," we recognized the possibility of our interpretation of this comparative word being different from your own, and we were glad for the opportunity afforded by Mr. Weick's visit to Dayton last summer to obtain his views as to how far away from the fractions given in the claim he considered "approximately" to reach. He expressed himself on this subject, and we gave him our data, which fell comfortably outside his estimated range, and we therefore seemed to agree that our airplane did not go far enough in the direction of your patent to fall within its scope.

We assured Mr. Weick that we felt that our trend would be in the direction of your patent rather than away from it, but it has not been so far, and, therefore, the status remains the same.

We did not supply Mr. Weick with written evidence of our design details, but volunteered to show him our pertinent data if he would visit our plant on the next day after our discussion, or to fly the airplane over to Wright Field where he was then occupied, and where his extreme familiarity with the subject would enable him to verify our representations which had formed the basis of our discussion. Mr. Weick phoned on the next day that he would be unable to give time to either of these routines.

In our recent discussion with Mr. Weick on this subject, we stated that there is a possibility that we might make a design change which would bring this airplane within the scope of your patent; and that in such case we would notify you of our desire to negotiate on the subject of a license, your proposal of which we have in our file for consideration.

For your further information—there is one of our tricycle airplanes in the possession of the C.A.A., at present situated near Washington on radio and blind landing experiments, which could, no doubt, be inspected by your personnel; and if you will advise which one of the seventeen claims of your patent, in your opinion, most nearly reads on this airplane, we will supply you with whatever data will assist in a definite determination of the question.

Yours very truly,

THE WACO AIRCRAFT COMPANY,

C. J. Brukner,
President.

Document 3-16(e), Weick & Associates, Inc., to The Waco Aircraft Company, Troy, OH, Attn: Mr. C. J. Brukner, President, 22 March 1939.

March 22, 1939.

The Waco Aircraft Company
Troy, Ohio
Attention: Mr. C. J. Brukner, President

Gentlemen:

We thank you for your letter of February 16th, giving us your impressions of the conclusions reached between yourself and Mr. Weick in your recent interview.

Mr. Weick is not in full agreement with all your conclusions. However, in view of your stand, we will let the matter await further developments as we are convinced that you will shortly come to the conclusion that you must make certain changes in your airplane which will place it definitely under the claims of our patent.

Very truly yours,

FRED E. WEICK & ASSOCIATES, INC.

Document 3-16(f), Weick & Associates, to Timm Aircraft Corporation, Grand Central Air Terminal, 1020 Airway, Glendale, CA, Attn: Mr. R. A. Powell, Vice-President, 22 March 1939.

March 22, 1939.

Timm Aircraft Corporation
Grand Central Air Terminal, 1020 Airway
Glendale, California
Attention: Mr. R. A. Powell, Vice-President

Gentlemen:

We acknowledge receipt of your letters of February 7th and 13th, advising us that you do not believe that your airplane comes under the various claims in our patent, and also advising us that you had been working on your steering mechanism prior to the date of the filling of the patent.

With regard to the patent date, the patent issued was merely a reapplication of an earlier patent filed in January, 1934, and, since we established a date much earlier than 1934 in connection with an infringement which we had to fight prior to obtaining our patent, we believe that we pre-date any work which you did in this direction.

With respect to the steering of the nose wheel, you will note that our patent claims cover the steerable nose wheel as well as the freely castering nose wheel. However, our claims as to the flap used in take-off refer to a combination of wheels which tends to reduce the angle of attack to less than its maximum when the airplane is on the ground, and do not take into account the method of locking or controlling the wheels.

Although you state that you have no difficulty in taking off without resorting to flaps, the use of flaps on take-off will greatly reduce the run of such an airplane, particularly under adverse field conditions, and, if your airplane is capable of taking advantage of this feature, it falls under the claims of our patent.

In view of the foregoing, we still contend that your airplane infringes on one or more of the claims of our patent and we, therefore, request that you enter into a license agreement, a copy of which we enclose herewith for your consideration.

Very truly yours,

FRED E. WEICK & ASSOCIATES, INC.

Document 3-16(g), Alpheus Barnes, The Wright Company, 11 Pine Street, New York, to Mr. Grover C. Loening, c/o U.S.S.* Mississippi*, Pensacola, FL, 16 Jan. 1915.

The Wright Company
11 Pine Street
New York, Jan. 16, 1914

Mr. Grover C. Loening
c/o U.S.S. Mississippi
Pensacola, Florida

My dear Loening:

No doubt you have read or heard of the decision of the Court in our favor. I enclose a copy of the decision so that you can see for yourself it is quite true.

Our company will control the manufacture and sale of machines in the United States absolutely.

Am off for Dayton Sunday and look forward to a very pleasant visit there as I feel sure things will now begin to hum.

Yours sincerely,

Alpheus Barnes

Document 3-16(h), Grover Loening, "What Might Have Been," in* Our Wings Grow Faster *(1935).

To understand what follows, a little insight into Orville Wright's personality is needed.

To begin with, let me make it clear that Orville is a scientist and a real engineer. On a flying field I once found Glenn Curtiss standing near one of his new planes. I asked him a simple question about the approximate area of a tail surface. Curtiss answered, "Oh, I don't know, but if it isn't right, the boys will fix it." And in that answer is the evidence that Curtiss was a promoter and not an engineer or even his own designer, excepting in a vague way. But if Orville was asked a similar question, he would bring out of his pocket a little memorandum book he always carried and tell you exactly, not approximately, the figures inquired about. He directed all the design work in the shop, even to small metal fittings, and many a time I had designed some detail and made a fine drawing of it, only to find that meanwhile

Orville had gone into the shop and, with one of his old trusted mechanics, such as Charley Taylor or Jim Jacobs, he would not only have designed the part, but had it made right there.

Factory organization was pretty difficult, but things got done as long as Orville was well and on the job. On the other hand, he would delay making an important decision and drive us all nuts trying not to disobey his orders on the one hand and yet not knowing what to do next.

The treasurer of the Wright company at this time was Alpheus Barnes of New York, who was at Dayton most of the time, and really also represented the New York group of directors and stockholders, Andrew Freedman, August Belmont, Cornelius Vanderbilt, Robert Collier, etc.

Barnes did all the bookkeeping work as well as advertising and negotiating of contracts and left the management of the plant pretty much up to me. He was a hearty, well-built, and genial character, smoking cigars continually and full of good stories. I quickly saw that Orville really didn't like him, but that did not bother Barnes much.

Orville was good at business, I thought, in that very few people could put anything over on him, but he certainly did not have any "big business" ideas or any great ambition to expand. He seemed to be lacking in push—which I attributed to two causes: his loss of Wilbur, and his becoming increasingly ill. In addition Orville and his sister Katherine had, preying on their minds and characters, the one great hate and obsession, the patent fight with Curtiss. It was a constant subject of conversation, and the effort of Curtiss and his group to take credit away from the Wrights was a bitter thing to stand for. There was a good deal of justification for it. Once Orville showed me the letter Curtiss wrote in 1907 or thereabout, agreeing "never to use for commercial purposes" the priceless data on wing surfaces, balance, etc., which the Wrights let him have for the use of a non-profit, scientific body known as the Aërial Experiment Association, fathered by Alexander Graham Bell—and from which Curtiss presently transformed his activities into an airplane company.

Orville sued Curtiss for revenge and prestige.

But the New York bankers interested in the Wright company sued Curtiss because they wanted to establish a monopoly. And Barnes would daydream with me of what it would mean if the Wright company would win the suit.

Suddenly one day the decision was announced. The Wright patent on the fundamentals of the airplane was valid, and the Herring-Curtiss Company were found guilt of gross infringement.

There was great excitement and anticipation, because the Wright company could legally have entered upon a totally different policy and could then and there have begun a monopoly of the field that would have been no less all-embracing than that of the Telephone Company. There were untold millions of dollars ready in New

York to be invested in such a trust. In no time Curtiss and what other companies there were could have been closed down or bought up, and we would have seen a totally different development of flying starting here and spreading to Europe exactly as the telephone monopoly did. When we look back on it, it might have been a better thing for aviation. Many destructive rivalries would have been stopped, and with the World War just getting ready to start, one hesitates to think what different a great rich legal trust might have made.

At any rate it did not happen because of one man—Orville Wright. With the winning of the suit, his revenge on Curtiss seemed satisfied, and all he wanted was tribute—royalties from everyone. He did not want to expand. He fought the New York interests. The antagonism between him and Barnes became acute. And the main thing was that nothing much was done, because Orville would not make decisions. He was a sick man, and few of us knew it.

Document 3-17(a-b)

(a) A. L. Klein, "Effect of Fillets on Wing-Fuselage Interference," *Transactions of the American Society of Mechanical Engineers* 56 (1934): 1-10.

(b) James A. White and Manly J. Hood, "Wing-Fuselage Interference, Tail Buffeting, and Air Flow About the Tail of a Low-Wing Monoplane," NACA *Technical Report* 482 (Washington, 1934).

The first major explanation of the value of wing fillets came in June 1932 at a meeting of the Aeronautics Division of the American Society of Mechanical Engineers at Berkeley, California. Arthur L. Klein, a professor of aeronautics at Caltech, explained how the addition of fillets removed virtually all of the aerodynamic disadvantages of low-wing monoplanes, making them just as efficient as high-wing designs. Klein's address, reproduced here in its entirety complete with the discussion that followed, marked the appearance of yet another "shelf item" in the reinvention of the airplane. His findings, based on GALCIT tests of the Northrop Alpha, made it clear that a smooth "fairing" of the joint formed where the wing met the fuselage would greatly improve not just the efficiency, but also the handling characteristics, of low-wing monoplanes.

As shown in the second document below, NACA research supported Caltech's findings. Not only would fillets help in the case of low-wing monoplanes, they would also aid more generally with buffeting and interference problems experienced by aircraft. The NACA tests being discussed in this document involved a series of tests in Langley's Full-Scale Tunnel on a small monoplane known as the "Doodlebug." The designer of the Doodlebug, James S. McDonnell, built the diminutive monoplane in hopes of winning the $100,000 prize in the Daniel Guggenheim International Safe Airplane Competition of 1929, which was based on the notion of producing a "Model T of the Air." The Doodlebug actually never entered the contest as it was damaged in a forced landing on the way to the contest. Anyway, the Great Depression effectively nixed the market for popular light planes. McDonnell approached the NACA with the idea of its buying the aircraft, saying that it would make an interesting subject for a number of difficult experiments. The NACA did purchase it, for $5000 (McDonnell's total venture had cost about $30,000, mostly financed by Philip Wrigley of chewing gum fame), and it made many tests with it both in the FST and in flight. The NACA engineers found that with wing fillets and the addition of an NACA cowling (specifically designed by Fred Weick), the little airplane experienced greatly reduced buffeting and aerodynamic interference.

The Doodlebug was a very effective and progressive design. It had Handley Page automatic leading edge slots, slotted flaps, and long-travel shock absorbers. As intended, it could fly in and out of very small places. At the 1930 National Air Races in Chicago, McDonnell put on a remarkable exhibition with the airplane. Right in front of the grandstand he staked out a circle 150 feet in diameter with little flags. He would take off from within the circle, fly around, and land with his entire landing run within the circle also.

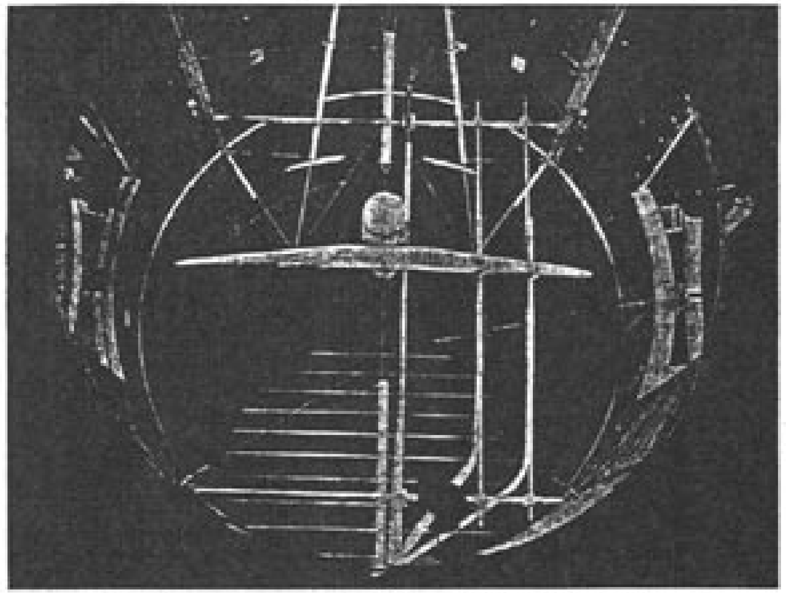


Fig. 1 View of Model as a High-Wing Monoplane in the Tunnel
(The three pitot-tube combs can be seen in the background.)

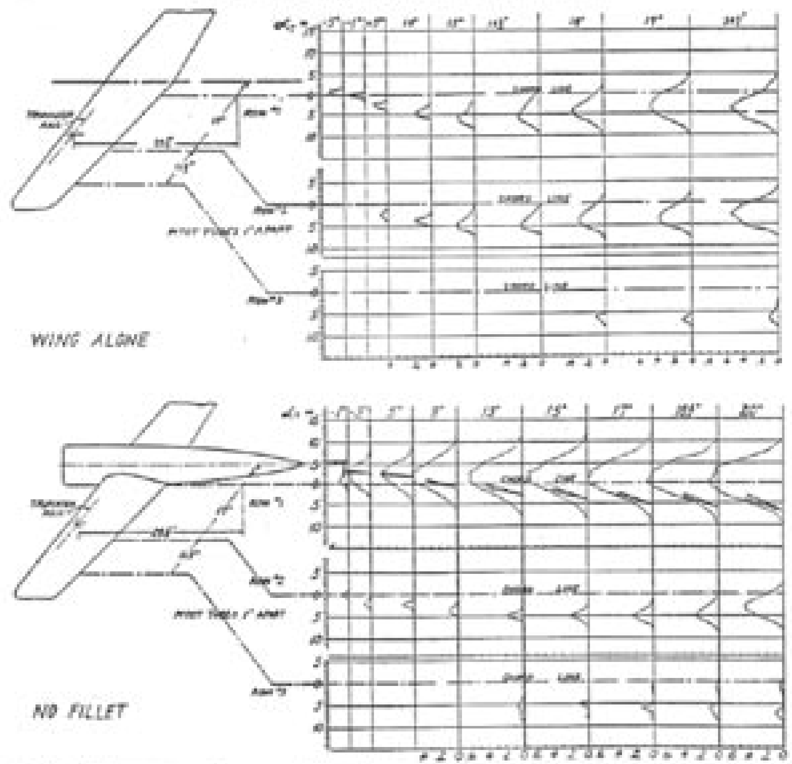


Fig. 2 Wake Distribution Behind the Wing Alone and the Wing and Fuselage With No Fillet
(The area between the vertical lines represents the energy lost.)

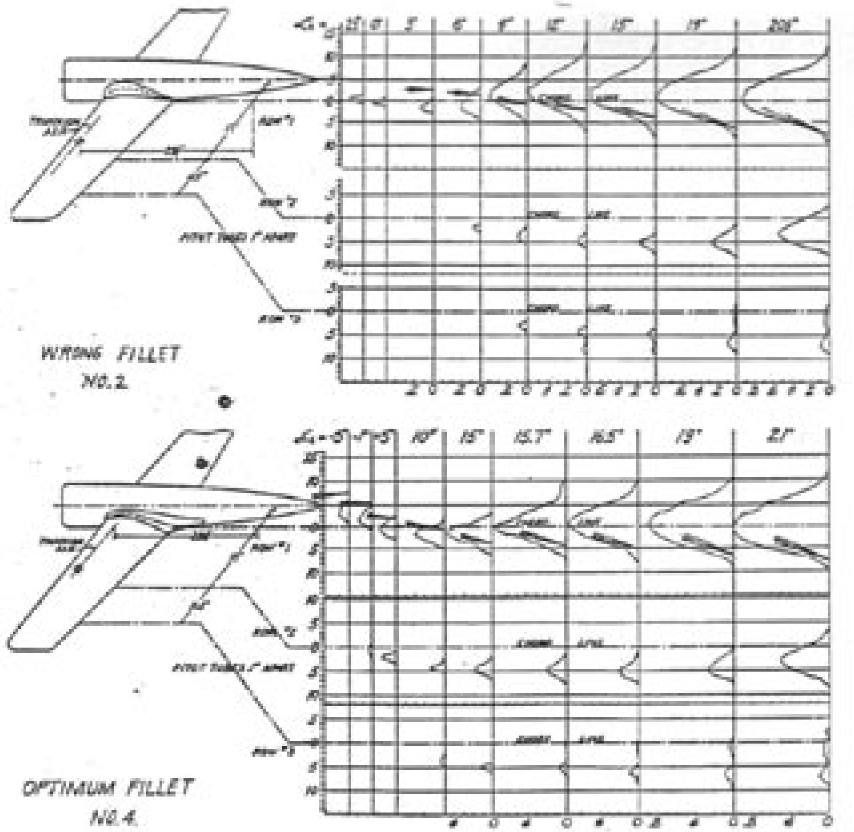


FIG. 3 WAKE DISTRIBUTION BEHIND FILLET No. 2 (WRONG FILLET) AND OPTIMUM FILLET No. 4

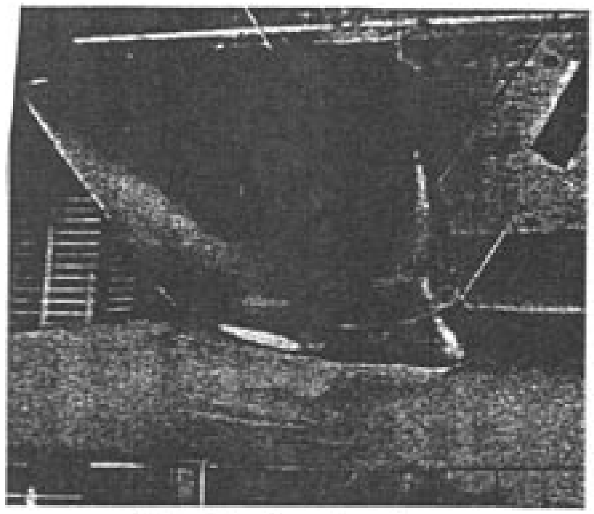


FIG. 4 THE WRONG FILLET No. 2
(Note that the model is inverted and the pitot tubes can be seen in the background.)

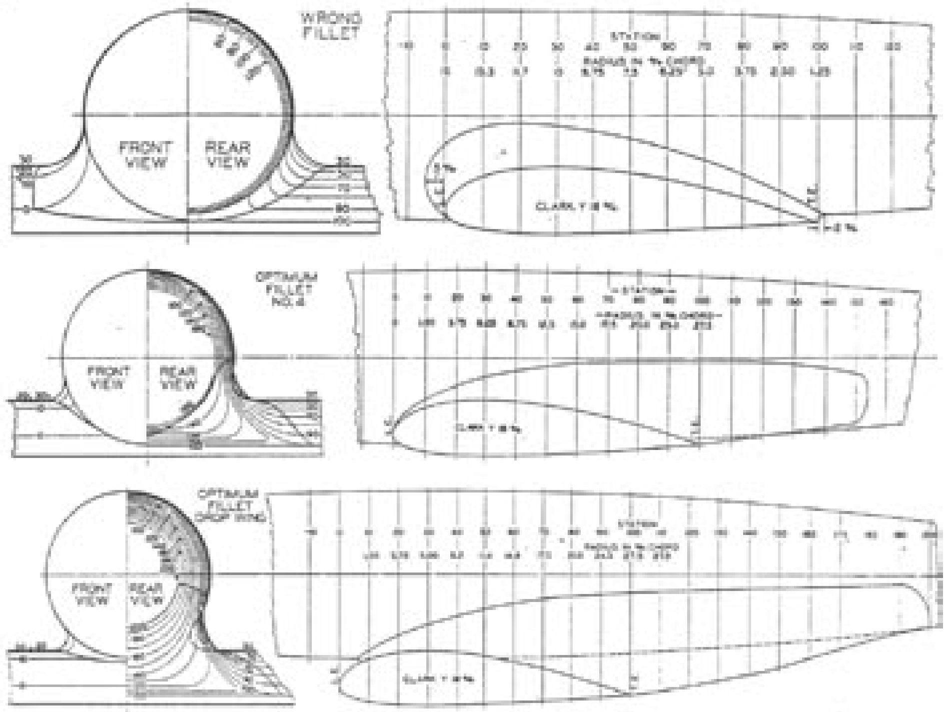


FIG. 5 LINES OF THE WRONG FILLET No. 2, OPTIMUM FILLET No. 4, AND FILLET D OF THE DROPPED WING

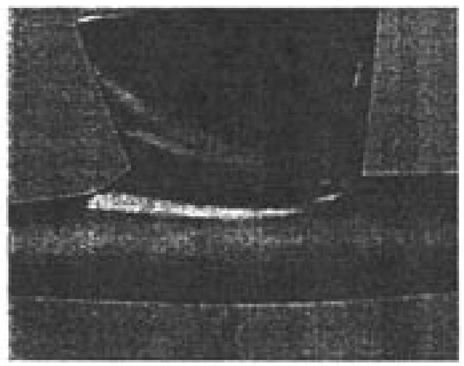


FIG. 6 FILLET No. 3

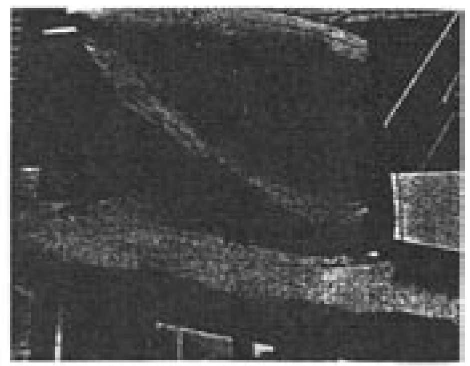


FIG. 7 WRONG FILLET No. 4

AERONAUTICAL ENGINEERING AER-56-1 5

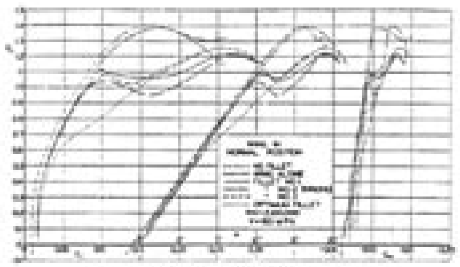


Fig. 8 Normal Low-Wing Configuration

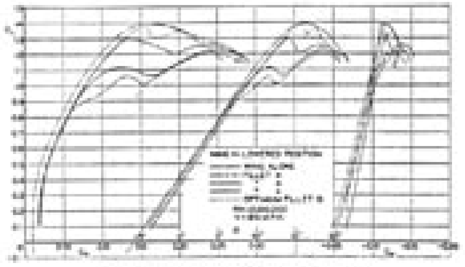


Fig. 9 Dropped-Wing Configuration

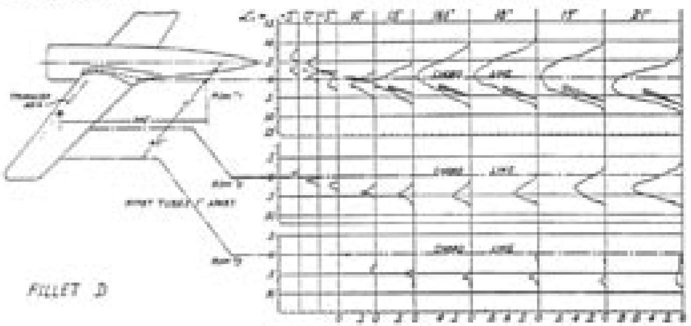


Fig. 10 Wake Diagram for Optimum Fillet D

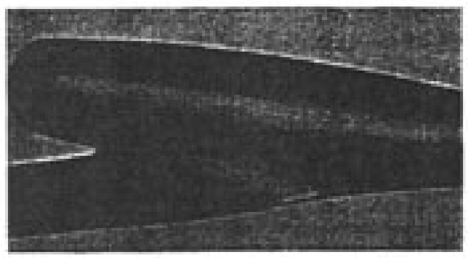


Fig. 11 Optimum Fillet D (Wing Lowered)

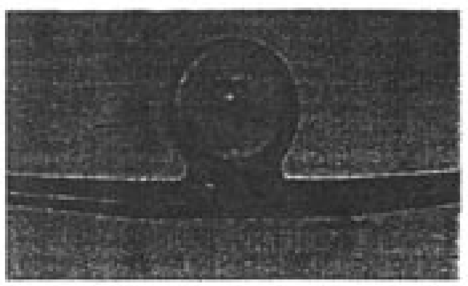


Fig. 12 Optimum Fillet D (Wing Lowered)

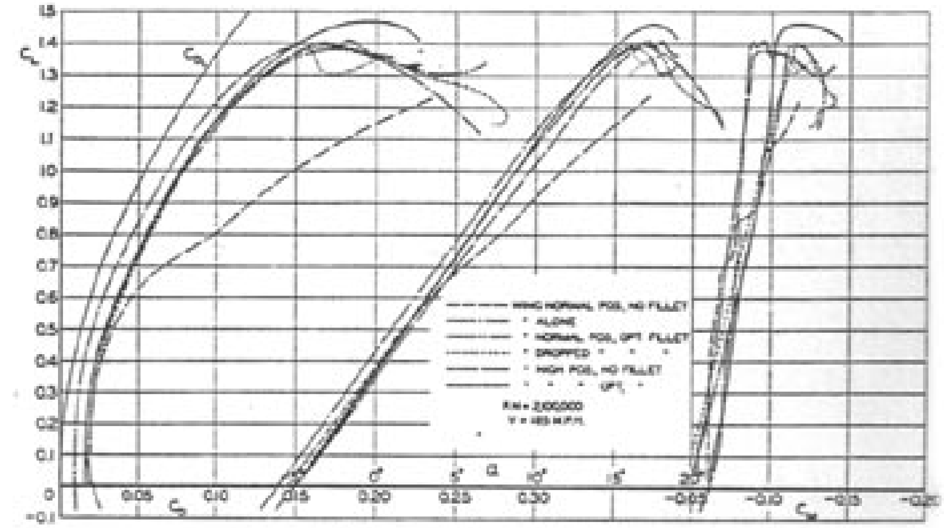


Fig. 13 Curves for Various Configurations

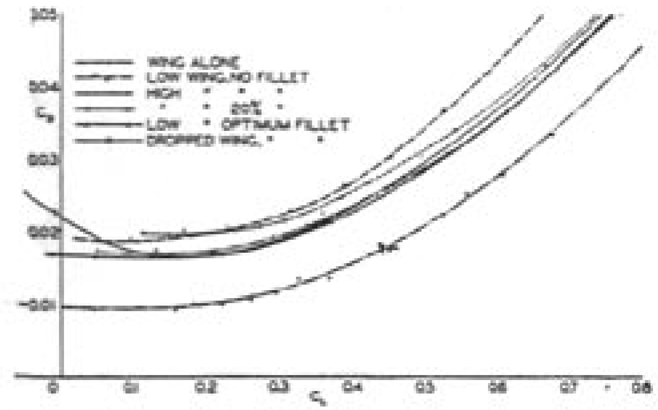


Fig. 14 Comparison of Various Configurations in the High-Speed Region Showing the Experimental Scatter

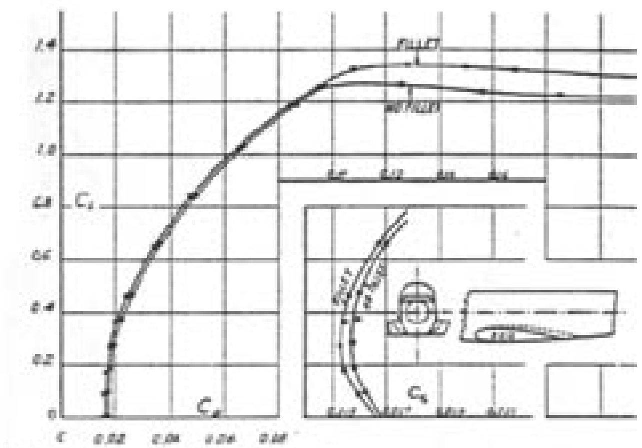


Fig. 15 Lift and Drag Polar Curves for the Case of a F
selage Wing Combination in Which the Fuselage Is Straig

Document 3-17(a), A. L. Klein, "Effect of Fillets on Wing-Fuselage Interference,"* Transactions of the American Society of Mechanical Engineers *56 (1934): 1-10.

EFFECT OF FILLETS ON WING-FUSELAGE INTERFERENCE
By A. L. Klein, Pasadena, Calif.

Mutual interference of wing and fuselage has been the subject of many previous investigations, but these have been on a very small scale. A comprehensive investigation into the case of a low-wing monoplane was thought to have important possibilities, especially as this type of airplane was known to have some aerodynamical peculiarities. In a preceding investigation it had been found that the addition of large fillets to the intersection of wing and fuselage would cause a great improvement and it was decided to extend these tests and to include the case of a high-wing monoplane for comparison.

Numerous investigations have been made on the mutual interference of wings and fuselages, but much of the preceding work has been done on a very small scale. It was thought that a comprehensive investigation into the case of the low-wing monoplane might be of importance, especially as this type of airplane was known to have some aerodynamical peculiarities. In a preceding investigation it had been found that the addition of large fillets to the wing-fuselage intersection would cause a great improvement; it was therefore decided to extend these tests and to include the case of the high-wing monoplane for comparison.

The model used for the investigation was a 1/6-scale model of the Northrop "Alpha," a Wasp-engined transport plane of approximately 5000 lb gross weight. The model had a span of 7 ft and a length of 52 in. The airfoil section was 19 percent Clark-Y at the root and 12 percent at the tip. The wing area of the model was 8.33 sq ft, and the root chord of the wing was 16.67 in. The wing was mounted with reference to the fuselage as shown in Fig. 5. The model as used consisted of a wing and fuselage only. This model was presented to the laboratory by the Northrop Aircraft Corporation, a unit of United Aircraft and Transport Company.

EXPERIMENTAL METHODS

The investigation was carried on by two distinct methods. The set-up for force measurements was identical with that described in ref. 2, while in addition three combs of pitot tubes were mounted behind the model for exploration of the wake behind the wing. These combs can be seen in Fig. 1, and also some of the pitot tubes can be seen in Figs. 6 and 7. The total pressure orifices were made of 1/8-in. brass tubing mounted on a steel tube. For static-pressure measurements, jackets containing side orifices were slipped over the total-pressure tubes. These tubes were connected to the multiple manometer, and observations were made of the wake distribution behind the set-up for various angles of attack. Records of the wake observations were made by taking shadowgraphs of the manometer on ozalid paper. These records were then reduced by plotting curves showing the loss of total head as percentage of the total head in the free stream. Figs. 2, 3, and 10 show curves of this type, and also show the relative position of the model and the combs. The position of the stabilizer and elevator for each angle of attack is indicated on the curves. The fillets used in the investigation were built up from physicists' soft wax, a compound of beeswax, Venice turpentine, and rosin, and modeled to templets. It was found that an almost glasslike surface could be given this wax by rubbing it with sandpaper dipped in kerosene.

WAKE OBSERVATIONS

Fig. 2 shows the wake losses behind the wing tested alone. Noting first the wake losses in row 1, it is seen that as the angle of attack increases, the center of the wake moves downward, as one would expect. When the wing reaches an angle of attack of 15 deg, the center of the wake has reached its lowest point, near which it remains until the angle increases to 16.5 deg, after which the wake moves noticeably upward. If one now looks at the force-measured curves for the wing alone (Fig. 8), one sees that the angle for maximum lift is 16.5 deg. In row 2 the behavior is similar to row 1; the downwash causes the wake to go downward until the angle of maximum lift is reached, after which the wake moves upward. In the case of row 3 an anomaly occurs; the downwash does not reach the maximum values at 16.5 deg, but continues to increase, and apparently stalling does not occur at this section of the wing until an angle of attack of approximately 20.5 deg is reached. The wake curves for small angles in rows 2 and 3 are omitted because the wake was smaller than the distance between the pitot tubes, so that no deductions could be drawn as to its magnitude and shape.

Below on the same figure is a similar set of observations for the model assembles as a low-wing monoplane with no fillet at the intersection between the wing and the fuselage. The maximum diameter of the fuselage was 9.5 in. Pitot comb 1 was mounted approximately halfway between comb 2 and the wing tip. The positions of the stabilizer and elevator are drawn in on the successive curves. It will be noticed that the wake curves for row 1 differ completely from those for the wing alone, while the wake curves in rows 2 and 3 are hardly distinguishable from those for the wing alone. The wake curves in row 1 are not only larger in area than those for the wing alone, but also the position of their center line indicates a much smaller downwash. It will be noticed that the center line of the wake never goes below the chord line, while for the case of the wing alone the center of the wake moves down below the chord line at small angles and remains there. These two conditions represent the limiting cases of the present investigation, the wing alone giving the condition for zero interference and the wing and fuselage with no fillet the case of maximum interference.

Now consider the case of a fillet that partly remedies the interference effect. Fig. 3 shows the wake distribution for what will be called the "wrong fillet." Fig. 4 is a photograph of this fillet, and its lines are shown in Fig. 5. In the case of the "wrong fillet," the wake curves in row 1 are similar to those for the wing alone up to an angle of 6 deg, but at and above an angle of 9 deg the wakes in this row are similar to those for the wing and fuselage with no fillet. The fact that this behavior is not observed at rows 2 and 3 indicates once more that the effect is largely localized in the region of the intersection of wing and fuselage. The force-measurement curves (Fig. 8) discussed later show that at the angle of 7 deg something happens to the flow. The change in the flow pattern from the wing-alone type to the no-fillet type is completely discontinuous. An observer watching the multiple manometer when the model is set at this critical angle sees either one pattern or the other, and the change from one to the other is sudden and complete. At this angle the manometer is in a very unsteady condition. The smaller flow pattern for the downwash and the larger for the trailing vortex replace each other rapidly on the tubes. These are no intermediate stable states; either the manometer is showing one pattern or the other, or it is changing rapidly. The method of watching the multiple manometer for the change in pattern was found to be extremely sensitive, as improvements could be made and the critical angle checked visually with an accuracy of ¼ deg. This made an exceedingly rapid method of observing the effect of modifications, as no computations were necessary, and all that was needed was to start the tunnel and to run the model through a range of angles of attack. The critical angles obtained in this manner checked very accurately with the breaks in the polar, lift, and moment curves. Fig. 3 shows the wake curves for a fillet of the optimum type. The critical angle in this case was 15.7 deg. The lines of this fillet are shown in Fig. 5, and it is illustrated in Fig. 7.

FORCE MEASUREMENTS

The force measurements were made with the normal three-component set-up. The tests were run at a water head of 39.24 cm, and as the root chord of the model was 42.35 cm, the Reynolds number for the tests was approximately 2,100,000. The accuracy of the tests can be judged from the points shown on the enlarged section of the polar (Fig. 14). It is not believed that variations in the maximum-lift coefficient of the order of 2 or 3 percent are of any significance, as it was found that almost imperceptible changes in the surface condition of the model would cause this much variation. This variation is in line with that found by the N.A.C.A. in large-scale tests (ref. 3).

Before beginning the investigation on the effect of fillets, lift, drag, and pitching moment, tests were run of the wing alone. These results furnish a convenient basis for the subsequent discussion.

NORMAL LOW-WING CONFIGURATION

Results of three component tests on the normal low-wing arrangement are shown in Fig. 8. Considering the curves in the case of no fillet, it is observed that at a C_L of 0.6 the polar starts breaking over sharply, the C_L against α-curve takes up a new slope, and the moment curve becomes irregular. Now referring to the wake-loss diagrams, it is seen that for an angle of 5 deg the downwash is much smaller for the wing and fuselage than for the wing alone. The wing-alone polar parallels the induced-drag parabola (aspect ratio 5.97). The polar curve of the wing-fuselage arrangement parallels the wing-only polar up to a C_L of 0.45, and then starts deviating from it. If one attempts to fit an induced-drag polar to this curve in the region from a lift coefficient of 1.0 to 1.2, one needs an aspect ratio of approximately 2.5, or somewhat less than one-half the aspect ratio of the complete wing. This leads one to suspect that the wing is acting as two monoplanes separated by the fuselage and that there is a trailing vortex on each side of the fuselage. The wake diagrams lead one to the same conclusion, as from them one sees that that downwash is practically the same at rows 2 and 3 as for the wing alone, while in row 1 the downwash has practically disappeared for all positive angles. It will then be assumed as a working hypothesis that the foregoing is correct and that the interference corresponds to the breaking up of the horseshoe lifting vortex into two side-by-side horseshoe vortices.

This aerodynamic picture enables one to account for the peculiar shapes of the various curves, and also shows why the minimum drag is not subject to much improvement.

Three fillets were then tested to determine the best type. These fillets all had a radius of 15 percent of the root chord. Fillet 1 was a uniform fillet of this radius throughout. Fillet 2 (wrong) had a 15 percent radius at the nose and tapered aft; its lines are shown in Fig. 5. Fillet 3 had a small nose and a radius of 15 percent at the trailing edge, and was of the same type as the optimum fillet shown in Fig. 5. A photograph of fillet 3 is shown in Fig. 6. All three of these fillets were a marked improvement over the unfilleted condition, fillet 3 having the greatest maximum lift and the greatest slope of its lift curve. As all of these fillets were a great improvement over no fillet, an elaborate program was undertaken to develop fillets of various types. It soon appeared that fillets 1 and 2 could not be much improved, while fillet 3 could be developed with considerable success. Numerous variations of this type were constructed, and the following rules were deduced from the tests:

1. Increasing the trailing-edge radius improves the maximum lift and prevents induced-drag losses at medium-lift coefficients; excessive trailing-edge radii increase the drag.
2. Increasing the radius at the nose increases the drag and decreases the maximum lift.
3. The fillet should taper as uniformly as possible from the nose to the trailing edge, and its maximum size should be as close to the trailing edge as possible.
4. The fillet should be washed out smoothly to the fuselage.

Fillet 4 is an optimum design of this type. Maximum lift has been sacrificed for low drag, and it will be noticed that there is only a trace of the former ill effects of the fuselage. The wake diagrams for this fillet are shown in Fig. 3, and it will be noticed that the break in the downwash does not occur until over 15 deg, while in fillet 3 the break occurred at 10 deg. It will be noticed also that with fillet 4 the slope of the lift curve is steeper than the corresponding curves for the wing alone. This effect can be easily explained, as in this case the fuselage contributes some lift, and the coefficients are calculated neglecting this effect.

LOW-WING CONFIGURATION, WING LOWERED

It was suggested by an aircraft constructor that an investigation of the case of a low-wing monoplane in which the wing passes entirely below the cabin floor would be of great interest. Accordingly, a block 2 in. thick was made and placed between the wing and the fuselage. This corresponded to lowering the wing 1 ft at full scale. It was impossible to take any observations of the unfilleted condition for this arrangement. The first fillet put on in this configuration had a 20 percent chord trailing-edge radius and had the nose of the wing faired forward and up to the fuselage. The results are plotted as fillet α in Fig. 9. Fillet b in the same figure corresponds to a similar fillet with the leading edge cut back. The fillet was then enlarged to 27.5 percent trailing-edge radius and the nose was undercut still more. The lines of this fillet D are given in Fig. 5. Fillet D was found to be the optimum for this configuration. Fillet e was an endeavor to secure still greater improvement. The fillet was hollowed out by decreasing the radii in front of the trailing edge of the wing. This hoped-for improvement was not realized, but the curve is included in the figure. The curves for case D and for the wing alone illustrate how perfectly the pernicious effect of the wing-fuselage intersection can be eliminated; it will be noticed on the polar that the profile drag is practically independent of the lift coefficient. It was found that, as in the case of the normal low-wing configuration, the radius at the nose should be a minimum.

HIGH-WING CONFIGURATION

The model was next arranged as a high-wing airplane, the wing being mounted so that it had the same angle of incidence as in the case of the low wing and so that its trailing edge was just touching the fuselage. The curves for the high wing with no fillet and high wing with fillets are shown in Fig. 13. The fillet had quite a large trailing-edge radius (20 percent) and was of the tapered type. The lift obtained with this fillet was the highest found, but was associated with a slight increase in drag. Fillets of intermediate radius and of various types, large at the front, small at the rear, constant radii, etc., caused very little change. The curves for intermediate radii, and in fact for all of the other variations, lie between the two curves mentioned. The most remarkable point in this group is the behavior of the polar of the no-fillet case in the vicinity of zero lift. It will be noticed that this curve departs markedly from the wing-only polar in the same manner, for negative angles, as does the low-wing no-filet polar at small positive angles. This illustrates the extreme sensitivity of the suction side of the wing.

GENERAL DISCUSSION

It has been seen from the foregoing that the aerodynamic disadvantages of the low-wing monoplane can be almost completely eliminated by proper design of the wing-fuselage intersection. The maximum-lift coefficient of the wing alone can be attained with either a high-wing or a low-wing configuration. The differences in drag between the high wing and the normal low wing are inappreciable.

In the case of the dropped low wing there is found to be some drag increase, but it is thought that this difference could be reduced by further investigation. The size of the optimum fillet in the case of a low-wing design can be decreased by the following means:

1. Making the fuselage small.
2. Keeping the distance of the trailing edge of the wing below the bottom of the fuselage small.
3. Using an airfoil of small top camber.

The minimum drag of the fuselage-wing combination can be best decreased by making the fillet at the leading edge of the wing as small as possible.

Buffeting. For the purposes of discussion, buffeting will be defined as follows: Buffeting is a violent shaking of the airplane and tail surfaces by aerodynamic forces at angles below the stall. Buffeting in this sense has been observed in the case of almost all low-wing monoplanes. If one considers the wake measurements and notices the critical angles, there is no difficulty in seeking what occurs.

The buffeting seems to be due to these three causes: first, the lift over the center section of the wing disappears, causing a decrease in the total lift available; second, the wake jumps from one side of the stabilizer to the other; and third, the stabilizer is then in the trailing vortex formed at the side of the fuselage. The position of the stabilizer is indicated on each of the wake drawings.

Mr. H. J. Steiger (ref. 4) has suggested that buffeting could be eliminated by cutting away the wing root and reducing the angle of attack at the fuselage. It is dubious if any such attempt would be satisfactory, as it would inevitably result in trailing vortices forming near the fuselage, and these vortices might cause a dangerous or uncomfortable tail vibration, in addition to their pernicious effect on the induced drag. It is easily seen that for a properly designed junction between the wing and the fuselage the lift must carry entirely across the span, since if there were no lift over that portion of the wing covered by the fuselage, the polar curve for the wing and fuselage would of necessity differ more from that of the wing alone than it actually does.

ACKNOWLEDGMENTS

The author wishes to acknowledge his gratitude to the entire staff of the laboratory for their assistance in performing the tests and in preparing this report. He is especially indebted to Dr. C. B. Millikan, Mr. W. H. Bowen, Mr. W. B. Oswald, and Mr. N. B. Moore for their efforts.

DISCUSSION

E. Ower. The author's results entirely confirm the ideas formed from some similar, although not so comprehensive, work for which the writer was responsible some time ago. An account of this work was given in a lecture read to the Royal Aeronautical Society in January, 1932, and the writer thinks that the explanation he then put forward of the type of interference for which fillets are found to be beneficial is worth repeating. He suggested that this interference occurs when the airstream has to expand at more than a certain rate if it is to remain in contact with the body and wing surfaces. A certain rate of expansion can be tolerated, but if the surfaces diverge from one another too rapidly, the flow detaches itself from them and a region of turbulence is set up which leads to a loss of lift and an increase of drag. A well-known analogous case is that of the outlet cone of a venturi tube—if the angle of this cone is too great, it does not "run full;" that is, the flow breaks away from the walls, with the resulting loss of efficiency of reconversion of the kinetic energy into static pressure.

This hypothesis was confirmed by the various tests made to investigate its truth, and it explains the author's results with fillets. For in his case the geometry was such, as indeed it is in most practical body-wing combinations, that the rate of expansion increased progressively from the maximum camber of the wing toward the trailing edge. Hence, as the author found, the best fillet increases in radius toward the rear of the wing. The writer's experiments were made with fillets of constant radius, but he did in fact predict that fillets of increasing radius toward the trailing edge would be preferable. The same line of reasoning indicates why fillets on the under surface of a high-wing combination are found to have very little effect; the divergence between the surfaces of the body and the wing is much less in such a combination than it is in low-wing positions. Moreover, in the high-wing position the pressure gradient along the lower wing surface is such as slightly to assist the flow to adhere to the surfaces, whereas in the low-wing position the pressure gradient on the upper surface tends powerfully in the opposite direction.

The author mentions the importance of preserving as far as possible the normal lift distribution along the span of the wing. This again agrees with views that the writer has expressed. This principle, together with that of avoiding regions of divergent flow, will be found to be of the utmost importance to the designer in his efforts to build high-performance aircraft. The designer is always more interested in direct proof than in speculation, and the author has provided such proof, whereas the writer, mainly through lack of time, was content to put forward ideas which needed corroboration by facts before they could be accepted with complete confidence.

Richard M. Mock. This is believed to be the most interesting piece of aerodynamic research in America published during 1932, outside of that of the N.A.C.A. It is unfortunate that the tests were not made with a running model propeller, as it is possible and likely that the slipstream affects the flow around the fuselage and especially over the wing-fuselage intersection. Therefore it is believed that the comparative drag figures are somewhat questionable. During take-off and climb with high angles of attack, the same would be true as affecting the lift coefficients, while for landing the propeller effect is negligible. The comparative maximum lift coefficients, without propeller, are very valuable, as this is applicable to the landing condition.

The writer differs from the author regarding his comparison with the high-wing monoplane. The high-wing monoplane which he used as a comparison showed almost 6 percent more maximum lift and about 3 to 4 percent less drag than the best low-wing arrangement. However, the writer is under the impression that a high-wing design, rather than have the wing resting on top of the fuselage with its trailing edge just touching the fuselage, will have less drag if the wing is sunk into the fuselage, so that the top of the fuselage meets the top of the wing about one-third or one-half chord back from the wing leading edge and the combination is carefully filleted. The top of the fuselage might be lowered in front of the wing to allow a clean leading edge, and the portion of the fuselage above the rear portion of the wing might have fillets of very large radius. This should reduce the frontal area and the drag still below that of the combination used, and if the leading edge is carefully faired, should not affect the lift other than increase it by directing the flow from the fuselage over the upper surface.

Regarding the best position for the low wing, it would be interesting to raise the wing, as on the Gee Bee racer, rather than lower it as was done. As the fuselage decreases in width near the bottom and the wing is cambered on top, a pocket or cavity is formed between upper surface of the wing, just in front of the trailing edge, and the lower surface of the fuselage. The air passing over the wing and over the fuselage must fill this pocket causing eddies and consequently drag. Therefore it is logical that by fairing over this cavity, as the author has done, the drag of the combination will be reduced. This could also be done by raising the wing slightly to where the fuselage is wider and then using a fillet, and also perhaps by changing the fuselage cross-section slightly so that an excessive fillet will not be necessary. Another means would be to have the wing, in front view, curve upward at the root, meeting the fuselage side at a right angle.

Of course, if the landing gear is attached to the wing, the wheel supports will be longer if the wing is raised. The increased length means slightly greater weight, and if the undercarriage is not retractable, the frontal area and consequently the drag will be increased. Lowering the wing from the optimum (filleted) low-wing position to the optimum (also filleted) dropped-wing position (Fig. 14) means an increase in drag of approximately 17 percent. With a fixed external landing gear, the two shorter wheel supports with the dropped-wing position might partially offset the increased wing-fuselage drag. With a fully retractable landing gear, the landing-gear resistance could be neglected, and only the best wing-fuselage arrangement considered, with the tail location and fillet varied to eliminate buffeting.

The effect of lift-increasing trailing-edge flaps on buffeting would be interesting.

The writer would appreciate having the author's opinion of the meaning of the double curve near the maximum-lift coefficient of the dropped-wing combination with optimum fillet D. He also would appreciate an opinion of the double wake behind the fuselage at −5 deg, zero, and +5 deg for the same wing-fuselage combination. Another pilot combination between 1 and 2 and still within the stabilizer span would have been interesting. The writer would like to know the drag coefficient of the fuselage at the various angles of attack so that it can be added to the drag of the wing alone and compared with the drag of the combination.

G. J. Klein.The early investigations into body-wing interference were, it is true, conducted at rather low values of Reynolds number. However, they disclosed very interesting results, particularly in the case of the low-wing monoplane, and certainly showed the need for further research at higher Reynolds number. In this connection, the present paper, together with an extensive series of experiments at the N.P.L. (R. & M. 1480 and R. & M. 1300), form a valuable extension and show that the general conclusions reached in the earlier work still hold at much higher values of Reynolds numbers. When we consider all this research together, we get a fairly accurate picture of the subject.

It is now definitely established that detrimental interference is due to burbling caused by an attempt to expand the airstream too rapidly in the angle between the side of the body and the surface of the wing. It is true that the presence of the body does change the lift grading of the wing and thus increases the induced drag, but unless this burbling occurs, this effect is very small. Actually the wings so modify the flow about the body that the body contributes appreciably to the total lift of the combination, and the resulting lift grading is not very different from the lift grading of the wing alone.

For the foregoing reasons it is sufficient to confine this discussion to the case where burbling occurs and to consider the factors involved. Obviously, the worst case would be a low-wing monoplane without fillets, having a highly cambered wing root and a body of small fineness ratio and of such a cross-sectional shape that the angle between the side of the body and the upper surface of the wing is small compared with 90 deg. This combination forms a pocket between the body and the wing near the trailing edge of the latter, into which the airstream cannot expand, even at the zero lift angle of attack of the combination. Increasing the angle of attack increases the difficulties in expanding the airstream, thus causing increased burbling. There are several methods of suppressing this burbling:

(1) The wing can be raised to a higher position on the body to eliminate the pocket effect.

(2) The body can be given flat sides, making an angle of 90 deg with the wing surface, as was done in the Schneider Trophy Racer S5.

(3) A fillet of increasing radius toward the trailing edge can be employed, as was done in the present paper.

In short, anything that eliminates this pocket effect over the useful range of angle of attack will result in a combination of body and wing that will be free of any undesirable interference effect.

An interesting result brought out by the paper is that the “double stalls” found in the earlier work at low values of Reynolds number are still present at much higher values. Undoubtedly the recovery from the first stall is due to a decrease in the extent of the burbling caused by the downwash from the nose of the body.

Probably the most important point brought out by the paper is that a properly designed low-wing monoplane can be just as efficient as a high-wing monoplane.

AUTHOR’S CLOSURE

Replying to Mr. Ower, the low-wing fillet described was first developed in our wind tunnel in May, 1931, and test-flown in June of the same year. The second Northrop Beta and all succeeding Northrop ships have carried this device. The author agrees entirely with Mr. Ower’s statements as to the necessity of eliminating most of the expansion of the wing fuselage intersection, as was first pointed out by Muttray (loc. cit.).

Replying to Mr. Mock, as he says, it is unfortunate that the tests were not made with a running propeller. The laboratory has under development a fuselage, in which is included an electric motor with complete dynamometer, for repeating the preceding investigation with a slipstream. The model when finished will have the proper scale horsepower and the proper ratio of propeller diameter to span. However, the author does not believe that the drag differences will be large in the case with slipstream and without, as the laboratory has been rather successful in predicting the performance and especially the high speed of the airplanes which it has tested. In a paper by Drs. Theodore von Kármán and Clark B. Millikan there are described the methods used by this laboratory in estimating the performance of the actual airplane. Due to the successful checks which have been secured, it is not very probable that the discrepancies in drag between power off and power on will be large.

Since the investigation of the fillet was undertaken as an engineering study rather than as a scientific investigation into the configuration for minimum drag of the wing and fuselage alone, we did not consider several of the cases mentioned by Mr. Mock, as the lowering of the wing in the case of a high-wing monoplane or the raising of it in the case of a low-wing monoplane would require a larger fuselage in order to maintain the headroom in the cabin.

The reduction in frontal area in the cases mentioned by Mr. Mock would be more apparent than real, as it would necessitate, if his suggestions were carried out, the enlarging of the fuselage in order to accommodate the passengers or other loads. The raising of the wing any great distance was impossible in this particular case as it would bring the floor level too high. The consolidation of a stressed-skin wing into beams in order to enable the passengers to place their feet below the top surface of the wing is exceedingly extravagant of weight, and it also causes the cabin to be encumbered with structural members which interfere with the free circulation of passengers or the stowing of freight.

The author does not believe that the position of the wing with reference to the fuselage is a vital matter, as what is gained in one place is lost in another. In the multimotor transport field there is a very interesting case. There are two modern low-wing bimotor transports of the same power loading and span loading, one with the wing passing completely below the cabin floor and the other with the wing beams passing through the cabin. The airplane with the wing completely below the cabin floor is much faster, in the neighborhood of 25 mph, than its competitor. The difficulties of designing a retractable landing gear increase with at least the square of the length of the members as does also its weight. The raising of a wing 6 in. or 8 in. farther above the ground will often make a satisfactory retractable landing gear almost impossible. The addition of lower surface flaps has been found in this laboratory to have no effect on the buffeting of a well-filleted airplane. The author believes that the double peak of the curve for the optimum fillet D is probably due either to asymmetries of the model or to asymmetries of the air stream. The model was known to have developed some aerodynamic twist. This twist raised the angle of attack of one side of the model above that of the other, and consequently one side stalled in advance of the other. This effect has also been found in the laboratory a number of times in rolling-moment tests. The author believes that the two peaks in the wake diagrams shown in Fig. 10, at −5 deg. and +5 deg, can be explained as follows: The upper one is probably the wake of the fuselage and the lower one that of the wing. When the downwash becomes large, the fuselage wake becomes merged into that of the wing. The drag coefficient of the fuselage has never been measured separately, so that Mr. Mock’s question on this point cannot be answered.

Replying to Mr. G. J. Klein, he is completely correct in his statement as to the three methods of eliminating the burbling in the wing-fuselage intersection. However, in the case of a straight-sided fuselage, the burble is not completely absent. The author would like to cite Fig. 15 as evidence in this case. The results in this figure were obtained with the Douglas transport fuselage, which is straight-sided in the region in contact with the wing and for a considerable distance aft of the trailing edge. The wing section used in this transport is the N.A.C.A. 2215. This section is of only 15 percent thickness, has a very small camber (2 percent), and furthermore the point of maximum camber is only 20 percent aft from the leading edge. This wing has therefore probably the smallest top camber of any section in common use.

Nevertheless, in the test shown, the difference between fillet and no fillet is quite perceptible. The model was tested with tail surfaces but less nacelles. The separation between the two curves shown on the larger plot in the figure has been exaggerated in the region of low lift coefficient in order to enable the two curves to appear separately on the reproduction. The smaller plot shows the experimental points in their true separation. In the low drag region the addition of the fillet reduced the minimum drag for the model approximately 1.8 percent and raised the maximum lift coefficient as shown from 1.27 to 1.34. For every high angle not shown in the plot, the curves cross each other, an effect the author is unable to explain, but at all usable angles of attack the filleted case is superior to that of the unfilleted.

The discussion of why some of the lift curves have double peaks has been included in the reply to Mr. Mock. However, the double stalls in the other sense, i.e., of curves that come to a maximum and then go on to a further maximum, are apparently due to the local burbling of the wing in the center, while outboard portions of the wing do not stall as soon, but continue to work at larger and larger angles of attack. These restricted portions of the span can then go to larger lift coefficients, as it is well known that wings of small aspect ratio can reach larger lift coefficients than normal wings. It has been found in our laboratory that the characteristic sharp break in the lift curve can be explained as an effect of this kind. We always find the sharp break in untapered wings. Tapered wings usually show the rounding top which we think means that the center of the wing stalls before the outboard portions. Evidence for this can be seen in Fig. 2 on the wake diagrams for the wing alone. In one very interesting case the laboratory found that this sharp drop-off of the lift curve was obtained with a tapered wing in which an auxiliary airfoil was used only over a part of the center section. In this case the sharp drop in the lift curve was obtained, not only in the laboratory but also in flight, while the airplane without the auxiliary airfoil stalled in the more usual manner. We therefore think that in this case the auxiliary airfoil held the flow on the center section to a higher lift coefficient than that which it would normally reach, and when it stalled, the entire wing stalled at once.

Since the foregoing paper was written, two confirmations have been published of the results, and numerous airplanes have been designed and flown with these devices.

Document 3-17(b), James A. White and Manly J. Hood, "Wing-Fuselage Interference, Tail Buffeting, and Air Flow About the Tail of a Low-Wing Monoplane," NACA* Technical Report *482 (Washington, 1934).

REPORT NO. 482
WING-FUSELAGE INTERFERENCE, TAIL BUFFETING, AND AIR FLOW ABOUT THE TAIL OF A LOW WING MONOPLANE
BY JAMES A. WHITE AND MANLEY J. HOOD

SUMMARY

This report presents the results of an investigation of the wing-fuselage interference of a low-wing monoplane conducted in the N.A.C.A. full-scale wind tunnel on the "*McDonnell*" airplane. The tests included a study of tail buffeting and the airflow in the region of the tail. The airplane was tested with and without the propeller slipstream, both in the original condition and with several devices designed to reduce or eliminate tail buffeting. The devices used were wing-fuselage fillets, an N.A.C.A. cowling, reflexed trailing edge of the wing, and stub auxiliary airfoils.

The use of proper fillets practically eliminated the wing-fuselage interference and greatly reduced the tail vibrations due to buffeting. An N.A.C.A. cowling reduced the buffeting and interference effects to unobjectionable magnitudes at angles of attack up to within about 3° of the stall. A large fillet alone gave the greatest reduction in buffeting effect, reducing the tail vibrations to one seventh their original amplitude, but the combination of the large fillet and N.A.C.A. cowling gave the best all-round results. This combination reduced the tail oscillations due to buffeting to one fourth their original amplitude, increased the maximum lift 11 percent, decreased the minimum drag 9 percent, increased the maximum lift/drag ratio of the whole airplane 19 percent, and increased the effectiveness of the elevator about 40 percent at angles of attack in the landing range. The reflexed trailing edge had a minor effect and the auxiliary airfoils in the best position tested were considerably inferior to the fillets. With the propeller operating, the interference effects were practically eliminated, even with the airplane in the original condition.

The elimination of the wing-fuselage interference slightly decreased the longitudinal stability of the airplane.

Records of the fluctuations in the dynamic pressure of the air stream at the tail show a prominent wake-fluctuation frequency of the order of magnitude of the natural frequency of the tail vibrations.

FIGURE 1.—The *McDonnell* airplane with large fillet in full-scale wind tunnel.

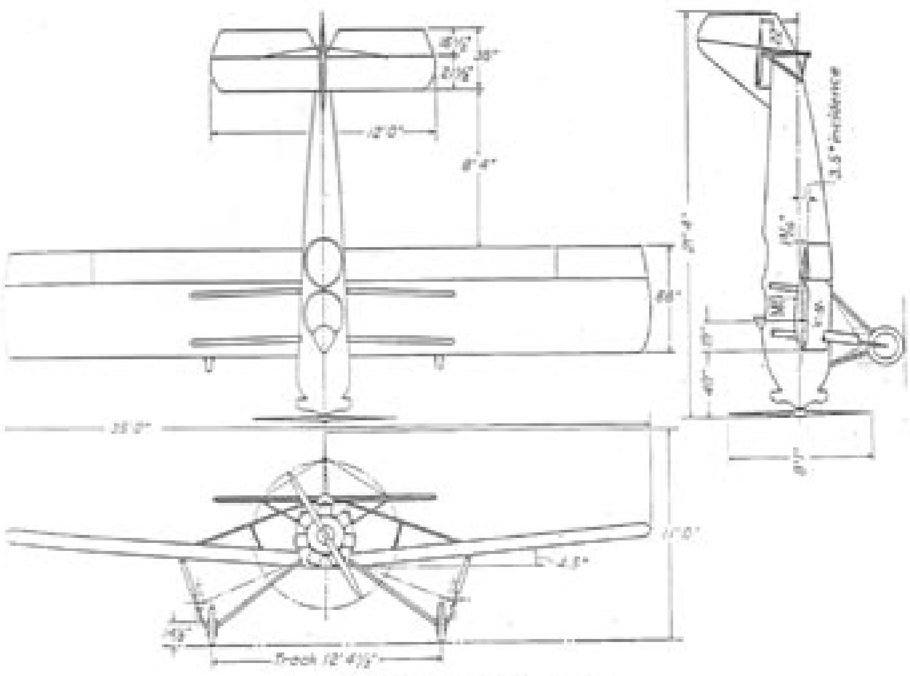


FIGURE 2.—Three-view drawing of the *McDonnell* airplane.

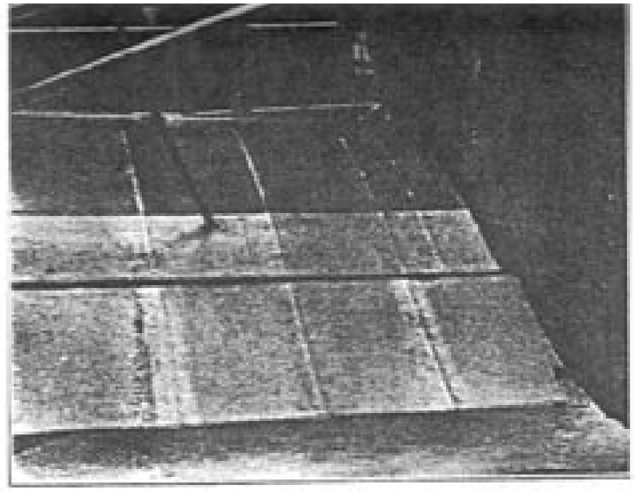


FIGURE 3.—Wing-fuselage intersection of *McDonnell* airplane.

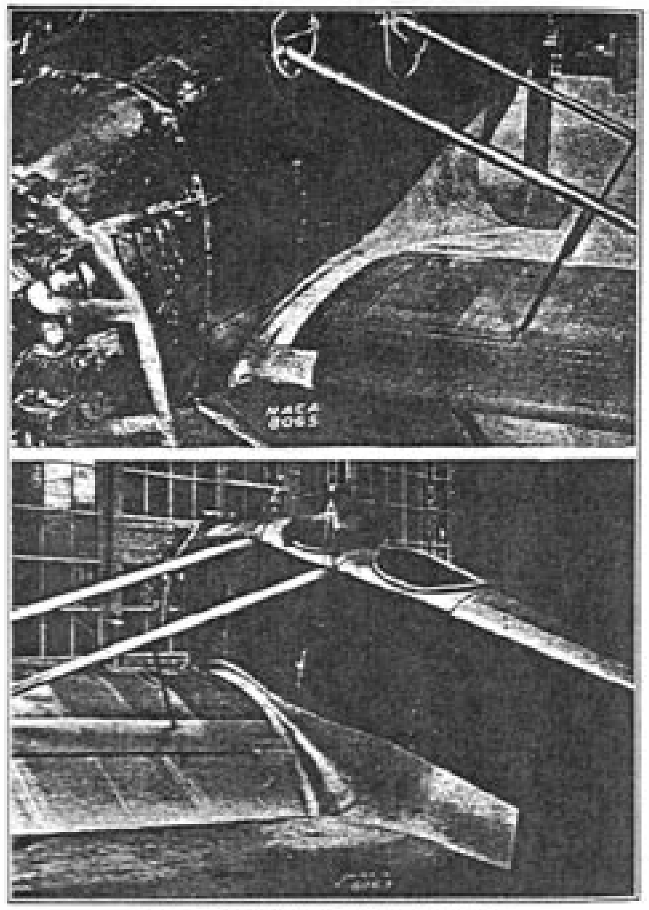


FIGURE 4.—Small fillet on *McDonnell* airplane.

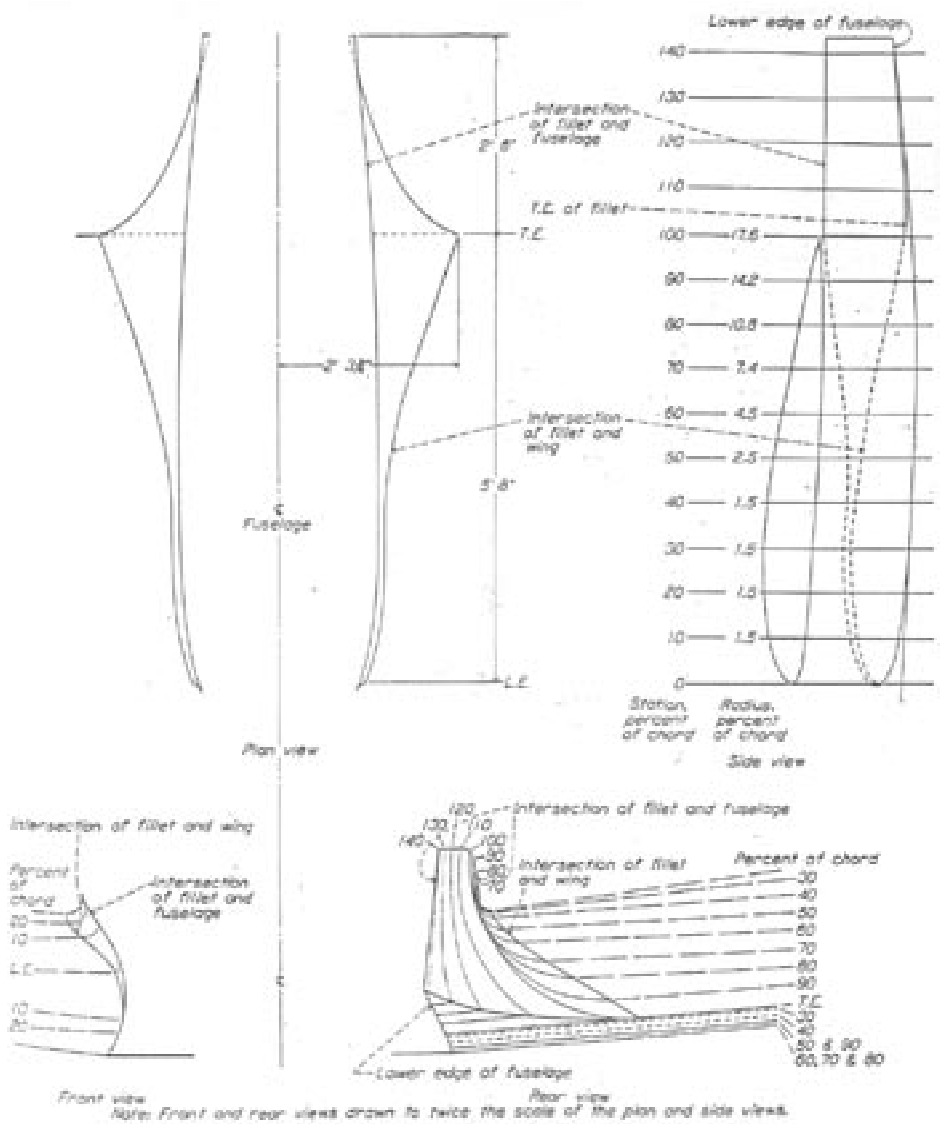


FIGURE 5.—Drawing of the small fillet.

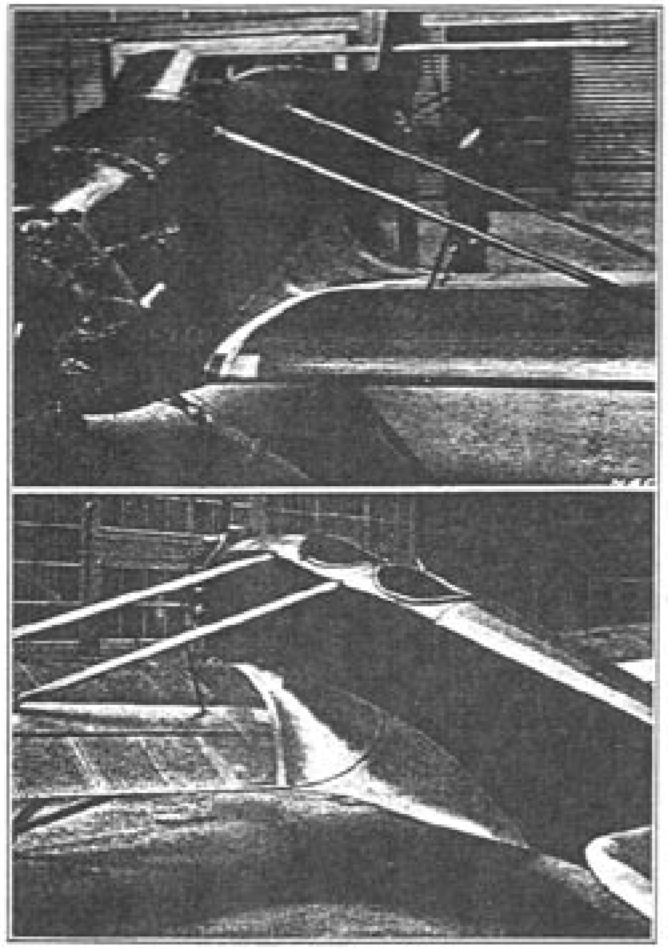


FIGURE 6.—Large fillet on McDonnell airplane.

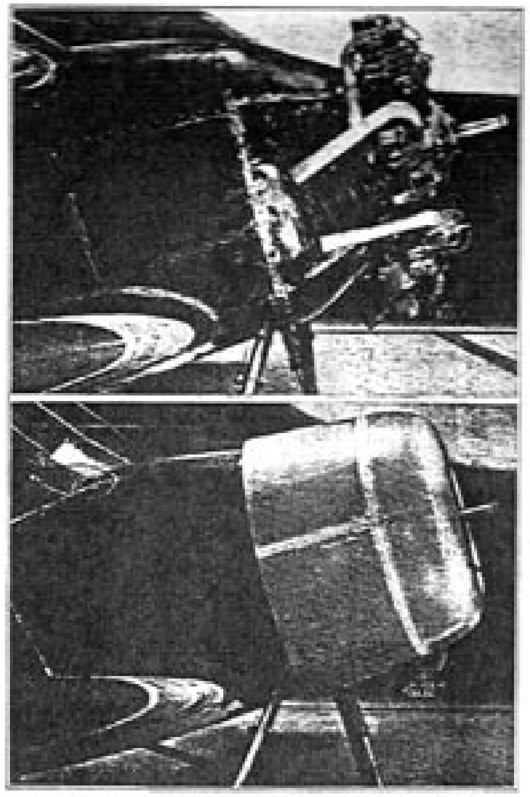


FIGURE [illegible].—Nose of McDonnell airplane in original condition and with N.A.C.A. cowling.

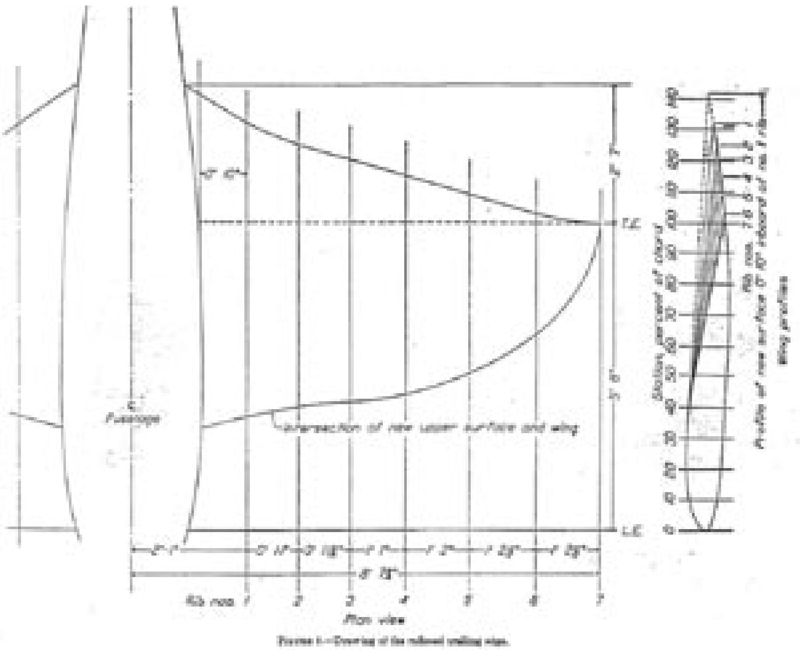


FIGURE [illegible].—Drawing of the reflexed trailing edge.

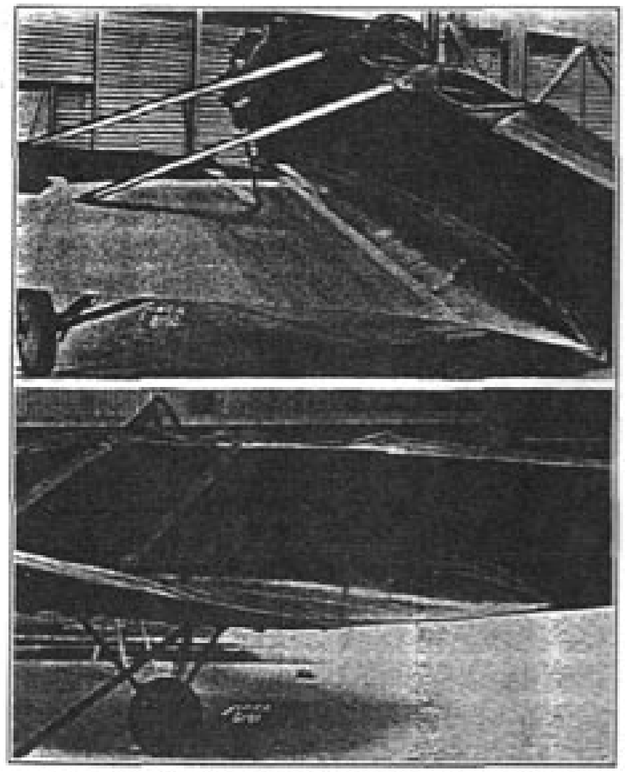


FIGURE 10.—Reflexed trailing edge with fillet on McDonnell airplane.

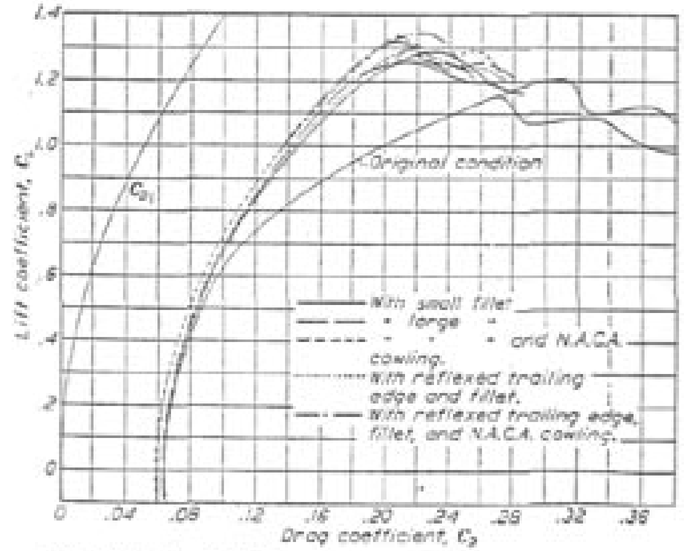


FIGURE 11.—Polars for McDonnell airplane with various fillets. Corrected for tunnel effects. Power off.

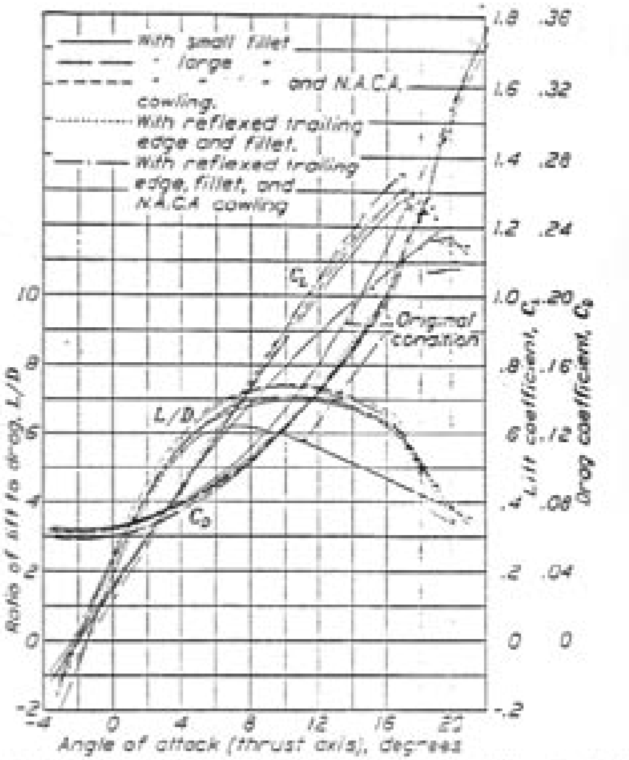
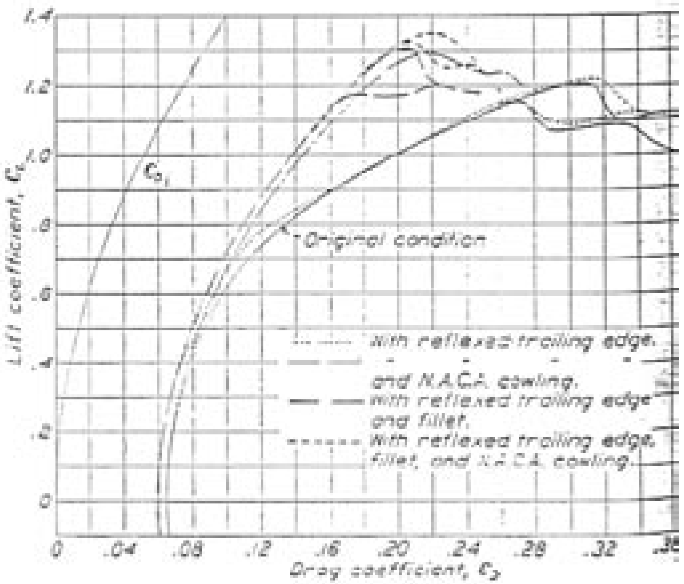


FIGURE 12.—Lift and drag of *McDonnell* airplane with various fillets. Corrected for tunnel effects. Power off.



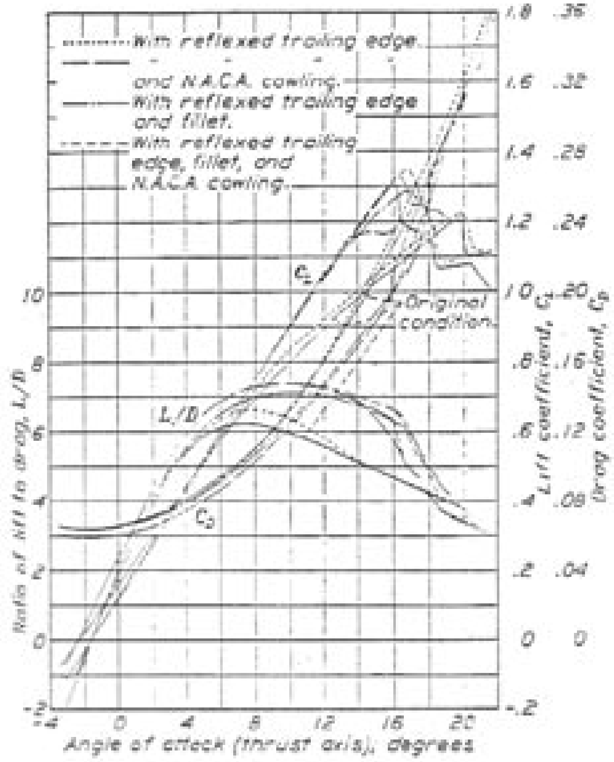


FIGURE 14.—Lift and drag of *McDonnell* airplane with reflexed trailing edge. Corrected for tunnel effects. Power off.

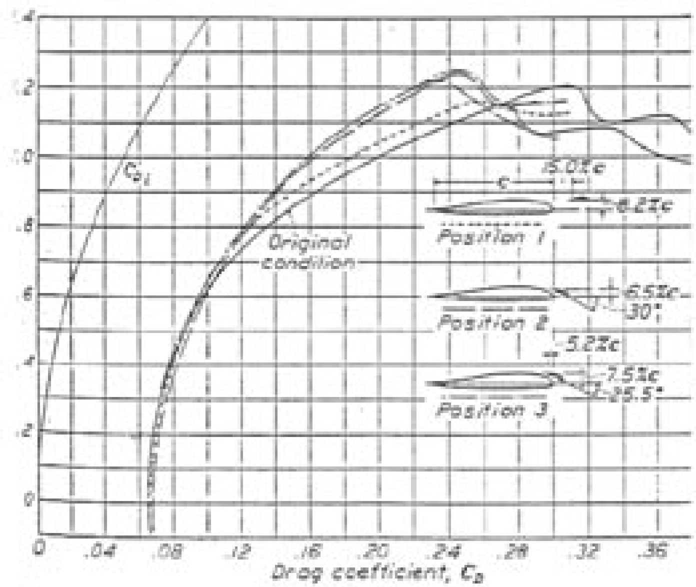


FIGURE 15.—Polars for *McDonnell* airplane with auxiliary airfoils. Corrected for tunnel effects. Power off.

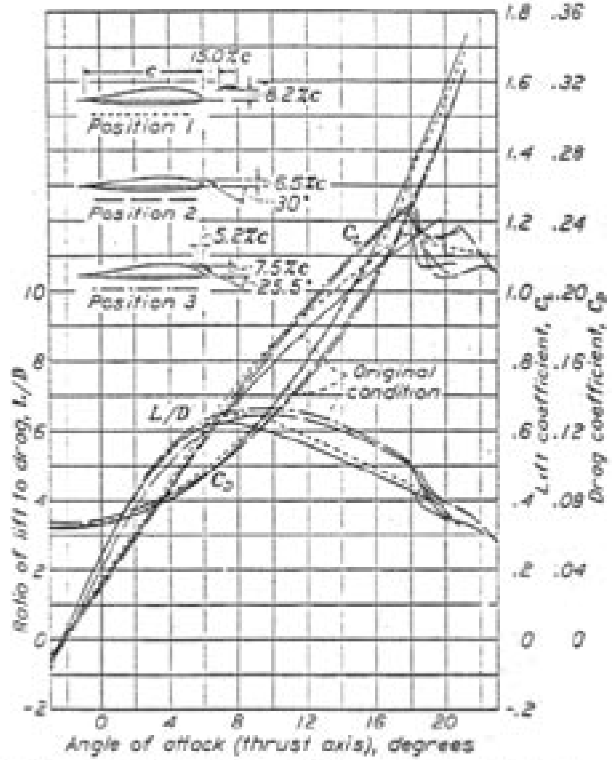


FIGURE 16.—Lift and drag of *McDonnell* airplane with auxiliary airfoils. Corrected for tunnel effects. Power off.

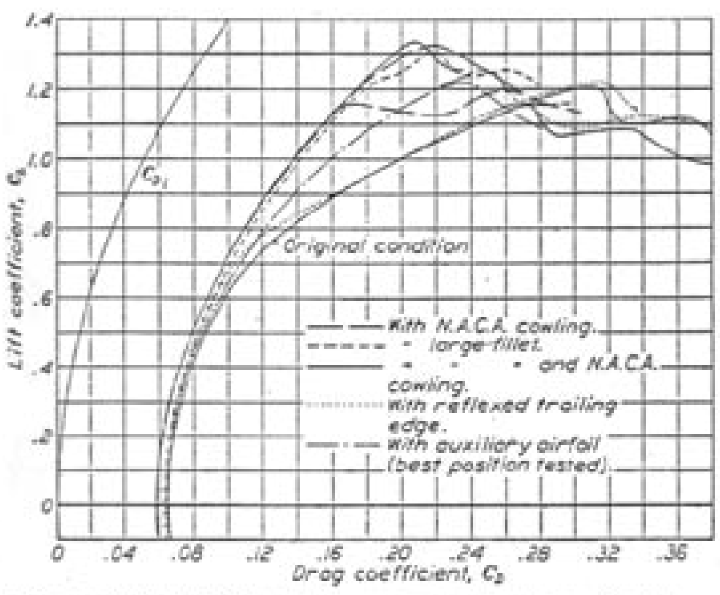


FIGURE 17.—Polars for *McDonnell* airplane comparing various devices. Corrected for tunnel effects. Power off.

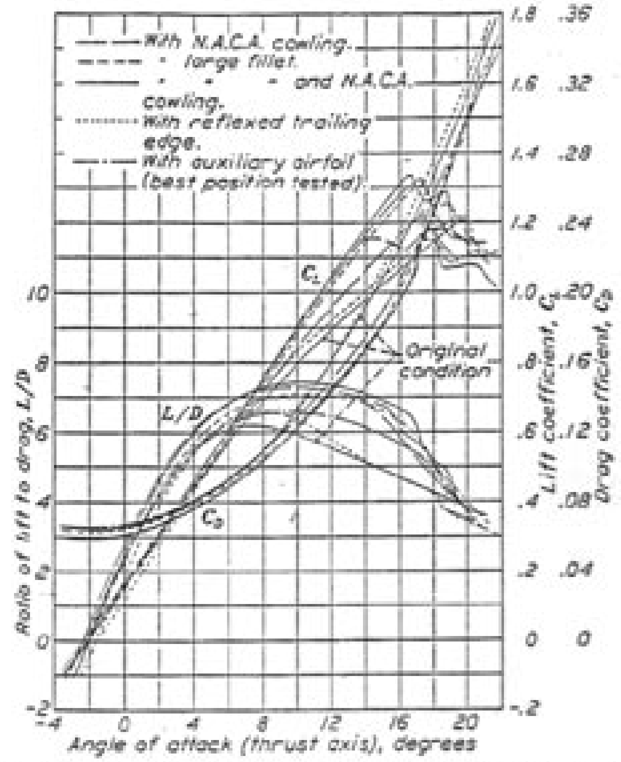


FIGURE 18.—Lift and drag of *McDonnell* airplane comparing various devices. Corrected for tunnel effects. Power off.

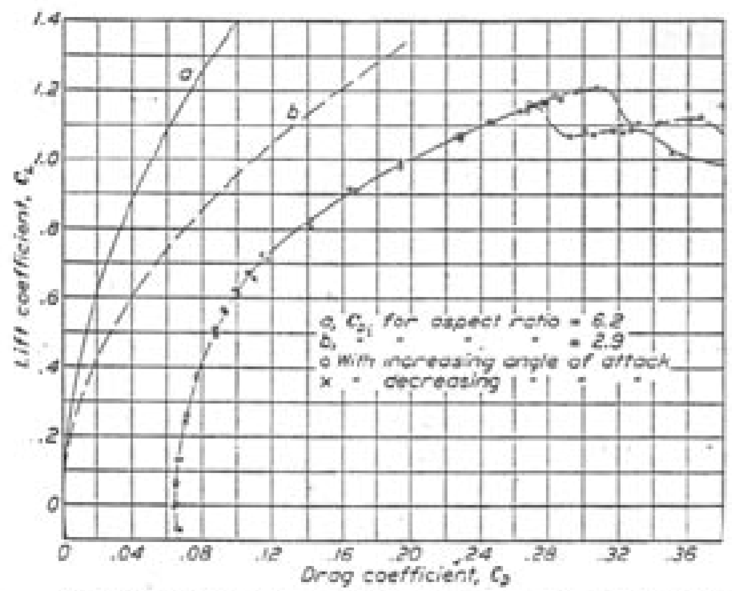


FIGURE 19.—Polar for *McDonnell* airplane in original condition. Corrected for tunnel effects. Power off.

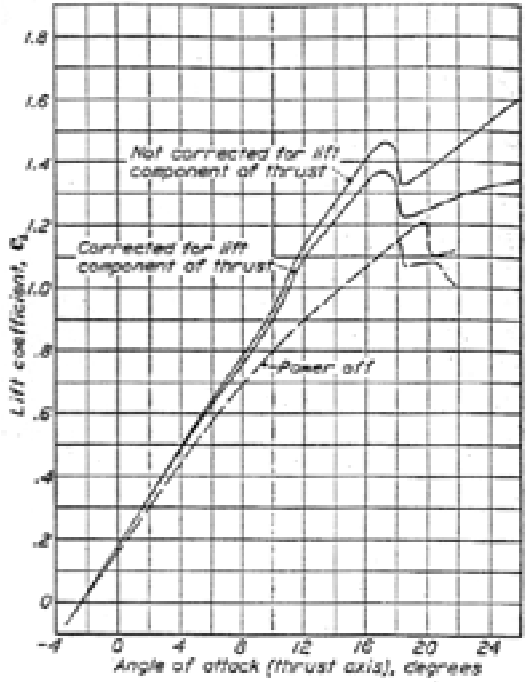


FIGURE 20.—Power-on lift of McDonnell airplane in original condition. Corrected for tunnel effects.

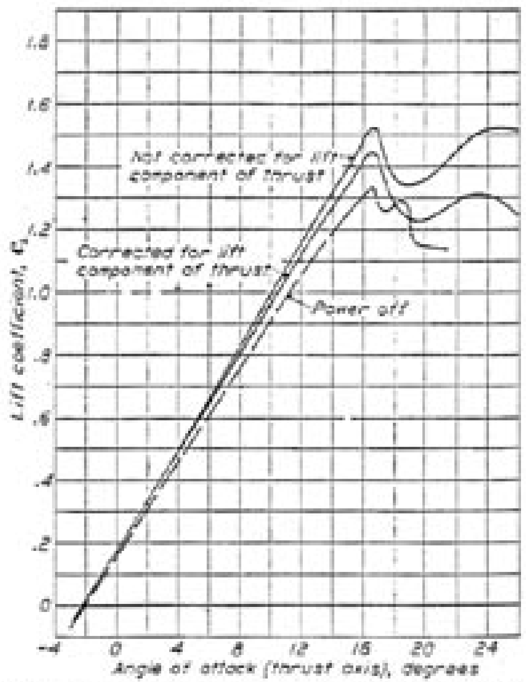


FIGURE 21.—Power-on lift of McDonnell airplane with large fillet and N.A.C.A cowling. Corrected for tunnel effects.

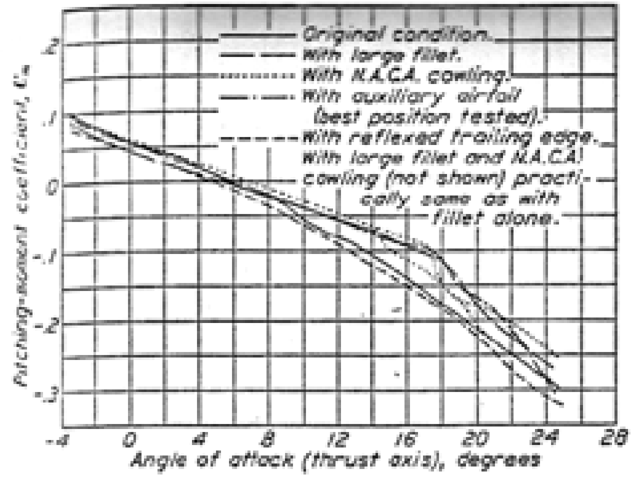


FIGURE 22.—Pitching moments of McDonnell airplane with various devices. Corrected for tunnel effects. Power off. Stabilizer 0.6° to thrust axis. Elevator 0° to stabilizer.

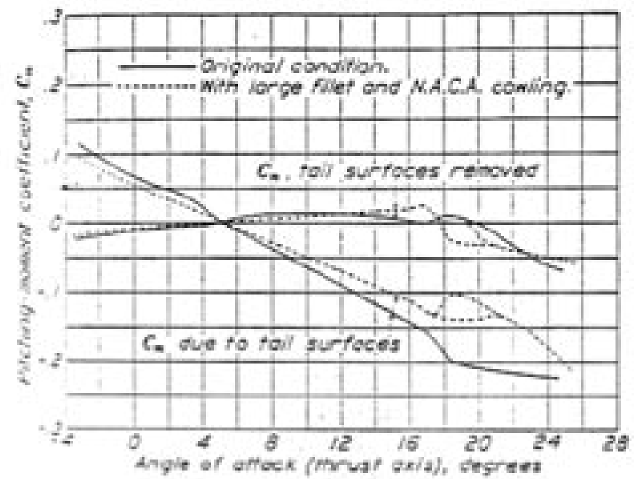


FIGURE 23.—Pitching moments due to tail of McDonnell airplane. Corrected for tunnel effects. Power off. Stabilizer 0.6° to thrust axis. Elevator 0° to stabilizer.

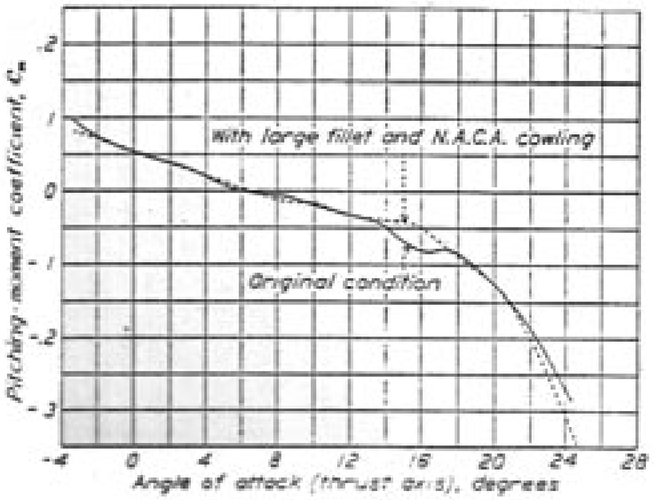


FIGURE 24.—Pitching moments of McDonnell airplane with power on. Corrected for tunnel effects. Stabilizer 0.6° to thrust axis. Elevator 0° to stabilizer.

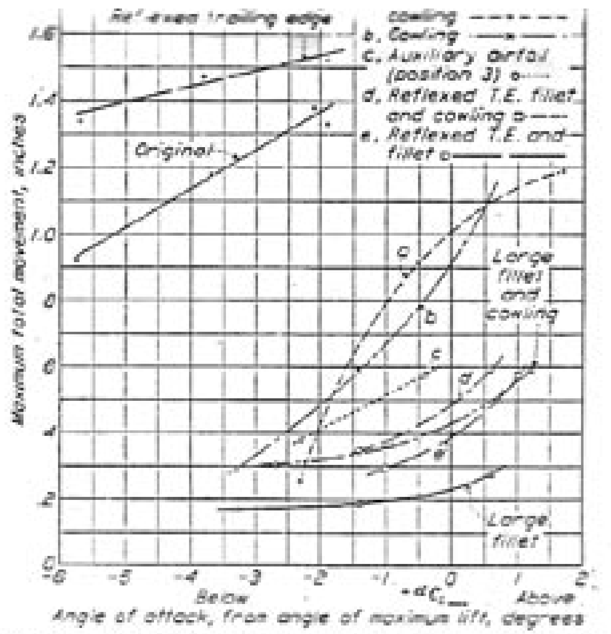


FIGURE 27.—Amplitude of stabilizer-tip movements under various conditions. Angle of attack corrected for tunnel effects. Power off. Air speed approximately 36 m.p.h.

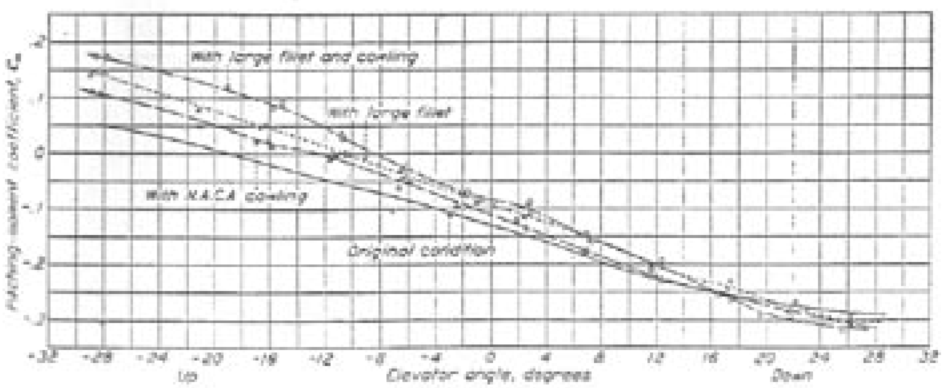


FIGURE [illegible].—Elevator effectiveness of McDonnell airplane with various devices. Corrected for tunnel effects. Power off. Angle of attack (thrust axis) = 15.8°.

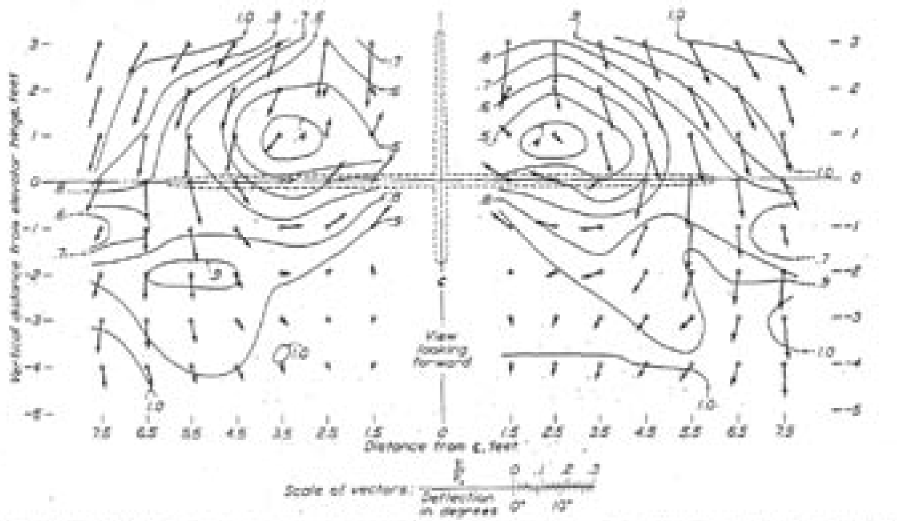


FIGURE [illegible].—Air flow at tail of McDonnell airplane in original condition, power off. Survey in vertical plane through elevator hinge line. The contours show the ratio of the dynamic pressure behind the airplane to the dynamic pressure in the free air stream. The vectors show the components of the velocity in the plane of the survey. Angle of attack (thrust axis) = 14.7° (corrected for tunnel effects). Lift coefficient = [illegible].

INTRODUCTION

The increasing use of low-wing monoplanes has emphasized the susceptibility of this type of airplane to detrimental interference at the intersection of the wing and fuselage. In addition to decreasing the aerodynamic efficiency, this interference often causes a loss of longitudinal control and a violent shaking, or buffeting, of the tail of the airplane by the eddying wake from the wing roots. Tail buffeting may become so severe in some cases as to endanger the tail structure. In at least one instance it was considered as a possible cause of the failure of a low-wing monoplane that broke to pieces in the air (references 1 to 4, inclusive).

Methods have been suggested for reducing or eliminating wing-fuselage interference and buffeting, and some tests have been conducted on small-scale models and in flight (references 2 and 5 to 9, inclusive). This report covers the results of tests conducted in the N.A.C.A. full-scale wind tunnel on a low-wing monoplane that was subject to tail buffeting. The tests included an investigation of the wing-fuselage interference and buffeting with the airplane in its original condition and with various devices installed to eliminate or reduce the detrimental effects. As the detrimental effects appear to be directly due to a premature breakdown of the flow at the wing-fuselage intersection, the devices were designed with a view to their ability to postpone this breakdown of the flow to the angle of attack at which the entire wing stalls. The devices tested were two different wing-fuselage fillets, an N.A.C.A. cowling, a reflexed trailing edge next to the fuselage, auxiliary airfoils of short span in three different positions, and various combinations of the above.

The value of the various devices was determined by visual observation of the air flow at the wing-fuselage intersection by means of strings; measurements of the lift, drag, and pitching moments of the airplane; records of the vibrations of the tail; and surveys of the direction and speed of the air flow at the tail of the airplane, including records of the fluctuations of the air speed. Observations were made both with and without the slipstream from the airplane propeller.

Part of the results given here have been previously published as a technical note (reference 10).

APPARATUS

Wind tunnel.—The tests discussed in this report were conducted in the N.A.C.A. full-scale wind tunnel. The wind tunnel, the balance for measuring the forces and moments, and the apparatus used for determining the air speed and direction at any point in the jet are described in reference 11.

Airplane.—The *McDonnell* airplane, a low-wing monoplane originally built for entry in the Daniel Guggenheim Safe Aircraft Competition in 1929, was chosen for these tests because it was reported by pilots to be subject to tail buffeting. Flight tests of the *McDonnell* airplane are described in reference 12. Figure 1 is a photograph of the airplane mounted in the wind tunnel; figure 2 is a 3-view drawing showing its principal dimensions; and figure 3 is a view of the intersection of the wing and fuselage. The airplane is equipped with a Warner Scarab engine having a rating of

110 horsepower at 1,850 r.p.m. The airplane is provided with movable leading-edge slots and trailing-edge, (FIGURE 1.—The *McDonnell* airplane with large fillet in full-scale wind tunnel.), flaps, but for these tests the slots were covered with doped fabric and the flaps locked in the neutral position. After preliminary tests had been made, a walkway that extended from the fuselage to 10 inches outboard and raised the top surface of the right wing five eighths of an inch above the normal profile from 15 to 69 percent of the chord was removed, and the gaps between the wings and fuselage, which were as much as 3 inches wide on the under side, were covered. The stabilizer was set at an incidence of 0.6° with respect to the thrust axis for all the tests and, except when elevator effectiveness was being measured, the elevator was locked in the neutral position.

Fillets.—The wing-fuselage fillets were designed to reduce the rate at which it was necessary for the air in this region to diverge in order to follow the surfaces. The radius was small at the leading edge and a short distance back started increasing smoothly to a maximum at the trailing edge, behind which the fillet was faired into the fuselage. The principal difference between the two fillets was in size, hence they will be referred to as the "small fillet" (figs. 4 and 5) and the "large fillet" (figs. 6 and 7). Another difference was that the small fillet had a constant radius from the leading edge back to 41 percent of the chord, whereas the radius of the large fillet began to increase at 6.6 percent of the chord back of the leading edge.

N.A.C.A. cowling.—The N.A.C.A. cowling (fig. 8) consisted of a hood that was placed over the engine and nose of the airplane without alteration being made in the original fuselage lines. The hood was designed in accordance with the information in reference 13, except that its cross section did not resemble an airfoil profile because it consisted of only one thickness of metal.

Reflexed trailing edge.—The modification of the wing root, herein called a "reflexed trailing edge" (fig. 9), was designed to decrease the incidence at the wing root. The lower surface of the wing, which had an upward curvature (N.A.C.A.—M6 section), was extended to the rear and a new upper surface formed of straight-line elements from the new trailing edge to the points of tangency with the upper surface of the original wing. The fillet tested in combination with this reflexed trailing edge (fig. 10) was similar to the large one previously described.

Auxiliary airfoils.—The auxiliary airfoils used in these tests were of the N.A.C.A. 22 section, had a 10-inch chord (14.7 percent of the main wing chord), and extended 30 inches from the fuselage on each side. They were tested in three positions near the leading edge of the wing (see fig. 15), the first position being similar to that found to be the optimum in the investigation reported in reference 14.

METHODS

Air flow at wing roots.—The air flow at the wing roots was studied by noting the behavior of a lightweight string on the end of a slender stick held by an observer in the cockpit.

Force and moment measurements.—The power-off lift, drag, and pitching moments were all measured with the propeller removed. The power-on measurements were made with the propeller turning at such speed that its thrust just balanced the drag of the airplane (due allowance being made for jet-boundary effect), thus simulating steady level-flight conditions. As the jet-boundary corrections could be only estimated beforehand, it was not feasible to adjust the engine speed so as to give exactly zero net drag. Therefore, three readings were taken at each angle of attack at three propeller speeds near the proper value and the value of lift for zero net drag was found from a plot of these points against net drag. All tests were made at an air speed of 55 to 60 miles per hour except in the case of the power-on tests, where at high angles of attack it was necessary to reduce the speed to keep the drag within the range of the available thrust.

Records of tail buffeting.—The vertical movements of the tip of the stabilizer were recorded on a moving film by means of an N.A.C.A. control-position recorder. From these records the amplitude and frequency of the motions of the tail surfaces were determined. The instrument was mounted on a solid base and connected to the stabilizer by an 0.008-inch diameter piano wire shielded from the wind by a steel tube. (FIGURE 2.—Three view drawing of the *McDonnell* airplane). The natural frequency of the piano wire and instrument was about 34 cycles per second, which is almost four times the highest frequencies recorded. Play and friction in the instrument caused errors in indicated amplitudes of the vibrations probably not exceeding one eighth inch.

During most of these tests the tail of the airplane was supported by a rigid A-frame fastened to the tail-post. In order to determine the effect of this rigid support, records were made of the movements of the stabilizer tip and the rear end of the fuselage while the tail of the airplane was free from external support, the airplane being prevented from turning about the main supports at the landing-wheel axles only by cables secured to the forward part of the fuselage.

Most of the records were taken at an air speed of approximately 58 miles per hour, but a few were taken at speeds between 35 and 60 miles per hour to determine the effect of change in speed.

Air flow at tail.—The direction and speed of the air flow at the tail in a vertical plane through the elevator hinge line were measured with all the tail surfaces removed, using the combined pitot-static, yaw, and pitch tube and auxiliary apparatus described in reference 11. In addition to the measurements of average speed and direction, several records were made with a recording manometer connected to the pitot tube to determine the frequency of the air-speed fluctuations in the wake from the wing roots and the relative magnitudes of the fluctuations at different positions near the tail. These records were not entirely satisfactory because of the large amount of damping in the long rubber tubes required to reach from above the air stream down to the pitot tube near the tail of the airplane. Consequently the true magnitude of the fluctuations cannot be determined from these records; however, some idea of the frequencies involved can be obtained.

RESULTS

Air flow at wing root.—The action of the string held in the region of the wing-fuselage intersection indicated that, except when the airplane was equipped with some of the most effective devices, the air flow over the upper surface of the wing began to break down near the intersection of the wing and fuselage and that the turbulent region spread laterally as the angle of attack was increased. With the airplane in the original condition the turbulent flow extended approximately 3 feet outboard from the fuselage at 14° angle of attack. The approximate angles of attack at which the air flow over the root of the wing first burbled, (FIGURE 3.—Wing-fuselage intersection of *McDonnell* airplane), when the airplane was equipped with the various devices with power off were as follows:

Original condition ----------------------------------5°.
Small fillet---12°.
Large fillet --15°.
N.A.C.A.cowling-----------------------------------14°.
Small fillet and N.A.C.A. cowling---------------17° (at stall).
Large fillet and N.A.C.A. cowling---------------17° (at stall).
Reflexed trailing edge ------------------------------7°.
Reflexed trailing edge and N.A.C.A. cowling-16° (at stall).
Reflexed trailing edge and fillet-------------------Above stall.
Reflexed trailing edge, fillet, and N.A.C.A. cowling --Above stall.
Auxiliary airfoil in position 1----------------------7°.
Auxiliary airfoil in position 2--------------------- 7°.
Auxiliary airfoil in position 3--------------------- 10°.

When the auxiliary airfoils were used, vortices trailing from their tips were evident. When the N.A.C.A. cowling was used, particularly in combination with any of the fillets and both with and without the slipstream, the action of the string indicated the presence of trailing vortices approximately concentric with the fillets.

The direction of rotation of these vortices was the reverse of what it would be for vortices corresponding to a loss of lift at the center section.

Lift and drag characteristics.—The power-off lift and drag data are presented in four groups of polar and lift and drag curves. The first group (figs. 11 and 12) compares the various fillets and fillet combinations; the second (figs. 13 and 14) shows the effects of the reflexed trailing edge alone and with the cowling and fillet; the third (figs. 15 and 16) shows the effects of the auxiliary airfoil in three positions; and the fourth (figs. 17 and 18) shows the effects of the cowling, (FIGURE 4—Small fillet on *McDonnell* airplane), alone and summarizes the other groups. In addition, a representative polar is shown with the experimental points (fig. 19). The theoretical induced-drag curve based on the geometrical aspect ratio of the wing (6.2) is included with each group of polars.

Power-on lift curves, corresponding to level flight, are presented for the original condition and for the condition with the large fillet and cowling (figs. 20 and 21). All the other conditions tested gave results practically the same as those for the large fillet and cowling. No means were available for determining the thrust of the propeller, so it was not possible to determine exactly either the effect of the slipstream on the drag characteristics of the airplane or what part of the total lift was due to the vertical component of the propeller thrust. An approximate correction for this vertical component of thrust was applied, however, in order to, (FIGURE 5—Drawing of the small fillet), make the difference between the power-off and power-on lift curves more nearly represent the effect of the slipstream; the lift curves are shown both with and without this correction. These approximate corrections were arrived at by computing, for each angle of attack, the vertical component of a thrust large enough to overcome the drag of the airplane without the slipstream.

All coefficients are based on the original wing area of 196.5 square feet. The added area due to the addition of the large fillet and the reflexed trailing edge amounted to about 2.5 percent and 7 percent, respectively.

Pitching moments.—Curves of pitching moments about the center of gravity plotted against angle of attack are shown for the power-off condition in figure 22. Curves of pitching moments with the tail surfaces removed and pitching moments due to the tail alone are shown in figure 23. Figure 24 shows the pitching-moment curves for two power-on conditions. The power-on pitching moments were found to be practically the same for all conditions. The influence of several of the devices on elevator effectiveness is shown by curves of pitching moment plotted against elevator angle for an angle of attack just below the stall (fig. 25). The pitching-moment coefficients are, (FIGURE 6.—Large fillet on *McDonnell* airplane), based on the original wing area (196.5 square feet) and the original mean chord of 5.62 feet.

Tail buffeting.—Typical records of the motion of the stabilizer tip are shown in figure 26. Curves of the maximum amplitudes of tail vibrations for various conditions of the airplane are shown in figure 27. Amplitude is here considered as the deflection between adjacent extremes of the up-and-down motion and is given in inches of motion normal to the plane of the stabilizer. The amplitude of stabilizer-tip movements with the propeller operating is not included in figure 27 because it did not vary consistently enough to permit the drawing of curves. Nearly all the maximum deflections measured with power on fell between 0.1 and 0.4 inch for angles of attack below the stall. The values in figure 27 were all obtained with the rear end of the fuselage rigidly supported. When it was free from external support the amplitude of stabilizer-tip movement was nearly doubled and the vertical movement of the rear end of the fuselage itself was only about one fifth as great as that of the stabilizer tip. Figure 28 shows the variation in amplitude with changes of air speed between 35 and 60 miles per hour. The natural frequencies of the stabilizer were as follows:

Vibrations per second

With rear end of fuselage rigidly supported--7.3

With rear end of fuselage unsupported--8.5

For each method of support the predominant frequency of the tail vibrations caused by buffeting was approximately the same as the corresponding natural frequency.

The stiffness of the stabilizer and fuselage was such that, when the rear end of the fuselage was externally supported, the stabilizer tip was deflected 1 inch by a force of 60 pounds concentrated at the tip.

Air flow at tail.—The surveys of the air flow at the tail are shown by dynamic-pressure contours and direction vectors (figs. 29 to 33, inclusive). The contours show lines of equal dynamic head expressed as the ratio of measured dynamic head to the dynamic head at the same point in the air stream with the airplane removed. The vectors show the component of the velocity in the plane of the survey, that is, normal to the tunnel axis. The length of the vector shows the magnitude of the component velocity v relative to the total velocity V_o in the direction of the flow at the point considered and therefore is also a measure of the angular deflection of the air flow from its initial direction parallel to the tunnel axis. When, as in this case, the angular deflections are relatively small, the scale of vector lengths can be divided so as to give directly the deflection in degrees in any direction from the tunnel axis by scaling the proper component of the vector. Thus, the angles of downwash and yaw of the air flow can be determined directly by scaling the vertical and horizontal components of the vectors. The surveys are presented with the vector scale graduated in terms of both v/V_o and the angular deflection from the tunnel axis. A specimen record of the fluctuation in dynamic pressure at the tail is shown in figure 34.

Wind-tunnel corrections.—All results except the velocity-component vectors shown on the surveys of air flow at the tail are corrected for tunnel effects.

DISCUSSION

Air flow at wing roots.—The visual observations of the air flow at the wing roots showed that the interference caused a premature stalling of the wing at that point. Several factors tend to cause this section to stall prematurely: The presence of the fuselage, which tapers to the rear and toward the bottom, increases the volume into which the air coming over the wing in that region must diverge; the side of the fuselage offers additional frictional resistance increasing the adverse pressure gradient; and the large drag of the engine absorbs much kinetic energy from the air and makes it less able to overcome the adverse pressure gradient. The observations showed that the disturbance started in this region at an angle of attack as low as 5° for the airplane in the original condition. The use of devices which either decreased the rate at which the air flow, (FIGURE 8.—Nose of *McDonnell* airplane in original condition and with N.A.C.A. cowling.), had to diverge or increased the kinetic energy of the air next to the fuselage postponed the breakdown of flow to much higher angles of attack.

Lift and drag characteristics.—A comparison of the polar for the airplane in the original condition (fig. 19) with the theoretical induced-drag polar for the whole wing (aspect ratio = 6.2) and for the portion at one side of the fuselage (aspect ratio =2.9) agrees with the observations of the air flow at the wing roots in indicating that even at relatively low angles of attack the smooth flow over the wing broke down next to the fuselage so that the part of the wing on each side of the fuselage tended to act independently as a wing of low aspect ratio.

The effectiveness of the fillets and N.A.C.A. cowling in preventing the premature break down of flow at the wing-fuselage intersection is attested by the straightness of the lift curves and the parallelism of the polars to the induced-drag polar as seen in figures 17 and 18. Both the large fillet and the N.A.C.A cowling postponed the breakdown of the flow to within 3° of the angle of maximum lift, although the double curve near maximum lift when the cowling was used alone indicates an unstable state or flow at high angles of attack. Figures 17 and 18 also show that the reflexed trailing edge increased the angle of attack at which the flow started to break down by about the same amount that the incidence of the wing at the root was changed (2° or 3°), but once the flow started to break down the reflexed trailing edge had little effect. The improvement due to the auxiliary airfoils in the best position tested was only about half as much as that due to the fillets or the N.A.C.A cowling. It is possible, however, that this is not the optimum position for the airfoils, as only three positions were tested.

When used alone the large fillet was found to give slightly better lift and drag characteristics than the small one, as shown by comparison of the two polars (fig. 11); but when used with the N.A.C. A. cowling the results were practically identical.

In addition to its effect on the wing-fuselage interference the N.A.C.A. cowling gave a large reduction in parasite drag. The minimum drag coefficient was reduced from 0.0637 to 0.0590 by the cowling; to 0.0625 by the large fillet; and to 0.0580 by the combination of large fillet and cowling.

The best lift and drag characteristics were obtained when the large fillet and N.A.C.A cowling were used together. The use of this combination eliminated most of the wing-fuselage interference, increased the maximum lift 11 percent above its original value, decreased the minimum drag 9 percent, and increased the maximum lift/drag ratio 19 percent.

The slipstream prevented a premature break down of the flow near the wing-fuselage intersection in all except the original condition and even in this condition the improvement was very great (figs. 20 and 21). In the original condition the lift curve begins to break over at almost the same angle of attack (about 6°) as without the slipstream; but as the angle of attack was increased, corresponding to a lower flying speed in level flight, the slipstream velocity became much greater relative to the air speed until it was sufficient to smooth out the flow, and at 12° the lift was

almost as high as when the large fillet and cowling were used. Beyond 12° the flow apparently started to the slipstream for maintaining the smooth flow, especially during landing.

Preliminary tests showed that the presence of the raised walkway next to the fuselage had no appreciable effect on the characteristics of the airplane equipped with the small fillet and that removing the walkway and covering the gaps between the wing and fuselage when the airplane was not equipped with any of the special devices had a negligible effect.

The maximum lift coefficient of the airplane in its original condition, as determined by these tests, was considerably higher than the highest value measured in flight with slots closed and flaps neutral (reference 12). This difference was due to the fact that in flight the pilot was not able to maintain steady conditions long enough to take satisfactory records at angles of attach above 16°.

Pitching moments.—Improving the air flow at the wing roots resulted in a slight decrease in longitudinal stability (fig. 22), due mainly to the increased downwash at the tail (fig. 23). The curves of pitching moments with elevator neutral and with power on presented in figure 24 show that there is very little difference in pitching moments between the various conditions with power on.

The effectiveness of the elevator (fig. 25) was increased by the devices that reduced the wing-fuselage interference, probably because of the higher velocity of flow over the tail (figs. 29 to 33, inclusive). Additional data taken at other angles of attack showed that the improvement extended over about the same angle-of-attack range as the corresponding improvement in lift and drag characteristics (from about 8° to beyond the stall). (FIGURE 9.—Drawing of the reflexed trailing edge).

Tail buffeting.—The effectiveness of the various devices in reducing tail buffeting is clearly shown in figure 27. The oscillations due to buffeting were reduced to amplitudes small enough to be considered unobjectionable throughout the range of normal flight attitudes by the use of the fillets, either alone or in combination with the N. A.C.A. cowling or the reflexed trailing edge. The use of the large fillet alone gave the least buffeting, reducing the oscillations to one seventh their original amplitude. The use of this fillet with the cowling, the combination giving the best lift and drag characteristics, reduced the vibrations to one fourth their original amplitude. The slipstream was practically as effective as the fillets.

In general, the various devices decreased the buffeting in about the same proportion that they improved (FIGURE 10.—Reflexed trailing edge with fillet on *McDonnell* airplane) the lift and drag characteristics. The N.A.C.A. cowling was an exception to this rule because whenever it (FIGURE 11—Polars for McDonnell airplane with various fillets. Corrected for tunnel effects), was used the buffeting was greater than would have been expected from the improvement in the polar. This excessive buffeting was probably due to the vortices mentioned in connection with the observations of the air flow at the wing root and seen on the survey of the air flow at the tail (fig. 32). (FIGURE 12.—Lift and drag of McDonnell airplane with various fillets. Corrected for tunnel effects).

The records of the stabilizer-tip movements (fig. 26) show the nature of the vibrations. It will be noted that the vibrations had a quite definite frequency, (FIGURE 13—Polars for McDonnell airplane with reflexed trailing edge. Corrected for tunnel effects), which was practically the same as the free-vibration frequency of the stabilizer. The amplitude, however, was so irregular that to an observer the motion looked like a haphazard shaking of the tail. There appeared to be very little deflection of the stabilizer and elevator as a beam, most of the deflection being due to twisting of the fuselage.

The vibrations of the stabilizer obtained under the conditions of these tests afford good comparisons between, (FIGURE 14), the degrees of buffeting under the various conditions tested, although the results of the special tests made with the rear end of the fuselage unsupported, (FIGURE 15—Polars for *McDonnell* airplanes with auxiliary airfoils. Corrected for tunnel effects. Power off), indicate that in actual flight the magnitude of the oscillations would be about twice as great as the values given in figure 27. The frequency is apparently dependent upon the natural frequency of the tail structure, which is slightly higher with the tail unsupported.

The severity of buffeting was shown to increase rapidly with increase in air speed between 35 and 60 miles per hour (fig. 28). (FIGURE 16—Lift and drag of *McDonnell* airplane with auxiliary airfoils. Corrected for tunnel effects. Power off.) It cannot be assumed, however, that this rate of increase would continue at velocities, (FIGURE 17—Polars for *McDonnell* airplanes comparing various devices. Corrected for tunnel effects. Power off.), above those investigated, as the relations may be affected by resonance between the natural frequency of the tail and the frequency of the buffeting eddies.

Air flow at tail.—The surveys of the air flow at the tail of the airplane substantiate the observations from the other data in regard to the effects of the wing-fuselage, (FIGURE 18—Lift and drag of *McDonnell* airplane comparing various devices. Corrected for tunnel effects. Power off.), interference on the air flow and lift distribution and indicate in more detail how elimination of the interference reduced tail buffeting. The lift distribution, (FIGURE 19—Polar for *McDonnell* airplane in original condition. Corrected for tunnel effects. Power off.), near the fuselage for the various conditions is indicated by the downwash vectors and also by the vertical position of the wake from the wing roots. For the original condition, prominent vortices next to the fuselage (figs. 29 and 30) show that the part of the wing on each side of the fuselage tended to act as a separate wing, (FIGURE 20—Power-on lift of *McDonnell* airplane in original conditions. Corrected for tunnel effects. And FIGURE 21—Power-on lift of *McDonnell* airplane with large fillet and N.A.C.A. cowling. Corrected for tunnel effects.), with its pair of tip vortices. These vortices produced an upflow of the air near the fuselage which probably increased the tail vibrations by causing part of the horizontal surfaces to be stalled. In the improved conditions, such as that with the large fillet (fig. 31) and with the N.A.C.A. cowling (fig. 32), the turbulent wake from the wing roots was greatly reduced. (FIGURE

22—Pitching moments of McDonnell airplane with various devices. Corrected for tunnel effects. Power off. Stabilizer 0.6° to thrust axis. Elevator 0° to stabilizer.) (FIGURE 23—Pitching moments due to tail of McDonnell airplane. Corrected for tunnel effects. Power off. Stabilizer 0.6° to thrust axis. Elevator 0° to stabilizer.) The power-on survey (fig. 33) shows that even in the original condition the slipstream practically eliminated the vortices due to the wing-fuselage intersection.

Judging from the air-flow surveys for the original condition, it would not be possible to reduce the tail buffeting materially by moving the horizontal tail surfaces upward for any reasonable distance (figs. 29 and 30). Lowering the tail surfaces about 2 feet would cause quite an improvement, but would bring the stabilizer down near the bottom of the fuselage—an impracticable location. In any case, since the interference that causes tail buffeting also causes a loss in aerodynamic efficiency, it appears best to cure the trouble at its source by methods such as those used in this investigation. (FIGURE 24—Pitching moments of *McDonnell* airplane with power on. Corrected for tunnel effects. Stabilizer 0.6° to thrust axis. Elevator 0° to stabilizer.)

The specimen record of the fluctuations in dynamic pressure at the tail (fig. 34) shows that although the fluctuations were very irregular they had some semblance of a definite frequency. It was very difficult to determine definitely either this frequency or the true magnitude of the fluctuations from the records taken, owing to the irregularity of the changes and to the large amount of damping introduced by the connecting tubes. In spite of these difficulties, however, after a careful study of the records, the following conclusions in, (FIGURE 25—Elevator effectiveness of *McDonnell* airplane with various devices. Corrected for tunnel effects. Power off. Angle of attack (thrust axis) = 15.8°), regard to air-flow conditions at the tail seem to be justified, although they cannot be considered as definitely proved: (FIGURE 27.—Amplitude of stabilizer-tip movements under various conditions. Angle of attack corrected for tunnel effects. Power off. Air speed approximately 58 m.p.h.)

1. The principal frequency of fluctuations in the wake from the wing-fuselage intersection for the airplane in the original condition was close enough to the natural frequency of the tail to indicate the possibility of resonance (fig. 34). These high-frequency fluctuations (7 or 8 per second) are of much greater magnitude, (FIGURE 28.—Variation in stabilizer-tip movements with changes in air speed. Power off. *McDonnell* airplane in original condition. Angle of attack (corrected for tunnel effect), 3.7° below α_{CLmax}), relative to the lower frequency changes than the record indicates because high-frequency fluctuations are damped much more than slow ones. (FIGURE 29.—Air flow at tail of *McDonnell* airplane in original condition, power off. Survey in vertical plane through elevator hinge line. The contours show the ratio of the dynamic pressure behind the airplane to the dynamic pressure in the free air stream. The vectors show the components of the velocity in the plane of the survey. Angle of attack (thrust axis) =14.2° (corrected for tunnel effects). Lift coefficient = 0.984.)

2. Improving the flow at the wing root increased the frequency of the eddies in the wake to approximately 50 percent greater than that for the original condition.

3. Over the range tested, from 37 to 58 miles per flour, the frequency of the fluctuations appeared to vary proportionally with the velocity.

4. In addition to the fluctuations with a fairly definite frequency there were also irregular and sudden "bumps."

It is difficult to say just how much of the buffeting motion was due to trains of oscillations set up by the bumps mentioned in item 4 and how much was due to the more regular air fluctuations of about the same frequency as the natural frequency of the tail. Undoubtedly, some of the reduction in buffeting for improved conditions of the airplane was due to the frequency of the eddies having been increased to a value well above the natural frequency of the tail.

CONCLUSIONS

The following conclusions are drawn from the tests on the *McDonnell* airplane. Differences in engine, fuselage shape, and wing section and location might modify the results for other low-wing monoplanes.

1. In addition to the presence of sudden changes or bumps, the eddying wake from the wing roots had a predominant frequency of fluctuation of the order of the natural-vibration frequency of the tail, although the fluctuations were very irregular, which suggests that the magnitude of the tail vibrations was very probably influenced to some extent by resonance effects. (FIGURE 34—Fluctuations in dynamic pressure at tail of *McDonnell* airplane in original condition, power off. Angle of attack (thrust axis) =14.2° (corrected for tunnel effects).)

2. Fillets without cowling reduced the wing-fuselage interference and tail buffeting to unobjectionable magnitudes throughout the range of normal-flight attitudes.

3. The N.A.C.A. cowling without fillets reduced the wing-fuselage interference and tail buffeting to unobjectionable magnitudes at angles of attack up to within 3° of the stall.

4. The reflexed trailing edge had a minor effect, slightly increasing the amplitude of tail oscillations due to buffeting.

5. The auxiliary airfoils, in the positions tested, gave some improvement but were considerably inferior to the fillets.

6. Buffeting was least when the large fillet was used alone. This fillet reduced the amplitude of stabilizer tip oscillations from the 1.37 inches obtained with the airplane in the original condition to 0.18 inch at an angle of attack 2° below the stall.

7. The combination of the large fillet and the N.A C.A. cowling gave the best all-round results. This combination reduced the total amplitude of stabilizer tip oscillations at an angle of attack 2° below maximum lift from the original 1.37 inches to 0.32 inch, increased the maximum lift 11 percent, decreased the minimum drag 9 percent, increased the maximum lift/drag ratio of the airplane 19 percent, and increased the effectiveness of the elevator about 40 percent at angles of

attack in the landing range.

8. The slipstream was practically as effective as the fillets in reducing tail buffeting.

9. The use of fillets or other devices for eliminating wing-fuselage interference slightly decreased the longitudinal stability of the airplane.

LANGLEY MEMORIAL AERONAUTICAL LABORATORY,
NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS,
LANGLEY FIELD, VA., December 13, 1933.

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{AQ1: Should figure legends be placed at the end of the text ?}

Document 3-18(a-c)

(a) Shatswell Ober, "Report on Wind Tunnel Model Tests, Ford Tri-Motor Plane—4-AT," 17 February 1927, Accession 18, Box 80, Folder "Wind Tunnel Report," Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

(b) Aircraft Engineering Department, Ford Motor Company, "Recent Design Changes on the 5-AT and their Effect on High Speed," undated (ca. 1929), Accession 18, Box 70, Folder "#6, Report on Fairing on 5-AT," Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

(c) "The Reminiscences of Harold Hicks," Aug. 1951, transcript, pp. 81-82, 106-108. Henry Ford Museum and Greenfield Village Research Center, Dearborn, Mich.

This trio of documents concerns the design of the Ford Trimotor—in common parlance, the "Tin Goose." This aircraft was a high-wing monoplane featuring an internally braced cantilever wing, fixed landing gear, and, as its name implies, three engines. In terms of basic configuration, the airplane resembled the Fokker trimotor, but the Ford transport was made entirely of metal, whereas the Fokker consisted of a mixture of woods, metal, and fabric. The airplane flew first in 1926 and stayed in production into 1933. It came in two versions, Model 4-AT and Model 5-AT, the latter carrying 13 to 15 passengers. A grand total of 200 of these aircraft were produced. They serviced airlines from coast to coast, at a normal cruising speed of a little less than 100 miles per hour. Aerodynamically, the transport was nothing special, with a relatively high drag coefficient ($C_{D,O}$) of 0.0471.

Nonetheless, the design reflected the growing ambition for aerodynamic streamlining. Both the 4-AT and 5-AT models were refined in wind tunnels. During January and February 1927, as we see in the first document below, engineer Shatswell Ober of the Aerodynamical Laboratory at MIT tested a model of the 4-AT transport for the Ford Motor Company. Although Ober's verbiage is sparse, his intention of aerodynamically refining the aircraft is clear.

The anonymously written "Recent Design Changes on the 5-AT and Their

Effect on High Speed," the second document below, related the intention of the Ford Aircraft Engineering Department to increase the high speed of their Trimotor design through aerodynamic means. The engineering staff weighed the advantages and disadvantages of design practices and innovations in terms of weight penalties, drag, and increase in speed, problems closely associated with the reinvention of the airplane discussed throughout this essay. It is clear from this document that Ford engineers were well aware of developments taking place elsewhere in the aeronautical community, even if they did not always use the same precise terminology.

The last selection involves the reminiscences of Ford chief engineer Harold A. Hicks, the man who oversaw the development of both Trimotor versions. This testimony from 1951 provides a wonderful insight into the design of the Ford Trimotor as well as the organizational environment in which it evolved. Hicks's comments are especially revealing because they shed light on the introduction of some of the key innovations of the airplane design revolution discussed in this chapter. NACA research on cowlings, airfoil sections, and engine placement all played a role in the design innovations incorporated by Ford.

Document 3-18(a), Shatswell Ober, "Report on Wind Tunnel Model Tests, Ford Tri-Motor Plane—4-AT," 17 February 1927.

FORD MOTOR COMPANY

AIRCRAFT ENGINEERING DEPT

DEARBORN MICH.

February, 1927

REPORT ON

WIND TUNNEL MODEL TESTS
FORD TRI-MOTOR PLANES 4-AT

To provide engineering data for the existing airplane and give information in regard to the aerodynamic effect of modifications, an extensive series of tests has been made on a model of the Ford 3-engine commercial monoplane type 4-AT. These tests were all made in the four foot wind tunnel at the Massachusetts Institute of Technology during January and February, 1927.

MODEL

The models were supplied by the Motor Company. They were well made with wings, tail, fuselages, nacelles and wheels of maple, and chassis, fin, and rudder of duralumin. The models were 1/40 full size, making a span of 20.6 inches with a net wing area of 67.8 square inches. Certain of the nacelle struts were omitted entirely in an effort to avoid excessive scale effect correction on these small parts. The original model 4-AT-3 has the Ford #2 wing, an open cockpit for the pilots, square bottom fuselage, and rounded top fuselage. Modifications consisted of two different wings, N.A.C.A. 81J and Ford #3, two different noses with closed cockpits designated as closed nose and long closed nose, a square top cowl and a small round cowl for the top of the square fuselage and a second fuselage rounded both top and bottom back of the wing. With the Ford #3 wing, additional bracing would be needed.

DIMENSIONS

It is convenient to have certain full scale dimensions. These have been enlarged from the model drawings.

Span overall (Ford #2 wing)	68.7 ft.
Net wing area (Ford #2 wing)	689 sq. ft.
Net wing area (N.A.C.A. 81J and Ford #3 wing)	677 sq. ft.
Mean wing chord (Ford #2 wing)	10.7 ft.
Aerodynamic aspect ratio	6.3
Horizontal tail area total	88.9 sq. ft.
Stabilizer area	49.8 sq. ft.
Elevator area	39.1 sq. ft.
Tail span	17.6 ft.
Aspect ratio	3.3
Distance C.G. to center of pressure of horizontal tail	31.1 ft.
Percent tail area of wing area	12.9%
Percent elevator area of tail	44%
Center of gravity located aft of leading edge at center	4.0 ft.
Below wing chord	1.33 ft.
Center of gravity percent of wing chord	32%
Power 3 Wright Whirlwind	600 HP
Wing loading (Ford #2 wing)	12.8 lb/sq/ft
Power loading	14.6 lb/H.P.
Gross flying weight	8800 lbs.

It is assumed that variations in weight and C. G. due to modifications are absorbed by variations in payload.

PROCEDURE

All tests were made in the four-foot wind tunnel at a wind speed of 40 miles per hour. The model was supported by a wing tip spindle tapering from 5/16" diameter to ¼" diameter. Conventional corrections for spindle drag and interference have been applied. The angle of attack is referred to the thrust line or the wing chord.

Tests were made as follows:

1. With original model 4-AT-3.
    a. Lift, drag and pitching moments with elevator neutral and three different stabilizer settings.
    b. Lift, drag, and pitching moments with elevator 25° up and two stabilizer settings.
    c. Lift, drag, and pitching moments with elevator 25° down and 12 ½ up and down at one stabilizer setting.
    d. Pitching moments without horizontal tail.
    e. Lift, drag, without tail and nacelles.
    f. Lift, drag, and moment, without tail, nacelles and chassis.
2. With modifications as noted, model complete with square fuselage:
    a. Lift, drag, and pitching moments with neutral elevator with:
        1. Closed nose round back cowl.
        2. Closed nose square back cowl.
        3. Closed nose small round back cowl.
        4. Closed nose without back cowl.

3. With Ford #2 wing and fuselage only:

    a. Lift, drag, with round fuselage and
        1. Open nose.
        2. Closed nose.
        3. Long closed nose.

        With square fuselage, long nose and
        1. Round back.
        2. Square back.

4. With model complete with Ford #3 wing, square body, small round back and long nose modified to fit wing:
    a. Lift, drag, and pitching moments with elevator neutral and one stabilizer setting.
    b. With these parts alone: Ford #2 wing
    N.A.C.A. 81 J wing
    Ford #3 wing
    Horizontal tail.

In addition some investigations of airflow around wing, fuselage, nacelles, and tail were made. Smoke was allowed to blow from a pipe into the airstream at different points, and the path of the axis of the jet of smoke was traced on a side view of the model. The wind speed was slow, either 15 or 20 miles per hour. These tests were made at several angles of attack with open and long closed nose. (The original airflow sketches were given directly to the Ford Motor Company's representative.)

COMPUTATIONS

To facilitate comparison with model tests of other airplanes, results have been expressed as coefficients. For tests with a fuselage in place, force coefficients have been found by dividing the force by the wing area, excluding the part above the body and the square of the wind velocity in miles per hour. The units are then lbs. per sq. ft. per (M.P.H.)^2. Pitching moments were referred to an axis through the center of gravity. The pitching moment coefficient K_M was found by dividing the moment by the net wing area, the mean wing chord, and the square of the air speed. These units are lbs. ft. per sq. ft. per (M.P.H.)^2 per ft. chord.

For tests of wings alone, the gross area was used. In addition, the center of pressure, defined as the point of intersection of the resultant force with the wing chord, was found. This is expressed as a fraction of the mean chord back of the leading edge of the wing at the center section.

Downwash was found by the method of moment differences only. The effective moment of the tail about the C.G. is found by the difference in pitching moments with and without tail plane. From the test of the tail alone, the moment of the tail without downwash is calculated. These curves of effective M_{CG} and calculated M_{CG} are plotted against angle of attack. Then at any angle on the effective M_{CG} curve, the downwash is the angular change necessary to give the same moment on the calculated M_{CG} curve.

RESULTS

The results are best considered by means of the plotted curves. Some tables of the more important data are also included. Table I gives full data for the original model, stabilizer 3 deg. elevator 0 deg., Table II complete model with Ford #3 wing, Table III lift and drag data for various fuselage and nose combinations, Table IV drag data of separate parts of original model. In general K_X and K_M are plotted against K_Y. Unless noted on the curves the Ford #2 wing was used.

DISCUSSION. LIFTS AND DRAGS

With neutral elevators the maximum lift coefficient on the original model is .00420, the minimum K_X .000198, for modified model with Ford #3 wing the maximum K_Y is .00364, the minimum K_X .000175, maximum L/Ds for these models are 9.5 and 10.2 respectively, and the ratios of maximum lift to minimum drag are 21.2 and 20.8. Some allowance must be made in the values for the plane with the Ford #3 wing for additional bracing. This might be about .000005 in K_X

at minimum drag, the maximum L/D would be reduced to about 9.9, and the ratio of maximum lift to minimum drag reduced to 20.8. Compared with tests on other models the lift with Ford #2 wing is high, but the drag is also high, the maximum L/D is good, and ratio of maximum lift to minimum drag fair.

With elevators turned up to trim at maximum lift, the maximum lift coefficient with #2 wing is .00400. The minimum flying speed corresponding is 57 M.P.H. Allowing a similar reduction, the minimum speed with #3 wing would be 62 miles per hour. Maximum speed estimated from minimum drag, 600 H.P., 80% efficiency for the two models should be 109 and 114 miles without allowance for scale effect. Speeds of about 115 and 119 miles per hour might be expected. Thus, the gain in high speed is nearly equal to the increase in landing speed.

CHANGES IN DRAGS

None of the various modifications of nose, fuselage, or back cowl have any serious effects on drag. The closed nose gives slight improvement, the long closed nose very slightly more, but from the open cockpit the improvement in K_X is only .000006, only 3%. The change in speed corresponding would be only 1 or 2 miles per hour. The form of back cowl has an almost negligible effect on the drag, except when the long closed nose is fitted, then the square cowl is not as good.

DRAGS OF PARTS

The total parasite drag of the model is affected by the wing section, as the total depth of fuselage varies with wing thickness. So with the Ford #2 wing, the total parasite at minimum drag is equivalent to 26.7 sq. ft. of flat plate, while with the #3 wing, it is 22.2 sq. ft. This difference is partly due to reduction in area of fuselage, partly to use of the better nose, and the rest to lessened interference. Against this reduction must be balanced about 1.0 sq. ft. of flat plate increase for extra bracing. The body has an effective drag of 8.7 sq. ft. with open cockpits, reduced perhaps to 7.5 sq. ft. with long closed nose. These flat plate areas are about 1/5 of the cross-sectional area. The chassis drag is some 9.8 sq. ft. of flat plate, naturally very high on account of the long struts. The nacelles together are equivalent to 6.4 sq. ft., about 1/3 of the total area to the outer ends of cylinders.

LONGITUDINAL STABILITY AND BALANCE

The curves of pitching moment about the center of gravity indicate ample longitudinal stability at all flying speeds, increasing as the speed decreases. The stability is not much affected by variations in back cowl, nose, or wing section.

With the Ford #2 wing, the open cockpit nose, and round back, a stabilizer setting of plus 3.0 deg. to the thrust line gives a trimming speed of 93 MPH. This occurs at an angle of attack of –1 deg. With closed nose, otherwise unchanged, the trimming speed is 94 MPH, practically the same. The form of back does affect the balance, however, the model with square back trims at 122 MPH, with the small round back at 98 MPH, with no cowl at 84 MPH. Except for the last, the stabilizer

adjustment is ample to control the trimming speed, if such cowls should be fitted on the ship.

With the Ford #3 wing, the long nose modified to fit that wing, the square fuselage, and round back, the trimming speed with stabilizer plus .2° is 95 MPH. The engine of attack is plus 1 ½ deg.

ELEVATOR CONTROL

From these tests the elevator control appears ample. Even with the stabilizer set to trim at cruising speed, only about 17 deg. up elevator is needed to give equilibrium at maximum lift. With the stabilizer set full tail heavy –5.6 deg., the ship trims at only 58 MPH with neutral elevator.

DOWNWASH

The effective downwash may be found for moderate lifts by the equation 2° plus 1775 K_Y. The 2° is due to flow along the fuselage. The slope coefficient 1775 is about .8 of the theoretical mean downwash for this aspect ratio. It is to be expected that the downwash will be modified by the body. There is no indication of a reduction in downwash near the stalling point.

TESTS OF WINGS ALONE

The wings when tested alone give some 12% less total lift than the complete model. Most of the difference is made up by the fuselage, as with wing and body alone the lifts are nearly equal to those for the complete model. The maximum lift coefficients for wings alone are Ford #2, .00373, Ford #3 .00336, N.A.C.A. 81J, .00312; the minimum drag coefficients .000064, .000054, and .000056. The maximum L/Ds are 16.3, 18.0, and 15.9. As the N.A.C.A.J. appeared inferior, it was not tested with a fuselage or complete model.

AIRFLOW TESTS

The airflow tests did not lead to any very tangible results. Beyond the burble point, there is a distinct flow forward just above the trailing edge, but from all that could be observed, this did not extend back far enough to "blanket" the tail seriously. If there were any serious "blanketing" it would appear as a marked reduction in effective tail moment.

CONCLUSION

The question of using the thinner Ford #3 with external bracing should be given most careful consideration, both in regard to estimates of aerodynamic advantage, and changes in weight and type of structure. As to the former, it appears that the Ford #3 wing is somewhat better.

The question of open or closed nose is largely one of comfort and convenience of pilot. Aerodynamically the long, well faired, closed nose is somewhat superior.

The type of back cowl must be selected from structural or other consideration beside aerodynamic.

In regard to the wing angle, it appears that with the Ford #2 the incidence of the wing to the thrust line could well be reduced from zero to –1 deg. or –2 deg., as the plane flies nose down, even when cruising. With the Ford #3 wing the present angle of zero degrees might be increased to plus 1°.

Aerodynamic Laboratory Staff,
By—Shatswell Ober
Mass. Inst. Of Technology

TABLE I

Wind Speed: 40 M.P.H.
Model Size: 1/40 Scale
February, 1927

α	Lift	Drag	L/D	X	Z	M_{CG}	K_Y	K_X	K_M
–8	–.024	.1553	–.15	.157	–.046	.185	–.00004	.000225	.000084
–6	.278	.1403	1.98	.169	.262	.145	.00041	.000204	.000066
–4	.566	.1360	4.16	.175	.555	.091	.00082	.000198	.000041
–2	.890	.1405	6.33	.172	.884	.040	.00129	.000204	.000018
0	1.190	.1470	8.09	.147	1.190	–.042	.00173	.000213	–.000019
2	1.493	.1652	9.04	.113	1.498	–.198	.00217	.000240	–.000090
4	1.782	.1887	9.44	.064	1.791	–.359	.00259	.000274	–.000163
6	2.054	.2144	9.57	–.001	2.065	–.494	.00298	.000311	–.000224
8	2.312	.2497	9.25	–.075	2.327	–.621	.00335	.000363	–.000281
10	2.524	.2863	8.82	–.157	2.536	–.751	.00367	.000416	–.000337
12	2.735	.3322	8.22	–.245	2.742	–.853	.00397	.000482	–.000385
14	2.831	.3788	7.47	–.317	2.839	–1.048	.00411	.000550	–.000474
16	2.904	.4466	6.50	–.371	2.91	–1.263	.00421	.000649	–.000572
18	2.736	.6010	4.55	–.274	2.78	–1.462	.00397	.000872	–.000661

OC = angle referred to thrust line
3°.0 = angle of tail plane to thrust line
Elevators neutral
Tested in Aeronautical Laboratory 4 Ft. Wind Tunnel at M.I.T.

Document 3-18(b), Aircraft Engineering Department, Ford Motor Company, "Recent Design Changes on the 5-AT and their Effect on High Speed," undated (ca. 1929).

SUMMARY

Results of tests for High Speed with various units recently designed for the 5-AT have been compiled for reference. A statement is made of the most important improvements and the high speed which might be attained if all these units were incorporated on one airplane. That there are possibilities for still further improvement is indicated by a statement of several lines of development, some of which have already been demonstrated as practical and others which have yet to be proven.

The following conclusions are drawn from study of speeds obtained from the 5-AT as listed in the table:

The job with wide fuselage has the same top speed as with the standard fuselage.

Raising the wing has little if any effect on high speed. (0.6 MPH increase recorded but may be due to different prop. combination.)

Long outboard mounts. (PX-17862) with more clearance between wing and nacelle shows about 5 ½ MPH increase.

Propeller combination of 3 blade outboard and 2 blade center shows a slight advantage over 2 blades for all props. Less than 1 mile per hour is consistently recorded.

About ½ MPH increase is recorded for improved streamlines on landing gear struts SK-275-277.

Tail surface fairing SKD-297 shows a decrease of about ¾ MPH.

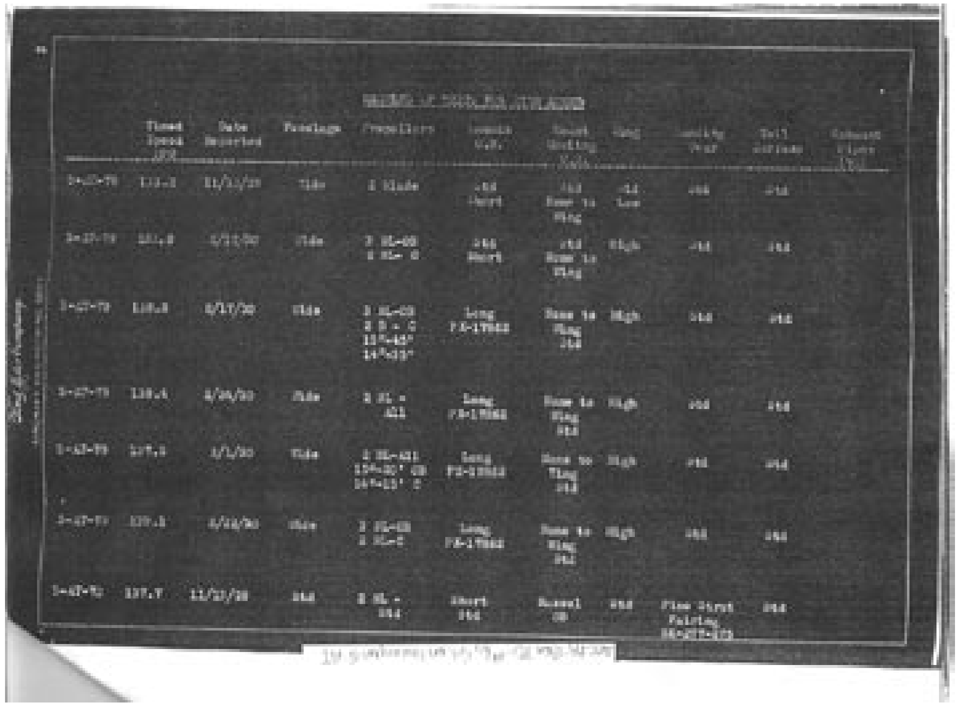
Nacelle to wing fairings (PX17799 revised) shows an improvement in high speed of about 3 ½ MPH. Note that this is on short mount.

Russel's outboard cowling shows about 3 ½ MPH increase.

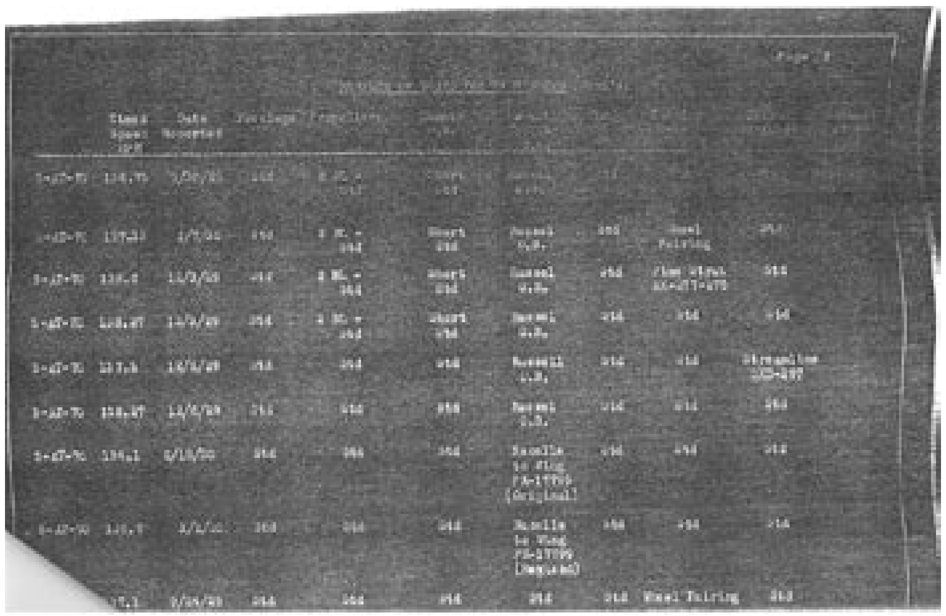
Landing gear wheel fairings show about 1 ½ MPH increase.

Possible Improvement in High Speed with all Changes Incorporated on one Airplane.

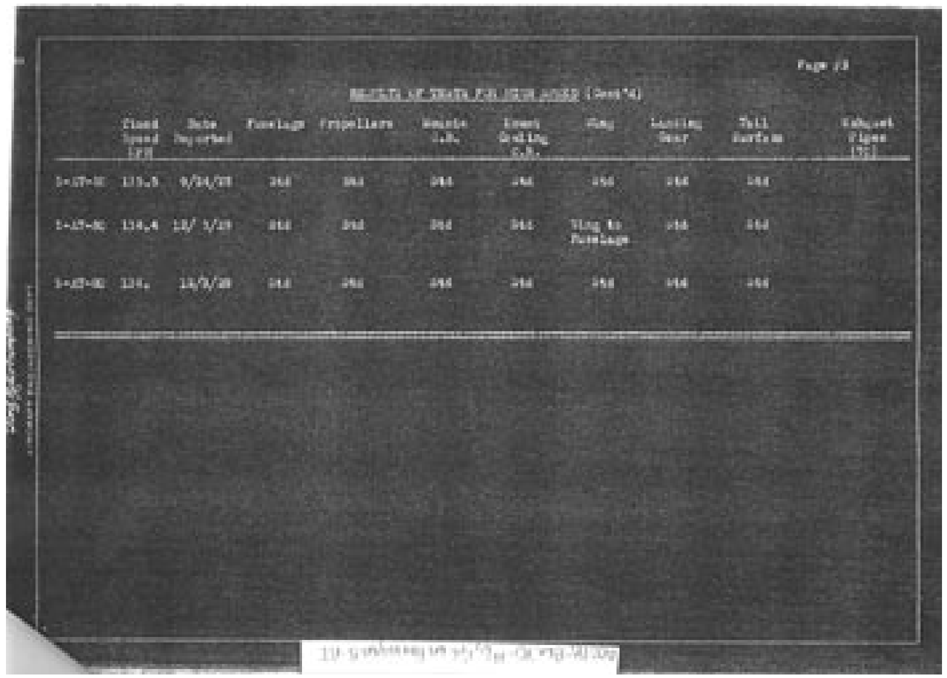
	Increase in speed	Increase in weight
Russell's outboard cowling and Mayo exhaust	3.45	50 lbs.
Landing Gear Strut Fairing SK-275 and 277	.33	10
Nacelle to wing fairing similar to PX-17799	3.80	20
Long engine mounts PX-17862	5.50	12
3 blade outboard and 2 blade center props	.60	64
Wheel Fairing	1.60	50
Wing to Fuselage Fairing	.40	30
5-AT standard airplane	133.30 M.P.H.	
Total	148.98 M.P.H.	236 lbs.



	Timed Speed MPH	Date Reported	Fuselage	Propellers	Nacelle C.G.	Smooth Cowling Nacelle	Wing	Landing Gear	Tail Surface	Exhaust Pipes
[illegible]	[illegible]	[illegible]	Wide	2 Blade	Std Short	Std Nose to Wing	Std Low	Std	Std	
[illegible]	[illegible]	[illegible]	Wide	[illegible]	Std Short	Std Nose to Wing	High	Std	Std	
[illegible]	[illegible]	[illegible]	Wide	[illegible]	Long [illegible]	Nose to Wing Std	High	Std	Std	
[illegible]	[illegible]	[illegible]	Wide	[illegible]	Long [illegible]	Nose to Wing Std	High	Std	Std	
[illegible]	[illegible]	[illegible]	Wide	[illegible]	Long [illegible]	Nose to Wing Std	High	Std	Std	
[illegible]	[illegible]	[illegible]	Wide	[illegible]	Long [illegible]	Nose to Wing Std	High	Std	Std	
[illegible]	[illegible]	[illegible]	Std	[illegible] Std	Short Std	[illegible]	Std	[illegible] Fairing [illegible]	Std	



	Timed Speed MPH	Date Reported	Fuselage	Propellers	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	Std	[illegible]	Short Std	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	Std	[illegible] Std	Short Std	[illegible]	Std	Wheel Fairing	Std	
[illegible]	[illegible]	[illegible]	Std	[illegible] Std	Short Std	[illegible]	Std	[illegible]	Std	
[illegible]	[illegible]	[illegible]	Std	[illegible] Std	Short Std	[illegible]	Std	Std	Std	
[illegible]	[illegible]	[illegible]	Std	Std	Std	[illegible]	Std	Std	Streamline [illegible]	
[illegible]	[illegible]	[illegible]	Std	Std	Std	[illegible]	Std	Std	Std	
[illegible]	[illegible]	[illegible]	Std	Std	Std	Nacelle to Wing [illegible] (Original)	Std	Std	Std	
[illegible]	[illegible]	[illegible]	Std	Std	Std	Nacelle to Wing [illegible] [illegible]	Std	Std	Std	
[illegible]	[illegible]	[illegible]	Std	Std	Std	Std	Std	Wheel Fairing	Std	



	Timed Speed MPH	Date Reported	Fuselage	Propellers	Nacelle C.G.	Smooth Cowling C.G.	Wing	Landing Gear	Tail Surface	Exhaust Pipes
[illegible]	[illegible]	[illegible]	Std	Std	Std	Std	Std	Std	Std	
[illegible]	[illegible]	[illegible]	Std	Std	Std	Std	Wing to Fuselage	Std	Std	
[illegible]	[illegible]	[illegible]	Std	Std	Std	Std	Std	Std	Std	

<u>Possibilities for Further Streamlining</u>

1. The use of fairing between the wing and nacelle gave a marked improvement on the short mount installation. The lengthening of the mounts giving more gap between nacelle and wing also improved high speed. The fairing of the long mount between nacelle and wing has not been tried as yet and seems to have possibilities for an added improvement. Would suggest that the radius where this fairing joins the wing be made as large as possible to properly join the airflow around the wing and the streamline. (Ref. PX-17799)

2. Mr. Manning's report of observations with strips of tape shows a very imperfect air flow around the nacelle. This condition is probably due to rotating airflow in the slipstream and the streamlining at somewhat of an angle with resultant direction of flow. Possibly we could gain an advantage by placing the streamlines on the struts at an angle to favor this resultant and thus change the direction of flow less abruptly.

3. The use of nacelles of greater length—diameter ratio is a possibility, although it has probably been previously investigated for our conditions. The length now given is of the Navy strut form, but recent propeller slipstream investigations show a lower pressure area than the remainder of the propeller disc, 15 ft. aft of the propeller.

4. The working of the exhaust manifold into an annular ring of streamline cross section with enough cowling effect to help to pay for at least its own resistance might help. Another idea along the same line is to have the exhaust outlet at the trailing edge of such an annular streamline manifold, thus throwing the exhaust gases in a direction to straighten out the flow around the nacelle. These gases, warmer than surrounding air, might set up a flow around the nacelle with beneficial lubricating effect on the much discussed boundary layer. In fact, it would be possible to approximate or better the effect produced by blowing air from slots on surfaces of bodies to decrease their resistance (German experiments about 1926).

All this has double possibility since it uses to advantage something which was formerly carried at a loss.

5. Further research on the correct way to fair the wing into the fuselage on top surface might have possibilities. Suppose experiments with 10-A models have already given considerable information on this subject.

6. Less abrupt projections around the cockpit and experiments with various cockpit enclosures seem a possibility for decreasing resistance. Decreasing the included angle of the windshield to less than 100 degrees will decrease fuselage drag—as per recent article in "Aviation".

7. Tail surfaces with more rounded tips and tapered and a possible increase in aspect ratio of stabilizer which would allow a decrease in area.

8. A considerable number of control horns and cables are exposed at present. Elimination of these projections would be an advantage.

9. A fuselage of elliptical cross section faired into the tail surfaces with large fillets would cut resistance considerably.

10. Fillets between wing and fuselage have of course been previously considered.

11. A method of changing the lift on a control surface (other than by breaking it on a hinge) such that it could be carefully faired into the fuselage is a possible future development. Slots, warping, external spoilers, and auxiliary airfoils are possibilities.

12. It is possible that a method of decreasing induced drag of a wing other than by increasing the aspect ratio can be developed, since it is a question of reducing the vortex at the tip.

Document 3-18(c), "The Reminiscences of Harold Hicks," Aug. 1951.

The 4-AT plane, although it was first powered with the J-4 engines, used the J-5 engines when Wright built them. Then George Mead, a man for whom I used to work down at McCode, and Fred Rentschler started the Pratt & Whitney Aircraft Company. They were previously with the Wright Company, and Wright didn't appreciate the fact, so they told me, that they saved Wright about a million dollars in one year. They didn't want to expand or something of that kind, so they pulled out.

They got Gordon Rentschler, Fred Rentschler's brother who was president of the National City Bank, to start them in business over in Hartford in the Pratt & Whitney Company's plant. They brought out the Wasp engine. Andy Willgoose really designed it. They came here about 1926, to get us to put the Wasp in our airplane.

We put the Wasp on the tri-motor plane, and of course that increased the performance. It upped the performance from about 114 miles an hour to about 128 miles an hour. That was before we had the N.A.C.A. ring cowls [SIC] on the job. That plane was known as the 5-AT.

Pressurized cooling was done on the big 10-AT, I think. It was a big ship we never flew, but this plane was equipped with liquid-cooled engines.

On the first Pratt & Whitney engines that we had, we designed cooling baffles that went directly across the cylinder head. In other words, we were testing sort of a cowl that was something like the final N.A.C.A. cowl before they started their cowling work. That was done largely to increase the speed of the ship. That it did, because the speed of the ship jumped up to 152 miles an hour.

Manning who had come with us at that time was our test pilot. He established the world's record for carrying a two-thousand kilogram load under a closed course. The wind shifted on the tri-angular course when he was flying the ship, enabling him to hang up a speed of about 160 miles an hour. The Ford tri-motored plane held that record for some time.

That special cowl was first used on the 5-AT. That was the one with the Pratt & Whitney engine.

We started putting the N.A.C.A. rings on all of our engines after the T.A.T. had bought a large number of the planes. The planes were brought in here for reconditioning. It was possibility in 1928 or 1929 that we started to put cowlings on all the engines. The top speed of the plane was increased from 128 to 152 or 154 miles per hour by these rings. No increase in fuel consumption ensued….

I have always felt that Tom Towle, more than anyone else, probably had more to do with the Ford tri-motor design initially than any one particular person, although it was a composite design by eight people. I myself worked up the landing gear in addition to supervising the design.

The Venturi type of exhaust was an idea that I had based upon air flow about a central exhaust tube. The idea was in connection with this low wing monoplane that we had worked upon, a fourteen place job that never materialized. It was because of Mayo's failure to give us the okay to proceed, he being interested in the 14-AT which was a large ship.

We were actually considering having externally braced wings, although it was a low wing job. We had two designs worked out. One was a fully cantilevered wing, and the other had two struts on each side as external bracing. We had figured that these tubes might easily be used as exhausts.

Al Esper, who was in charge of our wind tunnel activity in the old hangar building, had developed a hydraulic flow channel which was a return type using ink as the liquid in the channel and having aluminum powder which floated in a flecked condition on the surface.

By placing our models in this channel and the liquid being pumped around, we could see exactly then what the air flow would be like. There we worked out an exhaust system of fairing around a round tube that was truly remarkable. It was based entirely on Burnelli's theorem of flow where the energy is constant everywhere in the universe; that is, potential energy is converted into kinetic energy.

Nothing was ever done about that because that plane, of course, was never developed. It was just an interesting thing because it recalls the fact that we experimented in those early days with so many ingenious things that we could use and apply if we had to. We were always leading the aircraft industry at the time. The ideas would never get into practical operation because perhaps the immediate application for it never presented itself.

There was a freedom of thought and a freedom of opportunity to try things here in the early Ford Motor Company such as I have never seen before in any organization or since. It was truly a designer's paradise. Of course, there was always the treatment that you would get from Henry Ford that was to be feared perhaps. You knew that everything that you did was subject to immediate explanation and criticism.

For a person with any degree of courage, you never hesitated to go and try things out and do things because of the liberal attitude the Fords had on the working out of new ideas and new thoughts. They would take a personal interest in these thoughts.

You would always want to explain to them what you were doing and never attempt to conceal anything from them. You had to be very aboveboard with everything.

They would delve down into the detail. They would want to know everything about it, for example, the Venturi exhaust. As a matter of fact, you had to have everything on hand to explain to them; otherwise, they were so busy that they had little patience. If you didn't have things immediately available to show them, well, they had business elsewhere.

The wind tunnel was not particularly unique at the time the Ford Motor Company put its own tunnel in; there were wind tunnels then at that time. Because of the cramped space, we had to use a terrific amount of power and exhaust the air free into the atmosphere. That is, we couldn't build up the inertia of a continuously moving column of air.

We had a 250 horsepower electric motor up in the second story of the old hangar building that we connected with a large sirocco fan. We blew that air directly through the throat of our Venturi and in there used models about one-tenth size, as I remember. We used this for all of our aerodynamic work.

Al Esper was the technician who did the work at the wind tunnel. He was the one who was responsible for working out flow patterns, and he did an excellent job of it. The model making was done by James Lynch who is presently employed here in the Styling Department.

Document 3-19

"Lockheed 'Sirius' Monoplane," *Aero Digest* 16 (January 1930): 128.

The aviation trade journals played a vital role in announcing the introduction of aircraft that embodied new design principles—even those like the Lockheed Sirius that still retained elements of the old. In January 1930, *Aero Digest* magazine profiled the Sirius, a single-engine monoplane with fixed landing gear that was essentially a low-wing version of the Vega. But the plane incorporated Jack Northrop's innovative construction techniques, and even with its fixed gear and open cockpits, it reflected a streamline design. The Sirius was, in fact, quite similar to another sleek low-wing monoplane of the day, the Northrop Alpha, except that the Sirius was made entirely out of wood, whereas the Alpha was all metal. In 1929, Charles A. Lindbergh purchased a Sirius for his attempt to make a record nonstop flight from the West Coast to the East Coast. In 1931, he and his wife Anne Morrow flew the plane on their survey flight to Japan, via Alaska and Siberia.

Document 3-19, "Lockheed 'Sirius' Monoplane,"* Aero Digest *16 (January 1930).

LOCKHEED
"SIRIUS" MONOPLANE

The first of a new series of Lockheed planes, known as the Lockheed "Sirius," and named after one of the brightest stars in the firmament, is to go to Colonel Charles A. Lindbergh. Edward S. Evans, president of Detroit Aircraft Corporation, has announced that the Lockheed division will add the Lockheed Sirius type of plane to its regular line of models. Inasmuch as the various Lockheed models are named after stars, the name Sirius was selected as appropriate for the new type of plane.

First test flights of the Lockheed Sirius were made at the United Aircraft field near the Lockheed factory. Although no tests were run over a measured course, the Sirius is reported to have proved exceptionally fast, was highly maneuverable, showed good stability and landed at low speeds.

The Department of Commerce has paid tribute to Colonel Lindbergh's famed ship, the Spirit of St. Louis, in granting of the license of the new plane. It will be remembered that "WE" bore the identification NX-211 and the new ship will be known on Government records as NR-211. The change in the prefix on the number from X to R is due to new government regulations and will permit operation of the plane in countries outside of the United States. In its coloring Colonel Lindbergh's new ship is quite distinctive. The entire fuselage is black and has a wide gold stripe on the mid-section of each side and extending the entire length of the body. The wings, extending from the lower side of the fuselage, are painted orange-red, as are the control surfaces on the tail.

The streamline of the new ship has been especially well worked out. The engine is completely enclosed in a cowling of the type originated by the National Advisory Committee for Aeronautics. It is estimated that this cowling increases the speed of the plane fifteen miles per hour. Although the cowling fits closely over the engine and the nose of the plane, ample provision has been made for cooling. The Pratt & Whitney Wasp engine on the Lockheed Sirius develops 425 horsepower at 2,000 revolutions per minute.

The streamline effect is carried on throughout the fuselage from the engine back to the control surfaces. The ship embodies the Lockheed type of monocoque construction, resulting in freedom from external bracing to the wings, which would greatly hamper the streamline effect. Special low type wind-shields also have been fitted.

The two cockpits are located well in the rear. Both are equipped with complete controls, and the forward place has a number of special instruments. The two places are spacious and comfortable.

In the landing gear is found further evince of streamline effect. New Lockheed hydraulic shock-absorber struts are used in this construction. The wheels are almost entirely encased in a special streamline housing known as "Pants."

The wings of the plane are of low-wing cantilever construction. The mean chord is eighty-two inches and the wing area is 265 square feet.

Tanks for 320 gallons of gasoline are built in the forward part of the fuselage and an additional 120 gallons are carried in the wings. The oil tanks have a capacity of 28 gallons.

The Lockheed Sirius was primarily designed as a two-place sport plane of high performance. For average service it would carry two persons and their baggage with 150 gallons of gasoline over a range of approximately 1,100 miles. For Colonel Lindbergh's use, however, the extra gas tanks were installed, bringing the total capacity up to 440 gallons of fuel and increasing the range to 3,300 miles. This arrangement was specified by Colonel Lindbergh so that he may make extended flights without the long delays caused by landings for gasoline.

With the 3,300-mile range as desired by Colonel Lindbergh, there is a considerable increase in the gross weight, a condition which the Sirius was designed to meet. With tanks for 440 gallons of gasoline, the plane weighs 2,950 pounds. Gasoline weighs 2,640 pounds, and the twenty-eight gallons of oil, 210 pounds. Each person and equipment is allotted 200 pounds, and another 100 pounds is estimated for baggage. This results in a gross weight of 6,200 pounds, with the plane fully loaded for a flight of 3,300 miles.

Weights given below are for a standard production Sirius with a fuel capacity of 150 gallons and an oil capacity of twelve gallons.

SPECIFICATIONS

Span	42 feet 10 inches
Chord (mean)	82 inches
Wing area	265 square feet
Length overall	27 feet 6 inches
Normal fuel capacity	150 gallons
Normal oil capacity	12 gallons
Normal cruising radius	1,100 miles

Weights

Weight empty	2,950 pounds
Payload	500 pounds
Disposable load	1,450 pounds
Gross weight loaded	4,400 pounds

Document 3-20(a-c)

(a) "The New Martin Bomber: Model XB-907," Glenn L. Martin Company Engineering Report No. 240, 5 August 1932, Accession A1, Folder "B-10/char," United States Air Force Museum Research Division, Dayton, Ohio.

(b) Glenn L. Martin to Walter G. Kilner, undated (ca. 1933), Accession A1, Folder "B-10/(series)/his," United States Air Force Museum Research Division, Dayton, Ohio.

(c) Glenn L. Martin Company, "History and Development of the Martin Bomber," 20 January 1938, Accession A1, Folder "B-10/(series)/his," United States Air Force Museum Research Division, Dayton, Ohio.

This trio of documents concerns the development of the Martin B-10 bomber, one of the first aircraft in which all the various constituents of the airplane design revolution of the interwar period started to come together. A sleek, stress-skinned, all-metal monoplane, the B-10, which first flew in February 1932, delighted the U.S. Army Air Corps by reaching speeds that made all existing fighter planes obsolete. While the best fighters of the day could fly no faster than about 190 miles per hour, a B-10 could rush in to a target with a ton of bombs at 185 mph and speed away at 210. As historian Richard K. Smith has noted, as late as 1935, "there was not a fighter plane in the world that could catch a Martin B-10" ("Better: The Quest for Excellence," in *Milestones of Aviation* [New York: Hugh Lauter Levin, 1989], p. 255.). This came as a terrible shock to military establishments around the world as, up until this time, bombers could fly no faster than about 120 mph and cruise at only 95. None of these bombers possessed an all-metal structure; they were covered in fabric. None of them carried their bomb loads internally behind streamlined doors, as the B-10 did, and few of them had enclosed cockpits. And none of them had NACA cowlings and retractable landing gear like the B-10 did. All in all, Martin's integrated design of the B-10 led to a "combat airplane revolution" involving not just bomber design but also a quest for superior fighters.

Although one of the keys to the B-10's success was its two 775-hp Wright R-1820 engines, aerodynamic streamlining played a critical role. Besides utilizing an enclosed cockpit and retractable landing gear, the Martin Company also made good use of up-to-date NACA data on engine cowlings and the optimum placement of engine nacelles. The NACA's director of research, George Lewis, personally advised Glenn L. Martin sometime around 1930 to 1931 that Martin's new bomber design would not only fly significantly faster than its present 195 miles per hour, but would also land slower and more safely, if the engine's Townend ring were replaced by the NACA No. 10 cowl. Pratt & Whitney, the builder of the engine for the B-10, was contractually committed to using the ring. Martin eventually adopted the NACA cowling, increasing the airplane's maximum speed by 30 mph to 225 and also reducing its landing speed significantly. In 1932, the struggling company won the Collier Trophy for its B-10 design, and in the next two years, the army purchased more than 100 B-10s, rescuing Martin from the worst of the Depression. In fact, what the cowling did for the B-10's performance may well have been why Martin won the production contract from the army and why Boeing's B-9, which used the Townend ring, lost.

Perhaps the most illuminating document in this trio is the second one, a letter solicited by Major Walter G. Kilner of the Office of the Chief of the Air Corps around 1933, in which Glenn L. Martin explained the importance of his bomber design.

Document 3-20(a), "The New Martin Bomber: Model XB-907," Glenn L. Martin Company Engineering Report No. 240, 5 August 1932.

THE NEW MARTIN BOMBER

MODEL XB-907

CHARACTERISTICS AND FEATURES

The Glenn L. Martin Company
Baltimore, Maryland

August 5, 1932

GENERAL SPECIFICATIONS

AREAS

Wing, total, incl. ailerons	505 sq. ft.
Ailerons, rear of hinge	45.2 sq. ft.
Stabilizer	42.2 sq. ft.
Elevator (incl. 8.7 sq. ft. balance)	41.8 sq. ft.
Fin	20.1 sq. ft.
Rudder (incl. 5 sq. ft. balance)	25.9 sq. ft.

WEIGHTS

Gross Weight		10717 lbs.
Weight Empty		6967 lbs.
Structure and Fixed Equipment	4145	
Power Plant	2822	
Useful Lead—normal		3750 lbs.
Crew	600	
Fuel	1200	
Oil	105	

Equipment	102
Guns	276
Bombs (normal load)	1250
Bomb racks and controls	178
Pyrotechnics	39

PERFORMANCE—NORMAL FULL LOAD-1820-E-1 ENGINE

Maximum speed level—sea level—front cockpit covered	190 M.P.H.
Maximum speed level—8500 feet—front cockpit covered	201 M.P.H.
Maximum speed level—sea level—front cockpit open	186.4 M.P.H.
Maximum speed level—6000 feet—front cockpit open	197 M.P.H.
Maximum speed level—10,000 feet—front cockpit open	196.5 M.P.H.
Maximum speed level—15,000 feet—front cockpit open	190 M.P.H.
Take off speed—full load	76 M.P.H.
Take off speed—less bombs and ½ fuel	67.5 M.P.H.
Landing speed—full load	91.5 M.P.H.
Landing speed—less bombs and ½ fuel	83.0 M.P.H.
Rate of Climb—sea level to 6000 feet	1600 ft./min.
Time to 10,000 feet	7.8 min.
Climb in 10 minutes	12000 feet
Service Ceiling	20000 feet
Range—cruising speed (110 MPH) (200 gal.)	650 miles
Range—cruising speed (110 MPH) (300 gal.)	980 miles
Range—full speed (200 gal.)	370 miles
Range—full speed (300 gal.)	550 miles
Duration—cruising speed—200 gal.	5.9 hours
Duration—cruising speed—300 gal.	8.9 hours

GENERAL

Power Plant—2 Wright Cyclone R-1820-E-1 geared 1.58:1

Total power	1300 H.P. at 1950 R.P.M.
Airfoil	Tapered Gottingen 398
Aspect Ratio (S^2/A)	7.7

	Normal	Light	Overload	
Wing loading	21.2	17.36	22.4	lbs./sq.ft.
Power loading	8.24	6.74	8.70	lbs./H.P.
Useful Load	3750	1802	4403	lbs.

BOMB LOAD = 2500 LBS.

PERFORMANCE DATA

Engine—Cyclone R-1820-F 700 H.P. at 1900 R.P.M.

Wing Area increased to 600 square feet.

Gross Weight, lbs. (with 2500 lbs. of bombs)	12230
Wing area, sq. ft.	600
Span, ft.	70
Total Power H.P.	1400
Engine R.P.M.	1900
Propeller R.P.M.	1190
High speed, sea level M.P.H.	192
High speed, 8000 ft. (front cockpit enclosed by turret)	206
Stalling speed (less bombs and ½ fuel) M.P.H.	68
Rate of Climb (0 to 6000 ft.) ft./min.	1315
Time to 10,000 feet—min.	9.0
Time to 15,000 feet—min.	17.0
Service Ceiling—feet	21500
Wing leading—lbs/sq.ft.	20.3
Power loading—lbs./H.P.	8.7

BOMB LOAD = 1250 LBS.

COMPARATIVE PERFORMANCE

with

VARIOUS ENGINES AND INCREASED SPAN

Engines	Cyclone R-1820-E	9 cylinder R-1820-F	14 cylinder R-1535
Gross Weight	10717	10950	10950
Wing Area	505	600	600
Span—feet	62	70	70
Total power	1300	1400	1400

Engine R.P.M.	1950	1900	2500
Propeller R.P.M.	1235	1190	1670
High Speed—sea level	190	195	198
High Speed—8000 feet (front cockpit enclosed)	201	208	211
High Speed—8000 feet (front cockpit open)	197	—	—
Stalling Speed (less bombs and ½ fuel)	84	67.5	67.5
Rate of Climb— 0 to 6000 ft.—ft./min.	1600	1700	1750
Time to 10,000 feet—min.	7.8	7.0	6.7
Time to 15,000 feet—min.	15.5	13.5	13.0
Service Ceiling—feet	20000	23500	24000
Wing loading	21.2	18.25	18.25
Power loading	8.24	7.83	7.83

BASIC ARRANGEMENTS

The effect of summarizing all of the aerodynamic and structural improvements that have been developed during the last ten years may be somewhat surprising.

When Directive X-1670 for Bombardment Airplanes was issued to us for study a preliminary composition of the fundamental elements in modernized form was made, and the result indicated clearly that there would be no difficulty in exceeding the specified requirements.

The paramount importance of speed pointed out in the referenced Directive is readily appreciated, and was made a factor of first importance in the design program.

It is significant of new thought in bombing that the new Martin Bomber surpasses present standards of speed for pursuit, carrying three times pursuit useful load, but with only twice the power.

The limit of speed for bombing has by no means been reached, as will be apparent by comparison of performance data given above. The reduction of frontal area possible with 14 cylinder radial engines yields an increase of 12 miles per hour with only a minor increase of power and fuel consumption.

Still further increases of speed become immediately available with the mature

development of more powerful motors now in their experimental stages.

This unusual capacity for speed is the result of a complete analysis of the components of drag involved in bombing airplanes, and the results may be appreciated, without going into the mass of technical detail, by a simple comparison between the average drag characteristics of the series of bombers current during the past decade, and the drag characteristics of the new Martin Bomber. (Drag is expressed in the usual units of "flat plate" area.)

	Average of 6 Bombing Types	New Martin Bomber	Percent Improvement
Drag of wings	11.2	4.4	60.7%
Drag of Structure	35.6	13.6	61.8%
Total Parasite Drag	46.8	18.0	61.5%

The criterion of speed and cruising efficiency is the ratio of lift to drag, or briefly, the L/D. The new Martin Bomber has a maximum L/D = 11.4.

If the air resistance of the engines is deducted the L/D = 13.6, which compares with some of the best motor-less soaring planes that have achieved such excellent results in Germany. This fact is more remarkable in view of the relative strength factors; soaring wings being commonly designed to support about 20 lbs./sq.ft. of surface, while the Bomber wings have been static tested to 143 lbs./sq.ft.

As a result of its unusual aerodynamic finesses, the XB-907 is capable of extremely high speeds with the present available power. Power charts based on the present performance, with 650 H.P. engines show that the speed of 220 M.P.H. at 8000 feet can be readily obtained with 800 H.P. engines, and when 1000 H.P. engines become available a speed of 230 M.P.H. or more will be attainable.

A feature of XB-907 is its ready adaptability to a wide variety of engines. In its present condition it can mount either the Hornet 1860, the Cyclone 1820-E, the Cyclone 1820-F, or the twin Wasp 1535. For experimental testing the Cyclone 1820-E was installed, pending complete development of the 1820-F. The 1820-F engines will be installed ready for flight in the early Fall. Provision for other engines is made in the structure.

With the vastly increased speed there is a necessity to provide strength in proportion. The current specification of design load factors for bombing airplanes is 4.5 (Ref. page 171—Handbook of Instructions for Airplane Designers). With high speed and maneuverability in mind the XB-907 was built to factors of 8.5, (the same as observation planes) or nearly twice the strength of any other existing American Bombers.

Diving speeds of the modern high speed bomber are in excess of 250 M.P.H. and for pulling out from this speed the factor of 4.5 is inadequate and dangerous.

A twenty percent increase in the wing area of the new Martin Bomber is now being made to accommodate the 2500 pound load of bombs. This increased load can be carried with a load factor of more than 6.0 without change of the existing structure, due to the high strength provided in the basic structure in the original design.

The added wing area not only accommodates greater load but also provides a lower landing speed and greater climbing ability.

The climb of the XB-907 is excellent, and during flight trials at Wright Field it was officially found to be 1600 feet per minute with the original 1820-E engines. A substantial increase in climb is expected with the 1820-F engines.

Due to the elimination of external bracing and efficient planform of wing the additional area can be added with no material reduction of top speed. The drag of the added area at 200 M.P.H. is 96 pounds, and is equivalent to 51 H.P. The permanent engine installation (1820-F or R-1535) will provide a total of 100 H.P. more than the temporary 1820-E engines, which will amply provide for the slight increase in wing drag.

In addition to adaptability for various power plants and a complete range of load carrying capacity the new Martin Bomber has several predominating features which influence modern bombing.

Its relatively small size and high degree of maneuverability permit close squadron formation and concentration of protective fire. The size of squadron depends upon the mobility of the units. The lateral control of the new Bomber is very responsive to the touch and the ship can be wheeled through unusually small turning circles with a high degree of accuracy. All Air Corps pilots who have maneuvered the plane have remarked upon the accuracy of its turns, attesting the pronounced influences of scientific aileron proportions and proper balancing devices. In addition to aerodynamic balance of ailerons, all lateral moments are reduced to zero by special trailing edge vanes.

Particular attention is being paid to mass balance and a series of flight tests has been completed for the purpose of calibrating the dynamic action of the entire mass of the airplane in flight, permitting exact balance of lift and weight to reduce all pitching moments to zero.

Another basic feature commented upon by pilots is the steadiness of the plane upon its course in flight. This feature has been accomplished by extreme care in proportioning the dimensions of the fuselage and tail surfaces, and the components of directional stability have been made very positive for maintaining the desired course for sighting and releasing bombs with a minimum of effort.

The possibility which the new Martin Bomber offers for higher operating altitude will profoundly effect bombing practice. In conjunction with the C-4 sight, just recently perfected, bombing from 20,000 feet with heavy bombs is possible. It will be noticed from performance data (page 3) that the XB-907 is capable of a service ceiling of 23,000 feet with present engines and moderate supercharging, and there appears to be excellent prospect of realizing 25000 feet, which will put heavy bombing well beyond the reach of anti-aircraft fire and add materially to chances of arriving over large objectives undetected.

After discharging the bombs the XB-907 has all the speed and maneuverability of the finest two seat fighters. These qualities can be used to the utmost because of the high strength factors built into the airplane, adding qualities to its power of defense and ability to escape which are not present in the large, heavy bombing types.

The internal bomb racks are an important feature of the XB-907 for three reasons. The internal racks permit carrying the widest assortment of bombs from the smallest to the 1100 pound size, all inside the body. It eliminates the reduction of speed which is involved in any attempt to carry the same variety of bombs outside. It preserves the more delicate parts of the timing mechanism from mud and ice and particularly ice which forms in the cold higher altitudes. Other advantages of the internal bomb rack, such as facility for clearing jams in the releasing mechanism, or making readjustments during flight, are readily appreciated.

Possibilities for extending cruising radius of action are important. This basic feature has been incorporated in the new Martin Bomber by a unique arrangement made possible by the internal arrangement of bomb racks. The plane as now built has racks for 1250 pounds of bombs inside, and in addition racks for two heavy bombs (600 or 1100 lbs.) under the wings. By utilizing the upper portion of the internal rack to support a 250 gallon gas tank the fuel capacity can be increased to 550 gallons with 1200 to 1500 lbs. of bombs, and a cruising range of 1500 miles.

From the pilot's cockpit vision is afforded in all desired directions. It is impossible to approach the plane without being visible to the pilot. For formation flying his vision is entirely unobstructed.

Details of the bomber's cockpit have been worked out with extreme care. The bomber's operating position is very comfortable and he has easy access to all bombing controls. Each window in the front and sides of the bomber's compartment has been carefully determined to meet requirements, both before and during the sighting operation.

At speeds of 200 M.P.H. and over it is impossible to operate a gun from the conventional Scarf mount. Anticipating this difficulty the original nose design of the new Martin Bomber was so shaped and cowled that the front gunner received protection from the blast. However, the development of the larger type bomb sights (L and C type) required a greater depth of cockpit and the desired lines of the nose could be only partially maintained. The shelter afforded the front gunner was insufficient and research was begun upon the completely enclosed type of turret. This type has now been developed and will shortly be mounted on the XB-907 nose.

The turret consists of a "bee-hive" shaped enclosure of transparent, fire-resisting material upon an aluminum alloy skeleton frame. It is mounted on ball bearings and is easily rotated by hand, or locked in any position as desired. The gun carriage moves vertically in an open slot four inches wide, upon an accurately machined track built in the surface of the turret enclosure. The gun may be trained on any point in the entire forward hemisphere, and also about two thirds of the rear hemisphere. Practically the only limitation of fire is the physical parts of the airplane and propellers. The predominating merit of this turret is that it makes the gun actually usable in a convenient and practical manner when flying at high speed. It also provides complete protection for the bomber when he is operating his sight. The interior arrangement of the turret has been studied in a full size mockup, and so laid out that the bomber may transfer immediately from bombsight to gun, or the opposite without a second's loss of time. When seated at the bombsight he has only to raise his eyes to obtain full vision forward, up, to the side or to the rear.

The self-landing characteristics of the XB-907 have been generally remarked. From the pilot's viewpoint the approach is steady and the landing very easy and regular. All pilots who have landed the ship are unanimous in their description of the semi-automatic landing qualities of the plane. From the engineer's viewpoint these qualities have been obtained by careful analysis of the airplane's stability curves and arrangement of the wing planform to prevent premature stalling of either wing tip. Lateral control is present in positive quantity throughout the landing.

INTERIOR ARRANGEMENT

The design and arrangement of the body is an example of combined aerodynamic and structural efficiency.

The body structure is entirely new and it surpasses previous standards of weight/strength ratio by a considerable margin. During static tests it supported the greatest bomb load and loads equal to the breaking strength of the tail surfaces and the tail wheel, with scarcely measurable deflection. Its efficiency factor of weight to size is 1.11 compared to 1.33 for the best previous types. This remarkable gain is due to a novel type of construction known as the "restrained shell" monocoque. This type of body has no separate longitudinals. The full strength of the skin is derived by the use of a series of closely spaced tubular rings, and the backbone strength is contained in corrugated arches which run along the top and bottom of the body.

The XB-907 body lines were laid out after the manner of flying boat hull lines. Fluid air pressures at 220 M.P.H. approach the fluid pressures of water, corresponding in dynamic pressures, to water at a speed of about eight miles per hour. Advantage of this fact was taken in perfecting the airplane's stability characteristics and maneuvering qualities, and the body lines were proportioned accordingly.

In addition to the advantage of variety and protection of bombs on the internal racks, the body shape is ideal for comfortable and commodious radio and navigation installations. The radio compartment is a small enclosed room with ample space for all equipment to be arranged conveniently.

Excellent protection is provided for the upper rear gunner by the natural shape of the body and he is entirely out of the blast when training his gun through the entire upper hemisphere.

The lower hemisphere is covered by a really efficient floor gun mounting, and the upper slope of the underbody makes the floor gun particularly effective.

The shape of the body nose is ideal as a base for a gun mount, and it accommodates the turret enclosure without increase of drag, a feature which is impossible without suitable body lines.

Passage from both ends of the body to the bomb bay is permitted by the deep streamline shape of the body, and as a result access may readily be had in flight to all bomb gear, fuel valves, pumps, strainers, etc. The distribution of stresses in the body was examined in flight by structural engineers, and in carrying out this check-up various persons passed without difficulty from the rear cockpit, back to the stabilizer inside the body on several occasions.

In spite of its capacity for 2500 lbs. of bombs the XB-907 is small in overall dimensions. Although it is a monoplane the span, with increased wing area, is only 70 feet, five feet smaller than present service biplane bombers.

During flight tests the airplane was flown on one engine with full load. The particularly interesting part of this test was the powerful action of the rudder trailing edge vane. This vane is controlled from the cockpit at will. By its use the offset thrust of one-engine flight is completely balanced. In fact, there is no difficulty in making turns contrary to the live engine and this was done repeatedly during tests. This vane has a very fine adjustment in the cockpit and it can be used to compensate for drift in a cross-wind to a very exact degree.

The landing wheel retracting gear of the new Martin Bomber has been entirely successful from the beginning. A ratchet arrangement is provided on the operating handle to increase the pilot's mechanical advantage. The gear operates so easily that the ratchet is seldom used. The punishing tests given the landing gear during trials showed no effect whatever upon the landing gear. The operating arrangement in the cockpit has no complications of any kind as all the parts operate automatically and the pilot has only to set the operating handle for "up" or for "down," as desired. No wrong motions of the operator can jam or damage the gear in any way, and it has proven to be a highly successful device.

The wing construction is extremely strong. It consists of a backbone of corrugated dural of an average thickness of 1/16 inch, with substantial wing beams in the usual location. The form of the wing is maintained by substantial formers spaced rather widely. The entire wing assembly is simple and of few parts. The conventional multiplicity of small pieces is avoided. The bottom of the wing is screwed on and can be removed as desired, exposing the entire interior for inspection or repair.

The elimination of fuel tanks from the body avoids the unpleasant fumes of gasoline in the body and materially reduces the possibility of fire.

UTILITY OF THE BOMBER

The Martin ship is unique in its variety of possible uses. It was originally designed to be a fast day bomber. It not only exceeds the requirements for fast day bombing, but it is also able to meet all the requirements for heavy bombardment except the 60 M.P.H. landing speed, and in this characteristic the landing qualities of the ship are so far in advance of normal that the pilots' impression of the landing is very satisfactory. It will carry the normal heavy bombardment load of bombs (2000 lbs.) and with the 20% addition to wing area now being made it will carry the 2500 lb. bomb load and take off at less than 70 M.P.H. without difficulty. The Martin ship exceeds the high speed required for heavy bombardment by some 50 miles per hour, and the ceiling requirements by several thousand feet.

Furthermore, without bombs it has a maneuverability comparable with the two seat fighters.

The XB-907 has performance as an observation type far superior to requirements or existing types. Its large and roomy body affords excellent facilities for observation equipment and the complete vision which all members of the crew have is a decided asset.

Although no attempt was made in the design to develop characteristics suitable for a ground attack plane, it happens that the Martin ship fulfills the ideal attack requirements to a high degree. Its speed and maneuverability would be invaluable for this class of airplane. Its present guns can be trained to front or rear towards the ground, particularly the floor gun, and the mounting of additional guns can be readily accomplished.

The XB-907 with its internal bomb rack can carry all the smaller sizes of bombs. The use of this airplane as a raiding type has been suggested, carrying ten to twenty of the smaller bombs, up to the 100 pound size, making it a formidable weapon against troop trains, small bridges and similar objectives. Its speed and mobility will make it highly valuable for this work.

The excellent flying qualities, high performance and simplified construction of this airplane have resulted in a fundamentally new type of airplane which greatly surpasses existing airplanes and specification requirements in several classes. Its adaptability to existing and future engines speaks well for its further development. It brings into actuality performance and abilities for military use which have never existed before.

Photograph No. 7489 herewith shows the excellent accommodations afforded for radio with proper and adequate spacing of each item of equipment. A full size folding navigator's table is provided and the radio room is lighted for night navigation operations.

Pursuit escort will not be necessary for the XB-907 because the forward gun is made completely effective by the turret enclosure described on page 7. In combination with the upper rear gun and the lower rear gun this provides complete protection, even if the bomber becomes separated from its squadron. This eliminates the necessity of limiting the radius of action of long distance bombers because of the short range of the pursuit escort.

Document 3-20(b), Glenn L. Martin to Walter G. Kilner, undated (ca. 1933).

Maj. Walter G. Kilner,
Office of Chief of Air Corps,
Washington, D.C.

Dear Maj. Kilner:

Complying with your telephone request we are pleased to summarize the outstanding features of the New Martin Bomber recently submitted to and successfully tested by the Air Corps, and to review briefly this accomplishment as an event of historical significance.

On August 22, 1930 this design amongst others was presented to the Material Division at Wright Field and we were advised that if we felt confident of results the Engineering Division would give endorsement to the development of the airplane.

When the fully developed airplane was delivered to Dayton on October 7, 1932 its performance so far exceeded expectations that current directives, of not only bombing airplanes, but all other combat types as well, were forthwith discarded as obsolete.

For the first time in the history of military aeronautics the bombing airplane became independent of pursuit protection or attack.

The new Martin Bomber is the first twin engined airplane and the first bomber of the world to surpass the 200 M.P.H. speed mark. It is the fastest airplane of its size in existence.

This airplane places at the disposal of the United States Government a weapon of military strength unequalled anywhere else.

The brilliant demonstration of the new bomber in its acceptance trials opened a whole new picture of the tremendous potentialities of aircraft in military operations, and thereby brought aeronautics a long step nearer to its ultimate place in world affairs.

Not only in military operations, but in commercial transportation as well, the superlative benefits of this design are appreciated and in the space of a few months, already imitated.

The new Martin Bomber has broken away from conventional practice in almost every particular whether it be structures, aerodynamics, control, propulsive efficiency execution of detail.

Its wing construction for the first time rationalizes variations of applied air pressures with supporting strength, and for the first time a wing construction of this type has been proven in flight.

The construction of the body is one of those inventions that inspires the old comment—"Why wasn't it thought of before?"

The retractable lauding gear of the new Bomber functions in its perfected simplicity with the monotonous regularity of any common event.

The shake-proof cantilever tail surfaces are more readily recognized as an innovation. There is also something new in a twin engined airplane which flies "hands and feet off" with only one engine, as the new Bomber does for extended periods. This all important increase in safety is a vital contribution to military and commercial aeronautics alike.

The propulsive efficiency of the nacelle-propeller-wing combination may be appreciated in full contrast from the simple fact of propeller efficiency obtained at 88%, a degree of efficiency hitherto reserved for Schneider Cup Racers.

Air Corps officers have stated that the New Martin Bomber is the best designed and built airplane which ever came to Wright Field.

This Company has been and is continually engaged in the research for better aeronautics. During the years since the Old Martin Bomber we have steadily accumulated improved aerodynamic and structural elements.

The creation of such new elements opens the way to progress, but it is also conceivable and frequently demonstrated that a few such elements do not insure successful aircraft, nor does the unskilled aggregation of miscellaneous innovations produce lasting benefits.

The New Martin Bomber is proportionally advanced in all its elements. Its parts have such mutuality of improvement that the complete composition is more than a list of detail inventions. It has the power and perfection that comes from the blending of related parts, each perfect in itself—in a word, unity.

Document 3-20(c), Glenn L. Martin Company, "History and Development of the Martin Bomber," 20 January 1938.

HISTORY AND DEVELOPMENT OF THE MARTIN BOMBER
January 20, 1938
The Glenn L. Martin Company
Baltimore, Maryland
U.S.A.

Up to the year 1932 the Army's bombardment equipment consisted of large twin engined biplanes having a high speed of 114 m.p.h.

In 1932 the Martin Company re-entered the bomber field and submitted an airplane to the Army which so far surpassed the Army's equipment at that time, that the Martin Company was given the 1932 award of the Collier Trophy in recognition of this outstanding contribution to aeronautical science. This airplane (See Figures 1 and 2), which was originally known as the Model XB-907, incorporated new military design features which up to that time had never been used. Many basic aerodynamic and structural features were also incorporated in this model which made it the forerunner of the modern high speed commercial transports.

This model so far surpassed any previous airplanes of its type that it was found necessary by military authorities to change entirely military tactics as far as aerial warfare was concerned. Fundamentally, it was so far in advance of its day that since then major basic changes have not been required, however, the Martin Company has continuously incorporated improvements in structure, aerodynamics, power plant and equipment. Because of this policy, the airplane today is still outstanding in performance, striking power, and general utility. Added to this, the incorporation of modifications as a result of operating experience has made it one of the simplest airplanes to maintain in service. The general improvement may be noted by comparing Figure 1 to Figure 10.

The following outstanding flights have been made with this airplane besides the constant, day in and day out, routine flying.

(a) On August 24, 1935, Major General Frank M. Andrews, commanding officer of the G.H.Q. Air Force, accompanied by two assistants, flew his Hornet powered Martin B-12 bomber with Edo floats on a 2000 kilometer non-stop flight at 266 km. per hour. The course was laid from Hampton Roads, Va. to New York and return by way of Washington, D.C. Flying at an altitude of 3040 meters or more, General Andrews completed the second lap of 1000 kilometers in three hours 45 minutes and 13 seconds, and carried a payload of 1000 kg. made up of two big aerial bombs. At the speed of 266 km./hr. General Andrews broke three world seaplane speed records, for 1000 kilometers, one without payload, the second with payload of 500 kilograms and the third with payload of 1000 kilograms. (See Figure 4).

(b) A squadron of ten airplanes flew from Washington, D.C. to Alaska and returned. This flight covered a total of 12,000 km. While making this mass maneuver, the only incident of the flight was a forced landing caused by a piloting error. It was necessary to land in the water due to the excessively rough terrain in that section of the country. A water landing was made without any difficulty. The airplane was beached, repaired, and in two days continued the flight with the rest of the squadron. This particular incident indicates the ability of this airplane to meet all conditions. A notable record established as part of this flight was the photographing of 20,000 square miles of unexplored territory in three days.

(c) One of the later models was flown from the manufacturer's plant in Baltimore, Md., down through Central America, the west coast of South America, and across the Andes to Buenos Aires, a total distance of 13,000 km. without difficulty. The airplane on this flight carried a crew of four men, their personal equipment, and a full load of miscellaneous equipment.

(d) Recently a squadron of nine ships of this model was flown a total distance of over 13,000 km. on a tour over the Dutch East Indies. This flight, which included several long over water flights, was completed without any maintenance of any sort except refueling and the addition of engine oil.

The following is a chronological list of the outstanding features incorporated in the original Model XB-907 and added in succeeding models. The tabulated list enclosed gives more detailed information.

(1) Due to the high speed of this model it was found impossible to operate a flexible gun without wind protection for the gun and gunner. This was the first airplane in the United States to incorporate a gun turret which has since been incorporated on most military aircraft of its type. (See Figure 3).

(2) In order to keep the airplane aerodynamically clean and to protect the vital bomb rack mechanism from dirt, this airplane was the first one to incorporate bomb bay doors which completely enclose the bomb bay.

(3) In order to further increase the aerodynamic efficiency and to give added protection and comfort to the crew, enclosures were incorporated for each cockpit opening. These enclosures were designed to give easy access, to the cockpit and from the cockpit, and also give the rear gunner ample wind protection when operating the gun. (See Figure 6). The enclosure on the most recent model has been redesigned as one continuous structure covering both front and rear cockpits, with special arrangement so that the radio compass loop may be installed within it.

(4) The wings and horizontal tail surfaces were designed with water tight compartments to enable the airplane to remain afloat in case of an emergency water landing. This type of flotation requires no servicing and does not deteriorate or depend upon mechanisms for its operation, the floats being ready at all times without any action on the part of the pilot.

(5) Full cantilever wings and tail surfaces were incorporated which, while being aerodynamically efficient, contribute to simple maintenance due to the elimination of constant rigging and alignment which was necessary with previous airplanes. The wing construction is of a type developed through the cooperation of the Martin Company and Army engineers, which structurally has proven very sound from a strength, weight, stiffness, and manufacturing point of view. This type of wing construction has since been incorporated on Martin airplanes up to 62,000 lbs. gross weight with the same efficiency, thus demonstrating its basic soundness.

(6) The landing gear, which at the time of its inception, was an outstanding development for airplanes of its size, is cantilevered for side loads which tends to simplicity and cleanliness. The original main strut of the landing gear was of a twin oleo type later changed to a half fork to facilitate the changing of tires. It is of interest to note that the landing gear on this airplane, including the retracting mechanism, was of equal weight to the landing gear on previous bombers of the same gross weight which were not retractable nor cantilever.

(7) The development of the present power plant installation from the original has been the result of continuous testing and research. One of the most striking examples of this development is the engine cowling as may be

seen by comparing Figures 1 and 10. The latest design has been commented on by the engine manufacturer as being the most efficient type for properly cooling and regulating the temperatures of the most modern type of high horsepower engine. This cowling has also permitted the realization of a substantial increase in high speed.

(8) On the B-10B and succeeding models, wing flaps were incorporated to allow for increased gross weight without increasing the landing speed of the airplane. (See Figure 5). The prototype model was originally designed for 10,580 lbs. gross weight with an area of 550 sq. ft. which was changed to 682 sq. ft. on the succeeding airplanes. Since then the airplane's normal gross weight has been increased to 15,400 lbs. gross weight without affecting its flying characteristics. The increase in gross weight has been due to increase in fuel capacity and additional equipment which has recently been developed. To provide for this increase in gross weight the Martin Company has successively reanalyzed the structure of the airplane and incorporated changes in it to maintain full strength requirements. The most recent structural analysis has included the latest specification of the U.S. Army Air Corps on gust conditions combined with a diving speed of 300 m.p.h.

(9) On Model 139-WH2 and succeeding models, the rear section of the wings, that is from the rear spar to the trailing edge, which was fabric covered on preceding models is metal covered. This was done in the interest of maintenance as it eliminated the necessity of putting the complete airplane in the shop for replacing the fabric. The only fabric on the airplane at the present time is on the movable control surfaces which may be easily replaced with the spare surfaces when in need of recovering. This allows the airplane to be kept in service at all times.

(10) Many items of equipment have been incorporated in the later models, namely:

Gyropilots.
Constant speed propellers.
Deicers.
Abrasion shoes for horizontal stabilizer.
Supercharger regulators.
Latest types of radio and radio compasses.
Aerial camera.
Oxygen equipment.
Engine fire extinguisher.

Document 3-21

Jack Frye to Douglas Aircraft Corporation, "Attention: Mr. Donald Douglas," 2 August 1932. A facsimile of this letter appeared in *American Heritage of Invention and Technology*, Fall 1988.

One important link that cannot be overlooked within the context of the airplane design revolution of the interwar period—or airplane design at any time, for that matter—is the link between technical innovation and economic incentive. In the austere financial conditions brought on by the Great Depression, competition between the airlines, and between the aircraft manufacturers, grew extremely keen. This tight situation can be seen clearly in the document below; in it, Jack Frye, vice-president for operations for Transcontinental and Western Airlines (TWA), acknowledged in the summer of 1932 that, for his airline to survive, it needed to adopt a popular new transport airplane. TWA's aging fleet of Fokker trimotors were becoming more and more suspect in the eyes of the public and government regulators ever since the fatal 1931 crash of a TWA airliner carrying famous Notre Dame football coach Knute Rockne. Frye desperately wanted a new design like the Boeing 247, but the corporate connection between Boeing and United Air Lines blocked TWA from buying any of the new aircraft.

Document 3-21, Jack Frye to Douglas Aircraft Corporation, "Attention: Mr. Donald Douglas," 2 August 1932. A facsimile of this letter appeared in* American Heritage of Invention and Technology*, Fall 1988.

TRANSCONTINENTAL & WESTERN AIR INC.
10 RICHARDS ROAD
MUNICIPAL AIRPORT
KANSAS CITY, MISSOURI

August 2nd, 1932

Douglas Aircraft Corporation,
Clover Field
Santa Monica, California.

Attention: Mr. Donald Douglas

Dear Mr. Douglas:

Transcontinental & Western Air is interested in purchasing ten or more trimotored planes. I am attaching our general performance specifications, covering this equipment and would appreciate your advising whether your Company is interested in this manufacturing job.

If so, approximately how long would it take to turn out the first plane for service tests?

Very truly yours,

Jack Frye
Vice President
In Charge of Operations

JF/GS
Encl.

N.B. Please consider this information confidential and return specifications if you are not interested.

TRANSCONTINENTAL & WESTERN AIR, INC.

General Performance Specifications Transport Plane

1. Type: All metal trimotored monoplane preferred but combination structure or biplane would be considered. Main internal structure must be metal.

2. Power: Three engines of 500 to 550 h.p. (Wasps with 10-1 supercharger; 6-1 compression O.K.).

3. Weight: Gross (maximum) 14,200 lbs.

4. Weight allowance for radio and wing mail bins 350 lbs.

5. Weight allowance must also be made for complete instruments, night flying equipment, fuel capacity for cruising range of 1060 miles at 150 m.p.h., crew of two, at least 12 passengers with comfortable seats and ample room, and the usual miscellaneous equipment carried on a passenger plane of this type. Payload should be at least 2,300 lbs. with full equipment and fuel for maximum range.

6. Performance

Top speed sea level (minimum)	185 m.p.h.
Cruising speed sea level—79% top speed	146 m.p.h. plus
Landing speed not more than	65 m.p.h.
Rate of climb sea level (minimum)	1200 ft. p.m.
Service ceiling (minimum)	21000 ft.
Service ceiling any two engines	10000 ft.

This plane, fully loaded, must make satisfactory take-offs under good control at any TWA airport on any combination of two engines.

Kansas City, Missouri.
August 2nd, 1932

Document 3-22(a-b)

(a) Clark B. Millikan and Arthur L. Klein, "Report on Wind Tunnel Test on a Model of the Douglas Transport DC-1 with Various Modifications," GALCIT Report 119, 7 June 1933, GALCIT Wind Tunnel Test Reports, Boeing Company Historical Archives, Long Beach, California.

(b) Engineering Department, Douglas Aircraft Company, "Development of the Douglas Transport," Technical Data Report SW-157A, undated (ca. 1933-34), Folder AD-761184-05, Aircraft Technical Files, National Air and Space Museum, Washington, D.C.

This duo of documents reveals the unprecedented extent to which the results of fundamental wind tunnel data went into the design of the DC-1, information that soon after would be expanded into its later incarnations, the DC-2 and DC-3. Bailey Oswald, the Douglas Company's first aerodynamicist, arranged for wind tunnel tests of the DC series at the Guggenheim Aeronautical Laboratory at the California Institute of Technology, his alma mater. GALCIT's new 200-mph tunnel enabled a more direct approach to design, and Douglas's direct use of the tunnel helped bring the discipline of aerodynamics out into the open for manufacturers to exploit by showing how successful such an experimental approach could be. Douglas's wind tunnel tests of the DC-1 model resulted in the refinement of several important design elements. The inclusion of an aft sloping windscreen, an NACA cowl, retractable landing gear, cantilever monoplane wing, tail surfaces, and streamline fuselage all proved to be aerodynamically sound. In key respects, Oswald's test program set an important precedent; it was soon almost commonplace to be using wind tunnel testing in aircraft design. No longer would there be such dependence on intuition or on theoretical calculations alone. Systematic physical tests would be required for nearly all proposed designs. This proved to be one of the most critical, and lasting, contributions of the process of reinventing the airplane.

Document 3-22(a), Clark B. Millikan and Arthur L. Klein, "Report on Wind Tunnel Test on a Model of the Douglas Transport DC-1 with Various Modifications,"

GUGGENHEIM AERONAUTICS LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA
REPORT ON

WIND TUNNEL TESTS ON
A MODEL OF THE DOUGLAS TRANSPORT DC-1 WITH
VARIOUS MODIFICATIONS

PREPARED BY:

C. B. Millikan

A. L. Klein

Date: June 7, 1933

I. Introduction, General Description of Model and Tests.

This report describes the results of wind tunnel tests on a 1/11th scale model of the twin-engined Douglas Transport airplane DC-1, and also the results of several auxiliary wind tunnel tests on models of component parts of the airplane. The experiments were made in the closed working section of the 10 degree wind tunnel of the GALCIT (Guggenheim Aeronautics Laboratory at the California Institute of Technology)*. Practically all of the tests were made at a wind speed of about 185 m.p.h., corresponding to a Reynolds' Number based on mean wing chord of approximately 1,700,000. A few measurements were made at lower speeds to investigate scale effect, and certain pressure distributions were also run at lower speeds. The critical Reynolds' Number at which a 27 cm. sphere has a drag coefficient of 0.3 is about 330,000 indicating a wind stream with very low turbulence. The entire model was lacquered and rubbed down to a high polish.

During the course of the investigation many modifications were made. Table 1 gives the notation employed throughout the report in designating the various elements and configurations.

*cf. Clark B. Millikan and A. L. Klein: "Description and Calibration of 10-foot Wind Tunnel at California Institute of Technology". Trans. of A.S.M.E., Aeronautical Engineering, 1932-33.

TABLE 1

Notation Used to Describe Configurations Tested

Symbols		Configurations
W0	=	Original wing
Wt	=	Original wing with large tips
W1	=	Intermediate wing
W2	=	Final wing with straight trailing edge
F	=	Fuselage
N, N1,…N8, etc.	=	Various nacelles with wing-nacelle fillets
C	=	Chassis retracted
Cd	=	Chassis down in position for landing
T	=	Tail wheel
H1	=	Original horizontal tail surface (small area)
H2	=	Final horizontal tail surfaces (large area)
V1	=	Original vertical tail surfaces
V2	=	Final vertical tail surfaces
w1	=	Original auxiliary airfoil (large chord)
w2	=	Final auxiliary airfoil (small chord)
A	=	Free air ailerons
S1	=	FV1N8CTH1 = Original Standard configuration less wing
S2	=	FV2N8CTH2 = Final standard configuration less wing

S1, S2 are always associated with one of the wings, e.g. S1Wt, S2W2, etc. and unless otherwise specified correspond to stabilizer, elevator, and flettner all clamped at 0° setting.

S2W2 = S2W2w2 = S2W2 with auxiliary airfoil w2 set at –5°.

TABLE 2

Data on Various Surfaces

	W(sub 0)	W(sub t)	W(sub 1)	W(sub 2)	H(sub 1)	H(sub 2)	V(sub 1)	V(sub 2)	w(sub 1)	w(sub 2)
Area, S(ft.^2)	910	944	910	939	120	145.6	53.2	71.4	17.01	11.01
Span, b (ft.)	79	85	85	85	23.9	25.83	10.3	11.17	4.131	4.131
Aspect Ratio, AR	6.85	7.66	7.94	7.7	4.75	4.58	1.28	1.75	-	-
M.A.C., t(sub ft.)	12.22	12.06	11.71	11.86	-	-	-	-	-	-
Root Chord (ft.)	14.86	14.26	14.26	14.26	-	-	-	-	-	-
Rudder Elevator Area (ft.^2) (Total)	-	-	-	-	46	48	31.7	42.5	-	-
Flettner Area (ft.^2) (Total)	-	-	-	-	7.5	7.72	4.1	6.58	-	-

All dimensions correspond to full scale. The model was 1/11th full scale.

The wings were all similar in tapering from a 2215 root section to a 2209 tip section. They differed chiefly in area and plan form. The basic data for the four wings, the two horizontal surfaces, the two auxiliary wings, and the vertical surfaces, are given in Table 2 below. Dimensions are for the full scale airplane. Wing areas are gross areas with no allowance for the portion of the wing covered by the fuselage, while tail surface areas do not include any area in the fuselage. Mean aerodynamics chords were calculated by the Douglas Company. *SEE TABLE 2

A three view of the airplane in its original forms (S1W0, S1Wt, S1W1) is given in Fig. 1, and one in its final form (S2W2) is given in Fig. 2. Photographs of the models and of various details are given in Photos 1-8. The design data furnished by the Douglas Company and used in the performance and stability estimates are given in Table 3.

TABLE 3

W = Gross Weight = 17,000 lbs.

Po = Design Maximum Brake Horsepower = 2 X 712 = 1424 H.P.

No = Propeller R.P.M. corresponding to Po = 1340 R.P.M.

Assumed C.G. positions to which moments are referred are given in Figs. 1,2. If a is the distance of the C.G. behind the leading edge of the M.A.C.

a/t = 24.135 % for wing Wt

a/t = 20.0 % for wing W2

Note: The C.G. position was assumed to be the same for wings Wo, W1, and Wt.

1 X Tail Length = distance from elevator hinge to C.G. = 40.62° for S1

36.62° for S2

For details of the model and its various parts reference must be had to the following Douglas Co. drawings: 130150, 131198, 131255, 132309, 230678, 233258, 430679, 529827, 529828, 529973, 529974, 530113, 530148, 530234, 530325, 530447, 530448, 531062, 531371, 531404, 531949, 532086, 533596.

In particular drawings #430679 and 533596 give dimensioned three views of S1Wo and S2W2w2 with references to detailed drawings.

The tests were divided into the following broad groups:

1) Three component measurements on wings alone, including scale effect.
2) Effects of fuselage, nacelles, chassis, vertical and horizontal tail surfaces, and windshields; chiefly on lift and drag.
3) Effects of auxiliary airfoils.
4) Longitudinal stability and control.
5) Lateral (including directional) control.
6) Effects of bottom surface flaps.
7) Pressure distribution on an N.A.C.A. cowl.

Since nearly two hundred experimental Runs were made, no detailed list of the Runs will be included, but the results will be discussed in terms of the above grouping in section III.

II. Method of Making Tests and Calculations, and of Presenting Results: Notation.

The normal experimental setup is indicated schematically in Fig. 3. Unfortunately, at the time the tests were made, only five balances were available. Hence for lift, drag, and pitching moment investigations the two yaw balances were not used. For rolling and yawing moment tests the drag balance and one lift balances were moved into the positions of the two yaw balances. For this reason lift and drag could not be measured simultaneously with rolling and yawing moments. (This point it further discussed in III, 5 below).

The tare drag of the pyramid wire systems running to the wing trunnions (cf. Photo 2) was known from previous GALCIT investigations. The tare drag and moment of the sting, tail wire, and counterweight wire were determined by a) testing a wing alone with an auxiliary sting attached, and b) completely enclosing the tail and counterweight wires in streamlined windshields. The tare drag at the high speed attitude of the airplane was about 65% of the total parasite drag at this attitude. The tare moment was extremely small at all angles of attack.

All drags, angles of attack, and pitching moments were corrected by the Prandtl theory of tunnel wall interference to give free air conditions. Rolling and yawing moments and side force were uncorrected for the effect of wall interference.

All observations were reduced to the standard American system of absolute units (the notation for rolling moment, yawing moment, and side force is different from that recommended by the N.A.C.A.)

CL = Lift CD = Drag CM = Stalling

Moment

$\rho/2(V2s)$ $\rho/2(V2s)$ $\rho/2(V2s)$ t

Cs = Side Force CR = Rolling Moment CY = Yawing Moment

$\rho/2(V2s)$ $\rho/2(V2s)$ b $\rho/2(V2s)$ b

where

ρ = mass density of air (note: a correction was applied to the experimental observations so that in this formula ρ is to be taken as the free air density uncorrected for compressibility effects, at least up to 200 m.p.h.)

V = velocity
S = total wing area (See Table 2)
t = mean aerodynamic chord (See Table 2)

The conventions as signs are the same as those used by the N.A.C.A. and are as follows: Taking directions as the pilot sees them CM is positive when it tends to raise the nose, CS is positive when it corresponds to a force to the right, Cr is positive when it tends to lower the right wing, Cy is positive when it tends to move the right wing back. All control surface angles are positive when they tend to increase the lift (or side force) on the surface in question.

In certain cases the parasite drag coefficient CDp was determined (for the wing alone CDp = CDo = profile drag coefficient). For the wing alone the angle of attack for infinite aspect ratio was used, αo. The formulae employed in obtaining these quantities were:

$$\text{CDp (or CDo)} = \text{CD} - \frac{\text{CL2}}{\pi \text{AR}}$$

$$\alpha\text{o} = \alpha - \frac{\text{CL}}{\pi \text{AR}} \bullet 57.3 \text{ (degrees)}$$

where AR = aspect ratio = (span)2/area. It will be noticed that the lift distribution was assumed to be elliptical.

Unless otherwise specified pitching moments are referred to the appropriate C.G. position, cf. Figs. 1, 2. (Original Figures and Drawings not included herein) For wings Wo,W1,Wt the C.G. was assumed to be that of fuselage S1 shown in Fig. 1. For wing W2 fuselage S2 (Fig. 2) was used. In certain pitching moments are given with respect to a point 25% or 22% to the rear of the leading edge relative to the M.A.C., in which cases the figures are labeled so as to cause no confusion. The angle of attack is referred to the thrust axis throughout, even for the wing alone observations.

In making performance estimates and in comparing the effects of various modifications the equivalent parasite area has been used, where

f = equivalent parasite area = parasite drag = CDp S
$\rho/2(V2)$

Certain additional symbols are used in Section III which are there defined.

It should be mentioned that the plotted experimental points represent direct observations with no fairing, except that the tare drag results were faired before being subtracted from the observed total drags to give the final values.

III. Experimental Results and Discussion.

In view of the tremendous number of individual observations all results are given in the form of plotted experimental points and faired curves, practically no tabular data being given in the Report. Such detailed tabular data are available in the files of the GALCIT. The results are discussed below in accordance with the grouping previously mentioned.

1) Three Component Measurements on Wings Alone (Figs. 4, 5)

The results for the four wings reduced to infinite aspect ratio conditions are plotted in Fig. 4. CLmax for wing W1 is 4 or 5 percent lower than that for the other three which attain the quite normal values (for the GALCIT tunnel) of between 1.30 and 1.32. The rather low value of CDo for wing Wt is not especially surprising when it is considered that Wt was obtained from Wo by adding on large tips of comparatively thin section. The rather curious shape of the CDo curve between CL ~ 0.3 and 1.0 for wing W2 seemed a little suspicious when the results were finally plotted. Hence it was decided to make the series of Runs including 192 as a check on the earlier series including 152. Run 192 was taken ten weeks after Run 152, during which time the balance system had been completely shifted twice and the model completely disassembled and reassembled. The agreement of the two runs as indicated by the two sets of experimental points in Fig. 4 given an idea of the accuracy and reproducibility of the experiments. The value of CDomin = 0.0105 compares very satisfactorily with that obtained at the GALCIT and elsewhere with similar tapered wings. The pitching moment curves are unusual in two respects. First, all four curves indicate that CM is approximately constant in the normal flying range about a point between 1½ and 3 percent forward of the 25% point on the calculated M.A.C. Second, both of the more highly tapered wings W1 and W2 give a considerable and unusual instability just before the stall. With wing W2 this is especially pronounced. The reason for this anomalous behavior is not yet clear. It might be mentioned that a subsequent test at the GALCIT on a tapered wing with considerably more sweepback than W2 gave the same type of result but in a still more exaggerated degree. Whether the effect is due primarily to the large taper or to the sweepback cannot yet be stated.

As indicated by the Run numbers only wings Wo, Wt, and W1 were at first tested. From the results with them it was decided by the Douglas Co. to use wing Wt for the airplane, so that the subsequent Runs up to 150 were made with this wing. On the basis of the stability results discussed in section 4) it was later decided to change to wing W2 with which the subsequent tests were made.

In Fig. 5 infinite aspect ratio characteristics of wing W1 are plotted for three Reynolds' Numbers, in order to give an idea of the scale effect. The results at the lowest Reynolds' Number are considerably less exact than those at the higher velocities as is evidenced by the different amounts of scatter in the corresponding experimental points. Probably the most interesting features, the variation of CLmax and CDomin, are discussed in connection with other results of the same nature in section 8) below. In view of the similarity of the other wings tested to W1 it is probable that the scale effect for them all would be very similar to that shown for W1 in Fig. 5.

2) Effects of Various Modifications; Chiefly on Parasite Drag and CLmax

The first characteristic to be investigated in this category was the fuselage-wing interference. From Fig. 6 it appears that the addition of the fuselage and vertical tail surfaces to wing Wt causes an increase in CDp which is practically uniform over the entire flying range. Also CLmax for the combination is almost identical with that for the wing alone. Hence it appeared that the original fuselage-wing fillet was satisfactory in that there were no unfavorable interference effects. This conclusion was later verified in connection with wing W2 as is indicated in Fig. 9. Unfortunately in the latter case the horizontal tail surfaces could not be removed from the fuselage so that the additional drag and lift shown in Fig. 9 are due to the combination of fuselage interference and horizontal tail surfaces. Since the fuselage-wing fillet appeared to be so satisfactory no further modifications of it were investigated. A series of tests on wing Wt with fuselage and vertical surfaces at various Reynolds' Numbers gave results which are reproduced in Fig. 7. The scale effect is very similar to that given in Fig. 5 for wing W1 alone.

The wing engine nacelles were then mounted on the wing and the additional drag and interference effects noted. The original nacelles (N) gave rise to a considerable decrease in CLmax, to a fairly high ΔCDpmin and to a rather large decrease in effective aspect ratio (e). The first difficulty was ameliorated by the addition of fillets, but the other two did not respond to such simple treatment. A rather elaborate series of modifications was therefore investigated, including changes in cowling, nacelle shape and size, fillets, etc. The most important results are given in Fig. 8. The large variations caused by the various modifications are quite striking, as is the curve for NS corresponding to a case in which the engine and cowling were removed and the nacelle completed with a streamlined nose. It is, of course, possible that with power on the normal nacelle configurations might give curves approximating more closely to that for the streamlined nacelle. As a result of this series of tests it was decided that N4 was the best nacelle investigated and a new wooden nacelle, N7, was copied from it. The final nacelle, N8, which was used for all subsequent

tests was identical with N7 except for a small fairing at the wing intersection. The effect of the addition of the final nacelle N8 to wings Wt and W2 is almost identical as may be seen by comparing the curves of Figs. 6 and 9.

The parasite drag coefficients for FV1Wt and FV2W2 with various modifications are plotted in Fig. 10. In Fig. 11 are given the results obtained by replacing the normal wind-shield by a Fokker type (as used on the Boeing 247) and then by eliminating the wind-shield entirely and giving the fuselage a smooth, completely streamlined nose. From these two Figures the increase in parasite drag coefficient caused by adding various parts was determined and plotted in Fig. 12. The extremely small drag added by the horizontal tail surfaces H1, the tail wheel T, the chassis C, and the two types of windshield are noteworthy, as is the tremendous drag of the chassis in the unretracted position Cd.

The final parasite drags are discussed further in section 8 below.

3) Effect of Auxiliary Airfoils

In view of the beneficial influence which the addition of small auxiliary airfoils between the nacelles of the Douglas Dolphin had previously been found to exert, similar investigations were conducted on the present model. Two sets of auxiliary wings running between nacelles and fuselage were tested. Of these w1 had a large chord and w2 a smaller one. Both airfoils were tested at a series of angles of incidence to the fuselage axis.

The results for S1Wt with airfoil w1 (cf. Photos 4 and 7) are given in Figs. 13, 14. Referring to Fig. 13 it appears that for S1Wt the flow near CLmax is unstable so that the wing stalls prematurely and the value of CLmax is rather low. The addition of the auxiliary airfoil apparently stabilizes this flow condition and leads to considerably higher values of CLmax with only a very slight increase in CDp. It appears that a setting of about -5° is approximately the optimum. The influence on pitching moment is noticeable as is indicated in Fig. 14. The auxiliary airfoil decreases the static longitudinal stability perceptibly but not violently.

The model S2W2 was investigated with both auxiliary airfoils, the results being indicated in Fig. 15. The increase in CLmax for practicable settings of the auxiliary airfoil is not nearly so striking as was the case with S1Wt. This is largely because of the fact that at the stall the flow with model S2W2 is more stable than with S1Wt so that CLmax without any auxiliary airfoil is considerably higher in the former case. It appears that the smaller airfoil w2 is perceptibly better than the larger w1 so far as both CLmax and CDpmin are concerned. Again a setting of -5° seems to be about the best possible. From these tests it was decided to build the airplane using airfoil w2 set at approximately -5°. All subsequent tests demoted by S2W2' were made with this configuration.

4) Longitudinal Stability and Control

In view of the fact that the observed longitudinal stability of S1Wt (to be discussed below) was less than the designer had calculated, the moment coefficient for wing Wt alone was calculated about three axes. The results are given in Fig. 16 and indicate that CM is approximately constant about the 22% point of the M.A.C. instead of the 25% point as would be expected. This means that the conventional method of determining M.A.C. is not satisfactory for this wing and explains much of the discrepancy between calculated and observed stabilities.

One unusual feature of the model S1Wt was that the elevators of tail surface H1 were supported from the stabilizer on ball bearing hinges and were statically balanced by a counterweight forward of the hinge axis and inside the fuselage. Clamping arrangements were also provided so that tests could be made with the elevator free or with it clamped at any of a series of angles to the stabilizer. The pitching moment about the C.G. for S1Wt, with elevator free, as well as for Wt, is plotted in Fig. 6. It appears that the stability of S1Wt is small but positive for CL less than 0.8. Above this lift coefficient the model is first neutrally stable and then unstable until the stall. Because of this unsatisfactory behavior another wing, W2, was designed so as to have the same root chord as Wt and also to have the same fuselage connection as Wt, but so as to have its M.A.C. further to the rear and hence give greater stability. At the same time new horizontal and vertical tail surfaces (H2, V2) were constructed. Unfortunately it was not possible to wait for the completion of these surfaces in carrying on the experiments, so that most of the control surface investigations were made with S1Wt(i.e. with Wt and H1).

First a series of measurements was made with elevator free and various stabilizer and flettner settings. The results are given in Fig. 17. The first important conclusion to be drawn from this figure is that, with power off the airplane should trim hands off at approximately the high speed attitude for a stabilizer setting of 0° and a flettner setting of 0°. The next conclusion is that with fixed stabilizer (at 0°) the flettners have much more than enough effectiveness to enable the plane to be trimmed hands off at any attitude in the flying range. Fig. 18 gives the results of corresponding tests made with stabilizer, elevator, and flettner clamped rigidly at various angles. The results are self-explanatory and require no further discussion. In Fig. 19 are plotted the observed elevator angles as function of the angle of attack for various flettner angles with elevator free. Up to 10° throw the elevator angles have approximately the same numerical values as the flettner angles. The flettner effectiveness decreases at flettner angles above about 10° and apparently becomes very small at angles greater than about 20°.

The results of Figs. 17 and 18 have been replotted in Fig. 20, which gives at a glance the stability, trim, and control characteristics of S1Wt with elevator fixed and free. The auxiliary scales to the right, giving approximate origins for the CM scale for various stabilizer, elevator, and flettner settings, furnish a very simple means of seeing quickly what is the effectiveness of the various controls. It appears that the stability is too small to be satisfactory but all controls are extremely powerful.

The differences in CM, from Figs. 17 and 18, for the model with and without horizontal tail surfaces have been plotted in Fig. 21 as CMt pitching moment furnished by the tail. This figure enables a determination of the "tail efficiency factor"

to be made. This factor occurs in a formula for the pitching moment coefficient due to a tail surface which is used by C.B. Millikan in his courses on "Aerodynamics of the Airplane":

$$CMt = CMo' - \eta t \, \frac{1}{t} \, \frac{St}{S} \, \frac{1 - R/\pi(AR)}{1 + R/\pi(ARt)} = CL$$

where

CMt = pitching moment coefficient about the C.G. due to the horizontal tail surfaces.

CMo' = a constant depending on stabilizer and elevator settings.

1 = distance of C.P. of tail behind C.G.

t = mean aerodynamic chord of the wing.

St = total area of horizontal tail surfaces.

S = wing area

R = slope of curve of CL vs. ∝ (radius) for infinite aspect ratio.

~ 5.5 for all normal airfoil sections.

AR = effective aspect ratio of wing cellule.

ARt = aspect ratio of horizontal tail surfaces

ηt = tail efficiency factor.

The above formula is deduced theoretically for elevator fixed to take into account all downwash effects. The factor ηt is an empirical correction factor introduced to take account of the interference effect of a fuselage on the tail. The numerical value of ηt will naturally depend on the particular wing-fuselage tail combination under consideration, but an average value of between 0.75 and 0.80 has been recommended in general.

For the Douglas Transport S1Wt we have (from Table 2 and from Douglas Co. drawings scaled up to full size).

1 = 40.6 ft.	St = 120 ft. 2	S = 944 ft. 2
AR = 7.66	ARt = 4.75	t = 12.06 ft.
		R = 5.5

Hence from the formula

$$\frac{dCMt}{dCL} = -\eta t \, \frac{40.6}{12.06} \cdot \frac{120}{944} \cdot \frac{0.771}{1.386} = -0.241 \, \eta t$$

$$\therefore (dCMt/dCL)\text{theor.} = -0.241 \, \eta t$$

From Fig. 21 we have approximately for elevator fixed:

$$(dCMt/dCL)\text{exp.} = -0.15$$

Comparing these two results we find that for the S1Wt model as indicated by the Present tests:

$$\eta t = 0.62$$

This is a somewhat lower value than has been observed before in the GALCIT tunnel, and an explanation of this fact is not as yet apparent. One very interesting characteristic brought out by Fig. 21 is the remarkably slight decrease in tail effectiveness (i.e. in |dCMt/dCL|) which is introduced by changing from elevator fixed to elevator free.

In view of the unsatisfactory stability characteristics of S1Wt, a second wing W2, and a new set of horizontal tail surfaces H2, were constructed, as has already been mentioned. Unfortunately the stabilizer of H2 was built rigidly into the fuselage at a setting of 0°, so that experiments with wing W2 and horizontal ail surfaces removed could not be made. Also, because of the elaborate set of tests with S1Wt, no complete investigation of elevator and flettner effectiveness was undertaken with S2W2. In addition, the elevators of H2 were not statically balanced so that no elevator free measurements could be made. The pitching moments for wing W2 alone and for S2W2 (i.e. stabilizer 0°, elevator 0°, flettner 0°) are plotted in Fig. 9. It is seen that the stability of S2W2 is very good over the normal flying range (dCM/dCL = - 0.15) while even at the stall there is no actual instability.

The effectiveness of elevator and flettner in permitting trim at the stall is discussed later in connection with the measurements on bottom surface flaps (section 6). CM vs. CL curves are given in Fig. 34 in this connection for S2W2 and for S2W2w2 = S2W2'. Comparing the two curves one sees that the addition of w2 decreases the stability very slightly and makes the plane approximately neutrally stable for quite a range of angles of attack at the stall. The curves of Fig. 33 for elevator up indicate ample control to enable the tail to be gotten down for landing. The effect of flaps on stability and trip is discussed in section 6 below.

5) Lateral (including Directional) Control

As mentioned in section II, the number of wind tunnel balances available at the time of the tests was insufficient to permit six components to be measured simultaneously. With the normal "lift, drag, pitching moment" setup the rolling moment about an axis through the diamond shaped suspension frame (cf. Fig. 3) could be measured. It was at first assumed that this was identical with the rolling moment about the fuselage axis and a series of rolling moments with various aileron displacements was made with this normal setup. Later the drag balance and one lift balance were moved to the yaw balance positions. With the latter setup side force, yawing moment, and rolling moment could be determined. It was discovered that the side force was quite appreciable and this introduced rather considerable corrections in transferring rolling moments from the suspension frame axis to the fuselage axis. In addition a slight lack of symmetry in the model or in the airstream was found to cause measurable rolling and yawing moments with no aileron deflection. The final results given here contain correction factors introduced to take all of these elements into consideration and represent averages of Runs made with both balance arrangements.

The first set of observations dealt with the effectiveness of the conventional Frieze type ailerons whose outline is indicated in Fig. 1. Expressed in percent of the gross wing area these ailerons had an area of 11.2% behind the hinge axis or 14.1% overall. The results are given in Figs. 22 or 23. In plotting such aileron curves the convention has been adopted that solid lines correspond to moments of the desirable sign, while dashed lines correspond to reversal of control or to moments of undesirable sign. Fig 22 shows that the aileron effectiveness is good as would be expected from their rather large size. However, the effectiveness near the stall is not as large as Weick recommends for satisfactory control (Cr ~ 0.075 for CL = 1.0). The aileron effectiveness holds up quite well at the stall, no reversal appearing until $\alpha \sim 18°$. The yawing moments (Fig. 23) are favorable for "up aileron" at small angles of attack or large aileron angles. However, for "down aileron" the yawing moments have an unfavorable sign for all conditions tested. For a 2 to 1 differential, the normal arrangement of – 20°, + 10° has unfavorable yawing moments for angles of attack above about 7°.

Since it was planned to investigate the effects of bottom surface flaps a few experiments were next performed on free air ailerons. (A). The ailerons had a symmetrical profile, and an area of 5.85% of the wing area. Their location and shape is indicated in Fig. 24 where the chord line used as a reference axis for angles also appears. All aileron angles refer to the angle which this reference axis makes with the thrust line of the model. It was first determined that the minimum drag occurred with ailerons set at about +3° and this was accordingly chosen as the neutral setting. Since it appeared that plus angles (trailing edge down) gave reversed rolling moments, it was assumed that the aileron system had complete differential, i.e. only negative angles were considered. The results are given in Figs. 25 and 26. The rolling moments are quite satisfactory, comparing very favorably with those for the normal ailerons. The favorable yawing moments are of considerable magnitude. Hence the present wind tunnel tests show very promising characteristics for the free air ailerons. However, flight test results would be required before it could be ascertained whether or not the actual behavior in flight would be satisfactory.

A series of tests on rudder control was next undertaken. In Fig. 27 are plotted Cr and Cy as functions of rudder angle for flettner at 0°. These measurements refer to S1Wt, i.e. to the first vertical surfaces V1. The rolling moments are very small as was to be expected and the yawing moments are quite satisfactory, having nearly twice the maximum value which Warner states is desirable (E.P. Warner, Airplane Design, p. 444). The rudder effectiveness drops off very rapidly at rudder angles above 30°, so that there would be little purpose in designing for throws larger than ± 30°. The flettner effectiveness, shown in Fig. 28, is not quite so satisfactory. The largest rudder angle which can be obtained with rudder free is about 13° and the corresponding yawing moment is less than one half of that corresponding to a rudder angle of 30°. Fig. 29 gives the results of measurements with ailerons fairly well displaced and rudder set to counteract the yawing moment so produced. It appears that the rudder is able to neutralize the aileron yaw but without much favorable yawing moment to spare. For this reason, when the model was rebuilt with wing W2, a new vertical surface V2 was also installed. The results of tests on the complete model SW2' with full rudder angle and with flettner angles of 0° and 15° are given in Fig. 30. The rolling moments remain negligible, but the yawing moments are very large for both configurations and at all angles of attack.

6) Bottom Surface Flaps.

A series of six flaps (denoted by A, 100%, B, C, D, E) which were investigated is shown diagrammatically in Fig. 31. All were of the bottom surface, split trailing edge type, and all were tested at an angle of 45° to the chord line. Three configurations of the horizontal tail surfaces were tested at various times: elevator 0° flettner 0°, elevator -30° flettner 0°, and elevator -30° flettner +150°. The last two were investigated because it was desired to ascertain whether or not the plane could be trimmed at the stall with flaps down. The values of CLmax for the various configurations tested are given tabularly in Fig. 31. Curves of CL vs. α for some of the configurations are given in Fig. 32 and curves for CM vs. CL in Fig. 33. Flaps A and B give very similar results and comparing them with C it is evident that the extension of the flaps under the wing root gives a marked increase in CLmax and a considerable decrease in the undesirable diving moment. On the other hand the further extension under the fuselage as in D gives no appreciable increase in CLmax and only a slight further improvement in the diving moment. A comparison of flaps 100% and C shows that the Vee notch caused by the dihedral at the junction between center section and wing flaps, has no noticeable deleterious effect. The elevators have ample control to permit the plane to be trimmed at the stall with any of the flaps extending completely across the center section, but not with any of the other flaps. All of the pitching moment curves with flaps indicate a region of longitudinal instability from the stalling angle of attack (about 12°) up to about 15°. Hence if the airplane reached an angle of attack of about 12° with flaps down it would rapidly increase its angle of attack to about 15° and would be stable from this angle up. A very interesting and satisfactory result obtained with flaps 100%, C, D, and E is that the CM vs. CL curve with flaps down is very nearly an extension of the corresponding curve without flaps. This means that lowering the flaps in flight would be accompanied by only very small moments altering the trim. A complete set of curves of CD, α. CM vs. CL for the W2 wing only, SW2, SW2', and SW2' with flaps B, C, and E is given for comparative purposes in Fig. 34.

7) Pressure Distribution on N.A.C.A. Cowl

An investigation of the pressure distribution at the surface of the N.A.C.A. cowl on one of the engine nacelles was made using a multiple tube manometer. The setup is shown in Photo 8. A special N.A.C.A. cowling was built up having five pressure orifices on the inside of the cowl spaced along a fore and aft plane as shown in the pressure plots. Small copper tubes ran from these orifices to the rear outside of the cowl and were connected to the manometer by long rubber tubes. 4/9ths of the circumference from these orifices (so as not to interfere with engine cylinders) was

a similar row of pressure orifices on the outer surface of the cowl. The tubing for these orifices ran to the rear inside the cowling. In order to investigate a possible anti-symmetry in the flow the entire cowl was removed after one set of observations, was then rotated through 4/9ths of a revolution, and the observations repeated. The circumferential location of the orifices is indicated on Fig. 35.

The individual pressures were measured relative to an arbitrary origin which was approximately the static pressure in the wind tunnel working section. All pressures were then divided by the dynamic pressure q and plotted as vectors giving the ratio of p to q. Pressures measured at the outer surface are plotted as solid vectors in Figs. 35 and 36, inner surface pressures as dotted vectors. The sign of the pressures, i.e. whether suction or positive pressure is indicated by the arrows on the vectors. In view of the rather arbitrary origin for the measurement of the individual pressures, the algebraic sum of the two pertinent vectors should be used to give the net force on the cowl at any point. In other words the external and internal pressures should be considered together and not separately. From the two figures it is clear that there are considerably bursting forces acting on the cowl as well as a resultant force tending to pull the cowl forward into the propeller.

8) Performance Estimation

In this section estimates are made of the full scale values of the aerodynamic parameters entering into a normal performance calculations. These are extrapolated from the wind tunnel results given in the present report. Actual performance computations are not attempted in view of uncertainties in the final non-aerodynamic design parameters. The three aerodynamic parameters necessary for a performance calculation are CLmax, the airplane efficiency factor e, and the equivalent parasite area f. We discuss these individually below.

CLmax and Stalling Speed

In Fig. 37 are plotted values of CLmax vs. R for wing N.A.C.A. 2412 and model FV1Wt from results obtained at the GALCIT. The values of CLmax interpolated from the results of the present tests for S2W2' and S2W2' with flap E are also plotted. The two dotted curves represent the author's judgment as to a reasonable extrapolation of these results to full scale at the stalling attitudes. From the design data and these curves it appears that the full scale stalling speeds, Reynolds' Numbers, and CLmax's for the two cases are:

Uncorrected Model Results			Model Results Extrapolated to full scale from Fig. 37		
CLmax	Vs(mph)	R model	CLmax	Vs(mph)	R full scale
S2W2'	1.30	73	1.6 x 106	1.40	71 7.9 x 106
S2W2' with flap E	1.90	61	1.6 x 106	2.00	59 6.6 x 106

These are obtained from Oswald's Fig. 40, (W. Bailey Oswald "General Formulas and Charts for the Calculation of Airplane Performance", Technical Report No. 408, 1932.), assuming a wing loading of 17000/939 = 18.1 lb/ft2.

Efficiency factor, e.

The curve for S2W2' of Fig. 34 is used to obtain an estimate of e. This is done by constructing the induced drag parabola which gives, as nearly as possible, constant CDp over the flying range, and finding the effective aspect ratio ARe corresponding to this parabola, where CD1 = CL2/πARe

Then e is determined from e = ARe/AR = ARe/7.70

If one determines the parabolas which give CDp = CDpmin for a series of values of CL, and calculates e for each parabola one gets approximately the following

CL =	0.4	0.5	0.6	0.7	0.8	0.9
e =	0.91	0.88	0.85	0.82	0.78	0.74

Hence e is not constant but decreases continually as CL increases. This is apparently due to the unfavorable interference effects between wing and nacelles, since the variation of e is much less when the nacelles are removed. For power-on flight it is probably that these unfavorable effects are much reduced by the action of the slipstream. "e" is of principal importance for performance estimation in the calculation of maximum rate of climb at sea-level and absolute ceiling. For the present airplane a rough calculation shows that the first occurs at CL ~ 0.5, while the second occurs at CL ~ 0.8. Hence for power-on performance calculations a value of e of 0.85 should be conservative. We therefore take

e = 0.85

For single engined ceiling estimates it may be necessary to decrease this value somewhat.

Equivalent parasite area, f.

From Fig. 15 (Run 166) we get for the model S2W2'. CDpmin = 0.0208. We denote the value of f corresponding to this unextrapolated result by fm and obtain

fm = 0.0208 x 939 = 19.5 ft.2 ~ R = 1.6 x 106

The first estimate which is required deals with the effect of the increase to full scale Reynolds' Numbers on the equivalent parasite area. Karman has recently given a theory of turbulence skin friction for smooth surfaces which apparently gives a good approximation to experiment even for very large Reynolds' Numbers. This theory is presented in "Proceedings of the Third International Congress for Applied Mechanics", Stockholm, 1930, Vol. I, page 85. If we write

F = Cf (ρ/2) U2bl

Where

F = skin friction on a smooth plane surface of breadth b and length l (in the direction of flow)

U = undisturbed velocity far from the plate

ρ = fluid density

Cf = skin friction coefficient

R = Reynolds' Number = U1/V

Then Cf is given as a function of R in Fig. 38 which is reproduced from Fig. 6 of "Quelques Problemes Actuels de l'Aerodynamique" by Th. von Karman (Paris 1932).

We can use this theory in getting an estimate of the effect of Reynolds' Number on the drag of the model discussed in this report. We consider two limiting cases:

Case 1: The wing profile drag is assumed to arise entirely from turbulent skin friction for which Karman's theory is applicable. The remaining parasite drag is assumed to arise from eddy resistance and to have a drag coefficient independent of R. The characteristic length 1, is assumed to be the M.A.C.

Case 2: The entire parasite drag is assumed to be caused by turbulent skin friction. The characteristic length 1, is again taken as the M.A.C. since it is assumed that the large length of the fuselage and the small lengths of nacelles, tail surfaces, etc. will lead to an average effective length of about the wing chord magnitude.

For both cases we have:

CDpfull scale = Cf (Rfull scale)

CDpmodel = Cf (Rmodel)

where CDp is the portion of the parasite drag coefficient which is assumed to vary with R. We use subscripts 1, 2 to denote the two cases.

For Case 1 we take the wing profile drag

CDpomodel = 0.0105 Rmodel = 1.6 x 106

For Case 2

CDp2model = 0.0208 Rmodel = 1.6 x 106

From preliminary estimates we get Vmax ~ 200 m.p.h. The corresponding full scale Reynolds' Number is

Rfull scale ~ 22 x 106

From Fig. 38 this gives

Cfmodel = 0.0042 Cffull scale = 0.0027

Hence for both Case 1 and Case 2

Cf(Rfull scale) = 0.0027 = 0.64
Cf(Rmodel) 0.0042

In Case 1 we have now

CDpofull scale = 0.0105 x 0.64 = 0.0067

To this must be added the constant portion of CDp = 0.0208 – 0.0105 = 0.0103.

Hence finally

CDp1 = 0.0172 f1 = 16.1 ft.2

In Case 2

CDp2full scale = 0.0208 x 0.64 = 0.0133

∴ CDp2 = 0.0133 f2 = 12.5 ft.2

We must now estimate the effect of the lap joints and rivets of the actual airplane which were not present on the model. From N.A.C.A. Technical Note #457 one can estimate that the lap joints may increase the wing profile drag 3 or 4%, corresponding to a Δ CDp = 0.0003. From Technical Note #461, Fig. 6 it appears that the addition of a normal arrangement of rivets adds a drag to a smooth wing given by Δ CDp = 0.0018. For the fuselage and tail surfaces the presence of rivets, window depressions, etc., may be estimated to furnish an additional Δ CDp ~ 0.0009. Hence one may estimate the addition of rivets, roughness, etc., to furnish

Δ CDp = 0.0030 Δf = 2.8 ft.2

Collection the results we have

	CDp	f
Model results scaled up without modification	0.0208	19.5 ft.2
Model results scaled up and roughness included Case 1	0.0202	19.0 ft.2
Model results scaled up and roughness included Case 2	0.0183	15.3 ft.2

Case 1 is probably too pessimistic and Case 2 too optimistic, so that the value to be actually expected should lie between the two. It should be explicitly noted that no radio mast or antenna has been considered, nor has any other departure from the model configuration except rivets and surface roughness.

CONCLUSION

The above experiments were carried out from December 12, 1932 to May 25, 1933 under the direction of Drs. A. L. Klein and C. E. Millikan who were largely assisted by Messrs. N. B. Moore, Rosooe Mills, W. Bowen, A. Reed, and other graduate students at the GALCIT.

Guggenheim Aeronautics Laboratory
California Institute of Technology
Pasadena, California
June 7, 1933

Document 3-22(b), Engineering Department, Douglas Aircraft Company, "Development of the Douglas Transport," Technical Data Report SW-157A, undated (ca. 1933-34).

DEVELOPMENT OF THE DOUGLAS TRANSPORT

INTRODUCTION

After fourteen years of continuous, successful experience in building airplanes of all types for the United States Army, Navy, Post Office Department and Coast Guard and for private persons and foreign governments, the Douglas Aircraft Company started plans for the design and development of a high performance passenger airplane for airline use. Profiting by extensive experience in the design and production of aircraft, the Company decided to make an extremely thorough investigation of all factors, however minor, that might affect performance and passenger comfort. Before construction was started, hundreds of wind tunnel and structural tests were made in addition to an intensive mock-up investigation and studies and tests of special items, such as fuel systems, control mechanisms, heating, lighting and ventilating systems and sound control. When the various parts of the airplane were ready, they were each tested to show their static strength and freedom from vibration or flutter.

The finished airplane was, in all probability, subjected to more thorough flight tests than any other known type of passenger transport or even military airplane. Over two hundred flying hours and fifteen thousand gallons of fuel were used in making these flight tests. Not only were the usual tests for speed, stability and general performance made but also tests subjecting the airplane to dynamic loads in flight to prove its structural strength, to determine the best soundproofing practical, to eliminate vibration and to determine the effect of certain variables, such as different engine cowls, fairings, oil temperature regulators, propellers, wing and control surface flaps, engine cooling and power. In conjunction with these tests, several entirely new conceptions in flight testing were put into practice and a new technique for airline cruising operation was developed.

The development cost of the first airplane, including all research directly connected with the project, was approximately $325,000. In addition, the airplane incorporates a great amount of the experience obtained during airline operation of the highly successful single-engined Northrop transports, which represent an engineering and development cost of approximately $290,000.

It is desired to outline briefly in the following pages some of the work done in the development of the Douglas DC-1 and its successor, the DC-2. Tests are still being carried on daily, both on the ground and in flight, to improve and refine this airplane and make it a still more superior product, both from a manufacturing and an operating viewpoint.

AERODYNAMIC DEVELOPMENT

The aerodynamic design of the Douglas Transport was the subject of exhaustive study for a period of more than eighteen months. This study included aerodynamic calculations and wind tunnel and flight tests, which were carried out in a scientific and comprehensive manner. Through the correlation of these calculations and data, it was possible to predict and analyze the actual aerodynamic characteristics which were later obtained in service. The high degree of performance and safety offered by the Transport is the realization of features that have been thoroughly studied and tested in the wind tunnel and in flight.

The aerodynamic calculations were particularly concerned with performance and control at all attitudes of flight, both in normal and single-engine operating conditions. Special design of the controls, wing and fairing makes possible continued single-engine operation at high altitudes with sufficient controllability to ensure safety for meeting emergency conditions. The performance studies for obtaining the desired velocity, range and climb led to the choice of the bi-motor type with controllable-pitch propellers and high-lift wing flaps being adopted as best meeting the requirements of the high-performance airliner. The flaps give a gain in lift of 35% and a drag increase of 300%.

An extensive series of wind tunnel tests, including approximately 200 test runs, were carried out on a one-eleventh scale model of the Transport in the 200 mile-an-hour wind tunnel at the California Institute of Technology.

The large scale of the model and high speed of the tests were particularly valuable for this work. All items of the airplane affecting aerodynamic operation were tested with the view not only of obtaining the desired performance, stability and controllability, but also of perfecting each item to the greatest practical degree. Briefly, the investigation included tests on three complete wings with various modifications, various wings to fuselage fillets, tail surfaces, landing gears end tail wheels, several sets of ailerons, of normal and special types, six arrangements of high-lift wing flap devices, and other special arrangements. Tests on controllability and stability were made with controls both fixed and free. The lift and drag of the final model were tested at various Reynolds numbers in order to indicate the trend in passing to full-scale. The wind tunnel tests resulted in the final aerodynamic design providing an increased degree of performance with satisfactory stability and ample controllability for all normal and emergency conditions of flight.

It is interesting to note that some of the early models tested in the wind tunnel showed instability and that the tests revealed that it was necessary for satisfactory stability to have a hitherto untried arrangement of center of gravity, wing sweepback and general configuration. The actual airplane was built in accordance with this new plan of arrangement and the stability in flight proved to be exactly as predicted. If the wind tunnel tests had not been made, it is very possible that the airplane would have been unstable because ordinary investigation had indicated that the original arrangement was satisfactory.

The actual measured flight test results showed an excellent agreement with predicted performance in all phases and fully justified the extensive aerodynamic study and wind tunnel investigation. These flight data have further been used to modify aerodynamic features that indicated possible improvement, so that the final aerodynamic characteristics of the Douglas Transport are extremely satisfactory and very advanced for a transport airplane. In fact, the total resistance of the complete airplane is less than twice the resistance of the wing alone.

STRUCTURAL DEVELOPMENT

The studies of aerodynamics and general arrangement showed the desirability of having the engine nacelles well ahead of the wing leading edge. It was also found desirable to house the retractable landing gear within the nacelles. Sweeping back the outer part of the wing offered the advantages of getting the landing gear well forward of the center of gravity and having the center of gravity come well forward on the wing for stability. With these points in mind and recognizing the fact that the size and performance desired for this machine presented an entirely new problem, an exhaustive study of the various possible types of construction was made.

In developing a structure having the maximum strength and rigidity with a minimum of weight, it is preferable to design a wing with the material so distributed that there is no great variation in the stresses in the various parts. Such variation is apt to be caused by rigidly attaching very thin members, such as the skin, to very heavy members, such as spars or beams with heavy stresses, if very thorough and careful investigation of the distribution of loads, deflections, local stresses, etc., is not made. At the same time, the wing must have little or no torsional deflection, a minimum of vertical deflection, and no excessively large unsupported flat metal surfaces.

A first investigation showed that most metal wings were merely an adaptation of wooden designs in other material. However, the characteristics of wood and metal are quite different and, therefore, the design principles of one do not apply to the other. In a metal wing, having a thin skin rigidly attached to a heavy spar, sudden changes in cross section are apt to cause very objectionable stress concentrations. If precisely the proper proportions of material are not made, or if the designs of the various attachments are not exactly correct, there are apt to be cracks in the skin and popping of rivet heads due to the deflecting spars pulling against the skin.

In the Douglas and Northrop types of multi-cellular wing construction, there are a multiplicity of full length span-wise stiffeners, and the fact that they have no abrupt changes or "breaks" results in no concentration of stresses. With the centroids of the stiffeners located at the maximum distances from the neutral axis of the section, a most efficient structure for absorbing the bending load is obtained.

In a highly stressed airplane, torsional rigidity of the wing is of paramount importance in the prevention of wing flutter at high speeds and torsional deflection of the structure must therefore be kept to an absolute minimum. When under load, there will always be some vertical deflection but this must not be excessive since a wing with large vertical deflections might cause jamming of aileron controls and by no means inspires confidence in the passengers or pilots.

If unsupported flat metal surfaces are even moderately large, there is always a tendency for the middle of the surface to vibrate in flight even when there is no stress. This is termed "oil canning" and will, in time, cause fatigue in the sheet metal and in the rivets and cause rivet heads to work and to pop off. These unsupported flat surfaces continually drum and cause a noise that cannot be completely eliminated in a cabin because part is carried as vibration through the structure. Even when on the ground with the engines running, this "oil can" action and drumming is apparent. "Oil can" action should be differentiated from wrinkling in the skin. Wrinkling of the skin will be present in every metal wing with a flat metal covering taking stress. These wrinkles are deflections of the skin under load and ordinarily do not have any tendency to vibrate.

In determining the wing construction of the Douglas Transport, single, two, three and multi spar designs were considered as well as shell type and multi-cellular designs. .

After a thorough investigation of all types, the Northrop multi-cellular wing construction was finally decided upon. This type of structure consists of a flat skin reinforced by numerous longitudinals and ribs. The bending is taken by the combination of flat skin and full length stringers. Three main flat sheets or webs carry the shear loads and torsion and indirect stress are carried by the skin with frequent ribs preserving the contour and dividing the structure up into a number of small rigid boxes or cells. Since the major loads are carried in the outer surface of the wing as well as in the internal structure, an inspection of the exterior gives a ready indication of the structural condition. The unit stresses in the material are low and therefore the deflections are at a minimum giving a maximum in rigidity. This construction has proven to be a happy medium of those considered since it combines practically all of the advantages of each; namely, very small unsupported areas, extreme lightness for its strength and rigidity, also ease of construction, inspection, maintenance and repair. The Northrop wing being comparatively small, it is economical to have many of the stringers run from the top to the bottom of the wing as shear webs or spars. However, when the principle is carried out on a larger scale, as in the Douglas Transport with its deeper wing, it is more efficient to have only three shear webs or spars. Thus it was not necessary to evolve a new type of structure but merely to adapt a time proven type to the dimensions of the Douglas Transport.

In the fuselage, the structural problem was basically the same. However, the Douglas Company had had extensive experience in building metal monocoque fuselages. This experience, combined with that of the Northrop Company, resulted in the present fuselage construction. This construction consists of a smooth, stressed skin in contact with closely spaced over-strength bulkheads and numerous longitudinal stringers (either flanged members or extruded angles) as a rigid part of the skin passing through the bulkheads, thus all parts are securely attached together and the skin has very small unsupported areas.

The coast to coast airline, Transcontinental and Western Air, Inc., which has been using a fleet of Northrop mail planes in daily service with notable satisfaction, advised on the design of the Douglas Transport from an operator's viewpoint. The airline encouraged this type of wing, fuselage and tail construction principally because their actual experience of many thousands of flying hours in hard service with the Northrop mail planes showed that the maintenance costs of this type of construction are negligible.

CONCLUSION

It is gratifying to note that the time and expense of all the preliminary aerodynamic, wind tunnel, mock-up and design studies were more than justified by the results obtained. The performance, stability characteristics and wing deflections, as determined in flight, conformed almost exactly with the predicted results. In fact, no major changes were necessary in the arrangement of the various parts of the airplane. Similarly, other wind tunnel predictions were proven in flight to be accurate.

The superior strength and rigidity of the Douglas multi-cellular wing and all-metal fuselage construction has been proven in both static and dynamic tests as well as in service to more than justify the time and expense of the thorough investigation made of the structure. There is no doubt left regarding the strength and reliability of any part.

From the viewpoint of passenger and pilot comfort, the mock-up, soundproofing, heating and ventilating investigations have more than proven their value as shown in the quietness and comfort of this multi-engined transport.

To add further to the completeness and excellence of this airplane, carefully worked out maintenance aids have been so constructed and mounted as to provide for servicing and replacement with a minimum of time and expense. In fact, the complete power plant section, including the engine, propeller, oil tank and all cowling, may be completely removed in seventeen minutes.

In general, this airplane is the product of a painstaking study of all the problems concerned and a thorough and methodical investigation of every possible solution, combined with the extensive experience of the Douglas Company in producing a great quantity of experimental and production airplanes. The Douglas Transport takes the air fourfold in supremacy - in comfort, performance, safety and service - the luxury liner of the airways.

Document 3-23

"Douglas Airliner for Transcontinental Service," *Aviation* 32 (October 1933): 331-332.

The Douglas DC-series aircraft were marvels of their age. Major innovations in aerodynamics, structures, propulsion, and flight instrumentation came together in an elegant way to produce the world's most advanced long-range airplane. In this article from October 1933, *Aviation* magazine introduced the DC-1 to the aeronautical community. The article provides an excellent summary of the innovations employed in the DC-1 design.

Document 3-23, "Douglas Airliner for Transcontinental Service,"* Aviation *32 (October 1933).

Except for a few amphibians of the "Dolphin" class delivered for private use or for limited transport services, the output of the Douglas plant at Santa Monica, Cal., has to date been definitely militaristic. To meet Transcontinental & Western Air's requirements for new equipment, however, Douglas has entered the commercial field on a large scale. The delivery of the first of an order of twenty DC-1 transport airplanes marks the beginning of a new high-speed shuttle service between New York and Los Angeles over the TWA system.

Safety, speed, comfort and economy have been the keynotes of 1933 transport airplane design and an analysis of the specifications indicates that the new Douglas machine yields an exceptionally high rating on all four points. Under the head of safety may be listed such features as all-metal construction; a comparatively great amount of shock-absorbing structure ahead of and below the passenger compartment; high performance with one engine cut (an altitude of 9,000 ft. has been reported on one engine with full load); low landing speed (60 miles per hour at sea level), and steep gliding angles through the use of trailing edge flaps; the ability to land without damage (except to propellers) and with full braking facilities with wheels fully retracted; a wide range of vision from the pilot's cockpit (including a full view of the retracting undercarriage); quick-acting dump valve on fuel tanks; full fire-extinguishing equipment, and a full complement of the latest navigational and communication instruments.

High speed was forecast through exhaustive wind tunnel tests on interference and careful attention to aerodynamic cleanness. Engine nacelle positions follow N.A.C.A. recommendations for low drag and high propulsive efficiency. The ship shows a high speed of 188 m.p.h. at sea level, 210 m.p.h. at 8,000 ft. It cruises at 184 m.p.h. at sea level and at 190 m.p.h. at 8,000 ft., all figures being in excess of any so far reported for ships of its size and capacity. With the landing speed indicated above, the speed range of this airplane is worthy of note.

On the score of passenger comfort, nothing has been overlooked. The cabin, 6 ft. 3 in. high throughout and 5 ft. 6 in. wide, normally accommodates 14 passengers in 7 rows of two each, spaced 40 in. from seat back to seat back. For short hauls where extreme roominess is not essential (and where gasoline capacity can be reduced) accommodations can be installed for 21 passengers. The seats are of special Douglas design, fully adjustable, and are mounted individually on rubber to minimize direct vibration. Seat backs are reversible to permit passengers to sit face to face if desired. Since the cabin floor passes over the top of the wing structure, there is no obstruction of any sort in the cabin. Entrance is through a door on the left side of the fuselage in the rear. Aft of the door is a complete buffet, and beyond, a fully equipped lavatory.

Stephen J. Zand, acoustical engineer of the Sperry-Gyroscope Company, whose work with the new CurtissWright Condor has been previously reported (AVIATION, July, 1933) handled the sound-proofing of the new Douglas. By careful attention to rubber-insulating the engine mounts; by eliminating all direct contact between the structure and the cabin lining; by extraordinary care to eliminate all direct leaks (even to the extent of providing rubber gaskets for the cabin doors and designing special door locks without keyholes); and by a liberal use of Seapack (kapok processed into sheet form) and other sound-deadening materials in the 3 in. space between outer and inner shell, the average sound intensity in the cabin at cruising speed was reduced below 70 decibels—an outstanding achievement.

In connection with the sound-proofing development a complete ventilating and steam heating system has been worked out. Controlled ventilation is effected by admitting air through a vent in the nose of the fuselage and distributing it throughout the ship by ducts. A thermostatic control ensures that the temperature in the cabin will remain constant at 70 deg. With outside air temperatures going as low as –20 deg. F.

Economy results from good aerodynamic design (maximum performance from minimum power) and practical common-sense arrangement of component parts (for minimum maintenance and servicing expense). Exhaustive wind tunnel tests and careful calculations ensured acceptability on the first count, and an extensive specification of features required for minimum maintenance based on TWA's long operating experience guaranteed the second.

Getting down to structural details the entire machine is built of Aluminum Company of America's 24 ST and 24 SRT Alclad. The wing is of a cellular multiweb construction (similar to that used in the Northrop Gamma and Delta) and is tapered in plan and in thickness. N.A.C.A. Airfoil Section No. 2215 was used at the root, and No. 2209 at the tip. The central portion of the wing is built integral with the fuselage and serves as a mounting for the engine nacelles and the retractable landing gear. Outer wing panels are demountable by means of bolted joints. Two

main fuel tanks of 180 gal. each and two auxiliary tanks of 70 gal. each are mounted in the center section on each side of the fuselage.

The fuselage itself is a full monocoque with both the vertical fin and the horizontal stabilizer built integral. Longitudinal and directional trim are obtained by tabs in the rudder and elevators. The filleting of the wing and tail surface intersections has been carefully studied for minimum interference effect. All controls, including those for the trimming tabs, are internal.

Landing wheels retract upward and forward into the engine nacelles by a simple hydraulic mechanism. Counterbalancing the landing gear has made possible the use of hand operation only. Retraction is accomplished in 25 seconds and lowering in 20 seconds by means of a pump within easy reach of either pilot or copilot. When in the retracted position the axles rest in sockets attached to the main nacelle bulkhead. Hydraulic brakes with a differential control operating through the rudder pedals are provided. Shock absorbers are of the Douglas hydraulic type. A 42x15.00-16 tire is used.

The engine nacelles are monocoque except for the steel tube supporting structure forward of the firewall. The entire mount, including engine and all accessories, is quickly detachable and interchangeable right and left. Removal is facilitated by grouping all connections at the firewall and by using quickly detachable plugs for all electrical connections. Carburetor air intake is of sufficient capacity to prevent icing. Direct-cranking electric starters with shielded booster coils are used. Hamilton Standard controllable pitch three-bladed propellers are standard equipment.

Every possible feature for the comfort, convenience and safety of the pilots has been included. The control columns are located between the pilots and the outside walls so that there is no obstruction between the seats. Full instrument equipment has been installed on rubber-mounted vibration-proof boards. Under TWA specifications the latest type of Western Electric two-way radio has been installed including directional beacon receiver. All wiring for the radio or for other electric circuits is carried in aluminum conduit.

The ship is designed to accommodate either two Wright Cyclone F-3 engines (geared 11:16), each developing 710 hp. at 8,000 ft., or two Pratt & Whitney Hornet D geared engines with an output of 700 hp. at 6,500 ft. The general specifications as given by the manufacturer are: Length over-all, 60 ft.; span, 85 ft.; wing area, 948.6 sq. ft.; fin area, 28.9 sq. ft.; rudder area, 42.5 sq. ft.; stabilizer area, 97.6 sq. ft.; elevator area, 48.0 sq. ft.; aileron area (total), 86.8 sq. ft.; weight empty, 11,780 lb.; useful load (including radio and all equipment), 5,720 lb.; gross weight, 17,500 lb.; power loading, 12.3 lb. per hp.; wing loading, 18.5 lb. per sq. ft.

Document 3-24

Donald W. Douglas, Sr., "The Development and Reliability of the Modern Multi-Engine Air Liner," *The Journal of the Royal Aeronautical Society* (November 1935): 1010-1046.

In November 1935, Donald W. Douglas, then the president of the Institute of Aeronautical Sciences (since the 1960s, the American Institute of Aeronautics and Astronautics), addressed the Royal Aeronautical Society in London. What follows below is the section of his address concerning the major developments in aerodynamic design that had been helping to reinvent the airplane in the 1930s. In his speech, Douglas stressed the importance of single-engine operation within the demands of long-range operation, an issue that set the American flight environment apart from Europe and that served as a major stimulating factor in U.S. technological development.

Document 3-24, Donald W. Douglas, Sr., "The Development and Reliability of the Modern Multi-Engine Air Liner,"* The Journal of the Royal Aeronautical Society *(November 1935).

THE DEVELOPMENTS AND RELIABILITY OF THE MODERN

MULTI-ENGINE AIR LINER

(with special reference to Multi-engine Airplanes after Engine Failure)

By
DONALD W. DOUGLAS, Esq.
(President of the Institute of Aeronautical Sciences)

INTRODUCTION

Four essential features are generally required of any form of transportation: Speed, safety, comfort and economy. The airplane must compete with other forms of transportation and with other airplanes. The greater speed of aircraft travel justifies a certain increase in cost. The newer transport airplanes are comparable with, if not superior to, other means of transportation. Safety is of special importance, and improvement in this direction demands the airplane designer's best efforts.

SAFETY AND RELIABILITY

Statistics show that the foremost cause of accident is still the forced landing. The multi-engine airplane, capable of flying with one or more engines not operating, is the direct answer to the dangers of an engine failure. It is quite apparent, however, that for an airplane that is not capable of flying with one engine dead the risk increases with the number of engines installed. Hence, from the standpoint of forced landings, it is not desirable that an airplane be multi-engine unless it can maintain altitude over any portion of the air line with at least one engine dead. Furthermore, the risk increases with the number of remaining engines needed to maintain the required altitude. In general, therefore, the greatest safety is obtained from—

1. The largest number of engines that can be cut out without the ceiling of the airplane falling below a required value;
2. The smallest number of engines on which the airplane can maintain this given altitude.

For airplanes equipped with from one to four engines, it follows that the order of safety is according to the list below.

a. Four-engine airplane requiring 1 engine to maintain given altitude.
b. Three-engine airplane requiring 1 engine to maintain given altitude.
c. Four-engine airplane requiring 2 engines to maintain given altitude.
d. Two-engine airplane requiring 1 engine to maintain given altitude.
e. Three-engine airplane requiring 2 engines to maintain given altitude.
f. Four-engine airplane requiring 3 engines to maintain given altitude.
g. One-engine airplane requiring 1 engine to maintain given altitude.
h. Two-engine airplane requiring 2 engines to maintain given altitude.
i. Three-engine airplane requiring 3 engines to maintain given altitude.
j. Four-engine airplane requiring 4 engines to maintain given altitude.

Reliability and safety, however, depend upon other factors. These can generally be classified as the airline over which the airplane operates, and the aerodynamic design of the airplane. The airplane designer has control of reliability in so far as he is able to modify the multi-engine design to enable it to complete its flight over the necessary terrain after engine failure.

…performance has already been calculated or determined from flight test, Δ and l_t can be obtained from the normal full-power curves in Figs. 17 and 18.

2. The percentage drag increases resulting from the idling propeller and additional drag of the airplane are obtained from Figs 5 to 10, hence z is determined. Different values of z should be used for determining the various characteristics.

3. The principal performance characteristics are found from the normal values of the parameters l_p, l_s, l_t and Δ, and z by Figs. 16, 17 and 18. Alternately, the new values of the parameters can be calculated, or obtained through Figs. 13, 14 and 15, and z. Then the charts of Reference I are used to obtain the performance after engine failures.

AERODYNAMIC DESIGN

Achieving reliability in multi-engine airplanes is largely dependent upon the proper aerodynamic design of the airplane. It is necessary that special attention be given to the design of the airplane for flight with one or more engines out of operation. The multi-engine airplane should be definitely designed for this condition if the reliability consistent with the additional complication of many engines is to be realized. As had been previously pointed out, safety and reliability of the multi-engine airplane are not necessarily greater than that of a single-engine airplane, unless after engine failure it is capable of completing its mission in an efficient manner with adequate control.

Certain aerodynamic features are available which enable the engineer to design a multi-engine airplane that has good performance after engine failure. The principal considerations for obtaining most favorable operation are outlined below:

1. The rudder should produce sufficient yawing moment at the zero yaw attitude of flight to counterbalance the moments resulting from offset thrust, idling propeller drag, and any adverse yaw due to aileron action. The magnitude of this moment should be ample to provide for the lowest velocity intended for operation, usually near the stall.

2. The vertical surface should be designed as an efficient lifting surface based essentially on the criteria that would be used for judging a good wing design. It should be of high aspect ratio and so disposed that the efficiency of the surface is not impaired by adverse interference effects. The rudder angle necessary for producing equilibrium should not be too large because of the high parasite drag that might result.

3. A rudder flap is needed of sufficient size to hold the rudder in the required position with zero control force. Likewise, the ailerons and elevators require trimming flaps sufficient to enable the airplane to be flown "hands off."

4. Propellers should be of the feathering type in order to reduce their

drag when an engine fails. When turning against the torque of the engine, the negative drag of a propeller in high pitch is slightly less than when in low pitch. For higher pitch settings a free-wheeling propeller has considerably more drag than the feathered type, but less than when it is turning against engine torque. Reducing the propeller drag decreases the rudder size required and the drag of the rudder.

5. Span loading of the airplane should be as small as feasible (that is, large span and/or low weight) since span and weight have the greatest effect on ceiling of any of the airplane parameters. The problem of flight after engine failure is largely one of designing an airplane that has more than the usual capabilities in absolute ceiling.

6. The general "cleanness" of design of the airplane and efficiency of all component parts must be kept at their highest possible value, since all favorable and unfavorable effects are magnified when operating after engine failure.

7. The rudder moment required for balance should be investigated at the ceiling; especially for airplanes with highly supercharged engines, because aerodynamic force would there be reduced while thrust still remained large. In fact, bi-motored airplanes with highly supercharged engines will be found to require for single-engine operation a ruddersomewhat out of proportion to that which would normally be satisfactory.

All of these factors should be given due consideration if it is desired to achieve the reliability and safety factor possible when flying with one or more engines out of operation.

EXAMPLE OF USE OF METHOD

In order to illustrate the application of this method and the agreement between flight test results and calculations, the Douglas DC-2 Bi-motor Transport is used as an example since complete data on it are available.

Flight test results for full gross load reduced to standard performance conditions are as follows:

ON FULL POWER

High speed at 8,000 ft.	213 m.p.h.
Cruising speed at 75 percent. power at 14,000 ft.	200 m.p.h.
Maximum rate of climb at 8,000 ft.	1,000 ft./min.
Absolute ceiling	25,400

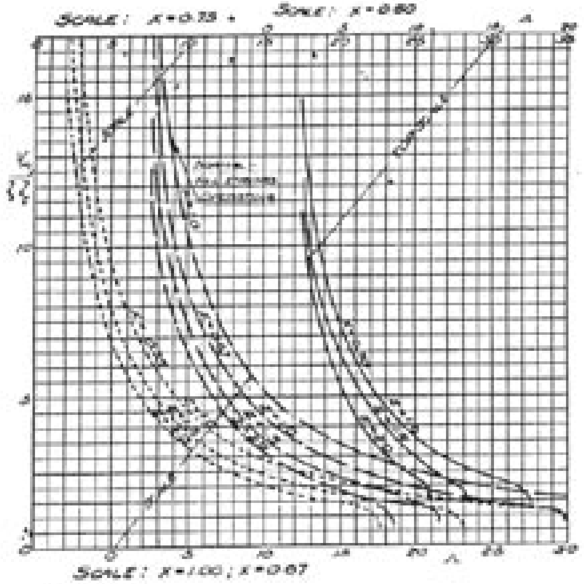


FIGURE 16.

$V_{m_x}/\lambda V_m$ AS A FUNCTION OF PARAMETER Λ AND PARASITE DRAG RATIO Z FOR MULTI-ENGINE AIRPLANES AFTER ENGINE FAILURE

V_m = MAXIMUM VELOCITY AT SEA LEVEL (MPH)

X = RATIO OF NUMBER OF ENGINES OPERATING TO TOTAL NUMBER AVAILABLE.

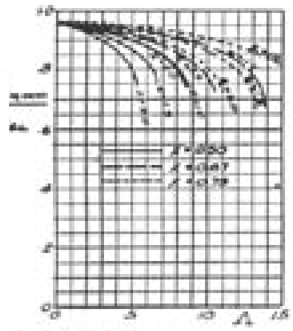


FIGURE 16a.

$R_{s_{max}}/R_{s_{o}}$ AS A FUNCTION OF PARAMETER Λ AND PARASITE DRAG RATIO Z FOR MULTI-ENGINE AIRPLANES AFTER ENGINE FAILURE.

X = RATIO OF NUMBER OF ENGINES OPERATING TO TOTAL NUMBER AVAILABLE.

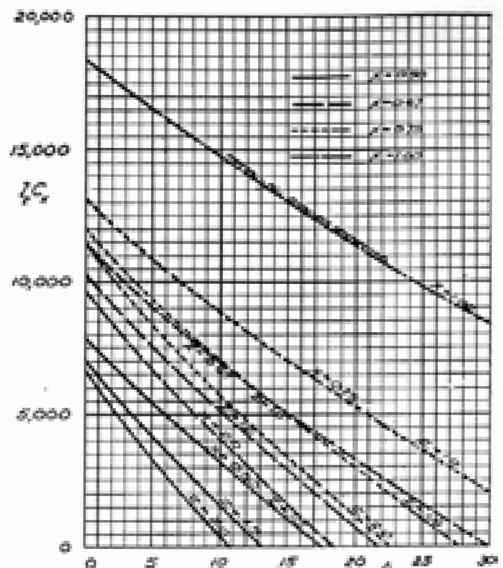


FIGURE 17.

λC_x AS A FUNCTION OF PARAMETER Λ AND PARASITE DRAG RATIO Z FOR MULTI-ENGINE AIRPLANES AFTER ENGINE FAILURE

C_x = MAXIMUM RATE OF CLIMB AT SEA LEVEL (FT/MIN)

X = RATIO OF NUMBER OF ENGINES OPERATING TO TOTAL NUMBER AVAILABLE.

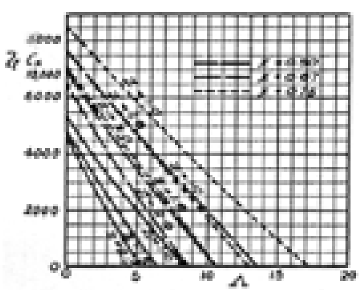


FIGURE 17a.

$\lambda C_{x_{max}}$ AS A FUNCTION OF PARAMETER Λ AND PARASITE DRAG RATIO Z FOR MULTI-ENGINE AIRPLANES AFTER ENGINE FAILURE

X = RATIO OF NUMBER OF ENGINES OPERATING TO TOTAL NUMBER AVAILABLE

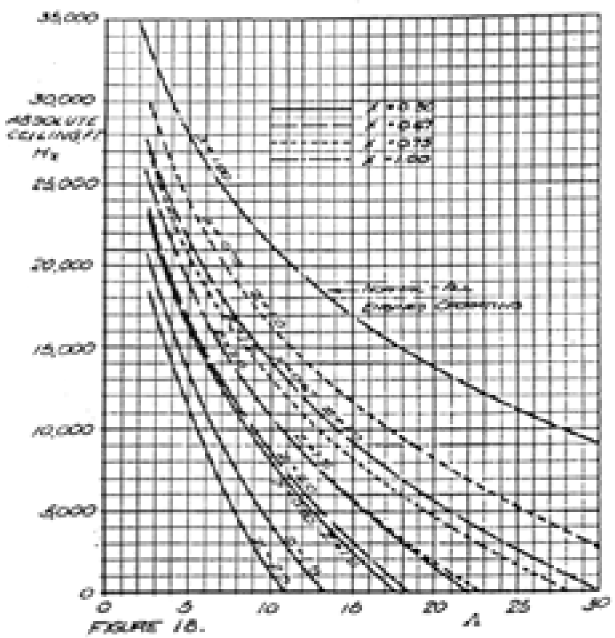


FIGURE 18.
ABSOLUTE CEILING H_1 AS A FUNCTION OF PARAMETER Δ AND PARASITE DRAG RATIO Z FOR MULTI-ENGINE AIRPLANES AFTER ENGINE FAILURE.
λ = RATIO OF NUMBER OF ENGINES OPERATING TO TOTAL NUMBER AVAILABLE

ON SINGLE ENGINE:

	Full Load:	Less 1,000 lb. Fuel. (Half Normal Fuel Load).
Absolute ceiling	9,500 ft.	10,800 ft.
Maximum rate of climb at 6,000 ft.	70 ft./min.	110 ft./min.

The performance on full power by reverse solution of the charts of Reference I yields the following performance parameters:

Δ_o = 6.55 (see Fig. 18, normal operation).
l_{to} = 10.4 (see Fig. 17, normal operation).
f = 18.0.

PROPELLER DRAG

For the 1,820 cubic inch engine, geared 16:11 at 9,500 ft.,

Q_t/r.p.m.$_p$ = 0.000057 X 1,820 (1 + 0.00002 X 9,500) 1.2/(11/16) = 0.216 (equation 15).

With the three-bladed 11-foot propeller at 113 miles per hour set 23° at 0.75 R. in low pitch and 32° in high pitch,

Q_n = (17,200 X 0.216)/(0.75 X 113 X 11.0^4) = 0.0300 (equation 20).

Then

T_c = –0.0136 (Fig. 8 for two-bladed propellers) for 23° blade angle.
= –0.0076 (Fig. 8 for two-bladed propellers) for 32° blade angle.

And

f_p = 3 X 11.0^2 X 0.0136 = 4.9 for 23° blade angle.
= 3 X 11.0^2 X 0.0076 = 2.8 for 32° blade angle.

Likewise, the drag of the propellers locked and feathered are found from Fig. 9 to be:

	Locked.		Free-Wheeling.		Feathered.
Angle at 0.75 R.	23°	32°	23°	32°	87°
T_c	–0.00225	–0.0020	–0.0072	–0.0035	–0.0013
f_p	8.2	7.3	2.6	1.3	0.5

Additional Drag

For the offset thrust of 9.0 ft. and tail length of 33 ft.,

F = (9.0/33) {2 +(1+9.0/9.0) (4.9/18.0)} = 0.69 at absolute ceiling, and

Δf / f = 0.28 (for efficient design, Fig. 10).

Total Parasite Drag Increase Ratio

z = 1 + (4.9/18) + 0.28 = 1.55 23° at 0.75 R.
= 1 + (2.8/18) + 0.26 = 1.42 32° at 0.75 R.

Similarly for other cases of propeller settings,

	Locked		Free-Wheeling		Feathered
Angle at 0.75 R.	23°	32°	23°	32°	87°
F	0.79	0.76	0.63	0.58	0.56
Δf / f	0.32	0.31	0.26	0.24	0.23
f_p / f	0.46	0.41	0.14	0.07	0.03
z	1.78	1.72	1.40	1.31	1.26

Absolute Ceiling

The absolute ceiling according to Fig. 18, for Δ_o = 6.55 and z = 1.55 is

H_1 = 9,500 ft. with inoperative propeller in low pitch.
H_1 = 10,500 ft. with inoperative propeller in high pitch.

Approximately 1,000 ft. increase in ceiling is obtained by setting the idling propeller in high pitch.

For the other propeller settings, the comparative single-engine ceilings are

	Normal Engine Torque		Locked		Free-Wheeling		Feathered
Angle at 0.75 R.	23°	32°	23°	32°	23°	32°	87°
Single-engine ceiling	9,500	10,500	8,400	8,700	10,500	11,100	11,500
Change in single-engine ceiling	—	+1,000	–1,100	–800	1,000	+1,600	+2,000

By dumping 1,000 lbs. of fuel, which is about half the normal fuel load, Δ_o is increased to

$$\Delta_o = 6.55 \text{ X } [17{,}000/18{,}000]^2 = 5.84 \ (–1{,}000 \text{ lb.})$$

and the normal single-engine ceiling increases 1,400 ft. from 9,500 ft. to

$$H_1 = 10{,}900 \text{ ft. (less 1,000 lb. fuel with propeller at 23°).}$$

OPERATION IN FLIGHT

Control of this airplane when operating on one engine was carefully studied with the result that it can be flown in this condition "hands off."

In one test a take-off was affected during which one engine was cut out after having traversed half of the take-off runway at 4,200 ft. above sea level. The airplane was climbed, flown at 8,300 ft. altitude, 1,000 ft. above the highest point of the transcontinental air line which is 7,300 ft. above sea level, and landed at the next regular airport, 5,100 ft. above sea level. The entire flight was made on the remaining engine without causing any unusual strain on the pilot or airplane. It is notable that on this entire single-engine flight an average terminal-to-terminal speed of approximately 124 miles per hour was made, which practically equals that formerly obtained with tri-motored equipment.

This illustration serves to demonstrate that a high degree of safety in operation of an airplane can be achieved by thorough study and application of the various principles and problems to flight after engine failure.

CONCLUSION

Reliability and safety in the operation of an air line are best served by multi-engine airplanes capable of completing under good control all required flights after engine failure.

In order that the airplane be capable of developing its full potentialities in flight after engine failure it is necessary that comfortable and sufficient rudder control be provided to maintain flight at approximately zero angle of yaw. The control must be sufficient to handle emergencies such as failure just after take-off, etc.

The performance of the airplane after engine failure can be calculated by a rapid parameter and chart method. The principal parameters required are the usual parasite, span and thrust horse-power loadings, l_p, l_s and l_t respectively, and parameter $\Delta = l_s l_t^{4/3} / l_p^{1/3}$, and parasite drag ratio z. Charts for finding z and the performance after engine failure are included.

The influence of the various factors affecting the performance after engine failure is readily seen. It is important that span be increased to a maximum, and that weight and propeller drag be reduced to a minimum. It is further desirable that the propeller offset from the airplane's plane of symmetry be as small as possible. The airplane should fly nearly at the attitude of zero yaw in order that drag remain low. The vertical surface should be of efficient aerodynamic design and have low drag with the rudder deflected.

For construction of the charts, all increase in drag of the airplane has been assumed to be of the parasitic type. This assumption has been found to be consistent with available flight test data. Should it be found desirable when sufficient data are available, all variations could be determined from the general relations, and charts developed therefrom.

Careful study of the various factors involved in flight after engine failure, particularly regarding control and performance, should enable the engineer to design an airplane that has control and performance satisfactory for any reasonable requirement. For multi-engine airplanes, the performance after engine failure should be regarded as a definite problem of design.

Outside of the purely mechanical reason that very large engines are not yet available, the principal justification for the multi-engine airplane is its possibility of continuing safe and satisfactory flight after partial engine failure. The importance of the problems of performance and control involved is, therefore, directly comparable to the basic problems of normal flight.

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3. Hartman, Edwin P. : Negative Thrust and Torque Characteristics of an Adjustable-Pitch Metal Propeller. N.A.C.A. Technical Report No. 464, 1933.

Mr. Douglas showed a film and added the following commentary:

I consider it a great honor to be here before you tonight on such an outstanding occasion to be able to join with your Society in again honoring the memory of Wilbur Wright. I bring to the members of the Royal Aeronautical Society the hearty and cordial greetings of the Institute of Aeronautical Sciences, our younger but similar association in America. Tonight I am not going to talk directly on the technical paper which I have submitted to you, but rather on the general subject of the developments in the air transport field in America. I not only feel that this might be less trying to many of you, but I am hopeful that the moving picture film I am about to have shown to you will so engross your attention that any defects in my talk will be unnoticed. As might be said in Hollywood, film by Fox, Warner and others—sound effects by Douglas!

The film to be shown is somewhat historical in that we will see at the start the first really successful air liners, namely, the early Fokker and Ford tri-engined planes. To touch but lightly on American air transport history, let me recall to you that it really started in 1926, when our Post Office Department transferred the operation of mail planes to private carriers. The revenues resulting to private concerns from this gave the needed subsidy to permit several capable concerns to embark seriously in the business of carrying passengers on regular schedules. Fokker and Ford made the first real contributions to the passenger equipment of these lines, and the records which these early air liners have to their credit is one to be proud of.

The stock market must be given some credit for the rapid increase in air travel in the early phase, since the money secured by the industry from Wall Street made possible a great expansion of facilities. In this period the development was most rapid in the ground organization of our large air carriers and in the navigational facilities supplied, in the main, by our constructive Department of Commerce. Thus, by, we will say, 1930, and in spite of an already hesitant tone in general business in America, the stage was set for the really rapid technological progress of the past five years.

It seems to me it has been the golden age of aeronautics for us—although not an age for gold in America!—an alert and ambitious military technical personnel played its part in accelerating the use of brains by our designers. A growing competition among our air lines spurred the development of faster and safer air liners. A need on the part of the manufacturers for a broader market for their products and an ability, represented in favorable balance sheets, to seek this with newer and better types, both military and civilian, added to the activity of this field. Early attempts at really fast airplanes, such as the original and successful Lockheed, pointed the way to others as to what might be done. But even beyond all this, and most certainly marking the time as one of real accomplishment, has been the fact that all agencies concerned in and contributing to aviation have been most alert, co-operative and constructive. Our engines have been developing at a pace to permit the airplane designer to raise his sights from time to time. Our instrument and radio people have aided tremendously by furnishing us with the means to fly in bad weather. Propeller makers have been most helpful and, in fact, I can say that the development of the variable pitch propeller to the practical point it has reached today, is probably the most fundamentally important development of this period. Without it many of our present air liners would be impracticable.

As the first modern multi-engine air liner development of this golden age we see on the screen the Boeing. This excellent ship with its high speed, twin instead of three engines, with its ability to continue flight on only one engine, was the first really modern transport. United Air Lines made use of these airplanes and was immediately successful in stimulating air travel. The public was offered flight schedules across our continent that attracted business on the basis of comfort, safety and speed. Other air lines were forced to seek new equipment and to try to still further increase speed and comfort. T. & W.A. embarked on an ambitious development program, enlisting the Douglas Company in the project. The result was our DC-2 type, which most of you have undoubtedly seen flying in and out of Croydon and which you will also see over some of our western mountainous country in the film. We of the Douglas Company are proud of this airplane, but we concede that we were but one agency in its development. From its inception we had the most excellent and wholehearted aid from the air lines. Pilots and maintenance engineers gave of their time and knowledge. The thought was developed that speed in the air was but one function to be solved in the solution of the equation of a successful ship. Speed of inspection and maintenance during fuelling stops, and studies to cut the time required to make replacements, were given great consideration. Safety was a factor which, of course, had to be developed to the utmost possible. Here we were furnished aid by the engine maker, the instrument people, the propeller designers and the wind tunnel and research agencies of our government and of our technical institutions. Flying, as our major air lines do, over great distances across high country in many places and through or over weather of the worst at times and continuation of flight at any point with one of our two engines stopped, was the prime requisite we had always before us in our design. As developed, the DC-2 actually took off after cutting one engine at a point 40 percent of the take-off run from a field about 4,500 ft. high, climbed on up to about 8,500 ft. and completed its flight across the highest part of the air line, some 300 miles, and all with full load.

Comfort was studied with care, and sound engineers developed efficient and practical methods of eliminating the formerly disagreeable and tiring noise of air transport. Heating and ventilation comparable to that found in modern buildings was affected after the aid of related industries was obtained.

The film also gives you a view of other interesting and remarkable achievements in American civil aviation. The Lockheed Electra has been built for and is being used very successfully by some of the air lines needing a smaller capacity airplane than the DC-2. The Lockheed has speed, comfort, and safety that are now demanded by our air travelers. An equally interesting and worthwhile air liner is the Curtiss Sleeper.

You will see on the screen a rather interesting view of the interior showing the comfortable quarters and a fair passenger turning in for a good sleep while en route over the American desert. I have had the surprising experience of seeing a mother settle her small family of two children, aged 2 and 4, for a quiet sleep at 10,000 ft. with the greatest of nonchalance. I felt at that moment that air transportation was really here and was spurred to plan more comfortable and roomy night quarters and facilities for our next developments.

I have spoken only about land transportation. The film will show you flying views of the latest and finest of our flying boats—the Sikorsky and Martin. These have been brought out by those two great concerns working closely with Pan-American Airways. They, in the Ford-Fokker era of land plane transportation, started their ambitious and highly successful foreign air lines with flying boats largely developed from naval boats. These latest pieces of equipment represent the last word as we know it in America for the safe and successful operation of overseas transport. With their four great engines it is difficult to imagine a flight failure due to engine trouble. Seaworthy hulls of large size seem adequate to permit of safe landings under all conditions likely to be met. Comfort is even better served in these commodious hulls than in the smaller bodies of our land planes.

The development of our modern air liner technically from 1926 to 1935 is quite apparent to the eye of the engineer. The development of the traffic in America that has spurred this on is revealed by a few statistics. In 1926 the total of commercial flying was about 4,000,000 plane-miles, a plane-mile being the equivalent of one plane flying one mile, whilst in 1934 this amounted to 41,000,000 plane-miles, a tenfold increase. The number of passengers carried increased from less than 6,000 in 1926 to 460,000 in 1934. Air mail volume increased from slightly over 200,000 lbs. in 1926 to an annual rate of 10,000,000 lbs. in the last quarter of 1934. Air express, an as yet practically undeveloped source of revenue, increased from 3,500 lbs. to 2,000,000 lbs. in this period. The time of flight from coast to coast of America has shown a great betterment in the past nine years, being now 16 hours against about 48 hours in 1926.

That we have achieved a marked gain in safety is proved by the records which show that whilst there was one fatality per 897,000 miles of air line operation of 1926, there was only one death for 6,817,000 miles in 1934. This improvement is most significant when consideration is given to the fact that we are now flying a great deal at night, which was not the case nine years ago and, furthermore, we are now operating in weather which even two years ago we would have regarded as impossible.

The story of the development of modern air liners would not be complete if we did not give credit to the effect of greater technical thought as applied to the flight of the air liner, rather than just to the design of the machines. Formerly the aeronautical engineer had completed his work when the final flight tests of his airplane were completed. From then on it was up to the pilot who, whilst vitally concerned with

the proper speed and altitude at which to fly, was not informed completely as to just what factors determined his best course. With the idea of realizing the ultimate in air line operations from the standpoint of speed and economy, many engineers and engineering pilots have necessarily collaborated. Results of their work in theory and flight tests have been made available to all and in such form that they are of great practical value and aid. Dr. Osborn, of our staff, and Pilot Eddie Allen have blazed the path along these lines, and further valuable information is being gathered by others inspired by this work. As can be seen from what I have said, we are only just learning to utilize the machine that we have developed. This brings me to the thought of the future development and I believe we should think for a moment at least of this. What lines do we wish to develop along speed, safety, comfort, size, efficiency? Speed to its ultimate, most certainly. Speed is the first thing that the air line ticket man has for a selling point. There may be many who fly on an air liner for the thrill or the change as compared with surface travel, but these will be fewer each year. There are undoubtedly many, and always will be, who fly because they prefer air travel on the grounds of cleanliness and lack of vibration, but speed must always be the greatest reason for traveling by air. Future development, therefore, must include the development of all the speed possibilities in air travel consistent with other necessary considerations. What the limit is, I cannot say. The aerodynamic engineer will, with some hesitancy, place a limit, but no one can be sure of it. I believe we should be cruising at 300 miles per hour in another ten years. I cannot say how; if I could we would be doing it now. Stratosphere or sub-stratosphere flight, while not to be developed necessarily primarily for speed, may open up avenues of development now not clear to us. Better utilization of slow landing devices will undoubtedly give us our next measurable increase in practical cruising speeds. There is undoubtedly some room for increases in overall aerodynamic efficiency. Possibly some expectations of somewhat better propulsive efficiencies is sound.

What of the development towards more safety? We must admit that whilst great strides have been made in this direction, more can be and must be done. We can reasonably expect to continue to improve the reliability of our engines, our instruments, our radio, our plumbing, and other vital gadgets, but at the moment we can only visualize a general betterment in the airplane itself by such changes as four-engined transport capable of flying on any two engines, rather than twin-engine airplanes. It seems at the moment that the most important advance in safety that we should be concerned in is in better protection against icing up, against electrical storms and in better development of blind flying and blind landing facilities. The icing problem is partially but inadequately solved at the moment. Its better solution may be mechanical and may also be along the line of greater altitude capabilities of the air liner. Hand in hand with this solution for icing goes the possibility that protection against dangerous electrical conditions may lie mainly in high altitude flight. Comfort development certainly is an open field and takes in the problem not only of comfort but sustentation of life at the high altitude that we may soon wish to fly at. Pressure cabins and/or free oxygen in the cabins are both being experimented with today. Our air lines are working diligently with our designers on these points and we look with confidence to some solution soon.

Our future developments from the standpoint of size seem, except in the case of trans-oceanic flying boats, to be controlled by traffic and comfort. Intrinsically, one of the advantages of air transport is its flexibility, so it would seem that size, beyond an efficient point from a structural standpoint and an economic one, is not to be expected to show any great change. Efficiency we must certainly hope to continue to improve, for air transport to be able to show an operating profit without depending on any subsidy. This will be improved by better pay-load percentages, lower first costs and lower maintenance charges. Already in America we find the operating costs of the modern ships quite markedly lower than the older and slower planes. Therefore, we need not fear that even with our speed development we shall cease decreasing costs.

Might I say that it seems to me that aviation is so tied with ground facilities of such a world-wide and international scope that the most important developments will come when we find means to more completely co-operate as nations. Also it appears that so much time and money is wasted because of the duplication of the same experiments in different countries, that something constructive should be considered at the present time. From this point of view military aviation seems to be a deterrent factor on more rapid and less costly progress. Why, however, can we not hope to take certain general aviation problems that do not involve anything intrinsically military, and allot them among the major nations: certain problems to one nation with more experience on a certain phase and certain problems to another? A complete interchange of the knowledge resulting would then follow. Surely this is possible, and if it is, the aviation industry, instead of being a potential breeder of war with its increasingly dangerous weapon, might become an agent in spreading understanding. I offer these last thoughts humbly and in the belief that engineers and scientists such as form the membership of the Royal Aeronautical Society, are able most fully to grasp the possibilities. A glorious future lies before aviation. Its development has only just commenced. May I hope that those interested in it here will believe that we in America extend to you our fellowship and promise of full co-operation for the common good.

The PRESIDENT: In asking Mr. Fairey to propose a vote of thanks to Mr. Douglas, he said that Mr. Douglas was indeed a remarkable man because he had given them three kinds of entertainment. First of all there was the scientific paper which they were all going to read in bed; then there was the film which they really ought to see again; and finally, there were the comments which Mr. Douglas had made in giving that film. In his quiet way Mr. Douglas had said many things in his talk which must not be lost, and therefore he was going to ask him to allow the Society to print his talk in the Journal separately from the lecture, which had already been printed, because personally he felt that was very necessary.

Mr. C. R. FAIREY (Past-President): They had listened to a very vivid, practical and interesting talk, but they all had in their hands the print of an equally interesting and even more important lecture, since in it Mr. Douglas proved that high speed in aviation could be obtained without any loss of economy, of safety, or of comfort, as the aircraft on the world's airways had already proved. It was to be hoped that in the hands of our designers this lecture would have some effects on the future of British air transport. If it led to the fact that we never heard again of that strange doctrine under the headline of "the fallacy of speed," it would have done something, although he was never quite clear himself as to whether the protagonists of that amazing doctrine were out to prove that aeroplanes ought not to go fast or to apologize because they did not do so. After all the basic ratio for aeronautical engineers, L/D, had a commercial as well as a technical significance. L was the income and D represented the outgoings, and anything we could do to improve L/D was all to the good of commercial aviation, and Mr. Douglas had certainly done that.

Everybody was grateful to Mr. Douglas for having traveled the long distance he had to deliver this lecture and also for the trouble he had taken in preparing it, and further, for giving them the opportunity of meeting him personally.

The vote of thanks was carried with hearty enthusiasm.

Mr. DOUGLAS, in acknowledging the vote of thanks, said the pleasure he had in coming to England to give this lecture had been enhanced by the great kindness shown to him and the indication that had been given that those who had listened to him had, at any rate, enjoyed the lecture to some extent.

Document 3-25(a-e)

(a) George W. Lewis, Director of Aeronautical Research, NACA, to Mr. R. D. Kelly, Research Supervisor, United Air Lines Transport Corporation, 5936 South Cicero Avenue, Chicago, IL, 3 Sept. 1937, copy in Research Authorization File No. 565, NACA Langley Historical Archives, Hampton, VA.

(b) John W. Crowley, Jr., Senior Aeronautical Engineer, to Engineer-in-Charge [Henry J. E. Reid], Langley Memorial Aeronautical Laboratory, Langley Field, VA, 5 Oct. 1937, copy in RA file 565, LHA.

(c) R. D. Kelly, Research Supervisor, United Air Lines Transport Corporation, Field Headquarters, 5936 South Cicero Ave., Chicago, IL, 6 Oct. 1937, to Dr. George W. Lewis, Director of Aeronautical Research, National Advisory Committee for Aeronautics, Navy Building, Washington, DC, 6 Oct. 1937, copy in RA file 565, LHA.

(d) Melvin N. Gough, Senior Test Pilot, Langley Memorial Aeronautical Laboratory, Langley Field, VA, to Engineer-in-Charge, "Suggestion that conference on 'Stalling' be held with interested personnel outside the Laboratory," 2 March 1938, copy in RA file 565, LHA.

(e) Edward P. Warner, New York City, to Dr. George W. Lewis, NACA, 6 Apr. 1938, copy in RA file 565, LHA.

In September 1937, NACA Langley performed stalling and icing studies with a DC-3 Mainliner passenger transport belonging to United Airlines. In order to warn the pilot of an approaching stall, the NACA engineers installed sharp leading edges on the section of the wing between the engine and fuselage. These sharp edges disturbed the airflow enough to cause a tail buffeting that could be felt by the pilot in his control column. When the pilot felt this buffeting, he knew that his airplane was approaching a stall and needed pilot correction.

Document 3-25(a), George W. Lewis, Director of Aeronautical Research, NACA, to Mr. R. D. Kelly, Research Supervisor, United Air Lines Transport Corporation, 5936 South Cicero Avenue, Chicago, IL, 3 Sept. 1937.

September 3, 1937.

Mr. R.D. Kelly,
Research Supervisor,
United Air Lines Transport Corporation,
5936 South Cicero Avenue,
Chicago, Illinois.

Dear Mr. Kelly:

Thank you for your letter of August 30. I am pleased that the DC-3 airplane will be available for the investigation of stalling characteristics sometimes near the end of September. The small instrument which the Committee has developed for indicating the stall is quite simple, and there will be no difficulty about making another for permanent installation on the DC-3 if you so desire.

In discussing the stalling characteristics with Mr. West, I mentioned the possibility of modifying the contour of the leading edge of the wing near the tip to obtain better stalling characteristics of the wing in front of the aileron. He forwarded to me a blueprint showing the planform and section of the tip portion of the wing of the DC-3. It was thought that we would suggest a section and size of spoiler for the leading edge for installation by you for the investigation. On further discussion of the question with members of our staff, it appears that it would be more desirable for the Committee to make such an installation, as in their opinion one or two positions should be tried for the modification of the wing section.

We have conducted an investigation for the Army Air Corps on a Consolidated two-seated fighter for the purpose of modifying the wing so as to indicate to the pilot the approach of the stall condition. In this particular case it was desirable to place a small sharp leading edge section on the wing near the center of the wing.

Sincerely yours,

G. W. Lewis
Director of Aeronautical Research.

Document 3-25(b), John W. Crowley, Jr., Senior Aeronautical Engineer, to Engineer-in-Charge [Henry J. E. Reid], Langley Memorial Aeronautical Laboratory, Langley Field, VA, 5 Oct. 1937.

Langley Field, Va.
October 5, 1937

MEMORANDUM For Engineer-in-Charge.
Subject: Cooperative tests with the United Airlines Company on the Douglas DC-3 airplane.

1. The tests that were conducted in cooperation with the United Airlines Company on their 21-passenger DC-3 airplane during the week of September 27 have been completed. The tests were primarily for the purpose of investigating the stalling characteristics of the machine and determining methods for improving the same so as to increase the safety of low-speed flight such as is encountered in take-off, landing, and approaches to landing. In addition, tests were made in which ice formations were simulated to determine the effect of ice formation such as is encountered in service on the ability of the airplane to fly and handle safely. Furthermore, tests were made of certain of the general flying and handling and stability characteristics of the airplane and also of the landing and take-off characteristics for the purpose of increasing our fund of information on these characteristics that we are currently accumulating for all modern airplanes for use in establishing what are desirable flying and handling qualities.

2. We found that the stalling characteristics, particularly in the condition in which the airplane is generally flown, i.e., with power on, was definitely undesirable and likely to be dangerous. The airplane stalled violently and with no warning to the pilot of the approach of the stall. The tip portion of the wing at the position of the ailerons stalled first so that lateral control was entirely lost at the final stall. Since in a machine of this type there is no necessity for stalling, it was decided that the most suitable solution would be to provide the pilot with a definite warning that he was approaching the stall so that he could avoid same. To accomplish this we installed sharp leading edges on the section of the wing between the engines and the fuselage. Those were so adjusted that at a few miles an hour before the stall was reached the flow on this part of the wing became disturbed and in passing over the tail surfaces caused a buffeting of the tail that could be felt by the pilot in the control column. These were very successful in definitely warning the pilot of the approach of the stall and were enthusiastically received by the United Airlines personnel. Tests showed that the sharp leading edges had no deleterious effect on the performance of the airplane in level or cruising flight nor did they appreciably increase the stalling speed flight. During the course of the other tests of this machine the development

of a stall warning indicator that mounted on the wing and indicated the approach of the stall to the pilot by means of light on the dash was carried out. While not perfected for this machine the development was carried sufficiently far to show that the principle involved was entirely practicable and that such a device would definitely increase the safety of flight on this machine. The pilots appreciated, in particular, the knowledge obtained that the stalling speed in a turn was higher than in steady flight. The United Airlines personnel, aside from the use of the sharp leading edges, expressed the opinion that they had learned more of the stalling characteristics of the DC-3 from these few cooperative experiments than in all their other flying of this machine.

3. In the tests to establish the effects of ice formation, pieces of sponge rubber were cemented to the forward part of the wing in positions that the United Airlines personnel, from their experience, were able to say was representative of that obtained in actual operation. The airplane was then flown with these in place and the performance of the airplane in steady flight and climb and the stalling speed were measured. In general, it was found that the speed in level flight and the stalling speed were not affected appreciably, but that the climbing performance was noticeably reduced. This was the first direct information obtained upon the effect of ice formation on the performance of the DC-3 and the United Airlines personnel felt that the information was of great value to them.

4. Our measurements of the general flying and handling and stability characteristics in conjunction with the observations of the United Airlines pilots we believe to be very valuable in that we now have a better appreciation of what the transport operators expect and are willing to accept in these respects. It was found definitely that the longitudinal stability was poor.

5. The landing and take-off characteristics measured constitute a valuable addition to our knowledge of these characteristics for large machines.

6. In summary, I believe the information that we received as a result of our tests of the aerodynamic characteristics of this the latest and largest land transport machine now in use, together with the direct knowledge that we received through discussion with the pilots of the problems of transport operation has been invaluable. On the other hand, according to the statements made by the United Airlines personnel, they were highly gratified with the information they have received in regard to a method of providing stall warning, the increased knowledge of the character of the stall, and the information of the effect of ice information on the performance of the airplane.

John W. Crowley Jr.,
Senior Aeronautical Engineer.

Document 3-25(c), R. D. Kelly, Research Supervisor, United Air Lines Transport Corporation, Field Headquarters, 5936 South Cicero Ave., Chicago, IL, 6 Oct. 1937.

UNITED AIRLINES TRANSPORT CORPORATION

Field Headquarters
5936 South Cicero Ave.
Chicago, Ill.

October 6, 1937

Dr. George W. Lewis
Director of Aeronautical Research
National Advisory Committee for Aeronautics
Navy Building
Washington, D.C.

Dear Dr. Lewis:

As you know, we took one of our DC-3 airplanes to Langley Field September 26th to be used there in the course of special stalling tests as per previous arrangements. We remained there all of last week and accumulated considerable data for your own organization and for ourselves. From our standpoint, this trip was extremely successful and we hope that the N.A.C.A. found the investigation to be beneficial to them also.

We were surely accorded every courtesy at Langley Field and the cooperation could not have been better. We feel that the amount of work accomplished was all that anyone could have asked for, particularly since it was the first time that our groups had worked together. We hope that such cooperative investigations may be continued in the future inasmuch as the information obtained should be of benefit to the entire commercial aircraft industry as well as to ourselves.

We have not completed our report of these tests as yet, but we plan to make immediate use of the information obtained by passing on some of the highlights to our pilot personnel at once. We know this information will be very interesting to them and that it will give them a better knowledge of the characteristics of this airplane. Therefore, they will be able to take advantage of those characteristics which were found to be particularly good and to avoid those which were shown to be somewhat more critical.

As you know, ice presents one of our greatest existing problems. Therefore, near the end of the week your people installed small blocks of sponge rubber upon the leading edges of the airplane in an effort to simulate ice deposits. This installation was made at our suggestion and the blocks were located as per our recommendation. By making flights with these in place, we incurred certain losses in performance which coincided with experiences which had been reported to us by our pilot personnel when they had encountered certain types of icing conditions. Had time permitted, we might have extended these tests still further, but inasmuch as the different types of ice formation are so varied, it seemed best to not carry the simulated tests further at this time. While these simulated ice tests were all of a very preliminary nature, they gave all of us a better understanding as to the good which might be accomplished by more thorough simulated tests.

Before we left Langley we had discussed the following arrangement with your engineers, which we hope you will see fit to approve. It was agreed that we should be able to furnish the N.A.C.A. with fairly accurate information concerning the various types of ice formations which are encountered; this information to be based upon actual questionnaires which will be furnished to our pilot personnel and upon actual observations which will be made by our Engineering and Research Organization. When this ice formation information is received, the N.A.C.A. can simulate it by means of plaster paris or any other satisfactory method and apply it to airfoils which can be tested in your wind tunnels. Thus, even the worst conditions can be carried throughout the complete range of flight analysis.

The N.A.C.A. stall indicator appears to us to have very great possibilities and we would like to encourage its development. It seems to us that this unit is about the only feasible method which can be expected to provide a sure warning ahead of the actual stalling condition, regardless of how that condition may be influenced by ice, load, accelerations, etc. While we did not consider the unit to be developed sufficiently that we should retain it on the airplane for service test, we are hopeful that it can be brought up to this point very shortly. We will keep the mounting attachment on our airplane in the hope that you will wish to forward the improved unit to us for further testing. We believe that these improvements can be made through the medium of wind tunnel tests.

We were very much interested in the moving pictures which were shown us concerning the behavior of tufts on the top surface of the wing during actual flight stalls. Some of these pictures were taken from our own airplane and if it is in line with your policy, we would like to purchase a print of these pictures just as soon as possible in order that we may exhibit them to our pilots and engineering personnel for educational purposes.

Again thanking you for this opportunity of working with the Committee and assuring you of our appreciation for the courtesies extended to us while we were at Langley Field, we are

Very truly yours,

R. D. Kelly
Research Supervisor

Document 3-25(d), Melvin N. Gough, Senior Test Pilot, Langley Memorial Aeronautical Laboratory, Langley Field, VA, to Engineer-in-Charge, "Suggestion that conference on 'Stalling' be held with interested personnel outside the Laboratory," 2 March 1938.

Langley Field, Va.,
March 2, 1938.

MEMORANDUM For Engineer-in-Charge.

Subject: Suggestion that conference on "Stalling" be held with interested personnel outside the Laboratory.

1. Much has been written and said by Laboratory personnel in the past 2 years regarding stalling and stall control, and its relative importance. Little has been heard from outside. Either they are awed or disagree. In talking with service personnel and others outside, I have heard <u>both</u> adverse and constructive criticism regarding our relative evaluation of the problem.

2. Feeling that the N.A.C.A. must <u>seek out</u> and be aware of the stand of others on the problem, I suggest that our future activities on the subject might best be guided by the understanding and changed viewpoints which might be gained as a result of a "stalling conference" held with members of the N.A.C.A. staff, representatives of the Army, Navy, Department of Commerce, and others interested.

3. It is suggested that such a conference be arranged.

Melvin N. Gough,
Senior Airplane Test Pilot

Document 3-25(e), Edward P. Warner, New York City, to Dr. George W. Lewis, NACA, 6 Apr. 1938.

EDWARD P. WARNER
New York City
April 6, 1938
Dr. George W. Lewis
National Advisory Committee for Aeronautics
Navy Building
Washington, D.C.

Dear George:

I have returned the copy of the DC-3 stall report that you were good enough to let me borrow. I've dictated some notes suggested by reading it; and if they're somewhat dogmatic in tone, that's to provoke an argument. Incidentally, I may be revealing an embarrassing ignorance of recent developments in the field; but I'll risk that.

The use of a sharpened leading edge on the portion of the wing inside the nacelle is apparently capable of improving the stalling characteristics, but that remedy always seems to me a very unsatisfactory one. Its object is to knock the top off the lift curve on the inner portion of the wing, so producing an early stall in that region, and giving the pilot warning by buffeting; but in so doing it reduces the maximum total lift on the wing, and increases the minimum speed, and it really does nothing fundamental about the stalling characteristics of the tip. What we should work for is not only a delayed stall at the tips, but a more gradual stall when it appears. The use of the sharpened leading edge in flattening off the lift curve inside the nacelle actually produces the shape of curve in that region that we would like to have out near the tips. The sharpened leading edge is then less a means of really improving the stall than a mere stall warning device.

The great virtue in the change of the leading edge of course is that it can be applied on existing ships. In all future development, the stall should be thoroughly investigated during the tests of the prototype, and if it is unsatisfactory, something more fundamental than a leading-edge change on the inner part of the wing should be required. Changes in mean camber near the tips are perhaps the most promising of the more fundamental modifications.

I observed with amazement the statement on page 10 of the report that with power off "the airplane is laterally controllable up to speeds 10 miles per hour above the minimum beyond the stall". If this is literally true, it suggests an immunity of the wing-tips from sudden stalling which is quite at variance with the indications gained in flight with power on; yet the introduction of power is ordinarily expected to have little effect beyond the scouring of the wing in the neighborhood of the nacelle, and the delay of stalling on the inner portion. I ordinarily expect, in other words, that the elimination of power will itself provide a stall-warning, rather than a stall-preventive, device. In this case it appears to be definitely stall preventive. That point would deserve further examination.

One factor in this difference of stall with the introduction of power is no doubt the change of the span-loading curve with the introduction of the slip-stream across the wing. The effect on span loading of bringing in power is similar to that of increasing the taper ratio, and that may be responsible for the sharpness of the tip stall. Even that doesn't seem an entirely satisfactory explanation; though I should be interested in getting the comments of some of your people, on the magnitude of the effect. If that's an important item, it's likely to be inherently worse on a four-engined than on a two-engined ship.

I was much interested in the longitudinal stability measurements on the ship, as recounted on pages 15 to 18 of the report, but the conclusions seem to me really to repeat the original premise, and therefore to be without relation to the special problem of the ship's stalling behavior. It is specifically stated that the tests were made "with the most rearward position of the center of gravity recommended according to the Douglas loading chart". The conclusion is then reached that the longitudinal stability is inadequate, being almost exactly neutral; but it is that very fact that determines where the most rearward position of the c.g. shall be located. It is a pity that some of these tests couldn't be repeated with a more normal center of gravity location. While I agree with all that is said about the undesirable effects of too small a measure of static longitudinal stability, I don't see any way of preventing that quantity from diminishing steadily towards zero as the c.g. is moved to the rear on any ship. We have then in any case to determine how small a degree of stability is acceptable, and that in turn determines how far back the c.g. may go. If, as your report suggests, the static stability characteristic is in fact a major difficulty here (on which point I am a little skeptical, as I understand that there's complaint of the stall even when the load distribution puts the c.g. well forward) the conclusion is not that there is an inherent fault in the airplane, but that the loading charts should be revised so that the c.g. will never be allowed to go as far back as its present rearmost position. I presume that that is what the report was really intended to imply.

In the matter of a stall-warning indicator I agree entirely with Mr. Jacobs (see page 25 of report) that the instrument as set up for these tests was probably more of an air-speed than an angle-of-attack indicator. Nevertheless the idea seems worth following up, for this general type of indicator has the advantage of seeming the only type that will give warning of the advent of a stall directly caused by local icing. If it were not for icing, I should prefer an indicator depending directly on measurement of the angle of attack, like the old Savage Bramson device, but when the ice forms on the wing it changes the conditions under which, and the angle of attack at which, the stall will appear. On the other hand the direct measurement of angle of attack has the advantage of indicating an approach to the condition of stall, whereas the boundary-layer device seems to be capable of indicating only at the time when a stall is actually developing.

Were it not for the special problem of icing, however, I should prefer a stall warning by mild buffeting to any sort of instrumental warning; and if the instrument is introduced it should be primarily to meet the special case of the modification of the stall condition by icing. Preferable to any sort of warning or indication, whether instrumental or inherent in the airplane, is a design of the wing which will not only retard the stall near the tip slightly, as compared with that at the root, but will also ensure as gradual a development of tip stall as possible, corresponding to a very flat top on the section-lift curve.

I was very much interested in the work that had been done on ice deposits as simulated by blocks attached to the wing; and especially in the conclusion that further work on this point could profitably be done in the wind tunnel. I am in full accord on that point.

In short, the stalling problem seems to present six questions.

1. Is it desirable to secure increased warning of the stall on the DC-3 by inducing buffeting with a sharpened leading edge on the wing roots?
2. Assuming that bad stall characteristics might develop on the DC-4, what line of experiment should be planned in advance for use in looking for a solution?
3. Specifically, can modifications of the section or angle of the outer wing be made which would smooth out the tip stall and render it more gradual, in case the flight test of the DC-4 should show that any improvement in characteristics is needed?
4. Is a deliberately induced buffeting a sufficient warning of approach to the stall (assuming that the entry into the stall cannot be made so gradual, and the lateral control be so well maintained, as to obviate the need for any warning), or should it be supplemented with an instrumental indication?
5. In connection with the effect of icing, is there anything to be gained by a change from the type of wing section now in general use to one which may have somewhat lower efficiency, but will be less sensitive to the effect of minor changes of form near its leading edge?
6. If buffeting gives a satisfactory warning of the approach of the stall when there is no ice, may the introduction of ice on the outer portions of the wings nevertheless change the stalling characteristic so that buffeting will become inadequate as a warning, and should be supplemented by an indicator acting directly in the region of danger near the tip?

Sincerely,
Edward P. Warner

Document 3-26(a-b)

(a) Excerpts from Lewis A. Rodert and Alun R. Jones, "Profile Drag Investigations of an Airplane Wing Equipped with Rubber Inflatable De-Icer," NACA Advance Confidential Report, Dec. 1939, copy in LaRC Technical Library.

(b) Excerpts from George W. Gray, Chapter 14, "Heat Against Ice," in *Frontiers of Flight: The Story of NACA Research* (New York: Alfred A. Knopf, 1948), pp. 307-329.

Perhaps an even more dangerous problem for airline operations was icing, a problem to which the NACA would commit considerable time and energy from the mid-1930s on. As historian Glenn Bugos has explained, icing was a critical systems-wide problem during this period of time:

> Ice caused aircraft to crash by adding weight and preventing the pilot from climbing above the icing clouds, so that the aircraft gradually lost altitude and slammed into the ground. . . . [I]ce accreted along the wing and tail leading edges disturbing lift and adding drag. Ice clogged the interstices of rudders and ailerons, preventing control and inducing buffeting. It changed the aerodynamic profile of the propeller, causing it to vibrate and exert less thrust per horsepower. It coated windshields, so the pilot flew blind. Ice made antenna wires oscillate and snap, and generated static that rendered useless most radio communication and navigation. It distorted pitot shapes, so that pilots got erroneous airspeed readings. And it clogged carburetors, suffocating the engine. Frequently, the pilot lost each of these systems—engine, wings, control surfaces, indicators, radio, sight—within minutes (Bugos, "Lewis Rodert, Epistomological Liaison and Thermal De-Icing at Ames," in *From Engineering Science to Big Science: The NACA and NASA Collier Trophy Research Project Winners*, ed. Pamela E. Mack (Washington, DC: NASA SP-4219, 1998), p. 32).

Although the problem of aircraft icing would never totally go away, the comprehensive study of the icing problem begun by the NACA, and continued by NASA, along with its development of different de-icing systems represents a major contribution of government-sponsored research to modern aviation. Its importance was recognized throughout the industry, for the first time in 1946 when Lewis Rodert of the NACA won the Collier Trophy for developing a thermal ice-prevention system.

The documents below concern the genesis of the NACA's icing research program, as well as the NACA's research into the stall problem that was experienced by the Douglas DC-3. In order to simulate the effects of ice formation on the DC-3's performance, the NACA engineers cemented pieces of sponge rubber to the forward parts of the wings, where ice was thought to form most often, and then measured the resulting changes in the plane's climb, cruise, and stalling speeds. In a December 1939 NACA Advanced Confidential Report on these tests, entitled "Profile Drag Investigations of an Airplane Wing Equipped with Rubber Inflatable De-Icer," Langley engineers Alun R. Jones and Lewis A. Rodert (the future Collier Trophy winner for his anti-icing work) emphasized just how big of an impact even a thin layer of ice could have on the lift, drag, and stalling of an aircraft.

It is not surprising to find, in the last document in this string, that aviation dynamo Edward P. Warner was very interested in both the stall and icing problems and how the two related, as he was a special consultant to Douglas Aircraft Company with particular concerns about the operation of large transports.

Although many aircraft systems were involved in the icing problem, the NACA's icing research definitely concerned aerodynamics. As ice built up on a wing, it destroyed lift and increased drag. Control and stability suffered as rudders and ailerons froze up. Propellers vibrated intensely and lost their thrusting power. Simply put, aircraft could crash—and many of them did.

The second document below is an excerpt from aviation writer George W. Gray's 1948 homage to NACA's wartime contributions, *Frontiers of Flight*. Gray examined the problem of aircraft icing and the various solutions that the NACA offered to alleviate it in the period from 1938 to 1948. The excerpt below does not include three (nonconsecutive) sections near the end of his chapter on icing. The first of these omitted sections concerned "Ice and the Engine" (pp. 320-323); the second, "Icing Problems of Jet Propulsion" (pp. 323-324); and the third, "Problems of Cooling and Ventilation" (pp. 328-29).

Figure 1.- Lockheed 12-A airplane equipped with inflatable de-icers.

Document 3-26(a), Excerpts from Lewis A. Rodert and Alun R. Jones, "Profile Drag Investigations of an Airplane Wing Equipped with Rubber Inflatable De-Icer," NACA Advance Confidential Report, Dec. 1939.

PROFILE-DRAG INVESTIGATION OF AN AIRPLANE WING

EQUIPPED WITH RUBBER INFLATABLE DE-ICER

By Lewis A. Rodert and Alun R. Jones

SUMMARY

The National Advisory Committee for Aeronautics has made profile-drag measurements in flight of a wing which was equipped with a rubber inflatable de-icer and to which various simulated ice formations were attached. Tuft observations at the stalling speed of the wing with the various drag conditions were made in order to determine the influence on the maximum lift coefficient.

The de-icer installation caused an increase of from 10 to 20 percent in the profile drag of the plain wing and reduced C_{Lmax} about 6 percent. Simulated ice, when confined to the leading-edge region of the de-icer, had no measurable influence upon the profile drag at the cruising speed. This ice condition, however, reduced the value of C_{Lmax} to about three-fourths that of the plain wing.

Simulated ice in the form of a ridge along the upper and lower de-icer cap-strips increased the profile drag by about 560 percent at cruising speed. This condition reduced the C_{Lmax} to approximately one-half that of the plain wing value.

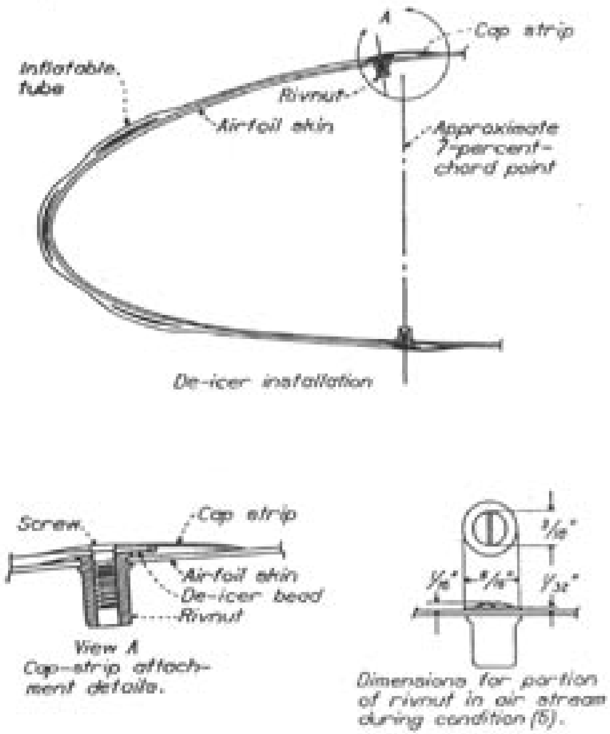


Figure 2. Rubber inflatable deicer details.

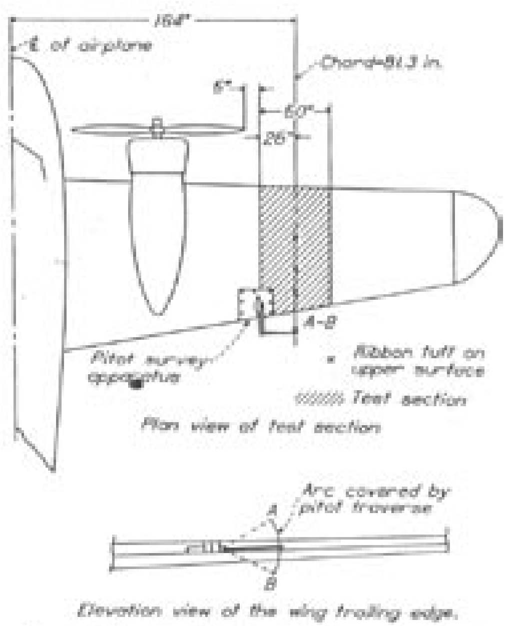


Figure 3. Location of test section and traverse apparatus.

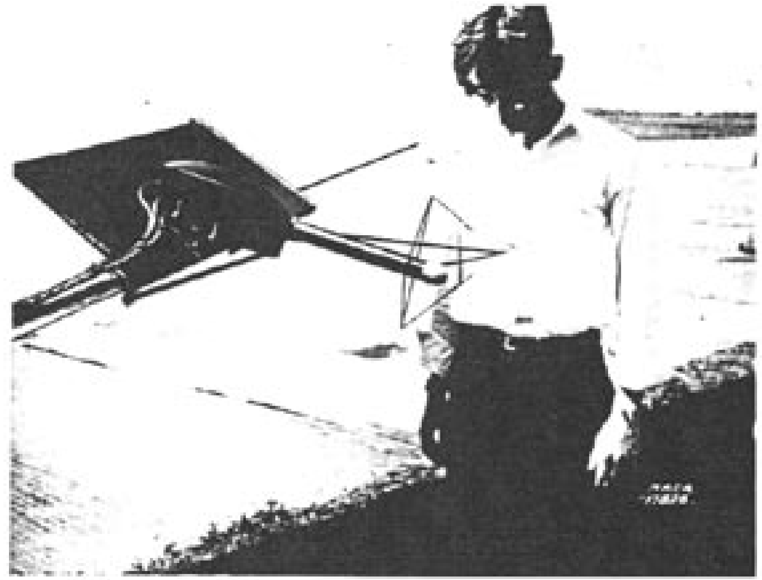


Figure 4.- Wake-survey apparatus installed on the trailing edge of the wing.

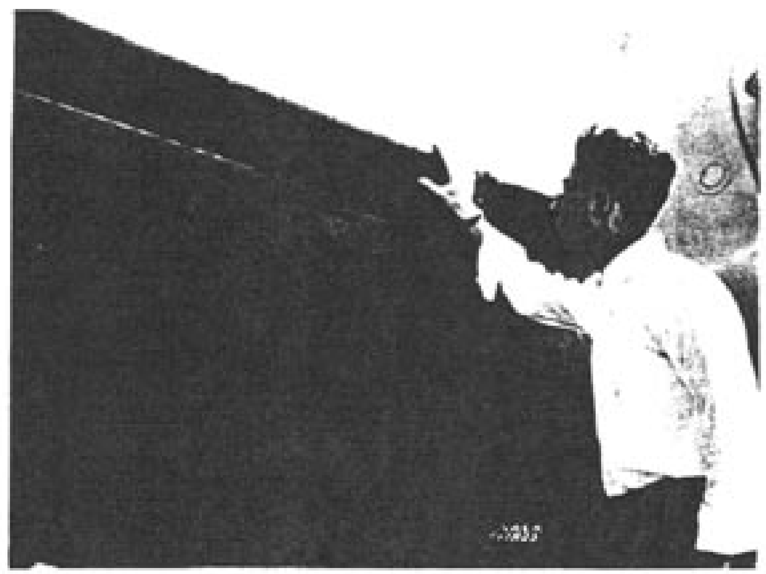


Figure 5.- Rubber inflatable de-icer installation. Drag condition (1).

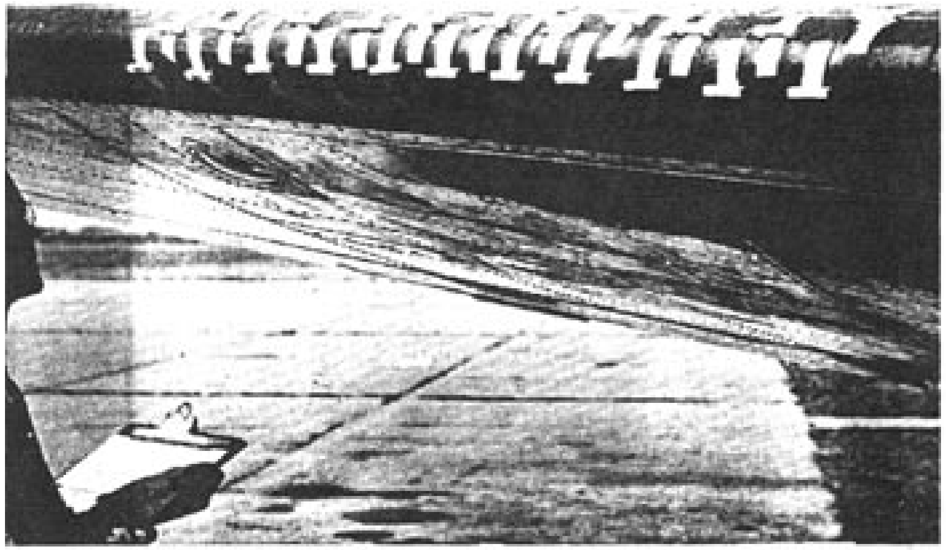


Figure 6.- Rubber de-icer with simulated ice formations on leading edge. Drag condition (2).

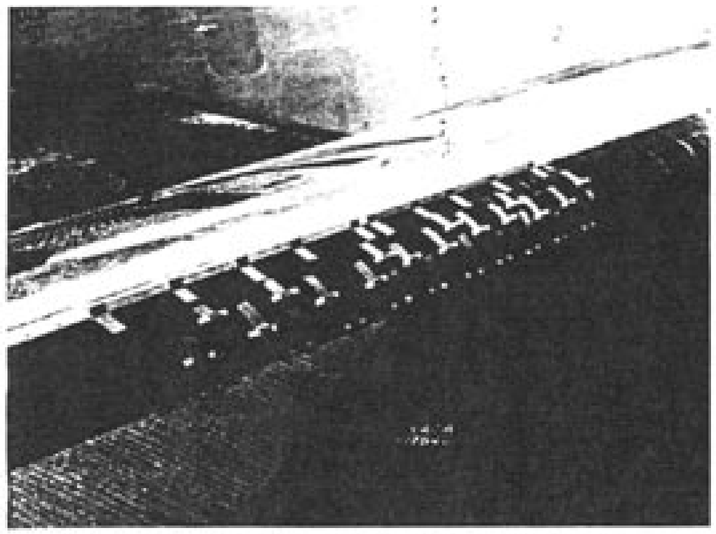


Figure 7.- Rubber de-icer with simulated ice formations on leading edge and with upper and lower cap strips. Drag condition (3).

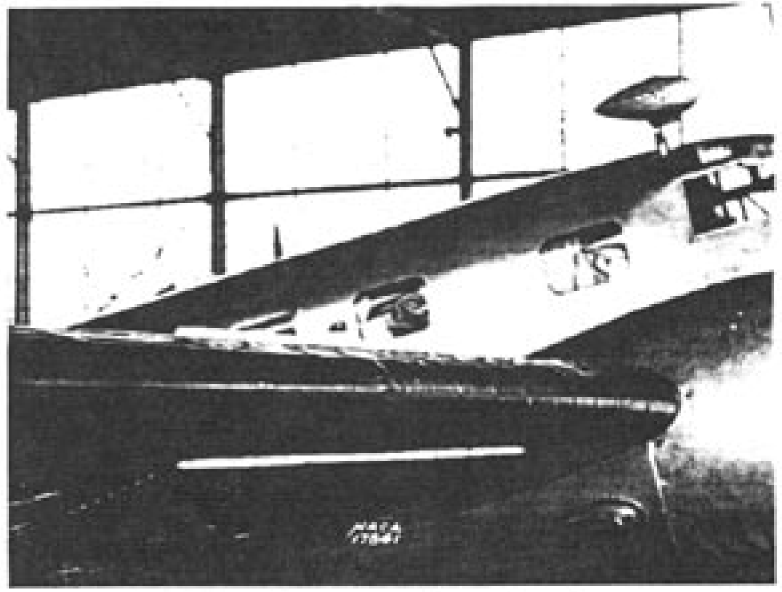


Figure 8.- Rubber de-icer with simulated ice only on cap strips. Drag condition (4).

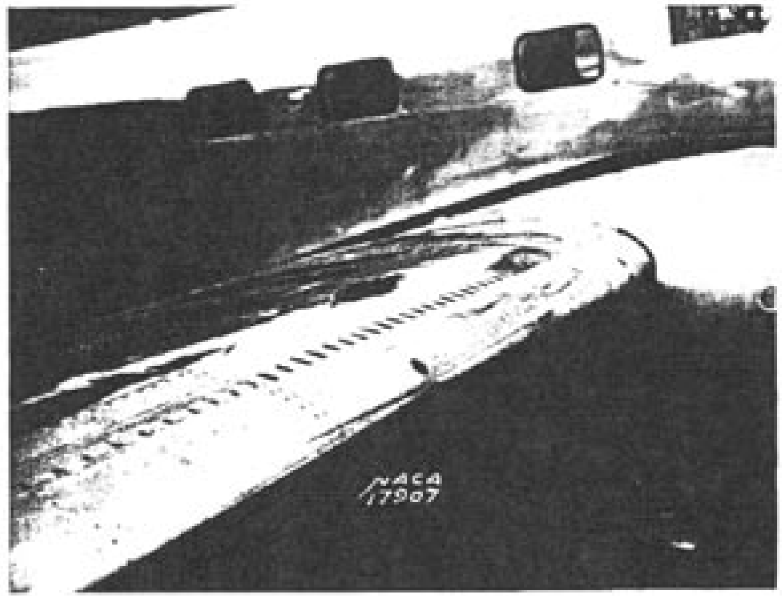
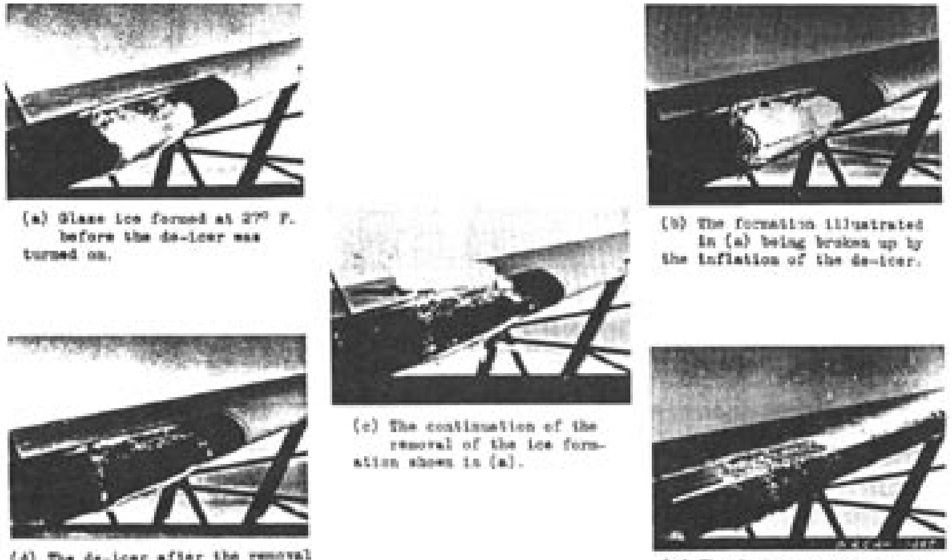


Figure 9.- Rubber de-icer removed and rivnuts faired. Drag condition (6). The unfaired rivnuts can be seen at the left of the test section.



(a) Glaze ice formed at 27° F. before the de-icer was turned on.

(b) The formation illustrated in (a) being broken up by the inflation of the de-icer.

(c) The continuation of the removal of the ice formation shown in (a).

(d) The de-icer after the removal of ice from the leading edge. Note the ice remaining on the upper and lower attachment cap strips.

(e) The de-icer in continuous operation under icing conditions at an air temperature of 27° F.

Figure 10.- The Goodrich de-icer in operation during a test at the N.A.C.A.

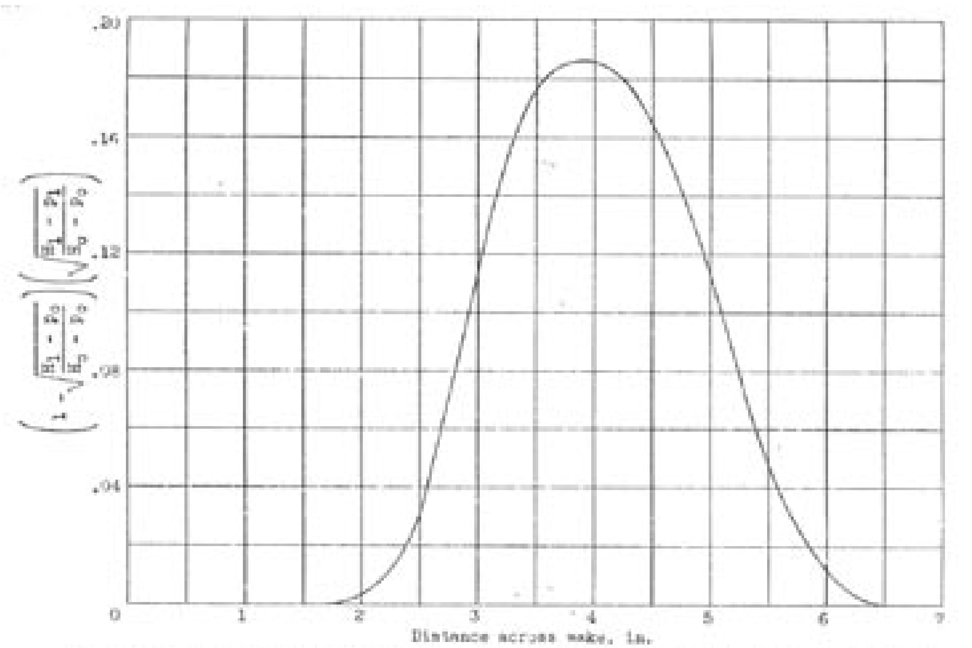


Figure 11.- Wake-survey curve for Lockheed 12-A wing with rubber de-icer and simulated ice on leading edge. Air speed, 175 miles per hour; chord at survey head, 81.3 inches.

INTRODUCTION

The large number of airplanes equipped with rubber inflatable de-icers makes this device of considerable importance in aircraft operation and performance. Inquires have frequently been made relative to the aerodynamic effect of the de-icer installation during flight in normal fair weather, and during icing conditions. A partial answer to these questions has been provided by previous investigations as reported in references 1, 2, and 3. In order to obtain more complete information, the National Advisory Committee for Aeronautics has made flight measurements of the profile drag of an all-metal wing as influenced, first, by a rubber inflatable de-icer installation and, second, by simulated ice formations on the wing loading-edge region. Observations were also made on the approximate effect that the ice simulations had upon the maximum lift coefficient.

The pitot traverse method of measuring profile drag presented a relatively simple and accurate means of obtaining the required drag information (reference 4). Information regarding the lift coefficient was obtained by making tuft observations at the stalling speed of the test section.

APPARATUS AND METHOD

The flight tests were conducted on a Lockheed 12-A airplane which was equipped with inflatable wing de-icers (fig. 1). The details of the de-icer installation are shown in figure 2. The Lockheed wing is an all-metal structure with brazier-head rivets and lap-jointed skin. The traverse apparatus was located on the airplane in the position shown in figures 3 and 4. The necessary mechanism was provided for the movement of the traverse head through the arc AB (fig. 3) in measured increments during flight. To obtain a profile-drag measurement, a survey was made of the stream pressures between the two points A and B which were, respectively, above and below the wing wake. A swiveling air-speed head, shown in figure 1, was placed at the end of a boom projecting from the front of the fuselage, the calibration of which was established by means of a suspended pitot-static head. Pressure recordings of the following were made: (1) dynamic pressure of undisturbed stream; (2) dynamic pressure in the wake at each position of the traverse head; and (3) difference between the static pressures at these two points.

Profile-drag measurements and stalling-speed observations were obtained with the test section in the following conditions: (1) with de-icer, no ice (fig. 5); (2) with de-icer and formations on the leading edge (fig. 6); (3) with de-icer and formations on the leading edge and the de-icer attachment cap-strips (fig. 7); (4) with de-icer and with formations on the cap-strips only (fig. 8); (5) plain wing without the de-icer and with the de-icer attachment rivets unfaired; and (6) with the de-icer removed and the rivets faired (fig. 9). The unfaired rivets of condition (5) can also be seen in figure 9.

Condition (1) corresponds to the state of the wing in normal operation when no ice is present. The simulations for conditions (2), (3), and (4) were obtained by fastening wooden blocks to the wing. The leading-edge blocks were 3/8 inch by 3/8 inch by 2 inches. The cap-strip pieces were triangular in section with ½ inch base and height, and ran continuously along the span of the test section. The simulated ice formations extended 50 inches along the leading edge of the wing. (See fig. 3).

Attention is invited to the fact that the de-icer apparatus on the Lockheed 12-A airplane was installed within the past year and, therefore, is similar in surface smoothness to other de-icers currently in use and installed prior to January 1939. It is understood that various modifications of the de-icer are now under consideration for use during the coming winter (1939-1940). These modifications vary in degree from rather minor changes of existing equipment to completely new equipment for installation on new airplanes and are intended to reduce or eliminate the discontinuity in the wing contour at the rear edge of the de-icer. It is also understood that an improvement in surfacing the de-icer is now possible. These modifications are intended to reduce the profile drag of the plain wing and to prevent residual ice accretions on the leading-edge and the cap-strip regions. Inasmuch as these modifications are still in the development stage, no data are available regarding their effectiveness in eliminating ice accretions.

The determination of the thickness, shape, and location of the simulated ice was based on unpublished reports of flight observations on a rubber de-icer in action and on replies to a questionnaire circulated to transport airline operators. According to these sources, two different types of failure to remove ice are common. Ice may remain in narrow ridges along the leading edge of the de-icer when in operation, or an accumulation of ice may gather on the upper and lower cap-strips, or both types of failure to remove ice may occur simultaneously. Pictures of such ice formations, which were taken by the N.A.C.A. during a recent flight investigation, are shown in figure 10. For condition (5), the de-icer was removed and the rivets (to which the cap-strips are attached) were not faired. This situation corresponds to the normal condition of the wing during the summer. The rivets were then faired (condition (6)) in order to approach as closely as possible the profile drag of the plain Lockheed wing.

Tests were made with each of the six drag conditions at speeds of 125 and 175 miles per hour. Additional tests at 140 and 150 miles per hour were made with drag condition (1). The corresponding Reynolds number range was from 6,500,000 to 9,000,000.

An approximation of the stalling speed of the test section for the various drag conditions was obtained from tuft observations. Ribbon tufts were fastened to the upper wing surface at the points indicated in figure 3. The air speed was decreased slowly until the ribbons indicated that the region of the test section had stalled. The speed at this instant was noted and was used to calculate C_{Lmax} for each condition. These tests were all made with the flaps up and the landing gear retracted.

TABLE I.- DRAG OF WING AS INFLUENCED BY RUBBER INFLATABLE DE-ICER AND SIMULATED ICE FORMATIONS

Drag condition	Description of condition	Test section profile-drag coefficient, C_{D_0}	Airplane lift coefficient, C_L	Reynolds Number	A Profile drag in percent of profile drag of plain wing	B Profile drag in percent of profile drag of wing with de-icer
(1)	De-icer, no ice	0.0105 .0106 .0113 .0116 .0124 .0126	0.23 .23 .31 .36 .44 .47	9.0×10^6 9.0 7.0 7.1 6.6 6.4	110 - - 120	100 - - 100
(2)	De-icer, and ice on leading-edge region	.0105 .0139	.23 .44	9.2 6.6	110 135	100 112
(3)	De-icer, and ice on leading edge and cap-strips	.0435	.23	9.1	458	414
(4)	De-icer, and ice on cap-strips only	.0455	.23	9.2	480	433
(5)	De-icer removed, rivets unfaired	.0099 .0111	.23 .45	9.0 6.6	104 107	94 89
(6)	De-icer removed, rivets faired	.0093 .0095 .0103 .0106	.23 .23 .45 .45	9.5 9.3 6.6 6.7	100 100	91 83

RESULTS AND DISCUSSION

The equation for the profile-drag coefficient as given in reference 4 may be written

$$C_{Do} = 2 \int_W (1 - \sqrt{(H_1 - p_o) / (H_o - p_o)}) (\sqrt{(H_1 - p_1) / (H_o - p_o)}) \, d(y/c)$$

Where:

y	is distance measured across wake.
c,	wing chord.
H_o,	total pressure of undisturbed stream.
p_o,	static pressure of undisturbed stream.
H_1 and p_1,	corresponding values in the survey plane.

Since $H_o - p_o$ equals the dynamic pressure in the undisturbed stream and $H_1 - p_1$ equals the dynamic pressure at the survey head, then $H_1 - p_o$ equals the difference in static pressures subtracted from the survey dynamic pressure. The three pressure readings taken in flight were sufficient, therefore, to evaluate the quantity

$$(1 - \sqrt{(H_1 - p_o) / (H_o - p_o)}) (\sqrt{(H_1 - p_1) / (H_o - p_o)})$$

for each value of y across the wake.

This quantity was plotted against y and, by multiplying the area under the curve so established by 2/c, the value of C_{Do} was obtained. A typical curve of the data is shown in figure 11. During the 125-mile-per-hour runs of conditions (3) and (4), the width of the wake was so great that it could not be traversed by the survey mechanism; therefore the values for C_{Do} were not obtained.

The results of the profile-drag investigation are shown in table I. The repetition of the data which will be noted for some of the drag conditions indicates that check flight tests were made for these conditions. The airplane lift coefficient during each drag test was calculated on a basis of the airplane wing loading and air speed. The profile-drag coefficient of the wing (at $C_L = 0.23$) with the rivets faired was found to be $C_{Do} = 0.0095$. This is assumed to be substantially the same as the profile-drag coefficient of the wing section without the rivets and is called the plain-wing profile drag. The profile drag of the test section at the various conditions is compared with the plain-wing profile drag in column A (table I) and with the profile drag of the wing with de-icer in column B.

Removing the de-icer had the effect of reducing the section profile drag about 5 percent, and fairing the rivets resulted in a further reduction of 5 percent. Placing simulated ice on the de-icer at the leading edge did not measurably increase the profile drag at cruising air speed, although as the speed was reduced this was not the case. At 125 miles per hour, the formations on the leading edge had caused an increase of 12 percent in the profile drag over that of the wing with a clean de-icer. Attaching a simulated ridge of ice on the upper and lower cap-strips and retaining the formation on the leading edge caused the profile drag of the wing at 175 miles per hour to become over four times its normal value. Removing the ice simulations from the leading edge but allowing the ridges on the cap-strips to remain caused a further increase in the profile drag at 175 miles per hour.

The results of the tuft observations at the speed of stall are given in table II. The maximum lift coefficient was calculated on a basis of the wing loading and the indicated air speed at the instant of stalling of the test section. The values for C_{Lmax} at the various drag conditions are compared with C_{Lmax} for the plain wing in column A (table II) and with C_{Lmax} of the wing de-icer in column B.

Obtaining the values for CL by tuft observations is admitted to be subject to error because of the effect of several factors which could not be evaluated in the present investigation. However, the lift data are thought to be of sufficient accuracy to show the approximate effect of the de-icer and the ice simulations on the lift coefficient. The de-icer installation without ice simulations reduced the value of C_{Lmax} below that for the plain wing by about 6 percent. Ice on the cap-strips reduced the C_{Lmax} to one-half the value obtained with clean de-icers.

Although the data were not determined for the landing conditions, the assumption is made that the lift coefficient during landing is similarly affected. A tendency of the airplane to roll during take-off and landing with only the 50 inches of test span affected indicates that the calculated data for C_{Lmax} are satisfactory approximations. The simulated ice formation on the cap-strips necessitated a landing speed between 110 and 115 miles per hour, whereas the normal landing speed is in the vicinity of 70 miles per hour.

The significance of the data in table II and the observations made during the tests as applied to the Lockheed 12-A airplane may be briefly stated as follows:

The increase in the profile drag caused by the de-icer or by ice formations which are confined to the leading-edge region on the de-icer result in a reduction of less than 1 percent in the cruising speed, assuming constant power. At the speed for best rate of climb, however, the increase in profile drag due to the ice on the leading edge will result in a material reduction in the rate of climb.

Although the clean de-icer causes only a slight reduction in C_{Lmax}, the presence of ice on the leading edge reduces this factor to the extent that the minimum safe landing speed in increased by about 14 percent above that of the clean wing. The minimum safe landing speed is calculated on a basis of the stalling speed of the wing cellule outboard from the engine nacelle to the wing tips and on which the de-icers are installed. According to similar calculations, ice on the de-icer attachment cap-strips increases the minimum safe landing speed about 50 percent above that for the plain wing.

CONCLUSIONS

1. The installation of the rubber inflatable de-icers increased the profile drag of the plain Lockheed wing by 10 percent at $C_L = 0.23$ and by 20 percent at $C_L = 0.45$.
2. The attachment of simulated ice to the leading-edge region of the de-icer resulted in a wing profile drag (at $C_L = 0.23$) which was not measurably different from that obtained with the clean de-icer. At $C_L = 0.45$, however, this condition increased the profile drag 35 percent above that of the plain wing and 15 percent above that of the wing with clean de-icer.
3. A formation of simulated ice along the upper and lower de-icer cap-strips increased the profile drag of the plain wing by approximately 360 percent at $C_L = 0.23$.
4. On the basis of tuft observations at the stalling speed, ice on the leading edge of the airplane wing may reduce the value of C_{Lmax} 25 percent, while ice only on the de-icer cap-strips may reduce C_{Lmax} 59 percent below that of the plain Lockheed wing.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics
Langley Field, Va., September 14, 1939.

Document 3-26(b), Excerpts from George W. Gray, Chapter 14, "Heat Against Ice," in **Frontiers of Flight: The Story of NACA Research** *(New York: Alfred A. Knopf, 1948).*

HEAT AGAINST ICE

Ice on the wing or tail adds weight, but more serious is its effect on the drag, lift, and pitching characteristics which may become so adverse as to render the airplane unmanageable. There is also the pilot's need to know at all times where he is, and that involves keeping windshields, cockpit windows, and radio antennas free. Heavy coatings on the antenna impose extra drag that sometimes tears it off. Even if the antenna is not broken, water freezing on the wire and insulators reduces the reception and transmission of the pilot's voice, and without a properly functioning radio the cloud-bound pilot is not only blind but deaf and dumb.

During the war, more than a hundred cargo planes of the Air Transport Command, flying from bases in India over the Hump to battlefronts in China, crashed in the Himalayas. Most of them were brought down by ice. In a single day in 1944, nine of these big Army transports, loaded with sorely needed supplies for the Allies' fighting forces, were lost.

Many of the fatal crashes of commercial aviation have been traced to this same cause. For years commercial transports have been equipped with anti-icing devices, but the apparatus in common use was designed to assist in meeting an emergency when it arises, not to prepare the plane for deliberate flight into ice clouds. If dangerous icing conditions are inadvertently encountered, transport pilots are instructed to turn back or land at a safe alternate airport.

"The greatest advance that can be made in air transport from its present level," said Edward P. Warner in 1946, "is not in speed, or even in economy, important as that is, but in regularity. When cancellations on account of weather are eliminated, or even reduced to a fifth or a tenth of their present number, air transport's whole status will be changed." It is difficult to see how aviation can come into its full destiny as an every-day competitor of steamships and railroads unless airplanes are made capable of flying in any weather.

MECHANICAL, CHEMICAL, THERMAL

Various ways of preventing or removing ice have been experimented with, but basically there are just three methods. In the familiar case of ice encrusting a doorstep, for example, if you hack it with a shovel and scrape it off, that's mechanical ice removal; if you sprinkle salt over the ice, and, by lowering its freezing point, cause it to melt, that's chemical; if you pour boiling water over the frozen surface, or wait for the solar rays to do the job, that's thermal. All three methods have been applied in aeronautics.

The mechanical method is represented by the rubber de-icer which was pioneered by the B. F. Goodrich Rubber Company in the early thirties. Its characteristic feature is an inflatable rubber covering which is placed over the leading edge of wing and tail. When freezing occurs, the pilot can bring the de-icer into action by turning a switch. Compressed air then alternately inflates and deflates the rubber surface, the ice is cracked and broken by these changes in shape, and the wind blows the fragments away. The de-icer has been progressively improved since its introduction in 1930, and by 1939 had become fairly common equipment on commercial transports and long-range military airplanes. There is no doubt that many a dangerous icing of wing or tail has been alleviated by this simple device. But ice rarely cracks off completely or smoothly, and the rough line of cleavage left by the broken ice may seriously increase the drag. Also, under severe icing conditions, the freezing may extend far back over the airfoil, beyond the influence of the pulsating rubber covering of the leading edge.

Chemical methods have been intensively explored both in the United States and abroad. Two schemes have reached application. One makes use of an anti-freeze liquid which when sprayed or exuded over a surface prevents water from freezing there. The liquid most commonly used is alcohol, and for propeller protection the alcohol is released onto the blades by a slinger ring mounted on the propeller hub. The flow over the blade is assisted by a feed shoe, a rubber pad covering the leading edge, and grooves in the pad distribute the alcohol to critical areas of the blade. The alcohol system has been applied also to the protection of carburetors and windshields, and for them as well as for propellers it has proved effective against moderate icing conditions. When the conditions are severe, however, ice can get the upper hand over alcohol.

The second chemical method is the device of coating surfaces with a lacquer or paste whose nature is such as to inhibit the formation of ice. For the most part the lacquers have been applied only to propeller blades, but some tests have been made of their use on the leading edges of wings, and in particular as a coating for the rubber de-icer which covers the leading edge. None of the lacquers provides permanent protection, however, although different compounds vary in their resistance to abrasion and erosion. In most cases, under severe icing conditions, the coating is worn off in an hour or so of operation.

Heat is the most obvious enemy of ice, and the NACA was early impressed with the desirability of trying it. In 1927, a study of the requirements for thermal ice prevention was begun at the Langley Laboratory by Theodore Theodorsen and W. C. Clay, and since then the researches have been expanded and extended to the two other laboratories. There is practically no phase of the airplane icing problem that has not received attention. Tests have been made, not only of thermal systems, but also of mechanical and chemical systems. Both Army and Navy have supported these efforts, and there has also been co-operation with many other government and private agencies, including the Civil Aeronautics Authority, the National Bureau of Standards, the Weather Bureau, operators of commercial airlines, and manufacturers of airplanes, engines, propellers, heat exchangers, and other components. Certain auxiliary researches were carried on under contract by the University of California and the Massachusetts Institute of Technology. Even the enemy contributed unwittingly to the effort, for a captured German airplane which embodied thermal protection for wings, tail, and propeller was sent to the Ames Laboratory for analysis, and certain of its features were adapted, improved and applied to American use.

RESEARCH ON THE GROUND AND IN THE SKY

The early studies at Langley concentrated on the wing. How to keep a wing free of ice was the first problem tackled, and models of wings were built with interior passages for conducting the heating medium. There was then no wind tunnel equipped with refrigeration, and so, when the time came to test these heat-protected models, the men at Langley decided to use the upper air as their refrigerating device. One of the models was placed on the wing of an airplane, mounted firmly a foot or two above the top of the wing on struts, the pipes were connected so that the heat could be turned on in an instant, and then the airplane took off for a high-altitude flight. When the desired height had been reached, water was sprayed on the leading edge of the model and in the below-zero temperature it quickly coated the edge with ice. Then the heat was turned on, and a camera recorded the melting of the ice and how long it took a given amount of heat to free the surface of its coating. In this way, working out changes in the laboratory, and testing them with artificially formed ice in the sky, the practical fundamentals of thermal ice-prevention were derived.

The first requirement of such a system is a source of heat, and that is freely available in the torrent of exhaust gases discharged by the engine. The second need is the ability of the surfaces likely to collect ice to conduct heat, and that need too is already met in the metal skins of wings, tails, and bodies, the glass transparencies of windshields, and the like. It would seem a simple matter to design a system for taking heat from the exhaust and conducting it to the surface where ice is prone to form. But anti-icing, like every other problem of safety, is conditioned by the necessity of keeping the structural weight as low as possible. Moreover, since the heat-conducting pipes must pass through spars, ribs, and other critical elements of the framework, there is also the necessity of guarding against any weakening of the structure.

The problem was attacked from many angles, first using steam as the heating medium. A boiler was placed in the path of the exhaust gases, and water boiled there supplied steam for the pipes leading to the front part of the wing. This arrangement was successful so far as the transfer of heat was concerned, but it added a lot of weight to the airplane. Moreover, the steam often leaked. So steam was abandoned, the boiler discarded, and the next scheme tried was that of piping the hot gases of the exhaust directly to the wing's leading edge. This reduced the weight to a minimum, and surely no heating system could be more direct. The idea was worked out by a group under Lewis A. Rodert, who then was in charge of the icing research program, and by 1938 results were so promising that a full-scale demonstration seemed called for. It was felt that the time had come to install the ice-prevention system in an airplane, seek out ice in the clouds, and determine under the critical conditions of flight the ability of the device to protect the airplane.

A thermal ice-prevention system is not something that can be bolted on, like a windshield wiper. By its very nature it partakes of the structure, and those who proposed full-scale flight experiments realized that to adapt an airplane for such research would involve ripping open the wings, body, and tail and rebuilding them with the added system enclosed and integrated with the structure, a project almost as expensive as the first cost of the airplane. Fortunately the previous experimental results were sufficient to impress both Army and Navy, and each agreed to finance a full-scale installation.

In 1939 the Navy provided funds to remodel a PBY Catalina with the wing-and-tail heating system built into it, and the Army did the same for a Lockheed 12 airplane. The Catalina was flight-tested by the Navy, with results so successful that almost immediately it was assigned to service in tough climates, eventually to Alaska. Meanwhile, the Lockheed 12 remained with the NACA and became the flying laboratory in which Rodert and his young men first tried out full-scale thermal systems under natural icing conditions.

SEEKING ICE IN THE CLOUDS

The reconstruction of the Lockheed 12 with wing and tail protection was completed in 1940, soon after the NACA broke ground for the Ames Laboratory in California. The Committee had already decided to transfer its icing program to the new laboratory on the West Coast; and so, toward the end of that summer, the Lockheed was flown from Langley to Moffett Field; and with it went the icing research staff.

One reason for the transfer was the need for more icing weather than the Virginia skies were wont to provide. The research at Langley too often had to resort to artificial means of imposing ice on the airplane. In California, the interaction of mountain ranges and ocean air provided an infallible mechanism for creating supercooled clouds. W. H. McAvoy and Lawrence A. Clousing were the pilots assigned to fly the airplane into these clouds, and Rodert and his researchers soon found plenty of ice to combat. The flights around San Francisco and along the Sierra Nevadas demonstrated that the heated wings would keep ice off. It was also shown that, if the wings were left cold for a while, only a few seconds heat were necessary to de-ice them. "This," says Rodert, "was the beginning which gave us assurance that eventually airliners would fly from Omaha to Chicago to New York and any other place, irrespective of ice clouds."

Sometimes the ice seekers got more than they were bargaining for. One Sunday morning the Lockheed was riding a storm along the Pacific coast north of San Francisco. "At about 9,000 feet altitude we came out of a large cloud and headed for another," relates Rodert. "By experience we had learned that the severest ice was found in the center of the cloud which looked the blackest. On this day the clouds were tremendous, towering to 17,000 feet. Turbulent air rocked and tossed us. As we headed to the next cloud in the eerie mist between the denser masses, a violent explosion shook us with a blinding flash. The lightning apparently was drawn out of the clouds by the path of our airplane and the blast went right through the craft, stem to stern. Our first thought was fire. We all sniffed. No smoke. Then—did we still have radio communication? Yes—as much as usual in a storm. We went over the airplane and found nothing out of order. After a little hunting for Moffett Field with our radio compass, we landed and were able to make an inspection of the exterior. The lightning had burned metal off the propeller, wing, and tail, but other than requiring minor repair on one or two places, our craft was intact. Airplanes have been struck by lightning many times without any evidence that such events, spectacular as they may be, result in catastrophe."

It was experiences such as this, and the danger of collision in that heavily traveled coastal area, that prompted the group to look for still other skies. Rodert and Cousing were Missesotans, and they remembered that a region northeast of Minneapolis was not traversed by airlines. They knew that a winter there would provide plenty of icing without the extra hazards of mountains and lightning. So in November of 1941 the Lockheed was flown to Minneapolis, and within five days the researchers were delighted with a rich harvest of data. They stayed on, and bucked the icy skies with their hot wings all winter. In January, the Consolidated-Vultee Company asked that the system be applied to their B-24 Liberator, and the Army turned over one of its Liberators for that purpose. Then it was requested that the anti-icing system be installed in the B-17 Flying Fortress. That gave the group two heavy bombers to experiment with, and after their heating systems had been built in, these airplanes were ferried from Ames to Minneapolis and flight tested at Minnesota Ice Research Base. This base, meanwhile, had been taken under the wing of the Army Air Forces and made the cooperative center for icing research, beginning with the winter of 1942-43. That winter there were seven pilots and seventy-five mechanics on duty at the Ice Research Base, in addition to research engineers; and thirteen airplanes with newly installed ice-prevention systems were tested.

In 1943, the Army turned over a C-46 airplane to the Ames Laboratory, and all the research techniques and devices that had been developed in the 9,000 pound Lockheed 12 were used on this 45,000 pound Commando—thermo-couples, strain gauges, air pressure orifices, and the necessary equipment to power the apparatus and record the results. The cabin provided ample space for the complicated equipment with its maze of wires, instruments, and 50,000 watt generator, and there was plenty of elbow room for the crew of three research engineers, two pilots, and a mechanic to move around in. The improvement of thermal systems was rapidly advanced by the use of this large airplane with its extensive instrumentation.

THREE WAYS OF TRANSMITTING THE HEAT

In the course of the flight experiments, several methods of transmitting heat to the surfaces were investigated. The scheme of piping hot gas from the engine exhaust, which had been substituted in the Lockheed 12 airplane for the earlier experimental steam system, was effective so far as heat conduction was concerned, and it added very little weight to the airplane, but in other ways there were disadvantages. In the first place, the hot vapors from the engine carry corrosive acids and other chemicals which have a destructive effect on aluminum and steel. In military airplanes, there was the additional risk that an enemy bullet might puncture the piping and release noxious fiery gases into the wing or fuselage. So the experimenters passed on to a new arrangement employing heated air, and this too was tried first in the Lockheed 12. The idea was to place a heat exchanger in the stream of the engine exhaust, use it as a furnace to heat fresh air drawn from the atmosphere, and then pipe this heated air into the leading-edge region of the wing.

The heated-air system was applied first to wings, and its value demonstrated there before attempts were made to serve other parts. The method was extended to the tail, to protection of the windshield, and finally to heating the cabin. In the case of the windshield, the transparency is made of two sheets of laminated safety glass separated by a space of less than half an inch. It is this intermediate space that receives the heated air. For satisfactory protection, the front of the windshield must be kept at 50 degrees F. How much heat is necessary to maintain that temperature depends on the airplane's speed, among other things. At a speed of 150 m.p.h., a heat flow of 1,000 British thermal units per hour per square foot of windshield surface is necessary. For protection of wing and tail, the system must be able to heat the forward ten percent of their surfaces 100 degrees F. above the outside temperature when flying through dry air speeds up to 200 m.p.h.; surfaces back of the forward ten percent should be heated one-fourth to one-half of this amount.

It was this type of thermal protection—the heated-air system—that the Ames Laboratory applied first to the Lockheed 12, then to the B-24, the B-17, and the C-46.

Compact, highly efficient heat exchangers were developed in the course of these experiments. A design built at the Ames Laboratory weighed only 30 pounds, occupied a cylindrical space eight inches in diameter and twenty-two inches long, and was able to put into the circulating air 315,000 British thermal units per hour—an average of more than 10,000 for each pound of its weight. The exhaust gasses enter the heat exchanger with a temperature of 1,500 degrees F., and there was a problem of finding metals refractory enough to stand up. Not only were they found, but the weight was kept so low that the complete system, including heat exchangers, connecting pipes, and all auxiliary equipment, adds no more than one-and-one-half percent to the gross weight of the airplane; in some cases much less. For example, an installation designed by the Ames group for a four-engine bomber of 60,000 pounds gross weight added 300 pounds, or only half of one percent. This compares with 230 pounds for the less effective inflatable rubber de-icer which breaks off ice after it is formed, and serves only wings and tail, whereas the thermal system is preventive, protects windshield as well as tail and wings, and in addition keeps the cabin comfortably warm.

Still a third method of getting heat from the exhaust to the surfaces has been tried out in the Lockheed 12 airplane, and with very promising results. In this system, air is taken from the atmosphere and mixed with ten to fifteen percent of its volume of exhaust gas, whereupon the mixture is conducted by pipes in the usual way to wing, tail, and windshield surfaces. The exhaust gas is so hot, around 1,500 degrees F., that even the small percentage is sufficient to deliver the mixture to the wing at 300 degrees F. The use of heat exchangers is dispensed with, which simplifies the installation as well as saves weight. Mixing valves are required, but for a four-engine airplane they add only about 20 pounds, to compare with 120 pounds for four heat exchangers. Use of the gas-air mixture thus promises a saving of about 100 pounds on an installation of this size. Experiments in an industrial laboratory and at Ames indicate that the corrosive effect for so small a percentage of exhaust gas is quite minor, indeed negligible when non-corrosive coatings are applied to the inner surfaces.

A still more recent series of experiments is investigating the use of electrical heating. The idea is to build resistance wires into the surfaces of wings and tails, or build them into pads which can be bonded to these surfaces—applying the principle of the electric blanket. For the windshield, the heat is supplied by passing current through a transparent conducting surface film on the windshield glass or through a network of fine electrical resistance wires embedded in the glass. These experiments follow earlier studies in the application of electricity to the protection of the propeller.

ICING RESEARCH IN THE WIND TUNNEL

At the time that headquarters for icing study were transferred from Langley to Ames, plans were being formulated for the Flight Propulsion Research Laboratory whose construction was begun the following January. It was early decided to equip the new research center with a specialized wing tunnel for controlled studies of icing on the ground, something that was needed to check results obtained in flight and also to prospect problems in advance of or independent of the flights studies.

A prime reason for placing the new tunnel at Cleveland was the presence there of the huge refrigerating plane which serves the altitude wind tunnel. This refrigerating plant in the largest "cold factory" in the world, and the installation is such that its full capacity can be turned to cooling either tunnel. Temperatures as low as -65 degrees F., and wind velocities as high as 320 miles per hour can be generated in the icing-research tunnel, and there are three test sections. One, a high-speed section measuring six-by-nine feet across, is used for studying ice problems in an engine, on a wing, windshield, antenna, or other part. The second test section, twelve-by-fifteen feet, is primarily for research on the propeller. The third section was planned to study icing problems of the rotation wing. Artificial blizzards and other freezing conditions can be turned on at will in the tunnel, and most of the icing situations likely to be encountered in flight can be set up in experiments and be investigated quite irrespective of the outside weather.

The tunnel was completed and came into use in 1944, with Wilson H. Hunter in charge. V-J Day thus found both Ames and Cleveland engaged in studies of ice formation and seeking methods of preventing it. The work of the two laboratories was closely correlated. At the height of the wartime effort in 1945 the group at Ames numbered 32 men, including such varied specialists as a chemist, a metallurgist, a physicist, a meteorologist, and an electrical engineer, in addition to mechanical engineers, pilots and mechanics. At the same time 49 men were engaged in icing research at Cleveland, most of them employed in the tunnel. Cleveland also makes flight studies. During the war it used a P-38 Lightning to investigate the effect of the turbo-supercharger on carburetor icing, and it is currently using two Army bombers, a B-24 and a B-25, in icing research. The B-24 has been fitted out as a flying laboratory to study turbo-jet icing; the B-25, with considerable instrumentation, is specializing on the effects on airplane performance of ice formations on propellers and other aircraft components.

PROPELLER PROTECTION

The propeller problem is different from that of wing, tail, and windshield, because it involves a rotating element. Ice on the whirling propeller, in addition to its damaging effect on thrust, can set up destructive vibrations. One requisite in designing an ice-prevention system is to guard against imposing any mass on the blades, or subtracting any, which would unbalance the propeller and be a possible source of resonance and vibration. It is of course essential that anti-icing have no adverse effects on propeller aerodynamics. Other objectives are economy of operation, safety of operation, and pilot convenience. The ideal would be a robot controlled by a thermostat that automatically turned on the system at the first encounter with ice; but this is yet to be.

During the war, the chief reliance was placed on fluid systems which flow alcohol or some other anti-freeze liquid over the blades, in the manner described earlier in this chapter. Not only military airplanes but the commercial transports made wide use of this method. A three-blade propeller, of the type commonly driven by 1,100-horsepower engine, requires about three quarts of alcohol per hour, and that involves carrying a reserve supply of the liquid. There are stories of airplanes lost when gunfire punctured and ignited their tanks of anti-freeze. Moreover, the presence of the feed shoe on the blade affects aerodynamic efficiency; the grooved surface has less thrust than the naked blade, and as a result the take-off requires a longer distance, the rate of climb is reduced, and the airplane's forward speed slowed down. With the trend to bigger broader blades of the paddle type turning fewer revolutions per minute, the adverse effect of the feed shoes on propeller performance becomes even more serious. The NACA has made many studies of fluid systems, first at Langley, and then at Ames, and its staff is convinced that for propellers, as for wings and tails, the best hope lies in the use of heat.

Several thermal systems have been projected, some using the principle of internal heating, others supplying heat only to the surface. For surface heating, two electrical systems are in the testing stage, one using a pad containing resistance wires, the other a special type of synthetic rubber which itself conducts the electric current, and in doing so generates its own heat. These thin heating pads are cemented over the leading edge of each blade, and their smoothness is such that the airflow characteristics of the propeller are not impaired by the added surface. However, the added surface does not remain smooth, for grit, sand, gravel, and other rough particles in the air abrade the pads and make frequent replacements necessary. It is this relatively short life of surface installations that has caused researchers to turn more and more to experiments with internal heating.

Internal heating requires special blades. The NACA's first experiments with this idea made use of hollow blades through which heated air was passed and released from holes in the blade tips. At first it was thought that the tip holes imposed an objectionable drag, but wind-tunnel studies have shown that the effect is inappreciable. The system requires a heat exchanger at the engine exhaust, and in the effort to eliminate this extra complication with its added weight the workers at Cleveland have been experimenting with a scheme for injecting fuel into the blade cavities, burning it there, and discharging the combustion gases through the tip holes, thus heating the blades directly by setting off a fire inside each of them.

Electricity has also been applied to internal heating. For this the blade must be built with a hollowed section under the area to be served, and into the hollow a rubber housing containing the usual electrical resistance wires in installed. Fortunately aluminum and steel, the usual structural materials for propeller blades, are good heat conductors, so any temperatures engendered within the metal quickly communicate their effects to the surface.

A principal drawback to the use of electricity is the large drain that must be made on the power system to generate the current. This is true of both internal and surface systems. Ames found that complete protection of each propeller of the Flying Fortress required about 5,000 watts. Fairly satisfactory protection for all but extreme icing conditions was had with only 2,400 watts per propeller. The former figure seems a heavy tax. It represents a total of nearly 28 horsepower for the four propellers. And, in addition to subtracting that much power from propulsion, it adds the weight of electrical generators which may total 120 pounds, for the four engines. Manufacturers are working on the problem of designing generators of lighter weight, and meanwhile the group in the icing research tunnel has investigated cyclic heating. Picture a system for a four-blade propeller in which the electrical current is on two blades for thirty seconds, then on the other two blades for the next thirty seconds, and so on alternately as long as protection is needed. Such an arrangement would require a smaller generator than a continuous system.

In one of its research flights, the C-46 at Ames had a quick demonstration of what ice can do to propeller performance. On this occasion it became necessary to shut off the electric heating from one of the two propellers while a new combination of instruments was being set up and plugged in. Within less than a minute ice began to form on the unheated propeller, and in the three minutes that were required to change the instrumentation the blade became so encrusted that the airplane speed dropped from 144 to 128 m.p.h. Energetic action was required from the pilot to keep the big airplane under control. Finally the new connections were plugged in, the electricity turned on, and—it was happiness to the crew to see the ice melt away. One minute of heating was sufficient to clear the blades, and the speed returned to 144 m.p.h.

POSTWAR FLIGHT RESEARCH

During the war, icing research was focused on obtaining results that could be quickly applied, and various approximate solutions were worked out. Many of the solutions came to application, and by 1946 thermal ice prevention systems were being installed as part of the standard design of a number of leading transport and cargo planes as well as of long-range bombers. The NACA, however, felt the need of more basic knowledge, and after V-J Day there was a return to fundamental problems. The primary objective was the establishment of more exact design data. It was recognized that an ice-prevention system should be especially designed for the airplane it is to serve, with no excess heating and no unnecessary weight added.

The size and shape of the wing, blade, antenna, windshield, or other part are necessary criteria for designing such a system, but so too are the operating conditions under which the airplane will be flown and the performance to be required of it. For example, its speed affects an airplane's capacity for collecting ice; so does its angle of attack; so does the altitude; and a whole series of weather items enter into the picture—factors such as the liquid content of the air, the water droplet size, the size distribution of droplets in icing clouds, and the temperature of the air. The postwar program has sought precise information on all these criteria, rigorously evaluating the influence of each on ice formation, and measuring the amount of heat required to protect a surface of given shape and size under every combination of circumstances. The end result will be design curves from which may be drawn the exact specifications for protecting any particular airplane.

Flight research has been the primary tool of these studies, and both the Ames and the Cleveland laboratories have been active, employing techniques that were largely pioneered at Ames. The C-46 remains the principal flying laboratory of the Ames group, with Alun R. Jones in charge. But instead of the wartime practice of going to the Ice Research Base above Minneapolis for its skies, the C-46 has been making flights over a triangular course of 1,000 miles—between San Francisco, Seattle, and Salt Lake City—with pilots of the United Airlines at the controls. The

Cleveland laboratory, for its research, has been using the B-24 and the B-25, and these planes are flown as far east as upstate New York and as far west as Fargo, North Dakota, wherever the weather offers the icing conditions best suited for the study. The Cleveland flight research is under the direction of Mr. Rodert, who became attached to that laboratory in 1946.

Much of the B-25 research was concentrated on studies of propeller icing, but the program also included careful checking of other parts as well. In the propeller studies, the rest of the airplane is kept free while the propeller is allowed to gather ice, and then measurements are made to determine how deposits on the propeller affect the performance of the airplane. The research is also concerned with how rapidly the propeller accumulates ice under a given combination of circumstances, how much heat and electric power are required to protect it, how much time is required to de-ice it, and so on. Wing icing is studied in the same way: all other parts are kept warm, and as the cold wing panel collects ice, instruments measure the effect of this on the performance of the airplane and determine in thermal units the minimum amount of heat needed for adequate protection. And so with the nose of the fuselage, the cowl rings, and other parts—the icing tendencies and performance effects of each are appraised in turn, and the indispensables of good protection are determined.

In the same way, Cleveland's other ice-seeker, the B-24, flying over the same regular airline routes between northern New York and North Dakota, has been used for special studies of two critical components: the antenna and the windshield. In the case of the former, a number of experimental antennae of different lengths were installed on the B-24 and set at angles ranging from 0 degrees to 90 degrees with respect to the direction of flight. Strain gauges mounted on the antennae measured the stresses set up by air drag. The increases in drag caused by ice encrustations are quite substantial. In a study in the icing research tunnel at Cleveland, more than 90 horsepower was required to overcome the drag of a heavily iced antenna, whereas without ice the antenna's drag presented only 30 horsepower. Small slender shapes, such as rods and wires, collect ice more rapidly than larger structures, and efforts have been made to design antennae in streamlined form.

While the ice-prevention specialists are concerned with the antenna because of its proclivity for collecting ice, the aerodynamicists are troubled with its drag. And as airplanes go to high speeds, this drag problem becomes increasingly serious. The idea of burying the antenna beneath the surface of the airplane has been experimented with in the effort to eliminate its drag. If installed within the metal skin of the airplane itself, an antenna ceases to function normally, since the metal shields it from radio signals; but there are schemes to make grooves or depressions in the airplane body or tail as housings for the antenna, and then fair these places over with non-conducting plastic. When this idea is successfully applied it will solve both the drag problem and the icing problem.

For studying windshield icing, the nose section of the B-24's fuselage was modified to provide for the installation of test panels of experimental windshields. These electrically heated panels were mounted at various angles of incidence and instrumented with the idea of determining the influence of shape, size, and angle. While the observer, sitting in what was the bombardier's compartment, watches and records these effects, automatic instruments measure the meteorological factors of the atmosphere through which the airplane is passing. In this way the amount of heat required to protect a windshield of given size, shape, and angle is correlated with the ice-making conditions of the air.

The postwar research contrasts with the wartime research in the greater exactitude of its measurements, and this in turn has been made possible by new and more refined instruments. Many of these new instruments were developed by Mr. Jones and his group at the Ames Laboratory, assisted by advice from the U.S. Weather Bureau. For example, it is extremely difficult to measure free-air temperature. Even in dry air such a feat is difficult, and to do it in a cloud of unknown moisture content required no end of ingenuity. "Similarly," remarked Mr. Jones, "the determination of the amount of free water in a cubic foot of cloud through which you are flying at 150 to 200 m.p.h. is a problem to be approached with respect." The co-operation of the Weather Bureau has been an important element of the icing research program, and Ames and Cleveland each has a Bureau meteorologist attached to its staff.

A primary object has been the development of a theory of thermal ice-prevention based on the observed and measured reactions of the heated surfaces to known meteorological conditions. A theory has been proposed by J. K. Hardy of the Royal Aircraft Establishment at Farnsborough, England, and during the war Mr. Hardy spent two years in research at the Ames Laboratory and demonstrated how the dissipation of heat in conditions of icing may be calculated from that in free air. It was interesting to find that his theoretical calculation of the heating requirements of the C-46 wing was in reasonable agreement with the standards adopted by the Ames researchers on experimental indications. Perhaps theory will not make too drastic changes in practice; but good design requires that an airplane be stripped to its essentials and carry no useless weight, and for such calculations a sound theory which has been thoroughly checked and confirmed by experiment is necessary.

The group at Ames has also investigated the effect of the thermal system's heating on the strength and durability of the airplane's structure. Temperature measurements taken of the inner skin, nose, ribs, baffle plate, and other "insides" of the C-46 wing show that they remained at a temperature below 300 degrees F., when the wing's ice-prevention system was operating efficiently in dry air at the conditions for which it was designed. If we assume that the wing will not get hotter than 300 degrees F., assuming also that it is made of the usual 24ST aluminum alloy, it appears that the designer must reckon on a reduction of 11 percent in the tensile strength of the structure.

Heat can also render metal more susceptible to corrosion. An extensive review has been made of the published information on this subject, and it would seem to indicate that so far as previous experience goes no such effect need be feared from 24ST sheet at temperatures below 300 degrees F., provided the sheet has been properly aged.

Heat causes metal to expand, and there is a question how the difference in temperature, for example, between the cooler outer surface and the hotter inner surface of a heater wing's skin may affect its internal stresses. Certain measurements made in the C-46 showed substantial increases in tension, compression, and shear stresses in the wing's leading edge. At some points these internal stresses amounted to 5,000 pounds to the square inch.

There are thus three metallurgical effects of heating to be considered: the effect of heating on the strength of the metal, on its corrosion-resistance properties, and on the internal stresses of skins and other parts whose outer surface is exposed to freezing temperatures. All three problems are receiving continued scrutiny at the Ames Laboratory.

Document 3-27(a-h)

(a) Richard V. Rhode, Associate Aeronautical Engineer, Langley Field, VA, to Mr. Charles Parker, Secretary, Aeronautical Chamber of Commerce, 300 Madison Ave., New York City, "Load Factors--Comments by Richard V. Rhode, NACA, for minutes of meeting held in room 240, Statler Hotel, Detroit, April 5, 1932," 14 Apr. 1932, copy in RA file 287, LHA (In pencil at top of letter, "This letter not sent.")

(b) H. J. E. Reid, Engineer In Charge, to NACA, "Program proposed by Mr. Rhode for determination of dynamic overstress of wings in gusts," 15 March 1933, RA file 287.

(c) G. W. Lewis to LMAL, "Research authorization on study of load and load distribution on commercial type airplanes," 5 Apr. 1933, RA file 287.

(d) Richard V. Rhode, Aeronautical Engineer, Langley Field, VA, to Engineer In Charge, "Proposed daily weather flights and gust tail load measurements on modern airplane," 6 June 1935, RA file 287.

(e) "Extracts from Letter Dated April 12, 1936, From Honorable Edward P. Warner to Dr. J. S. Ames, National Advisory Committee for Aeronautics, With Comments of Dr. D. M. Little, U.S. Weather Bureau," 12 May 1936, RA file 287.

(f) Edward P. Warner, 122 East 42nd St., to Dr. J. S. Ames, NACA, Navy Buillding, 18 Apr. 1936, RA file 287.

(g) Dr. G. W. Lewis, Director of Aeronautical Research, NACA, to Mr. L. V. Kerber, Chief, Aircraft Worthiness Section, Bureau of Air Commerce, Department of Commerce, Washington, DC, 26 Feb. 1938.

(h) Interview transcript, Richard V. Rhode to Michael D. Keller, Hampton, VA, 10 Jan. 1967, pp. 1-6, copy in LHA.

This string of documents on the early days of the NACA's gust loads research represents another major nexus in the airplane design revolution—one linking aerodynamics, meteorology, and structures. It also shows the importance of instrumentation—in this case the V-G (Velocity-Gravity) recorder, an instrument that recorded speed and the acceleration along one or more of the principal axes of an aircraft. This was not the first—and certainly not the only important flight instrument; by that time, there were also accelerometers, strain gauges, pitot tubes, and motion-picture cameras. But the V-G recorder certainly proved to be an essential instrument, one that provided key data guiding aircraft design. In essence, it was aviation's first "black box."

NACA Langley engineers Richard V. Rhode (pronounced "Road-ee") and Henry J. E. Reid (Langley's Engineer-In-Charge from 1926 to 1940) invented this crucial new recording instrument in 1929-30. What led to their design was a series of thoughts about the relationship between atmospheric gusts and airplane aerodynamics. Flight researchers investigating gusts noticed that when an airplane encountered a gust, the airplane behaved just as if it had changed its angle of attack. Changes in angle of attack changed lift, causing the airplane to move vertically; analysis showed that how much of this vertical "acceleration" took place was a function of the airplane's speed and the degree to which the angle of attack changed. Rhode and Reid recognized that the relationship between velocity and acceleration due to gravity could be computed—better yet, the aircraft itself could be used to measure the gusts encountered. The NACA engineers also understood that larger aircraft were generally less maneuverable than smaller ones. Both large and small aircraft would inevitably encounter gusts brought on by thunderstorms and other bad weather, but it was undoubtedly more critical to design the right amount of strength into large aircraft, transports in particular, because their greater size and versatility resulted in more strenuous gust loads. With these conclusions in mind, Rhode and Reid built their first V-G recorder. They screwed on the instrument, which was about the size of an old-style alarm clock, to a firm structural part of the airplane near its center of gravity. It then automatically recorded the greatest accelerations

encountered by the airplane in relation to its speed. The last document in the string below is an excerpt from a 1967 interview with Rhode, in which he recalled the invention of the V-G recorder.

Though developed for NACA flight research, the V-G recorder became ubiquitous. A number of U.S. airliners began carrying them in the 1930s, including several trans-continental passenger planes. The famous Clipper flying boats of the era all carried V-G recorders on their flights across the Pacific to Hawaii, the Philippines, and China. In World War II, thousands of U.S. and Allied aircraft (a number of the instruments were lent to the British Air Ministry early in the war) came equipped with the instrument. The V-G's contribution to airplane design proved substantial. When the scratchings on the blackened disks of the instrument were examined, sometimes after months or even years of service, the influence of gusts and all the other vicissitudes of flight were recorded precisely in terms of speeds and inertial loads. In doing that, the V-G recorder not only told the story of what happened to an aircraft in an accident or fatal crash but also generated data that was extremely helpful in aircraft design. Before the end of the 1930s, there was enough data on gust loads for U.S. aviation regulators to establish a design criterion of 55-feet per second "effective gust velocity."

As this string of documents suggests, the NACA pioneered the field of gust loads research during this period. In 1936, it purchased two light airplanes—a Fairchild and a Stinson Reliant—and with them Langley flight researchers studied the structure of what they called the "natural gust." In 1937 Langley also started to design a ground facility in which the effects of gust gradient on aircraft response could be measured. Not completed until after World War II began, this facility consisted of a vertical jet of air for simulating a gust, a catapult device for launching dynamically-scaled models into the air jet, curtains for catching the model after it traversed the gust, and instruments for recording the model's responses. By 1940, the NACA had elicited several different means by which to improve an airplane's response to gusts, including gust "alleviators" that did such things as affect longitudinal control through variations in elevator angle.

Without question, gust loads research offered information critical to the design and operation of all American aircraft designed from the mid-1930s on.

Document 3-27(a), Richard V. Rhode, Associate Aeronautical Engineer, to Mr. Charles Parker, Secretary, Aeronautical Chamber of Commerce, "Load Factors," April 14, 1932.

April 14, 1932.

Mr. Charles Parker, Secretary,
Aeronautical Chamber of Commerce,
300 Madison Ave.,
New York City

Dear Mr. Parker:
Subject: Load factors—comments by
Richard V. Rhode, N.A.C.A., for minutes of meeting held in room 240.
Statler Hotel, Detroit, April 5, 1932.

We believe the line of attack being followed by the Department of Commerce is proper and rational in so far as the correlation of the loads with the speed and lift coefficients is concerned. The question of the value of the total loads that are likely to be applied in flight is, however, still entirely unsettled and probably will remain so for sometime to come.

It is my opinion that the stresses in gusts will ultimately be found to be the important factor for the time design of civil airplanes. I do not believe that stresses in the ordinary maneuvers exceed the maximum gust stresses. A great deal of accelerometer data show very conclusively that the "common garden variety of maneuver" does not impose accelerations greater than about 3g even for single and two-seated airplanes. On the other hand, accelerations in gusts as high as 3g have been measured. These accelerations, however, do not necessarily represent the stresses. It is a fact that in the more serious gusts—those with relatively steep velocity gradients—there is an excess stress due to the dissipation as internal work in the structure of the kinetic energy gained by the wing as it deflects with respect to the main mass of the airplane. This "dynamic overstress" may vary from 0 to 50 percent or more of the stress due to the applied acceleration considered as a static load. Thus if an acceleration of 3g is measured in a gust, the stress may be anywhere from that caused by a static load factor of 3 to that caused by a static load factor of 4-1/2 or more. The exact overstress depends on the gradient of the gust and the flexibility of the wing structure, as well as on the speed of flight and the gust intensity. These matters at present are very imperfectly understood, and no attempt to calculate them with precision is justified because of the lack of sufficient definite information on gust gradients.

The N.A.C.A. has purchased an instrument for measuring structural deflection in flight and also strain gauges suitable for use on airplanes in flight. It is hoped that by making use of such instruments in conjunction with a combined air-speed accelerometer developed by the N.A.C.A. some definite conclusions regarding gust stresses may be formulated from the statistical evidence gathered.

The most enlightening information on loads at present in the possession of the N.A.C.A. is applicable almost entirely only to military aircraft and in particular to aircraft required to undergo the most severe military tactics. This information has not been published, although the N.A.C.A. contemplates doing so in the near future. From the standpoint of the design of civil aircraft this information will be of value only as applied to a special acrobatic or unrestricted category. I think it will demonstrate rather clearly that the critical load factors are determined primarily by physiological and psychological considerations of the pilot and that a logical design will proceed from a knowledge of the applied load factor coordinated with certain characteristics of the airplane, such as speeds and wing loading. I am tempted to believe, from the limited data available thus far, that for all practical purposes the probable maximum applied load factor will be found constant or nearly so throughout a range of speeds from V_{max} to $V_{terminal}$, with the low-angle load factors somewhat higher, perhaps, than the high-angle factors. Only further statistical data, however, will definitely prove or disprove this opinion.

Yours very truly,

Richard V. Rhode,
Associate Aeronautical Engineer.

RVR.IT

EWM

Approved:

H. J. E. Reid
Engineer-in-Charge.

Document 3-27(b), H. J. E. Reid, Engineer-in-Charge, to NACA, "Program proposed by Mr. Rhode for determination of dynamic overstress of wings in gusts," March 15, 1933

Langley Field, Va.,
March 15, 1933.

From LMAL
To NACA

Subject: Program proposed by Mr. Rhode for determination of dynamic overstress of wings in gusts.

1. There is forwarded for your information and approval a memorandum prepared by Mr. Rhode suggesting a program of flight tests involving measurements of accelerations and wing stresses in gusts. Mr. Crowley is of the opinion that this work is an essential step in the interpretation of the load factors they are attempting to establish, particularly on transport airplanes. He feels that this program will enable us to maintain an advanced position in this field of work, as compared with the work of other laboratories.

2. Our investigation of load factors and gusts involving the use of V-G recorders on commercial airplanes has all been charged to R.A. 287, but it is realized that we have gone beyond the scope originally intended in this authorization since R.A. 287 provides for an investigation on our Fairchild cabin monoplane. It is believed that R.A. 287 should be replaced by a new research authorization, or that the method of procedure should be revised somewhat as follows: "Flight tests are to be made on a number of commercial airplanes in which pressure distribution, stresses, and load factors will be measured as may be desirable. These flights will be made in all weather conditions and over all types of terrain. The main source of information will be transport airplanes in regular operation on established airways."

H. J. E. REID

Engineer-in-Charge

Document 3-27(c), G.W. Lewis, Director of Aeronautical Research, to LMAL, "Research authorization on study of load and load type distribution on commercial type airplane," April 5, 1933.

Washington, D.C.
April 5, 1933.

From NACA
To LMAL

Subject: Research authorization on study of load and load type distribution on commercial type airplane.

Reference: LMAL letter, March 15, EWM.DD, paragraph 2.

1. Consideration has been given to the question of reference as to the suitability of continuing to charge to Research Authorization No. 287 the work on the study of load factors and gusts on commercial airplanes in actual transport operation by the use of the N.A.C.A. V-G recorder.

2. In view of the fact that the work the Committee is actually doing on this problem at the present time comes within the scope of the title and purpose of the investigation as stated in the research authorization, it is believed unnecessary to request a new authorization or an extension of No. 287 to cover the very different procedure being followed in conducting the investigation. Since it was not possible at Langley Field with the Fairchild cabin monoplane to obtain information regarding the load factor on commercial airplanes under the varying conditions encountered in actual operation, approval was given by this office, as within the scope of administrative discretion, to the extension of the procedure to airplanes in operation in commercial air transport and the study of accelerometer and V-G recorder records obtained in such operation.

G.W. Lewis.
Director of Aeronautical Research.

Document 3-27(d), Richard V. Rhode, Aeronautical Engineer, to Engineer-in-Charge, "Proposed daily weather flights and gust tail load measurements on modern airplane," June 6, 1935.

Langley Field, Va.,
June 6, 1935

MEMORANDUM For Engineer-in-Charge.

Subject: Proposed daily weather flights and gust tail load measurements on

modern airplane.

1. The increasing importance of the atmospheric gust as it effects many phases of aeronautics makes it desirable to prosecute research on gust phenomena more intensively than we have done in the past. There seems to be no question that transcontinental transport flights will be carried out at altitudes upwards of 10,000 feet. The question of the applicability of our transport V-G data, which have been obtained at relatively low altitudes, at these higher altitudes therefore arises. It is also of increasing importance to know the relations between gustiness and types of weather in addition to the relation with altitude.

2. While we have attempted in the past to obtain some of these relations by (a) asking transport pilots to fill out information forms to accompany the acceleration records, and by (b) cooperating with the Weather Bureau in obtaining acceleration records on their daily airplane flights at Cleveland, these attempts have proved unfruitful. In the first case, pilots have failed to cooperate, and in the second case, suitable accelerometer equipment for the conditions of the job was not available.

3. I believe, however, that we should and can obtain useful information along the lines mentioned by making more or less regular daily flights at Langley Field with certain equipment as set forth below.

4. It is proposed, therefore, to use a modern airplane carrying our optograph equipment modified to take certain pertinent measurements as follows:

a. Acceleration.
b. Air speed.
c. Temperature.
d. Pressure.
e. Humidity.

Certain other measurements may be desirable, but those above are fundamental

to the main research. The airplane would be climbed daily to 17,000 or 18,000 feet with some few short level flight runs at intermediate altitudes. It would be necessary to equip the airplane with blind flying instruments, including radio, as are now used by all military and commercial airplanes, so that cloudy or overcast conditions would not interrupt the flights. Oxygen equipment is also desirable.

5. It is proposed also to extend our gust tail load measurements to this airplane, tasking measurements along with the measurements listed above. The present gust tail load project on the O2H was undertaken primarily to test the feasibility of the method, with the understanding that the O2H was not entirely suited to the job and would be useful for only a limited period. It was anticipated that, if the method proved feasible, the tail measurements would be continued on a more suitable airplane. It is believed that the preliminary results on the O2H indicate both the feasibility of the method and the importance of gust loads on tail surfaces, and hence justify further research on a modern airplane.

6. The airplane desired is one of modern type and performance. From the standpoint of the tail load measurements, it should be similar to modern and probable future transport types; via., a low-wing tapered monoplane. This requirement is not essential from the standpoint of the acceleration and weather measurements. It is therefore proposed to start the weather and acceleration measurements on the XBM-1 airplane; this work can proceed immediately. It is also proposed that the committee take steps to procure a fast low-wing monoplane on which to continue the weather measurements and also on which to make simultaneous measurements of gust tail loads. A Vultee transport would be ideal, although one of the new Northrop attack airplanes ordered by the Army would be suitable. In this connection it seems worthwhile to point out that if a cabin job were procured, it could also be used to carry on the present functions of our Fairchild FC-2W2, which is rapidly becoming obsolete and worn out. The installation required for the researches herein mentioned would not interfere with the cabin space or use of the airplane for transport work.

Richard V. Rhode,
Aeronautical Engineer.

Document 3-27(e), Extracts from letter dated April 12, 1936 from Edward P. Warner to Dr. J. S. Ames, National Advisory Committee for Aeronautics, with comments of Dr. D.M. Little, United States Weather Bureau.

EXTRACTS FROM LETTER DATED APRIL 12, 1936, FROM HONORABLE EDWARD P. WARNER TO DR. J. S. AMES, NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS, WITH COMMENTS OF DR. D. M. LITTLE, UNITED STATES WEATHER BUREAU

"The most important of aerodynamic problems from the transport point of view seem to be those lying midway between aerodynamics and meteorology, and perhaps deserving the special notice of the Subcommittee on Meteorological Problems as well as that of the main Aerodynamics Committee. Specifically, it is of the greatest importance that all possible information be secured, and that existing knowledge be extended with all possible rapidity, on the subject of the structural effects of atmospheric disturbances. The type of indirect analysis of gust structure upon which Mr. Rhode and the flight section at Langley Field are currently engaged (seemingly in a comparatively small way) has a potential value that cannot be exaggerated. The transport operators in attendance at the recent meetings assured their fullest cooperation in the installation of instruments on their airplanes, and in the keeping of such records as would enhance the value of the instruments' indications and facilitate their interpretation. I am supplementing discussion at the meeting with a letter to each member of the Committee urging further that they make it a practice to submit to the N. A. C. A. a report of any actual experience with possible structural implications that any of their pilots may encounter, such as an unusually violent bump, or an apparent rolling, pitching, or yawing moment of extraordinary intensity due to atmospheric disturbances. If, in all such cases, proper records can be secured of the attendant conditions, it will put in hand a powerful tool for correlating structural effects with identifiable meteorological phenomena and also for determining the degree in which the worst conditions that happen to have been recorded in a somewhat limited amount of recording can, in fact be taken as representative of the worst that need ever be anticipated in design."

Perhaps the Weather Bureau could assist in this by suggesting that V-G recorders be installed on the airplanes taking upper air soundings. We would even undertake to change the glass slides once a day if Mr. Rhode would be interested in getting individual records for each sounding for use in correlating it with the meteorological data.

"There is another matter that touches both on meteorology and on aerodynamics—the question of ice formation and its effects. I am sending you a separate memorandum on the general subject of the ice problem, and the possible organization of research thereon, but there are some obvious aspects that are almost purely

aerodynamic. One useful job for the full-scale tunnel would be a study of the effect of de-icers, both in their deflated and in their operating condition, on the drag and the stalling characteristics of wings. Another would be an investigation of the temperature drop of air flowing around a wing. Where moisture in the air forms as ice on an aerodynamic surface, it is presumably the result of a change of temperature on contact with that surface. If there is to be any possibility of using heat effectively in the prevention of ice, data are needed on the exact conditions under which heat ought to be used and the exact amount that ought to be applied. Even if heat is considered out of the question, the investigation of temperature changes would still be worthwhile as providing a basis for determining the probable locations of ready ice deposit. Experience indicates that certain types of control surfaces, for example, accumulate ice, while others do not. It suggests the need for particular care in the design of slotted controls, the more especially as de-icers can hardly be used in a slot. While of course much of this work can only be done in a refrigerated tunnel where practical tests of actual ice deposit can be made, there is more that can be done in any kind of a flowing stream if ice formation can be redefined in more general terms as a thermodynamic problem."

Considerable icing data are being accumulated by the Weather Bureau in its program of upper air soundings. We have in mind tabulating and correlating the data with the meteorological records at a later date. The work can probably be started next year as soon as other pressing projects are out of the way. It will then be possible for Mr. Samuels to bring up to date his work on the meteorological aspects of the icing problems.

"Another meteorological problem on which the transport operator badly needs light is the range of variation of wind conditions in the upper air. While the Weather Bureau is badly handicapped by having to depend on visual observation of its balloons, it is my understanding that more data exist than have ever been put into a usable form, and that the funds for personnel for interpreting and qualifying them have been lacking. I hope that the Aerodynamics Committee or its meteorological Subcommittee can take an active interest in sponsoring that work, and do whatever lies within its power to encourage its active prosecution."

A detailed summary of all pilot balloon observations (Army, Navy and Weather Bureau) made in the United States previous to 1923 was published in the Monthly Weather Review Supplement No. 26. This summary was based on levels above the surface and confined to the area east of the Rocky Mountains. Another summary showing wind rises, resultant winds and average velocities for individual stations by months and for the levels 750, 1500, 3000, and 4000 meters above sea level is being prepared and the first section covering the eastern third of the United States has been published in Monthly Weather Review Supplement No. 35. The data for the remainder of the country will be ready for publication in about six months. A third summary of winds at high levels (6, 8, 10, 12, and 14 kilometers) by seasons and for individual stations is now being prepared and will probably be completed in about three months. Considerable work has also been done in tabulating the frequency of different upper air wind velocities, grouping them by directions and the velocity limits as recommended by the International Commission for Air Navigation. It will require about one year to prepare these data for publication. Upper air wind data have been tabulated for all ocean areas with a view to publishing a summary of winds over the oceans and for island and coastal stations. A great amount of work is yet to be done on this project, however, and without outside assistance, it will probably require two or three years to complete it. It is proposed also, when time permits, to compute and publish in great detail, upper air wind data for the stations having the longest records. Another study which should be undertaken is the grouping of upper air winds along with other meteorological elements according to different air masses.

Document 3-27(f), Edward P. Warner to Dr. J. S. Ames, National Advisory Committee for Aeronautics, April 18, 1936.

CONFIDENTIAL
COPY
ONE HUNDRED TWENTY-TWO EAST FORTY-SECOND STREET
NEW YORK CITY

April 18, 1936

Dr. J. S. Ames,
National Advisory Committee for Aeronautics,
Navy Building,
Washington, D. C.

Dear Dr. Ames:

Among the subjects considered at the recent meeting of the special Sub-committee on Problems of Transport Construction & Operation were one or two which I believe might be of interest to the Committee on Structures & Materials, and might provide it with some suggestions for lines of work which would have a special value for air transport. Though the special sub-committee took no formal action to the extent of requesting specific researches by resolution, I am taking the initiative as its chairman in rendering a report of the apparent sense of the meeting, and in including also some points of which lack of time forbade any discussion at the meeting, but of the significance of which I am persuaded by personal conference outside the meeting with members of the group who were in attendance and with others active in the transport field.

Distinctly the most important of questions is that of load factor due to atmospheric disturbances. It has been our practice up to the present time to assume a constant factor, based on an assumed constancy of vertical gust. With the development of very large airplanes and of airplanes having a large part of their weight distributed along the wings, the simple assumptions of the past become less tenable. The possibility of change of stress or the introduction of new types of loading through the action of gusts having linear dimensions small compared with those of the airplane becomes important. In this connection it is important also that we have some direct appreciation of the actual extent to which the effective strength of an airplane is increased by the suddenness with which the gust load is applied. I am not speaking now of the dynamic overloading factor but of its converse, the well-known ability of structures to stand a larger load very quickly applied and very promptly removed than they can sustain under continued applications. The more complex the internal organization of the structure and the larger the part played by internal friction during deformation, the more important that factor becomes. At the present time structures are conventionally designed for a factor of safety of 1.5 and for a gust velocity of 30 feet per second. We have evidence that gusts approaching 50 feet per second have actually been encountered at times, yet I am personally cognizant of no single case of apparent loading beyond the yield point in a modern monocoque structure. The reconciliation of these apparently conflicting facts ought to be of marked interest to the Structures Committee.

Of marked interest also is the question of the structure of fuselages carrying internal pressure. I do not propose that for any specific investigation under the Committee's auspices at the present time, for I understand that the Army has it in hand and is carrying forward as rapidly as possible. Nevertheless it might be a subject that the Committee would find interest in discussing, and upon which some cooperative grouping of researches by the various governmental agencies may in due course be desirable.

The matter of structural loading in gusts of course involves structures and aerodynamics jointly. I have already suggested its reference, from a slightly different point of view, to the Aerodynamics Committee.

Sincerely,
Edward P. Warner

Document 3-27(g), C. W. Lewis, Director of Aeronautical Research, to Mr. L.V. Kerber, Chief, Aircraft Airworthiness Section, Bureau of Air Commerce, February 26, 1938.

February 26, 1938

Mr. L.V. Kerber
Chief, Aircraft Airworthiness Section,
Bureau of Air Commerce,
Department of Commerce,
Washington, D.C.

Dear Mr. Kerber:

I have referred to our laboratory for the consideration of Mr. R. V. Rhode your letter of February 9, 1938, inquiring as to the extent to which the values of gust-load factors for airplanes of high wing loading recommended in his recent report have been confirmed by data obtained from N.A.C.A. V-G recorders installed in commercial airplanes.

Mr. Rhode states in reply that the factors recommended in the report cannot be said to be confirmed by evidence from the V-G records, nor can they be said to be not confirmed. Unfortunately, variations in the maximum effective gust velocities determined from the V-G records depend to so great an extent upon operating policies of the air lines routes flown, the number of hours of flying time recorded on each airplane type, and unreliability and inconstancy of the weather, that the effects of airplane variables are completely masked.

It is the Committee's viewpoint that the program of the collection of records from N.A.C.A. V-G recorders provides a general view of gust-load conditions under which the various transport airplanes operate. We believe it is important to continue to install the recorders in airplanes of new types, especially those that differ appreciably from previous or current types in important respects such as wing loading or size, but we have not yet obtained data as a result of these records that could be called comparable in respect to the external variables. It is therefore the opinion of the Committee that theoretical and gust-tunnel investigations are needed to supplement the information obtained on gust loads by analysis of V-G records, so that the effects of such variables as wing loading, gust gradient, stability, and other factors may be determined.

With such a procedure the V-G data will ultimately provide a basis for the establishment of a datum or normal level of conditions and the investigations conducted at the Committee's laboratory will provide the information required for a rational application of gust-load criteria to different airplane types. Mr. Rhode's report is therefore considered to be a preliminary step toward this objective. The material in this report may be said to be confirmed by results obtained in the Committee's gust tunnel to the extent that such results apply, but, as indicated above, it cannot be said to be confirmed by the V-G data, for the reasons stated.

Sincerely yours,

C. W. Lewis

Director of Aeronautical Research

Document 3-27(h), Interview of Richard V. Rhode by Michael D. Keller, January 10, 1967.

January 10, 1967
RVR: meh

1. Q. What, generally, was the sequence of events which led to the development of the VG and the VGH recorders?

A. The need for obtaining measurements of gust load factors on airplanes in actual service operations was recognized by NACA during the late 1920's. The earliest of measurements were made with a modified version of a commercially available instrument known as a Tetco TIM recorder. I believe the initials TIM stood for Time in Motion, because the instrument was normally used on vehicles such as trucks to record the periods during which the truck was idle or operating. A stylus, which reacted to vibrations or jarring of the vehicle, impressed a record of its motions on a circular waxed paper disc, which had a 24-hour time scale on it. The NACA converted this instrument to an accelerometer of suitable sensitivity, and, when properly installed near the center of gravity of an airplane, the instrument would register accelerations of the machine normal to the plane of the wings. It was calibrated to read acceleration in g units.

Although the converted Tetco TIM Recorder served to give some idea of the accelerations or load factors during service operations, it was not very accurate and data on acceleration alone were not enough to describe the structural loading situations of importance. Measurements of airspeed were also required. In maneuvering flight, for example, a given acceleration occurring at low speed is associated with moderate or high angle of attack of the wing, whereas the same acceleration occurring at high speed is associated with a low angle of attack of the wing. Since the pressure or load distribution on the wing is normally greatly affected by angle of attack, it was of great importance that airspeed and acceleration be simultaneously measured to permit interpretation of the records in terms of angle of attack or load distribution.

The importance of this aspect of aircraft structural loading was greatly illuminated as a result of analyses of acceleration and airspeed records taken by the NACA on an open time scale during dive tests of the U.S. Navy's first dive bomber, the XBM-1, a biplane built by the Glenn L. Martin Company in Baltimore. In analyzing the records from these tests, Mr. Eugene E. Lundquist plotted the data on a chart in which the scale of ordinates was acceleration normal to the wing chord and the scale of abscissas was indicated airspeed. Parabolic lines, each representing a condition of constant lift coefficient (or pressure distribution) could be drawn in on

this chart. With such a chart, the loading situation could be seen at a glance.

Because of these qualities, the chart became of great importance as a means of representing the critical or design loading conditions for aircraft both civil and military airworthiness requirements.

In this same time period, it became evident from studies of gust loading on aircraft that simultaneous values of normal acceleration and indicated airspeed measured in flight could be converted to "effective gust velocities" when certain simple characteristics of the airplane were known and applied to the derivation. Such effective gust velocities could, within limits, be used to determine the loads on other airplanes having different characteristics and flying at other speeds than the one on which the measurements were made. There were, however, no instruments available at that time which could be used in service operations to measure simultaneous values of acceleration and airspeed. The NACA recording accelerometers and airspeed meters, which were commonly used in flight tests at the Langley Laboratory, employed optical systems and light beams to trace their records on photographic film. They were delicate instruments and required constant maintenance and frequent re-calibration. Moreover, the operating time period for one loading of the film drums was in all cases much too short for use in service operations. All in all, it was not practicable to employ instruments of this type, even if modified to permit long-period use, in service operations. The thought of collecting and analyzing "miles of records" in order to determine the critical loading conditions that occurred only rarely in service also acted as a strong deterrent to the development of an airspeed-acceleration recorder that would register against time with an open time scale.

The unique instrumentation requirements of the gust load problem, viz., (a) the determination of simultaneous values of airspeed and acceleration in rough air for conversion to "effective gust velocities;" (b) the necessity for measurements over long periods of time on many airplanes in service operations; and (c) the requirement for simple and rugged instrument characteristics, when considered in the light of the Lundquist acceleration-airspeed chart, led me to suggest that a simple instrument, which would register acceleration against airspeed, be devised for use in determining the desired gust-load measurements.

H.J.E. Reid, Engineer-in-Charge of the Langley Laboratory, and a skilled instrument designer and technician, undertook to devise the required instrument which he designed to employ a thin, flexible steel stylus to scratch a record on a small smoked-glass plate. The light pressure of the stylus against the smoked glass resulted in sufficiently low friction to reduce error to acceptable small limits. The original instrument has been modified to improve the damping means and the balancing of the linkages, but today's version is still the same size and shape as the original and employs the same basic principle. Mr. Reid and I jointly held the patent on this instrument.

The instrument was given the name VG recorder for brevity and euphony at my suggestion. "V" is the standard symbol for airspeed and "G" is the capitalized version of the standard symbol, g, for the acceleration of gravity.

The Lundquist acceleration-airspeed chart was, for the same reasons, called the VG diagram, although it later became more properly known as the V-n diagram, because "n" is the symbol for normal load factor.

Although the VG Recorder filled the great need for a simple practical instrument to measure critical loading conditions in aircraft operations, it was not capable of providing much desired detail respecting aircraft operating and loading conditions. A practical instrument capable of measuring the three quantities acceleration, airspeed, and altitude against time on a reasonably open time scale for long periods was still much to be desired.

At a later period, Mr. Philip Donely, who was then Chief of the Gust Loads Branch of the Aircraft Loads Division, pushed for perfection of an optical system and other qualities which would make the desired instrument practical and, working with Mr. Edmond Buckley (now AA/Tracking and Data Acquisition) and others of the Langley Instrument Research Division, finally succeeded in developing the fine instrument now known as the NASA VGH Recorder.

Both the VGH Recorder and the VG Recorder continue to be used as complementary devices for the measurement of aircraft load statistics.

2. Q. How did relations of the laboratory personnel with the people of Hampton change over the years? Why?

A. The laboratory personnel at Langley have always consisted in part of local people and in part of outsiders. In general, the professional employees have come from outside the community and I assume the question relates primarily to these persons and to the people of Hampton who were not employed or who had no friends or relations employed at the Laboratory.

There is no question in my mind that, in the earlier years of the NACA, the professional employees of the Langley Laboratory were, on the whole, not warmly welcomed into the community.

Document 3-28 (a-h)

(a) Floyd L. Thompson, Associate Aeronautical Engineer, to Engineer-in-Charge [Henry J. E. Reid], NACA Langley, "Preliminary study of control requirements for large transport airplanes," 14 July 1936, with attachment "Suggested Requirements for Flying Qualities of Large Multi-Engined Airplanes," copies in RA File No. 509, Langley Historical Archives, Hampton, VA.

(b) Edward P. Warner to Mr. Leighton W. Rogers, Aeronautical Chamber of Commerce, Shoreham Building, Washington, DC, 19 Nov. 1936, copy in Research Authorization (RA) file 509, LHA.

(c) William H. Hernnstein, Jr., Assistant Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Visit of Mr. E. P. Warner to the Laboratory, December 3, 1936," 3 Dec. 1936, copy in RA file 509, LHA.

(d) Eastman N. Jacobs, Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Mr. E. P. Warner's visit to the Laboratory to discuss airplane tests—December 3, 1936," undated, copy in RA file 509, LHA.

(e) Hartley A. Soulé, Assistant Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Visit of Mr. Warner to flight section on June 14, 1937," 15 June 1937, copy in RA file 509, LHA.

(f) Floyd L. Thompson, Associate Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Tests of flying qualities

of Martin M-10B airplane," 14 July 1937, copy in RA file 509, LHA.

(g) Edward P. Warner, 11 West 42nd Street, New York City, to Dr. George W. Lewis, NACA, Navy Building, Washington, DC. 28 Dec. 1937, copy in RA file 509, LHA.

(h) Robert R. Gilruth, "Requirements for Satisfactory Flying Qualities of Airplanes," NACA Advanced Confidential Report, Apr. 1941.

For a thorough analysis of the torturous 25-year path leading to flying-quality specifications for American aircraft, one must read Walter Vincenti's chapter, "Establishment of Design Requirements: Flying Quality Specifications for American Aircraft, 1915-1943," in *What Engineers Know and How They Know It: Analytical Studies from Aeronautical History* (Baltimore and London: The Johns Hopkins University Press, 1990).

Below, readers will find several of the documents on which Vincenti based his analysis of the NACA's initiation of a systematic research program into flying qualities, which the government organization began from 1936 to 1937 in response to the request for help from Douglas consultant Edward P. Warner on the DC-4.

A few of the major figures in the history of NACA/NASA research appear as authors in this document string. Although hardly a household name, Hartley A. Soulé (born 1905) ranks as one of the great aeronautical minds within the NACA. A New York University engineering graduate who came to Langley in 1926, Soulé made significant research contributions in the field of stability and control. In the late 1940s and 1950s, he served as the leader of the NACA's Research Airplane Project, responsible for high-speed experimental airplanes like the Bell X-1 and Douglas D-558. He went on to major management responsibilities with the Mercury program.

A much better known figure is Eastman N. Jacobs (born 1902), a University of California-Berkeley graduate who came to work at NACA Langley in 1925. As readers will see in the next chapter, Jacobs played *the* central role in NACA airfoil development in the 1930s and early 1940s, until his abrupt (and rather mysterious) departure from the lab (and from the American aeronautics community almost entirely) late in World War II.

Floyd L. Thompson (born 1898), who came to work at Langley in 1926, served as director of NASA Langley Research Center from 1960 to 1968. Among the many

roles he performed during the venturesome early years of the U.S. manned space program, Thompson chaired the committee that looked into the 1967 space capsule fire on the ground at Cape Canaveral, which killed three astronauts.

Although not nearly as prominent, William H. Herrnstein (born 1905) was another productive researcher with the NACA. After graduating from the University of Michigan's aeronautical engineering program in 1927, he came to work at Langley, serving in Fred Weick's PRT section. In this capacity, he became deeply involved in the PRT's low-drag cowling and engine nacelle placement tests, among other investigations. He worked at Langley into the NASA period.

A University of Minnesota aeronautical engineering graduate, Robert R. Gilruth (born 1913) took over the NACA's flying qualities research program when Soulé moved to wind tunnel duties almost immediately after arriving at Langley from Minnesota in 1937. (Vincenti's chapter discusses Gilruth's role at length.) Gilruth went on to become the head of the NACA's Pilotless Aircraft Research Division at Wallops Island, Virginia, the head of Project Mercury, and the first director of the Manned Spacecraft Center (later Johnson Space Center) in Houston, Texas.

Gilruth's April 1941 NACA report on the "Requirements for Satisfactory Flying Qualities of Airplanes," the last document in this string, is, without question, one of the most impressive papers in all NACA literature. As Vincenti noted in his analysis of the report, "To arrive at his results, Gilruth reviewed the mass of flight data and pilot opinion to see what measured characteristics proved significant; he also considered what was reasonable to require of an airplane and its designers." Although originally a classified report, in 1943, the NACA reissued it as *Technical Report No. 755*. "As the first generally accepted set of flying-quality requirements," the work quickly became an "oft-cited milestone in its field" (Vincenti, "Establishment of Design Requirements," p. 92). In an interview conducted with Gilruth 45 years later, in 1986, Gilruth looked back at what he had done: "I boiled that thing down to a set of requirements that were very straightforward and very simple to interpret. [The] requirements went right back to those things that you could design for" (Interview of Robert R. Gilruth by James R. Hansen, Kilmarnock, VA, 10 July 1986, copy of transcript in LHA). The requirements that Gilruth laid out in his report gave "comprehensive coverage" to longitudinal stability and control, lateral stability and control, and stalling characteristics. Within each, there were numerous subdivisions and each subdivision "stated a requirement, usually in a number of parts, gave a simple explanation of the reasons for it, and briefly discussed the design considerations necessary to its achievement" (Vincenti, p. 93). Aircraft designers were thus supplied with clear and precise quantitative limits for all the basic "flying qualities," ones based on extensive testing. To many in the airplane industry at the time, it seemed remarkable that "the same requirements could be applied (with minor exceptions) to all current types and sizes of airplanes," but that was what Gilruth's NACA work provided. Before long, such requirements became something "taken for granted" (Vincenti, p. 92), a normal part of the new design paradigm generated by the airplane design revolution of the interwar period.

For an even fuller treatment of the contents of Gilruth's classic report, see the entirety of the Vincenti chapter.

One cannot leave the topic of Gilruth's report without making a final observation. It is tempting to see much of the basis for Gilruth's later success as head of NASA's Mercury program already reflected in his brilliant laying out, two decades earlier, of satisfactory flying qualities for American aircraft. Historians and others have given too little credit to previous aeronautical experience, and previous NACA research experience, for the early successes of NASA, particularly in the manned space program.

Document 3-28(a), Floyd L. Thompson, Associate Aeronautical Engineer, to Engineer-in-Charge [Henry J. E. Reid], NACA Langley, "Preliminary study of control requirements for large transport airplanes," 14 July 1936.

Langley Field, Va.,
July 14, 1936.

MEMORANDUM For Engineer-in-Charge.

Subject: Preliminary study of control requirements for large transport airplanes.

Reference: (a) NACA Let. Jan. 14, 1936, CW:MW.
(b) LMAL Let. May 4, 1936, FEW.T.

1. Some time ago, the Honorable Edward P. Warner recommended that the N.A.C.A. undertake a study of the control requirements of large transport airplanes, which was authorized in letter of reference (a). In reference (b) an outline of the work to be undertaken in this research was given. The study indicated in section (a) of that letter has been competed and the results are attached hereto.
2. The first part of these results is entitled, "Suggested Requirements for Flying Qualities of Large Multi-Engine Airplanes." These requirements are set down in forms such as one would use in preparing specifications for the flying qualities. They are similar in some respects to the suggested specifications prepared by Mr. Warner and submitted to Mr. Weick for comment, as noted in a letter from Mr. Warner to Mr. Weick December 22, 1935. An attempt to write down the requirements in a detailed form such as this is beneficial, in that it helps to crystallize ideas regarding what items are important and indicates wherein data are lacking concerning quantitative values for various items.
3. The second part of the results is entitled, "General Program of Tests of Airplane Flying Qualities." This program indicates the type of tests required to obtain the various items specified in the previous section on requirements, and also indicates the instruments required.
4. The next step in the investigation will be to carry out section (b) of letter of reference (b). This work will consist of a trial of the methods outlined in the program on one or more of the airplanes available at the laboratory. It is expected that a Stinson Reliant will be available for the tests some time after the first of August. In the meantime, attention will be given to the design of any special instrument required.

Floyd L. Thompson,
Associate Aeronautical Engineer.

SUGGESTED REQUIREMENTS FOR FLYING QUALITIES OF LARGE MULTI-ENGINED AIRPLANES

Note: The chief purpose of these suggested requirements is to show what items are believed to be important. Although numerical limits have been stated, they are, for the most part, considered to be quantitatively unreliable, owing to the present lack of data concerning what constitutes satisfactory flying qualities.

I. Longitudinal stability and control characteristics:
A. Longitudinal stability:
With elevator free the airplane shall be dynamically stable throughout the speed range for all loading conditions. The period of the longitudinal oscillations should never be less than 40 seconds and the damping should be sufficient to reduce the amplitude of the oscillation to one-fifth the original amplitude in four cycles.

1. Longitudinal stability is required for the following conditions:

   a. Flap up–full power–entire range of trimming speeds.

   b. Flap up–level flight–entire range of trimming speeds.

   c. Flap up—power off—entire range of trimming speeds.

   d. Flap up–any three engines operating or the outboard engine on one side and the inboard engine on the other side cut out. Entire range of trimming speeds with aileron and rudder trimming tabs adjusted to give lateral balance.

e. The same conditions should be met with the flap down except that the high speed shall be limited to the placarded speed.

B. Elevator control:

1. Range of elevator control:

The range of elevator control should be sufficient so the following conditions may be met:

a. It shall be possible to maintain steady flight at any speed from the designed probable diving speed to the minimum speed. This condition shall be met with any loading condition and with any power condition with the flap up.

b. With the flap down it shall be possible to maintain steady flight at any speed from the placarded speed to the minimum with any loading and any power condition.

c. If the conventional landing-gear arrangement is used it shall be possible to land the airplane power off with the flaps either up or down with the loading condition that gives the most forward position of the center of gravity in a three-point attitude without aerodynamic bouncing. It shall be possible for the same loading condition to hold the tail wheel down while braking vigorously enough to give a deceleration of 0.3 g during the landing run down to a speed of 30 m.p.h. With the loading condition that gives the most rearward position of the center of gravity it shall be possible to raise the tail wheel off the ground in the take-off run by the time a speed of 30 m.p.h. is attained.

d. If a tricycle landing-gear arrangement is used it shall be possible with the loading condition that gives the most forward position of the center of gravity to raise the nose wheel off the ground in the take-off run by the time a speed of 30 m.p.h. is attained.

2. Variation of elevator angle with speed:

a. The curve of equilibrium elevator angle against speed for every setting of the trimming tab both power on and power off for all loading conditions

and with the flaps up and down shall be smooth and shall everywhere have a positive slope.

3. Range of elevator control forces:

With every setting of the trimming tabs, with every loading condition either power off or power on, and with the flaps up and down, it shall be possible to fly the airplane within the ranges given for the elevator control in the previous section with a range of stick forces of no greater than 100 pounds.

4. Variation of elevator force with speed:

a. A curve of stick force required for steady flight plotted against the speed of flight for every tab setting and every loading condition with either power on or off or flaps up or down shall be smooth in form without discontinuity or sudden changes of curvature. Its slope shall be everywhere negative throughout the specified speed ranges and shall nowhere be less than ¼ pound per mile per hour.

5. Variation of elevator forces with throttle setting:

a. The force required on the stick to overcome, without change of tab setting or speed, the effect of any change in engine operating condition from full power to fully throttled shall not exceed 100 pounds.

6. Elevator trimming tabs:

a. It shall be possible to trim the airplane at a low enough speed so that no greater force than a 30-pound pull will be required in meeting the landing requirements previously given.

b. It shall be possible to trim the airplane at its maximum speed.

7. Effectiveness of elevator control:

a. As an indication of the effectiveness of the elevator for maneuvering the airplane, it shall be possible to obtain an acceleration of 0.8 of the design load factor at any speed with the elevator alone with the airplane originally trimmed for cruising speed without applying a force of more than 200 pounds and not less than 60 pounds to the control stick.

b. As low speeds down to 10 miles per hour above the minimum where the theoretical maximum acceleration approaches 1, it shall be possible to change the attitude of the airplane in space with respect to its transverse axis in either direction by 5° in 1-½ seconds through use of the elevator alone.

II. Lateral stability and control characteristics:

A. Lateral stability:

The airplane shall be laterally stable for the same conditions for which longitudinal stability is required. The period of the lateral oscillations should not be less than 20 seconds and the damping should be sufficient to reduce the amplitude of the oscillation to one-half of the original amplitude in two cycles.

Between the minimum trimming speed and the minimum speed the airplane shall show no negative dihedral effect nor any autorotative tendencies.

From the minimum speed to the limit of the elevator control there shall be no sudden development of marked autorotative tendencies nor any sudden change in the lateral stability characteristics.

1. Limits for roll due to sideslip (dihedral effect);

a. The dihedral effect should be sufficient so that when the ailerons are freed immediately after putting the machine into a 15° bank, and using the rudder to avoid a change of heading, the angle of bank shall be reduced to 2° within 15 seconds and with the loss of altitude of not over 300 feet. This condition shall be met for all conditions for which longitudinal stability is required.

b. The rolling acceleration accompanying abrupt rudder displacement shall be less than one-half of the yawing acceleration.

B. Aileron control:

1. Aileron power:

The aileron power shall be sufficient so that the following conditions can be met:

a. At a speed of 70 miles per hour with flaps down or 80 miles per hour with flaps up, it shall be possible to bank the airplane 15° in 2-½ seconds with the ailerons alone, and at 120 miles per hour or higher the same angle of bank shall be obtained in 2 seconds.

b. At a speed 2 miles per hour above the stall with the flaps down it shall be possible to bank the airplane 10° in 2 seconds with the ailerons alone.

c. The aileron effectiveness shall be proportional to the aileron deflection.

2. Aileron forces:

a. The force required to obtain the aileron reactions given should not exceed 50 pounds applied tangentially at the rim of the wheel.

b. The aileron force shall be approximately proportional to the aileron deflection.

3. Yaw due to ailerons:

a. At speeds above 10 percent in excess of the minimum speed the ailerons should not produce a yawing acceleration greater than 0.1 the acceleration in roll. At speeds below 10 percent in excess of the minimum the acceleration in yaw should be less than one-fifth of the acceleration in roll.

4. Aileron trimming tabs:

The range of power of the aileron trimming tabs should be sufficient so that the following conditions may be met:

a. It shall be possible by the use of the tabs to balance the airplane against the dissymmetry of loading corresponding to full gas tanks on one side of the center of gravity and empty gas tanks on the other.

b. It shall be possible by the use of the tabs to compensate for any rolling tendency accompanying steady flight with asymmetrical power and loading conditions.

C. Rudder control:

1. Rudder power:
The power of the rudder control should be sufficient so the following conditions may be met:

a. It shall be possible during steady flight at 70 miles per hour with the flaps down or 80 miles per hour with the flaps up, as well as at any higher speed, to produce a change of heading of 15° in 3 seconds by use of the rudder alone. At 120 miles per hour the same change shall be possible in 2 seconds.

b. At 2 miles per hour above the stall with flaps down it shall be possible to make flat turns up to a change of heading of 10° in 2 seconds.

c. At 20 miles per hour above the minimum speed as well as at any higher speed it shall be possible to hold a straight course with the wings laterally level with both engines on either side cut out and those on the other side operating at full rated power.

d. With any three engines operating or one outboard and the opposite inboard one cut out, it shall be possible to hold a straight course on the ground down to 50 miles per hour with the flaps either up or down.

e. The rudder effectiveness shall be proportional to the rudder deflection.

2. Rudder forces:

a. It shall be possible to obtain the rudder reactions given without applying a force greater than 180 pounds on the rudder pedals.

b. The rudder force shall be proportional to the rudder deflection.

3. Rudder trimming tabs:

The range of power of the rudder trimming tabs should be sufficient so the following conditions may be met:

a. Down to 10 percent in excess of the minimum speed, it shall be possible by adjustment of the trimming tabs to fly straight with any three engines operating or one inboard and the opposite one outboard cut out with no force on the rudder pedals.

b. Down to 20 miles per hour above the minimum speed with the flaps down, it shall be possible with the trimming tabs to reduce the force on the rudder pedals required for straight flight with both engines on either side cut out and those on the opposite side operating with full rated power to 30 pounds.

D. Combined operation of rudder and ailerons:

1. It shall be possible to enter a 45° banked turn at 140 miles per hour in 5 seconds without the rudder force exceeding 100 pounds or the aileron force exceeding 75 pounds. The same limitations on forces which apply to a 30° banked turn at 200 miles per hour entered in 4 seconds. It shall be possible to make normally banked turns up to a 15° bank at speeds within 5 miles per hour of the minimum with the flaps either up or down. It shall be possible with the flaps either up or down to fly the airplane steadily for at least 10 seconds up to the limit of the elevator control if the elevator control is sufficient to stall the airplane. It shall be possible at speeds above 10 percent in excess of the minimum to maintain a steady sideslip with an angle of bank of 20°.

GENERAL PROGRAM OF TESTS OF AIRPLANE FLYING QUALITIES

Longitudinal Stability and Control

Longitudinal Stability – (Section I-A)*

Item — Period and damping of oscillations.

Procedure — Elevator free:— Trim airplane for desired speed and then push stick forward until airplane gains 5 to 10 miles per hour. Release the stick and record the ensuing oscillations as the airplane returns to a steady state at the original trimming speed.

Elevator fixed: — Repeat tests as before except that after setting up a disturbance the stick is returned to the original setting or neutral point and held during the ensuing oscillations.

Observations or records — If recording instruments are used, it is only necessary for the pilot to start the instruments at the start of the oscillations and to turn them off when steady flight is reestablished. If indicating instruments are used, it is necessary to observe and record the air speed at successive peaks of the oscillations and to determine the time between successive peaks for four oscillations.

* The reference in brackets and similar references throughout this program apply to "Suggested Specifications for Flying Qualities of Large Multi-Engined Airplanes." This referencing system is employed to show the correlation between the flight program and the items to be specified.

Document 3-28(b), Edward P. Warner to Mr. Leighton W. Rogers, Aeronautical Chamber of Commerce, Shoreham Building, Washington, DC, 19 Nov. 1936, copy in Research Authorization (RA) file 509, LHA.

November 19, 1936.

Mr. Leighton W. Rogers
Aeronautical Chamber of Commerce
Shoreham Building
Washington, D.C.

Dear Mr. Rogers:

I have just heard from George Lewis that Mr. Clayton visited Langley Field on November 13th to discuss the test work being done by the National Advisory Committee for Aeronautics that bears on airplane design requirements, and that you are planning a meeting of your own people on December 3rd to discuss some of these matters.

My special purpose in writing is to say that I expect to be in the east myself by December 1st and stay for a couple of weeks thereafter, and that I should very much like to attend the meeting of the 3rd, if an invitation for me to be present would be consonant with the meeting's purposes. I am, as you know, very deeply interested in all questions bearing on transport design and regulations; I have been particularly concerned with promoting testing of the flight qualities of commercial airplanes at Langley Field; in connection with the development of the DC-4, I have lived very close to all manner of questions of specifications and regulation; and I was designated as chairman of the special National Advisory Committee for Aeronautics subcommittee on the aerodynamic problems of transport operations (which, despite its title, has not by any means been limited to aerodynamics in its considerations).

I shall probably be staying here through next Monday or thereabouts. If you want to reach me after that, I would suggest that I be addressed at the Harvard Club, 27 West 44th Street, New York City.

Sincerely,

Edward P. Warner.

Document 3-28(c), William H. Hernnstein, Jr., Assistant Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Visit of Mr. E. P. Warner to the Laboratory," December 3, 1936.

Langley Field, Va.
December 3, 1936.

MEMORANDUM For Engineer-in-Charge.

Subject: Visit of Mr. E. P. Warner to the Laboratory December 3, 1936.

1. Mr. Warner visited the Laboratory on the morning of December 3 primarily for the purpose of discussing the program of measuring flying qualities of airplane and the progress we have made to date on the flight tests contemplated with the Committee's Stinson in connection with this program.

2. A meeting was held in Mr. Reid's office which was attended by Mr. Warner and Messers. Reid, DeFrance, Crowley, Jacobs, Thompson, Soule, and Herrnstein of the Laboratory staff. Mr. Warner suggested that airfoils of 30 to 40 percent thickness be tested for possible future use on transport airplanes of high wing loading and high aspect ratios.

3. Mr. Crowley stated that the Stinson was being prepared for its part in the program of measuring flying qualities of airplanes to check out the method to be used when we receive various airplanes for flight tests. He stated that the instrument installation would call for some special instruments but that many of the standard instruments in the airplane would be used and the readings photographically recorded. Instruments to be added to the airplane would include control-position and control-force recorders, two turnmeters, an accelerometer, and an air-speed recorder.

4. Mr. Warner inquired as to how long it would take to make an instrument installation in any airplane furnished. Mr. Thompson thought it would take two days at the most and probably less time. He added, however, that a different control force recorder would have to be used than that which is now being used in the Stinson as it takes too long to install it.

5. Mr. Warner wanted to know how long a pilot could sustain forces on the controls. He was already familiar with our work done on the maximum force a pilot can exert on the controls. Mr. Crowley replied that such measurements had been made at the Laboratory and also added that the pilots complained of aileron forces on the stick of 10 pounds as being excessively heavy, whereas they can exert a maximum force of about 30 pounds. Mr. Warner asked how the maximum wheel forces a pilot can exert check up with his specifications on this subject. No one knew exactly but the data are all on record.

6. Mr. Crowley stated that with the artificial horizon the flight section hoped to be able to discard most of the recording instruments provided the horizon proved to give accurate enough results. He also added that the Stinson would probably be flying next week. One difficulty with using the Committee's Stinson as a guide in the measurement of flying qualities of larger airplanes to be tested later is that the machine rolls very rapidly as compared to larger airplanes.

7. Mr. Warner stated that very complete wind-tunnel tests made by the Douglas Company showed that they could meet any of his specifications easily with the exception of control on the ground in take-off with unsymmetrical power conditions. The tests showed that about two times ordinary rudder size would be needed to meet this specification. Mr. Warner also stated that he thought it a common error to overestimate the safety of twin-engine and 4-engine airplanes. In most cases the twin-engine airplane, which is the worst, needed about 90 miles per hour forward speed to enable it to be held on straight course after one engine was cut out.

8. Mr. Warner inquired if the control-position recorder to be used would be mounted in the cockpit. Mr. Crowley answered that it would.

9. In answer to a question by Mr. Warner, Mr. Soule answered that no flight routine had yet been settled for these tests. Such a routine would depend upon test results obtained with the Stinson and further work. It was agreed that this work should have a broad base at the start particularly.

10. In regard to control at the stall and warning of the stall, Mr. Warner stated that the Douglas DC-4 was specified to have the lift spoiled at the center section so as to maintain lateral control on the tapered wing, and that any buffeting resulting from this was to be felt through the controls only and not by the passengers through the structure.

11. Mr. Jacobs expressed concern about the ability of manufacturers to meet lateral control specifications when partial span flaps were to be used on tapered wings. He was afraid that the tips of these wings would always stall first due to the flaps.

12. Regarding torsional stiffness of the wings, Mr. Warner stated that the big problem now was to get a lateral control that would not subject the wing to torsion. He said that the devices such as slot-lip ailerons were out of the picture at present because of icing difficulties. Regarding the use of heat for the prevention of ice formation, Mr. Warner observed that ice would form on heated surfaces when the outside temperature is much below freezing when no ice would be present if heat were not applied.

13. Mr. Warner asked if in our studies of long-range flight, neglecting the wing factor, we had determined that it would be economical to fly at high altitudes. Mr. Jacobs replied that he was not sure but doubted if there was any advantage from the pure-range standpoint. He said that there would be an advantage in speed but it would not be as large as was commonly supposed.

14. Mr. Soule pointed out the difficulty in adapting the present control-force device to all types of wheels and sticks with which test airplanes will be supplied. It was decided that the Laboratory would have to obtain advance information on the control columns, etc., of the airplanes to be tested. Mr. Warner said that there is nothing definite to date as to what airplane would be the first to undergo such tests.

15. Mr. Warner observed that in all the large future airplanes the control will probably be irreversible, doing away with any feel in the controls of the airplane.

16. Following this meeting Mr. Warner visited the individual activities of the Laboratory.

W. H. Herrnstein, Jr.,
Assistant Aeronautical Engineer.

Document 3-28(d), Eastman N. Jacobs, Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Mr. E. P. Warner's visit to the Laboratory to discuss airplane tests—December 3, 1936."

Langley Field, Va.
Undated

MEMORANDUM For Engineer-in-Charge.

Subject: Mr. E. P. Warner's visit to the Laboratory to discuss airplane tests—December 3, 1936.
Reference: LMAL Let. (L.F.1) Dec. 3, 1936, IHD. EWM.

1. A conference between Mr. Warner and Messrs. Reid, Crowley, Thompson, Soule, and Jacobs was held mainly to answer questions of Mr. Warner about the proposed tests and the instruments developed for the work (covered in Mr. Herrnstein's memorandum of reference). Later, the instruments were inspected. Many related problems were discussed at length; for example, lateral control and high-lift devices, servo controls, ice formation, sub-stratosphere flying, etc. The main discussion was hampered by the fact that we had no copy of Mr. Warner's specifications for the new transport to refer to. Later, I had a chance to go over the specifications with Mr. Warner, the flight section group, and Mr. Gough, and also had an opportunity to discuss our airfoil work with Mr. Warner while I drove him down to Old Point to get the specifications. The part about stalling characteristics, the part in which I was particularly interested and the only part I had time to consider in detail, seemed to me essentially reasonable and definitely desirable.

2. As I understand it, however, the vital part of Mr. Warner's proposal as far as the Committee is concerned, is that we develop a short flight-check procedure for commercial airplanes. Mr. Warner is undoubtedly right in contending that every effort should be made to shorten the procedure to an extent that the airplane is

required for only about one week. On this basis, it would probably be possible to flight-check many new types. This object is of vital importance to the Laboratory, because a familiarity with new types will tend to get us out of the dark with regard to the practical effects of the application of new developments.

3. It is my personal opinion that the flight-check procedure might be improved and shortened by going over the subject carefully with the pilots so as to take full advantage of their qualitative observations during a few preliminary flights. I agree with the flight section personnel, however, that we will be in a better position to discuss the final procedure among ourselves and with Mr. Warner after the preliminary tests on the Stinson have been completed.

4. Mr. Warner offered suggestions that will be followed up on both our airfoil and our fuselage work. He wants us first to investigate further variations of airfoil thickness distribution, and second, the best practical maximum thickness on the simplifying assumption that a span to thickness ratio of the order of 60 must not be exceeded. He criticized the fairness of our fuselage models and suggested that we test fairer modifications. He also thought that we have not gone to sufficiently low fineness ratios on our fineness ratio series. I agreed that we should extend it if the tests show that we have not definitely gone beyond the optimum.

Eastman N. Jacobs,
Aeronautical Engineer

Document 3-28(e), Hartley A. Soulé, Assistant Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Visit of Mr. Warner to flight section on June 14, 1937," 15 June 1937.

Langley Field, Va.,
June 15, 1937.

MEMORANDUM For Engineer-in-Charge.

Subject: Visit of Mr. Warner to flight section on June 14, 1937.

1. Mr. Warner visited the flight section on June 14 to discuss the section's work pertaining to the measurement of flying qualities. He was familiar with the status of the project and the more important results. His object on this visit was to inspect the results from the Stinson and B10-B (Martin bomber) airplanes in order to familiarize himself with the extent of the results, details of presentation, and the time required for the tests. He was also interested in the precision of the measurements and agreement of runs under supposedly identical conditions as an indication of the practical tolerances when specifying the various factors.

2. The inspection took about 3 hours. Mr. Thompson was present about half this time. During the discussion accompanying the inspection Mr. Warner made several comments and suggestions, the most important of which was that for pull-ups for large airplanes we should investigate the flight path for the representative cases to assure ourselves that the maximum normal acceleration is indicative of the attitude gained in the maneuver in a given time. He is interested in the delay between the control movement and the upward motion of the airplane. After going over all the figures Mr. Warner requested permission to look over the text of the memorandum on the Stinson tests. This was granted and he spent about a half hour reading the memorandum. He made note of further suggestions but as time was not available for more discussion, he intends to submit these in writing.

Hartley A. Soulé,
Assistant Aeronautical Engineer.

Document 3-28(f), Floyd L. Thompson, Associate Aeronautical Engineer, to Engineer-in-Charge, NACA Langley, "Tests of flying qualities of Martin M-10B airplane," 14 July 1937.

Langley Field, Va.
July 14, 1937.

MEMORANDUM For Engineer-in-Charge.

Subject: Tests of flying qualities of Martin B-10B airplane.

Reference: (a) NACA Let. Mar. 29, 1937, L.F.
(b) LMAL Let. Apr. 3, 1937, WHH. DLC.

1. The investigation of the flying qualities of the Martin B-10B airplane obtained on loan from the Army Air Corps in accordance with letter of reference (a) has been completed. The airplane made available for the tests was no. 34-34 and was one of those attached to the 20th Bombardment Squadron. It was made available to the Committee for tests on May 1 and was released to the Army on July 2. Throughout this period the airplane was housed in the Army hangar and was serviced by Army personnel, and at various times it was made available to Army personnel for flights, when such flights did not interfere with the test program. The actual test program required approximately 26 hours of flying, most of which was done during June. The installation of instruments and some preliminary flights were made early in May, but because of the Engineering Research Conference and because of the necessity of repairs and routine checking of the airplane, no tests were made during the latter part of May.

2. The B-10B airplane is the second one with which tests have been made in the investigation of flying qualities. The program followed was in essential agreement with that submitted with letter of reference (b), but with minor modifications suggested by experience gained in testing the single-engine Stinson monoplane, the first airplane used in this investigation.

3. In the original plan for this investigation it was intended to develop a method wherein the measurements could be obtained, as much as possible, by means of indicating instruments as well as by means of the recording instruments usually employed by the Committee. A certain amount of development along this line was followed in the test of the Stinson, but further development was handicapped with the B-10B airplane because of the arrangement of that machine. The use of the indicating instruments requires that an observer have access to the pilot's cockpit whereas in the B-10B airplane the observer does not have access to the pilot's cockpit and can only communicate with the pilot by means of the interphone system. It should be noted, however, that although in the tests of the Stinson the elevator and aileron forces were recorded, with the B-10B an indicating system was used, the observations being made by the pilot. Regardless of the system used for making measurements testing is greatly handicapped when the observer does not have access to the pilot's cockpit.

4. Equipment used in the tests performed satisfactorily with the exception of the control-force indicator. This instrument, as arranged at present, is suitable for the measurement of steady forces but maximum recording hands required in determining suddenly applied forces did not work satisfactorily. The instrument is now undergoing further development in order to make possible the determination of suddenly applied forces as well as steady forces.

5. With the B-10B airplane it is possible to vary the throttle settings, the propeller pitch, the flap position, and to raise or lower the landing gear. In addition, the center-of-gravity position is a variable. Thus, as for nearly all modern machines, there are a great number of possible conditions of the airplane to consider in arranging the test program, but after consideration the program was restricted to 5 conditions of the airplane representative of the following regimes of flight:

(1) High-speed flight.
(2) Climbing flight.
(3) Power-off flight.
(4) Take-off.
(5) Landing.

The complete series of tests was made with a center-of-gravity position corresponding approximately to the rearmost position recommended for the airplane. Because the airplane exhibited static longitudinal instability with this position of the center of gravity, a portion of the tests related to the longitudinal stability was repeated with the airplane loaded for the most forward center-of-gravity position.

6. The data for the tests have not been completely evaluated but some points of general interest can be stated. With the rearmost center-of-gravity position, the airplane with power on possessed longitudinal static instability but with the power off the airplane was stable. This instability was eliminated when the center of gravity was shifted to the most forward position. In the latter case, however, the trimming tab did not have sufficient power to balance the airplane within the permissible speed range with the flaps lowered. The airplane had very little effective dihedral, as was evidenced by the slow leveling off when a wing was dropped and slipping toward the lower wing developed. With one engine operating at full power and the other engine idling, the power of the rudder tabs was insufficient to balance the turning moment due to the asymmetric thrust at any speed. With the tab set to assist the pilot straight flight could be maintained down to a minimum speed of 90 miles per hour. At this speed the rudder force required was of the order of 160 pounds. A complete report on the results of these tests will be made as soon as possible.

7. As a result of the experience gained, first with the Stinson monoplane, and second with the Martin B-10B airplane, it is felt that the procedure has been fairly well perfected. Some further development of instruments and procedure will be required, but in general it is believed that from now on the major point of interest will be the actual results obtained, rather than the perfection of procedure. It is believed that in machines wherein the observer has access to the pilot's cockpit, the complete program can be carried out in approximately one month. Some advance notice, however, is required to permit the preparation of instruments; in particular, the construction of parts necessary to adapt the control-force recorder to the particular airplane involved. Drawings of the control wheel installation should be made available at least two weeks in advance of the delivery of the airplane for testing.

Floyd L. Thompson,
Associate Aeronautical Engineer.

Document 3-28(g), Edward P. Warner, 11 West 42nd Street, New York City, to Dr. George W. Lewis, NACA, Navy Building, Washington, DC, 28 Dec. 1937.

EDWARD P. WARNER
11 West 42nd Street
New York City

December 28, 1937.

Dr. George W. Lewis,
National Advisory Committee for Aeronautics,
Navy Building,
Washington, D. C.

Dear George:

As you know, the completion of the DC-4 is near at hand. Naturally it will be necessary to run a most exhaustive series of tests on a ship of such exceptional size and cost before deciding on a production policy; and as far as possible the factor of difference of personal opinion ought to be eliminated from the results of those tests.

Furthermore, from a scientific point of view it would in any case be desirable that we check with particular care on the flying qualities of this ship, in view of the attempt that it represents to put flying-quality specification on a quantitative basis.

For both of those reasons, I am very hopeful that it will be possible for the Advisory Committee to participate in the testing, and to provide for the use of your recording instruments and the techniques that you have developed in their use.

The tests will of course be made at Santa Monica; nor would it seem entirely practicable to take the machine to Langley Field for this specific purpose, at least until a number of months after its construction and after the first and most urgent need for instrumental test methods would have passed. Could it not then be arranged for one or two members of your staff, competent in the use of the flight recording instruments, to go to Santa Monica and remain there as part of the test crew for the duration of the test period?

While I do not definitely speak for any of the air lines, and have no authority to commit any of them, I should think that it would be possible to meet whatever expenses might be involved in such a transfer of personnel; but in any event I am sure that the results that you would get from such a participation would be of the greatest interest, and I have no doubt that the parties involved in the test would be glad to agree that a substantial proportion at least, if not the whole, should be considered available for early publication by the Committee.

I of course make this suggestion with less hesitation than I might otherwise feel because of the cooperative nature of the project. In participating to increase the value of the tests and to increase the accuracy of the immediate appraisal of the qualities of the ship, you would be rendering service not merely to a single air line, {AQ1}but substantially to the entire air transport industry; since the companies directly involved in the DC-4 purchase and presumably all equally anxious to get the most reliable information possible upon the ship, represent nearly 80% of the mileage flown and nearly 90% of the passenger traffic handled under the American flag.

In this same connection of flying-quality testing, which is obviously going to assume increasing importance as airplanes increase in size, it occurs to me that it ought to be possible at the same time to increase your background for the judgment of test results, and to render an immediate service to the Army and Navy, by applying the technics of measurement that you have been developing on the Stinson and the Martin to the Boeing XP-15, and to the new Sikorsky and Consolidated four-engined patrol boats.

Would it be appropriate to suggest to the Army and Navy personnel concerned the availability of your new procedure and its possible interest in connection with these large aircraft, or would it be better that I take the initiative by corresponding with the Service personnel? Personally I should think that you could better take it up directly, unless you have some fixed principle against seeming to promote the use of the Committee's services in a particular instance in advance of their being asked for.

I should think that another very useful application of the same methods would be on the flying model which Martin has been using recently to predict the behavior of the big boat for which he has just received a Navy order. Aside from being of considerable interest to Martin and to the Navy, in its greater delicacy of indication of existing flying qualities and of suggestion of the changes that perhaps should be made in the full-scale ship to ameliorate them, direct measurement of flying qualities in this instance would be of obviously immense scientific interest in establishing a specific quantitative correlation between the behavior of the flying model and that of the final machine, and in showing just how much scale effect there is in these matters in free flight.

I shall be very much interested in knowing what you think about all this. Please let me know if I can be of any assistance.

Sincerely,
Edward P. Warner

Document 3-28(h), Robert R. Gilruth, "Requirements for Satisfactory Flying Qualities of Airplanes," NACA Advanced Confidential Report, Apr. 1941.

REQUIREMENTS FOR SATISFACTORY FLYING QUALITIES OF AIRPLANES

By R. R. Gilruth
Langley Memorial Aeronautical Laboratory
April 1941

INTRODUCTION

The need for quantitative design criterions for describing those qualities of an airplane that make up satisfactory controllability, stability, and handling characteristics, has been realized for several years. Sometime ago, preliminary studies showed that adequate data for the formulation of these criterions were not available and that a large amount of preliminary work would have to be done in order to obtain the information necessary. It was apparent that flight tests of the flying qualities of numerous airplanes were required in order to provide a fund of quantitative data for correlation with pilots' opinions.

Accordingly, a program was instituted which covered the various phases of work required. The first step involved the development of a test procedure and test equipment which would measure the characteristics on which flying qualities depend. Thin phase of the work is reported in reference 1, although since that time the test procedure has been expanded and modified on the basis of additional experience and several changes have been made in the equipment used.

Another phase of the investigation has involved the measurement of the flying qualities of a number of airplanes. The procedure used has been in general in accord with that described in reference 1. At the present time, complete tests of this nature have been made of 16 airplanes of varied types. These airplanes were made available largely by the Army and more recently by private companies at the request of the Civil Aeronautics Board. In addition, flying-qualities data of more limited scope have been obtained from time to time on a number of other airplanes, the tests of which covered only particular items of stability and control but which, nevertheless, augment the fund of data now available.

A third phase of the investigation, one which also has been pursued throughout the duration of the project, has involved the analysis of available data to determine what measured characteristics were significant in defining satisfactory flying qualities, what characteristics it was reasonable to require of an airplane, and what influence the various design features had on the observed flying qualities.

In order to cover this work adequately, a number of papers dealing separately with the various items of stability and control are necessary. Several such papers have been prepared or are in preparation at the present time. Detailed studies of all items will require considerable time for completion, but it is believed that the conclusions reached to date are complete enough to warrant a revision of the tentative specifications set forth in reference 1. As opportunity for additional analysis occurs, it would be desirable to cover the individual requirements at more length than is possible at this time. As a result of further studies, it may also be desirable to revise again the flying-qualities specifications given here.

In addition to the actual specifications, the chief reasons behind the specifications are discussed. Wherever possible, interpretation of the specification is made in terms of the design features of the airplane unless the subject is covered in reports of reference.

In formulating the specifications, every attempt has been made to define the required characteristics in easily measurable, yet fundamental terms. It was necessary to consider all stability and control requirements in arriving at each individual item because of the varied functions of the individual controls and the conflicting nature of many of these functions.

The specifications require characteristics that have been demonstrated to be essential for reasonably safe and efficient operation of an airplane. They go as far toward requiring ideal characteristics as present design methods will permit. Compliance with the specifications should ensure satisfactory flying qualities on the basis of present standards, although as additional knowledge is obtained it may be possible to demand a closer approach to ideal characteristics without in any way penalizing the essential items of performance.

FLYING-QUALITY REQUIREMENTS

It has been convenient to present the flying-quality requirements under the following individual headings. They appear in the report in this order.

I. Requirements for longitudinal stability and control:
   A. Characteristics of uncontrolled longitudinal motion.
   B. Characteristics of elevator control in steady flight.
   C. Characteristics of elevator control in accelerated flight.
   D. Characteristics of elevator control in landing.
   E. Characteristics of elevator control in take-off.
   F. Limits of trim change due to power and flaps.
   G. Characteristics of longitudinal trimming device.

II. Requirements for lateral stability and control:
   A. Characteristics of uncontrolled lateral and directional motion.
   B. Aileron-control characteristics.
   C. Yaw due to ailerons.
   D. Limits of rolling moment due to sideslip.
   E. Rudder-control characteristics.
   F. Yawing moment due to sideslip.
   G. Cross-wind force characteristics.

H. Pitching moment due to sideslip.

I. Characteristics of rudder and aileron trimming devices.

III. Stalling characteristics.

These requirements pertain to all flight conditions in which the airplane may be flown in normal or emergency operation, with the center of gravity at any point within the placarded limits, some of the specifications are based on the behavior of the airplane at some specified airspeed. The airspeed in such cases shall be taken as the indicated airspeed. Where minimum airspeed is referred to, unless otherwise stated, it shall be taken as the minimum airspeed obtainable with flaps down, power off.

With the exception of part III of the requirements, which deals exclusively with characteristics at or close to the stall, the requirements pertain to behavior of the airplane in the range of normal flight speeds at angles of attack below that at which the stall would occur.

In the specifications which follow, the lower limits of the control-force gradients are specified in terms of the ability of the controls to return to trim positions upon release from deflected positions. This is a very desirable characteristic because it assures {AQ2} a control friction sufficiently low in comparison with the aerodynamic forces to allow the pilot to feel the aerodynamic forces on the controls. However, some additional interpretation of the specifications is necessary, because no control system can be made entirely free of friction and, therefore, there will always be some small deviation from return to absolute trim. At the present time, it is not possible to fix the allowable limits for these deviations. It is known, however, that controls reasonably free from friction, as measured on the ground, have satisfactory self-centering characteristics in the air as long as there is a definite force gradient. For elevators, force gradients as low as 0.05 pound per mile per hour have been satisfactory when the friction was small. For relatively small airplanes such as fighters, trainers, and light airplanes, it appears that about 2 pounds of friction in the elevator control system and 1 pound in the aileron represent an upper limit. In several cases, where push-pull rods with ball bearings were used throughout the control system, friction in both elevator and aileron systems has been found to be under 1/2 pound.

For large airplanes not intended to maneuver where visual or instrument references are always available, self-centering characteristics are not believed to be essential, although they are very desirable. In these airplanes, control friction should be kept as low as possible, although there is indication that considerable more friction can be tolerated. A representative amount of control friction for a transport or medium bomber would be about 10 pounds in the elevator system and 6 pounds in the ailerons.

Irreversible controls have somewhat similar characteristics to controls with high friction; that is, they are not self-centering and therefore tend to destroy control feel. They are not considered desirable, although on very large airplanes where the rates of deviation from steady flight are slow they have been used successfully on ailerons.

I. Requirements for longitudinal stability and control.

<u>Requirement (I-A):</u> Characteristics of uncontrolled longitudinal motion.

When elevator control is deflected and released quickly, the subsequent variation of normal acceleration and elevator angle should have completely disappeared after one cycle.

<u>Reasons for Requirement (I-A)</u>: The requirement specifies the Degree of damping required of the short-period longitudinal oscillation with controls free. A high degree of damping is required because of the short period of the motion. With airplanes having less damping than that specified, the oscillation is excited by gusts accentuating their effect and producing unsatisfactory, rough-air characteristics. The ratio of control friction to air forces is such that damping is generally reduced at high speeds. When the oscillation has appeared at high speeds as in dives and dive pullouts it was, of course, very objectionable because of the accelerations involved.

The short-period oscillations involve variations of the angle attack at essentially constant speed and should not be confused with the well-known long-period (phugoid) oscillation, which involves variation of speed at an essentially constant angle of attack. As shown by the tests of reference 2, the characteristics of the latter mode of longitudinal motion had no correlation with the ability of pilots to fly an airplane efficiently, the long period of the oscillation making the degree of damping unimportant. Subsequent tests have not altered this conclusion. The case of pure longitudinal divergence of the airplane (static instability) will be covered later under requirements of the elevator control in steady flight. No requirement for damping of the long-period phugoid motion appears justifiable at the present time.

<u>Design considerations</u>. – A theoretical analysis of this problem (unpublished) has shown that the damping of the control-free (short-period) oscillation is dependent chiefly on the magnitude of the hinge-moment coefficient of the elevators and on the mass balance and moment of inertia of the control system. The analysis shows that the damping is improved by increasing the hinge-moment coefficient, increasing the mass balance, and reducing the moment of inertia. The introduction of friction damping in the control system should, of course, also be effective although control friction is very undesirable for other reasons.

<u>Requirement (I-B)</u>: Characteristics of elevator control in steady flight.

1. The variation of elevator angle with speed should indicate positive static longitudinal stability for the following conditions of flight:

a. With engine or engines idling, flaps up or down, at all speeds above the stall.

b. With engine or engines delivering power for level flight with flaps down (as used in landing approach), landing gear down, at all speeds above the stall.

c. With engine or engines delivering full power with flaps up at all speeds above 120 percent of the minimum speed.

2. The variation of elevator control force with speed should be such that pull forces are required at all speeds below the trim speed and push forces required at all speeds above the trim speed for the conditions requiring static stability in item 1.

3. The magnitude of the elevator control force should everywhere be sufficient to return the control to its trim position.

4. It should be possible to maintain steady flight at the minimum and maximum speeds required of the airplane.

Reasons for Requirement (I-B): Items 1 and 2 require positive static stability for flight conditions in which the airplane is flown for protracted lengths of times, or where opportunity exists to establish a trim speed so that stable characteristics can be realized. Positive static stability at this time is not considered particularly helpful to a pilot at very low speeds with full power on or with flaps extended with full power on, because of the large trim changes due to power usually experienced. The conditions are classed as emergency conditions because in actual operation they are entered suddenly from approach conditions, where relatively little power is used. In these cases the elevator force and position changes, due to applied power and change of flap setting, are usually far greater than any inherent stable or unstable force or position gradients which exist due to the degree of static stability present. For these reasons, static stability in these conditions is not considered essential, at least not until trim changes due to power are reduced to much lower values than are experienced at the present time. The magnitude of allowable trim change due to power and flaps, is covered later in Requirement (I-F).

In other conditions of flight, however, static stability is regarded as an essential flight characteristic. Item 1 pertains to the elevator-fixed condition. This requirement ensures that the airplane will remain at a given angle of attack or airspeed as long as the elevator is not moved, and provided that disturbed motion of the airplane is not left uncontrolled for long periods of time. Positive stability eliminates the need for constant control manipulation in maintaining given conditions and, furthermore, simplifies the control manipulation when a speed change is desired, because the direction of control movement required to start the rotation in pitch corresponds to that required to trim at the new angle of attack. A negative slope to the elevator-angle curve is a necessary requirement for elevator control feel, and the degree of control feel increases as the variation of elevator angle with angle of attack is increased (reference 4). 1n general, it may be said that the variation of elevator angle with angle of attack should be negative and as large numerically as is consistent with other requirements of elevator control.

Item 2 requires that the elevator-free, static longitudinal stability shall always be positive. This specification ensures that the airplane will not depart from a trim speed except as a result of definite action on the part of the pilot.

Item 3 requires that the elevator control be self-centering a characteristic which is necessary for the attainment of control feel.

The reason for item 4 is obvious.

Design considerations. - A detailed analysis of the static longitudinal stability characteristics of various airplanes and the influence of various design features on the observed characteristics, is given in reference 3.

Requirement (I-C): Characteristics of the elevator control in accelerated flight.

1. By use of the elevator control alone, It should be possible to develop either the allowable load factor or the maximum lift coefficient at every speed.

2. The variation of elevator angle with normal acceleration in steady turning flight at any given speed, should be a smooth curve which everywhere has a stable slope.

3. For airplanes intended to have high maneuverability, the slope of the elevator-angle curve should be such that not less than 4 inches of rearward stick movement is required to change angle of attack from a C_L of 0.2 to C_{Lmax} in the maneuvering condition of flight.

4. As measured in steady turning flight, the change in normal acceleration should be proportional to the elevator control force applied.

5. The gradient of elevator control force in pounds per unit normal acceleration, as measured in steady turning flight, should be within the following limits:

a. For transports, heavy bombers, etc., the gradient should be less than 50 pounds per g.

b. For pursuit types, the gradient should be less than 6 pounds per g.

c. For any airplane, it should require a steady pull force of not less than 30 pounds to obtain the allowable load factor.

Reasons for Requirement (I-C): Item 1 of this specification requires that sufficient elevator control should be available to execute maneuvers of the minimum radius inherent in the aerodynamic and structural design of the airplane. Since the curvature of the flight path is directly related to the normal acceleration, it is obvious that the attainment of either the maximum lift coefficient or the allowable load factor is the limiting condition.

Item 2 is a requirement for stability in turning flight. Airplanes that do not meet this requirement tend to "dig" and overshoot desired accelerations in maneuvers, even though every use is made of visual and instrument references.

Item 3 specifies the amount of stability required of an airplane which must be maneuvered at or close to maximum lift without resort to visual or instrument references. It has been demonstrated by tests of several pursuit airplanes that longitudinal stability and control characteristics as specified are necessary for airplanes that require a high degree of control feel. The provision of such characteristics also reduces the time required to change angle of attack in entering rapid turns or zooms due to the simplified control manipulation associated with a definitely stable airplane.

The linear stick-force gradients specified in item 4 are, of course, very desirable as an aid to the pilot in obtaining the accelerations desired.

The numerical limits specified for the force gradients in item 5, are such that the minimum radius may be readily attained in any airplane. For pursuit types, gradients greater than 6 pounds per g were considered heavy by pilots. For airplanes where the load factor is lower, such as bombers, transports, etc., which are not required to maneuver continuously, a gradient of 50 pounds per g is not excessive. To ensure against inadvertent overloading of the structure, the 30-pound lower limit (item 5-c) is necessary. For pursuit airplanes with allowable load factors of 9, this lower limit would correspond to a gradient of about 4 pounds per g. For airplanes with lower load factors, such as bombers, transports, or light airplanes, the gradient in pounds per g would be proportionately higher.

Important design factors. - In turning flight, due to the curvature of the flight path, a stabilizing effect is obtained which increases the slope of the elevator-angle curve over that obtained in straight flight. The stick forces required to maintain a given lift coefficient are considerably greater than those for straight flight, however, because the elevator angles are higher and because they are obtained at greater speeds. For this reason, it is necessary to specify the upper limit of elevator-force gradients only for accelerated flight.

A linear relation between stick force and normal acceleration is always obtained provided the elevator-angle curve and hinge-moment coefficient curve have linear variations with angle of attack and deflection, respectively.

Requirement (I-D): Characteristics of the elevator control in landing.

l. (Applicable to airplanes with conventional landing gears only.) The elevator control should be sufficiently powerful to hold the airplane off the ground until three-point contact is made.

2. (Applicable to airplanes with nose-wheel type landing gears only.) The elevator control should be sufficiently powerful to hold the airplane from actual contact with the ground until the minimum speed required of the airplane is attained.

3. It should be possible to execute the landing with an elevator control force which does not exceed 50 pounds for wheel-type controls, or 35 pounds where a stick-type control is used.

Reasons for Requirement (I-D): For airplanes with conventional landing gears, the three-point attitude usually corresponds closely to that for the development of minimum speed for landing. In addition, an airplane alighting simultaneously on main wheels and tail wheel, is less likely to leave the ground again as a result of possessing vertical velocity at the time of contact.

The reason for item 2 is obvious.

The limits of allowable control force in landing were determined from considerations of the pilot's capabilities. The limit forces given are 80 percent of those which a pilot can apply with one hand to the different control arrangements with the control 12 inches from the back of the seat. (See references 4 and 5.)

Design factors. - The requirements of the elevator in producing three-point or minimum speed landings are by far the most critical from a standpoint of control power. Flight-test data show that low-wing monoplanes with flaps down require about 10° more up elevator to land than to stall in comparable conditions at altitude. Without flaps this increment due to ground effect is not so great, and with high-wing monoplanes without flaps the landing frequently requires less elevator than the power-off stall at altitude.

Requirement (I-E): Characteristics of elevator control in take-off.

During the take-off run, it should be possible to maintain the attitude of the airplane by means of the elevators at any value between the level attitude and that corresponding to maximum lift after one-half take-off speed has been reached.

Reasons for Requirement (I-E): The attitude of an airplane for optimum take-off characteristics depends upon the condition of the runway surface. On smooth, hard surfaces with low-rolling friction the shortest take-off run is obtained with a tail-high attitude. Where rolling friction is high, however, it is advantageous to maintain an attitude which gives high lift.

Design considerations. - Adequate control of the attitude angle during take-off depends more on the proper location of the landing gear with respect to the center of gravity than on the characteristics of the elevators themselves. This requirement certainly is not critical from a standpoint of elevator control. An airplane that has sufficient tail volume to be stable and sufficient elevator control to perform three-point or minimum-speed landings should meet this requirement easily, as long as the main landing-gear wheels are properly located.

Requirement (I-F): Limits of trim change due to power and flaps.

1. With the airplane trimmed for zero stick force at any given speed and using any combination of engine power and flap setting, it should be possible to maintain the given speed without exerting push or pull forces greater than those listed below when the power and flap setting are varied in any manner whatsoever.

a. Stick-type controls - 35 pounds push or pull.

b. Wheel-type controls - 50 pounds push or pull.

2. If the airplane cannot be trimmed at low speeds with full use of the trimming device, the conditions specified in item 1 should be met with the airplane trimmed full tail-heavy.

Reasons for Requirement (I-F): It is desired that emergency manipulation of flaps or throttles do not require simultaneous adjustments of the trimming device. The force limits specified are approximately 80 percent of the maximum that a pilot can apply with one hand. The one-hand limit is necessary to allow the adjustment of throttles, flaps, or trimming device while complete longitudinal control is maintained. It is, of course, desirable that the trim changes be less then the limiting values given. The ideal condition would be one where the stick forces required for trim were not influenced by the position of the flaps or throttles.

It is also desirable that the control position required to maintain a given speed or lift coefficient be independent of the power and flap position insofar as possible. It is not, however, believed reasonable or necessary to specify any definite limits at this time.

Design factors. - Because of simultaneous changes in downwash, dynamic pressure at the tail, and pitching moment of the airplane less tail, the trim change produced by variations of power and flap setting are very difficult to predict. Several of the effects, however, have opposite signs, so that with sufficient care it should be possible to restrict the trim changes to a reasonably low value. Wind-tunnel tests of a powered model of the design under consideration would be a great help if not an absolute essential in this connection.

Requirement (I-G): Characteristics of the longitudinal trimming device.

1. The trimming device should be capable of reducing the elevator control force to zero in steady flight in the following conditions:

a. Cruising conditions - at any speed between high speed and 120 percent of the minimum speed.

b. Landing condition - any speed between 120 percent and 140 percent of the minimum speed.

2. Unless changed manually, the trimming device should retain a given setting indefinitely.

Reasons for Requirement (I-G): It is, of course, desirable to be able to reduce the elevator force to zero in conditions where the airplane must be flown for protracted lengths of time. It is also desirable to be able to establish a trim condition within the allowable speed limits of the airplane so that release of the controls will not put the airplane in a dangerous position.

The reasons for item 2 are obvious.

II. Requirements for Lateral Stability and Control.

Requirement (II-A) : Characteristics of uncontrolled lateral and directional motion.

1. The control-free lateral oscillation should always damp to one-half amplitude within two cycles.

2. When the ailerons are deflected and released quickly, they should return to their trim position. Any oscillations of the ailerons themselves shall have disappeared after one cycle.

3. When the rudder is deflected and released quickly, it should return to its trim position. Any oscillation of the rudder itself shall have disappeared after one cycle.

Reasons for Requirement (II-A): Because of its relatively short period, the lateral oscillation must be heavily damped. It is not logical to specify limits for the period of the oscillation because the period is dependent on factors covered by other specifications and also because the period is dependent on the size, speed, and weight of the airplane. The amount of damping specified in item l has been obtained with all satisfactory airplanes tested.

Items 2 and 3 of the requirement (II-A) are included to ensure stability in the behavior of the lateral controls themselves.

Attention is called to the omission of a requirement for spiral stability. Tests have shown that the lack of spiral stability has not detracted from the pilot's ability to fly an airplane efficiently. In fact, it is very difficult to determine whether an airplane is inherently spirally stable or not, because divergence will occur with a spirally stable airplane if perfect lateral and directional trim do not exist or if slight asymmetry in engine poser occurs in a multiengine airplane. For these reasons a large amount of inherent spiral stability would be required to ensure against lateral divergence under control conditions.

Since it appears that the degree of spiral stability or instability is inconsequential or at least of doubtful importance under actual conditions, it is desirable to avoid any such requirement because the design conditions for spiral stability conflict with other factors known to be essential in the attainment of satisfactory flying qualities.

Design considerations. - The theory of dynamic stability has been rather extensively developed from a mathematical standpoint. The charts of reference 6 make the calculation of the dynamic characteristics a relatively simple matter, provided the stability derivatives are known. In general, however, the stability derivatives are not known and cannot be estimated to a reasonable degree of accuracy, particularly with power on.

On the basis of experience, however, it appears that the damping requirement is now a critical design condition. There is every indication that when other requirements of fin area and dihedral are met, the uncontrolled lateral motion will be satisfactory.

Items 2 and 3 of the requirement (II-A) are dependent, as was the elevator-free motion (requirement I-A), on the control-hinge moments, mass balance, and moment of inertia of the control systems.

Requirement (II-B): Aileron-control characteristics (rudder locked).

1. At any given speed, the maximum rolling velocity obtained by abrupt use of ailerons should vary smoothly with the aileron deflection and should be approximately proportional to the aileron deflection.

2. The variation of rolling acceleration with time following abrupt control deflection should always be in the correct direction and should reach a maximum value not later than 0.2 seconds after the controls have reached their given deflection.

3. The maximum rolling velocity obtained by use of ailerons alone should be such that the helix angle generated by the wing tip, pb/2V, is equal to or greater than 0.07.

where: p maximum rolling velocity, radians/second

b wing span

V true airspeed, feet per second

4. The variation of aileron control force with aileron deflection should be a smooth curve. The force should everywhere be great enough to return the control to trim position.

5. At every speed below 80 percent of maximum level-flight speed, it should be possible to obtain the specified value of pb/2V without exceeding the following

control-force limits:

a. Wheel-type controls: ±80 pounds applied at rim of wheel.

b. Stick-type controls: ±30 pounds applied at grip of stick.

Reasons for Requirement (II-B): Item 1 of this requirement states an obviously desirable condition for any control; i.e., that the response shall be proportional to deflection.

Item 2 is designed to eliminate controls that are unsatisfactory from a standpoint of lag in the development of the rolling moment, or controls in which the initial rolling action is in the wrong direction.

Item 8 was obtained by correlation of pilots' opinions and measured characteristics for some 20 different airplanes of various types and sizes (reference 7). It was found that pilots judged the adequacy of their lateral control on the basis of the helix angle generated by the wing tip of the airplane. Airplanes giving values of pb/2V less than 0.07 were always considered unsatisfactory.

Item 4 is a requirement for self-centering characteristics of the lateral control. This is a necessary condition for satisfactory control feel.

The specifications of item 5 was determined by the limitations of pilots in applying forces to the lateral controls. Lower forces are, of course, desirable.

Design considerations for Requirement (II-B): Item 1 represents a normal characteristic of flap-type ailerons, provided they are not deflected beyond the range where effectiveness is linear. Certain spoiler-type ailerons, however, have been unsatisfactory because of their failure to meet this requirement. In these cases, the variation of effectiveness with deflection was either markedly nonlinear or such that appreciable movements of the control about the neutral point were required before the ailerons became effective.

Item 2 also is met by all conventional flap-type ailerons. Again, however, certain arrangements of lateral controls that depend on spoiler action have proved unsatisfactory because of lag or incorrect initial development of the rolling moment. Detailed information on various satisfactory and unsatisfactory spoiler types may be found in reference 8 and later reports on the subject.

The specification of the helix angle, pb/2V ≥ 0.07 of item 3, corresponds approximately to requiring a rolling-moment coefficient, C_2, of 0.035 or greater. Actually, since pb/2V is equal to the ratio of the rolling-moment coefficient to the damping-moment coefficient, C / C_{2p}, a criterion in terms of C_2 alone is not strictly applicable. The damping-moment coefficient tends to decrease with increased taper of the wing and to increase with increased aspect ratio. However, for the aspect ratios and taper ratio likely to be used, the criterion considered in terms of rolling-moment coefficient alone, $O_2 \geq 0.035$ should be satisfactory. In several types tested, particularly the very large airplanes, control-cable stretch resulted in a very serious loss of aileron effectiveness. There is also indication that wing twist under the torsional loads applied by ailerons, should be considered in an interpretation of the rolling-moment coefficient required to obtain the specified value of pb/2V.

Item 4 sets the upper limit for aileron control friction since the ability of a control to center itself depends on the ratio of the inherent force gradient to the frictional force.

The control-force limits of item 5 are, of course, critical at the high speed specified. This requirement can be met by using existing design methods without servo control or mechanical booster systems except, perhaps, for the very largest airplanes that appear at this time.

Requirement (II-C): Yaw due to ailerons.

With the rudder locked at 110 percent of the minimum speed, the sideslip developed as a result of full aileron deflection should not exceed 20°.

Reasons for Requirement (II-C): Aileron yaw is responsible not only for annoying heading changes as a result of the use of ailerons, but also for a reduction of aileron effectiveness unless the rudder is carefully manipulated to eliminate the sideslip induced. This latter effect is also dependent on the rolling moment due to sideslip (dihedral effect).

The requirement for aileron yaw expressed in this manner clearly separates satisfactory characteristics from those considered unsatisfactory by pilots and, moreover, has the merit of relating the factors responsible for aileron yaw in a fundamental manner. The limiting condition of 20° sideslip seems surprisingly high, but the number of satisfactory airplanes that develop sideslip angles substantially this great, cannot be ignored. The requirement, however, is written to cover the critical low-speed conditions. At cruising speeds, comparable tests would give sideslip angles of the order of 5°.

Design considerations. - The sideslip due to ailerons is chiefly dependent on the aileron yawing moment, the yawing moment due to rolling, the dihedral effect, and the directional stability of the airplane. Compliance with the requirement depends mainly on the provision of sufficient directional stability, since the aileron yawing moment and the yawing moment due to rolling are determined by the aileron power. Of course, the designer has some control over the adverse aileron yawing moment through the use of differential in the control system and by increasing the profile drag of the up aileron. These effects, however, are generally small in comparison with inherent yawing moments due to ailerons and rolling velocity, which are always adverse in sign.

* The measured sideslip angle on which this and subsequent specifications are based, should not be confused with the angle of bank of the airplane. The angle of sideslip is simply that given by a vane free to pivot about a vertical axis and align itself with the relative wind.

The required amount of directional stability is simply that which will give an equilibrium of the yawing moments at or below the angle of sideslip specified. The adverse aileron yawing moments can, of course, be determined in the wind tunnel.

The yawing moment due to rolling for wings of various planforms is given in the charts of reference 9.

Requirement (II-D): Limits of rolling moment due to sideslip (dihedral effect).

1. The rolling moment due to sideslip as measured by the variation of aileron deflection with angle of sideslip should vary smoothly and progressively with angle of sideslip, and should everywhere be of a sign such that the aileron is always required to depress the leading wing as the sideslip is increased.

2. The variation of aileron stick force with angle of sideslip should everywhere tend to return the aileron control to its neutral or trim position when released.

3. The rolling moment due to sideslip should never be so great that a reversal of rolling velocity occurs as a result of yaw due to ailerons (rudder locked).

Reasons for Requirement (II-D): Item 1 ensures that the roll due to rudder will always be in the correct direction and that any lateral divergence will not be of a rapid type. It is also a necessary but not a sufficient condition for the ability to raise a wing by means of the rudder.

Item 2 was required to ensure that the rolling moment due to sideslip will be of the correct sign with controls free. The ability of the control to self-center here again is a requirement for control feel.

The reason for item 3 is obvious.

Design considerations. - Wind-tunnel data showing the effects of flaps, wing planform, and fuselage-wing arrangement on the rolling moment due to sideslip are given in references 10 and 11. These results are generally substantiated by flight test. With single-engine, low-wing airplanes, however, the dihedral effect in sideslips made to the left sometimes became negative at low speeds with power on, even though it was satisfactory with power off or with power on at higher speeds. Low-wing monoplanes generally required from 4° to 8° more geometric dihedral angle than high-wing monoplanes, to obtain the same effective dihedral effect. On airplanes with the trailing edges of the wing swept forward, flaps reduced the effective dihedral and where the trailing edge of the wing was a continuous straight line, flaps had little or no effect on the dihedral effect.

In order to meet item 2, the friction in the aileron control system must be low and the aileron required to overcome the rolling tendencies in the sideslip (dihedral effect) must exceed that at which the ailerons would tend to float due to the spanwise angle-of-attack variation.

The upper limit of the rolling moment due to sideslip (item II-D-3) is dependent on the yaw due to ailerons (item II-C-1) and the power of the aileron control (item II-B-3).

Requirement (II-E): Rudder-Control characteristics.

1. The rudder control should everywhere be sufficiently powerful to overcome the adverse aileron yawing moment.

2. The rudder control should be sufficiently powerful to maintain directional control during take-off and landing.

3. The rudder control should be sufficiently powerful to provide equilibrium of yawing moments at zero sideslip at all speeds above 110 percent of the minimum takeoff speed under the following conditions:

a. Airplanes with two or three engines: With any one engine inoperative (propeller in low pitch) and the other engine or engines developing full rated power.

b. Airplanes with four or more engines: With any one engine inoperative (propeller in low pitch) and the other engines developing full rated power.

4. The rudder control in conjunction with the other controls of the airplane provide the required spin-recovery characteristics.

5. Right rudder force should always be required to hold right rudder deflections, and left rudder force should always be required to hold left rudder deflections.

6. The rudder forces required to meet the above rudder-control requirements should not exceed 180 pounds (trim tabs neutral).

Reasons for Requirement (II-E): The reasons for these various items are obvious. Item 1 must, of course, be met if satisfactory turns are to be made at low speeds unless, of course, the directional stability is very great. Item 2 represents one of the most important functions for rudder control, although if a tricycle landing gear is used it becomes much lees important.

Items 3 and 6 should ensure adequate control over a symmetric thrust following engine failure subsequent to take-off. It does not seem necessary to retain directional control below the speed specified because of the probability that lateral instability due to stalling would set in first. The 180-pound force limit specified, is about 90 percent of the maximum that an average pilot can apply.

Design considerations. - The rudder power needed to meet item 1 of the above requirement, can be determined in the same manner that the directional stability required by aileron yaw was found (requirement II-C).

In at least one instance, item 2 of the above requirements was met without any rudder control. This was accomplished by using a tricycle landing gear and by eliminating the rudder-position variation with speed and power. However, due to the inherent instability of conventional landing gears, a certain amount of rudder control during take-off and landing will always be required when this arrangement is used, even though the rudder-trim change due to power or speed, were eliminated. Just how much rudder is needed here is not known. The efficiency of the brakes, type of tail wheel (lockable or free-swiveling), and the magnitude of the inherent ground-looping tendency undoubtedly enter into the problem. Also, in landing, the stalling characteristics of the airplane may have an important bearing. On the basis of data on hand, however, it appears that a rudder control that is sufficiently powerful to meet the other requirements outlined should generally be satisfactory from a standpoint of ground handling.

Items 3, 4, and 5 do not appear to require additional discussion.

Requirement (II-F): Yawing moment due to sideslip (directional stability).

1. The yawing moments due to sideslip (rudder fixed) should be sufficient to restrict the yaw due to ailerons to the limits specified in requirement (II-C-1).

2. The yawing moment due to sideslip should be such that the rudder always moves in the correct direction; i.e., right rudder should produce left sideslip and left rudder should produce right sideslip. For angles of sideslip between ±15° the angle of sideslip should be substantially proportional to the rudder deflection.

3. The yawing moment due to sideslip (rudder free) should be such that the airplane will always tend to return to zero sideslip regardless of the angle of sideslip to which it has been forced.

4. The yawing moment due to sideslip (rudder free with airplane trimmed for straight flight and symmetric power) should be such that straight flight can be maintained by sideslipping at every speed above 140 percent of the minimum speed with rudder free with extreme asymmetry of power possible by the loss of one engine.

Reasons for Requirement (II-F): The reasons for item are covered in discussion under requirement (II-C).

Item 2 of this requirement states a desirable characteristic for any control; i.e., the response should be proportional to the deflection.

Item 3 is designed to ensure satisfactory directional stability, particularly at large angles of sideslip where vertical tail stalling has frequently led to trouble. This requirement follows directly from the results of reference 12.

Item 4 is included to prevent the directional divergence following an engine failure from being excessively rapid. Although the ability to fly with rudder free on asymmetric power, is probably not in itself important, it is undoubtedly strongly related to the rate of divergence, and therefore the required quickness of action on the part of the pilots when this emergency occurs.

<u>Design considerations</u>. - The directional stability required to fulfill item 1 has been discussed under requirement (II-C).

General discussion of the factors that determine the fin area required to meet items 2 and 3 of this requirement, is given in reference 12. However, the interference effects of wing-fuselage position, vertical tail arrangements, etc., are so great that wind-tunnel tests would appear a necessary aid to design for these requirements. Since the directional stability at large angles of sideslip, however, is related to the manner in which the flow breaks down on the vertical surfaces and on its effect on the floating characteristics of the rudder, the scale of the test should be kept as great as possible.

<u>Requirement (II-G)</u>: Cross-wind force characteristics.

The variation of cross-wind force with sideslip angle, as measured in steady sideslips, should everywhere be such that right bank accompanies right sideslip and left bank accompanies left sideslip.

<u>Reasons for Requirement (II-G)</u>: Under normal conditions in a sideslip or skid, a force is produced which acts toward the backward-lying wing tip. Since this actual angle of sideslip cannot be observed by the pilot, the cross-wind force developed

allows appreciation of the fact that sideslip exists because of the lateral acceleration which occurs. In steady sideslips the cross-wind force is balanced by a component of the weight of the airplane, so that an angle of bank results. The greater the cross-wind force the greater is the angle of bank. An approximate relation between angle of bank ϕ and the crosswind force may be written as follows:

$$\phi = \sin^{-1} \frac{\text{cross-wind force}}{\text{weight of airplane}}$$

In addition to providing the pilot with "feel" of the sideslip or skid, the lateral attitude from which it is possible to recover with the rudder alone (without permitting a heading change) is directly related to the magnitude of the cross-wind force. Obviously, a positive dihedral effect is also necessary for the performance of this maneuver, but the fact remains that turning toward the low wing will always occur if the lateral attitude from which recovery is attempted, exceeds that which can be held in steady sideslip with full rudder.

For these and other reasons, large values of cross-wind force are desirable and more rigid specification than that given would lead to better flying qualities. On the other hand, it is not known whether this could be done without increasing the drag of the airplane.

None of the airplanes tested to date has failed to meet the requirement as written. It is included, however, because there is indication on the basis of wind-tunnel tests, that some future designs may actually develop cross-wind force of opposite sign to that normally experienced. Obviously, this condition could not be tolerated.

<u>Requirement (II-H)</u>: Pitching moment due to sideslip.

As measured in steady sideslip, the pitching moment due to sideslip should be such that not more than 1° elevator movement is required to maintain longitudinal trim at 110 percent of the minimum speed when the rudder is moved 5° right or left from its position for straight flight.

<u>Reasons for Requirement (II-H)</u>: A pitching-moment change due to sideslip is undesirable because it requires that the elevator as well as the rudder must be coordinated with the ailerons. Also, since sideslip of considerable amounts may be carried inadvertently, a marked variation of pitching moment with sideslip will tend to produce inadvertent angle-of-attack changes. The condition is critical at high lift coefficients, so compliance with the specifications given should automatically ensure satisfactory characteristics at higher speeds.

<u>Design considerations.</u> - It is believed that the change in pitching moment with sideslip occurs as a result of the downwash change experienced by the horizontal tail as it moves from behind the wing center. In most cases, the moment produced is a diving moment because of the relatively high concentration of downwash at the wing center due to the propeller or partial-span flaps. It has also been noted that the magnitude of the pitching moment due to sideslip, progressively decreased as the angle of attack was reduced, presumably because of the corresponding reduction of downwash angles.

Requirement (II-I): Power of rudder and aileron trimming devices.

1. Aileron and rudder trimming devices should be provided if the rudder or aileron forces required for straight flight at any speed between 120 percent of the minimum speed and the maximum speed, exceed 10 percent of the maximum values specified in requirements (II-B-5) and (II-E-6), respectively, and unless these forces at cruising speed are substantially zero.

2. Multiengine airplanes should possess rudder and aileron trimming devices sufficiently powerful in addition to trim for straight flight at speeds in excess of 140 percent of the minimum speed with maximum asymmetry of engine power.

3. Unless changed manually, the trimming device should retain a given setting indefinitely.

Reasons for Requirement (II-I): The reasons for the items listed above are obvious.

Requirement (III): Stalling characteristics.

1. The approach of the complete stall should make itself unmistakably evident through any or all of the following conditions:

a. The instability due to stalling should develop in a gradual but unmistakable manner.

b. The elevator pull force and rearward travel of the control column should markedly increase.

c. Buffeting and shaking of the airplane and controls produced either by a gradual breakdown of flow or through the action of some mechanical warning device, should provide unmistakable warning before instability develops.

2. After the complete stall has developed, it should be possible to recover promptly by normal use of controls.

3. The three-point landing attitude of the airplane should be such that rolling or yawing moments due to stalling, not easily checked by controls, should not occur in landing, either three-point or with tail-first attitude 2° greater than that for three-point contact.

Reasons for Requirement (III): The items of this requirement are in keeping with all others given; i.e., it demands all that can be obtained with existing knowledge and yet is sufficiently rigid, so that any airplane that complies with the specification will be reasonably safe in terms of our present standards. Since there is never occasion in the normal operation of an airplane for a pilot to stall intentionally, such characteristics that provide warning of the stall are given first importance. If the warning is unmistakable, the relative violence of the actual stall loses much of its significance because it would then occur only as an intentional act on the part of the pilot and at a safe altitude. Item 2 is included to ensure that recovery from an intentional stall can be promptly made.

Item 3 is an outgrowth of some experience in studying ground-handling problems. In most cases, poor stalling characteristics are troublesome in landing because of wing dropping either during the actual landing flare or after the airplane has alighted during the landing run. In other cases the ring stall has influenced the flow

at the vertical tail in such a manner that powerful yawing moments have developed. Unless the stall itself can be made to develop in a gentle manner, the cure for these characteristics can be effected by preventing the occurrence of the stall altogether in the landing maneuver.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va.

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Document 3-29(a-d)

(a) C. H. Dearborn, "Full-Scale Wind-Tunnel Tests of Navy XF2A-1 Fighter Airplane," NACA Confidential Memorandum Report, 17 May 1938, RA file 603, LHA.

(b) H. J. E. Reid, Engineer-in-Charge, LMAL, to NACA, "Drag-reduction investigation on XP-40 airplane," 7 Apr. 1939, RA file 637, LHA. Includes attachment from Clinton H. Dearborn, Aeronautical Engineer, LMAL, "Test Program on XP-40 Airplane in Full-Scale Wind Tunnel," undated.

(c) Don R. Berlin, Chief Engineer, Curtiss Aeroplane Division, Curtiss Wright Corp., to Major Carl F. Greene, Materiel Division, Liaison Officer, NACA Laboratory, Langley Field, VA, 4 Aug. 1939 [originally classified "Confidential"], RA file 637, LHA.

(d) C. H. Dearborn, Aeronautical Engineer, LMAL, to Engineer-in-Charge, "Investigation for determining means of increasing the high speed of the XP-41 airplane," 8 June 1939, RA file 672, LHA.

As one can see in the following documents involving its drag reduction program, the NACA followed a very systematic method of experimental parameter variation. First, its engineers examined an airplane in detail, identifying those external features most suspected of causing unnecessary drag. They then made the airplane as aerodynamically clean as possible, by carefully removing protuberances like the radio antenna and using putty or tape to cover holes and leaks and to reshape irregular surfaces such as the cockpit canopy. Following this, they mounted the plane in the test chamber, and measured its drag at various wind speeds. In this faired and sealed condition, the airplane naturally proved to have less drag than the original body, but it was impossible for this pristine shape, with essential parts covered up or removed, actually to fly. The wind tunnel workers returned the plane to its service condition, item by item, and evaluated the change in drag caused by each action.

In the case of the cleanup tests of the Seversky XP-41 in late 1939, Langley studied the drag of the airplane in 18 different configurations. The data indicated that the changes in drag values corresponding to the steps of the cleanup process were generally small, amounting to only a few percent of the total drag coefficient and thus involving only small speed changes. Taken together, however, increments like these often resulted in impressive gains in total performance.

Document 3-29(a), C. H. Dearborn, "Full-Scale Wind-Tunnel Tests of Navy XF2A-1 Fighter Airplane," NACA Confidential Memorandum Report, 17 May 1938.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

CONFIDENTIAL MEMORANDUM REPORT
For the
Bureau of Aeronautics, Navy Department
FULL-SCALE WIND TUNNEL TESTS OF NAVY XF2A-1 FIGHTER AIRPLANE

By C. H. DEARBORN

INTRODUCTION

At the request of the Bureau of Aeronautics, Navy Department, an investigation has been conducted in the full-scale wind tunnel on the Navy XF2A-1 fighter airplane to determine: (1) its minimum and high-speed drag coefficients, (2) reductions in drag to be obtained by improving the fairing of various parts of the airplane, and (3) probable points at which carbon monoxide enters the cockpit.

This report presents as Part I the wind-tunnel investigation, and as Part II recommended changes in the airplane which would involve major alterations to the design.

PART I - WIND-TUNNEL INVESTIGATION TESTS

The Navy XF2A-1 airplane was mounted on the full scale wind-tunnel balance as shown in figure 1. The propeller was removed and all control surfaces locked in their neutral positions for the tests. Inasmuch as the time allotted for the performance of the tests of this airplane was very limited, it was necessary to reduce the amount of tunnel testing to a minimum.

First a force test was made on the airplane in its normal condition over a small angle-of-attack range, including the minimum and high-speed drag angles of attack at four test speeds from 60 to 100 miles an hour. From this test it was ascertained that an angle of attack of 1.05° approximates the mean angle between the reported

high speeds for sea level and an altitude of 15,200 feet, and all further force tests were therefore confined to this angle of attack.

In order to study the general flow conditions, wool tufts were placed at frequent intervals over the fuselage. To determine points of possible entry of carbon monoxide into the fuselage, tufts were placed at the edge of the opening for the arrester gear hook at the end of the fuselage and at the opening around the tail-wheel assembly. An observer with a fine wool tuft on the end of a rod was stationed in the cockpit to note the entry of air around proximity of the cockpit. The observation of the tufts was carried out at a tunnel speed of 60 miles an hour, and, in addition to visual observation, moving pictures were made.

Next a series of force tests was conducted on the airplane at the high-speed attitude and for the same four tunnel speeds as noted for the standard condition but with the following modifications to the airplane:

(1) With streamline fairings over the engine exhaust stack to obtain an indication of what drag reduction might be made by altering the stacks (fig. 2).

(2) With cover plates over the gaps between the top of the wheels and the ends of the landing-gear strut fairings in addition to change (1) (fig. 2).

(3) With landing wheels removed and flush cover plates installed over wheel wells (fig. 2).

(4) Changes covered by items (1) to (3) were removed, and modifications made to the engine cowling, which included refairing the carburetor intake scoop, the oil-cooler intake scoop, the openings at the forward ends of the blast tubes, and fairing out of the indentation at the forward end of the fuselage at the point where the blast tubes leave the fuselage. Figure 3 shows the modification to the oil-cooler scoop, figure 4 shows the modifications made on top of the cowling, and figure 5 presents sections of the changes in fairing which were accomplished by the addition of plasticine. The sections in figure 5 were patterned after low-drag engine cowls developed in the high-speed tunnel and during the modifications plasticine was added until the turbulent wakes noted by the behavior of the tufts for the original condition disappeared. The openings at the forward ends of the blast tubes were sealed as the greater part of the disturbing effect at these points appeared to be due to the flow of air from within the cowl through the areas between the blast tubes and openings in the cowl. As there was no flow of air through the blast tubes, the arrangement tested appears to simulate a possible service installation.

(5) With gun sight removed to determine its drag.

(6) The break in the contour of the cockpit enclosure at the end of the movable section was faired out by the addition of a sheet-metal patch. This change was made in the presence of changes (4) and (5).

(7) The final change consisted of adding a rounded windshield, which is shown with item 6 in figures 6(a) and 6(b). (Original figures not included herein.)

At the completion of the force tests, surveys were made in the wake of the left wing by means of a comb of total-head and static tubes. The measurements were made at the five spanwise stations shown in figure 7 for a test speed of 90 miles an hour and for angles of attack corresponding to zero lift and to the high-speed condition. Surveys were repeated at stations 4 and 5 with the ailerons sealed in order that the drag due to the aileron slot might be determined.

RESULTS AND DISCUSSION

The tuft observations are summarized in figure 8. It will be noted that in general the flow is satisfactory over the fuselage with the exception of the areas in back of the air-intake scoops, small cowls at the forward ends of blast tubes, and the exhaust stacks. A reproduction of several frames of the moving-picture film in figure 9 shows that the flow at the wing-fuselage juncture is very satisfactory. The flow at the top of the engine cowling, however, was particularly turbulent. The reproduction of the moving-picture film covering this area in figure 10 unfortunately does not give as clear a picture of this condition as might be desired, and it is therefore recommended that recourse be taken to the projection of the moving-picture film. It will, however, be noted in figure 10 that there is considerable movement of the tufts from one frame to another as for example may be seen by the changed attitude of the tuft marked in the upper frame. One of the greatest disturbing factors at the forward edge of the cowling appeared to be the flow of air outward between the blast tube and the cowling.

The observer in the cockpit during the tuft observation reported air flow into the cockpit at the following points, which may be locations at which carbon monoxide enters the fuselage.

TABLE 1

Locations	Nature of inflow
Case and link ejector chutes	Slight inflow
Flap indicator control	Slight inflow
Opening in side of fuselage for aileron control tube	Strong flow
Flow from rear end of fuselage	Steady flow
Bomb sight windows.	No flow or so small as not to be discernible
Gas drain tubes.	No flow or so small as not to be discernible

The principal point of air flow out of the cockpit was at the gap at the end of the movable portion of the cockpit enclosure. The tufts placed around the edges of the opening at the end of the fuselage for the arrester hook and around the opening for the tail-wheel assembly showed a infinite inflow of air into the fuselage.

The force-test results for each individual run (figs. 11(a) to 11(h)) show that the scale effect on the lift and drag coefficients is small within the speed range of the tests. The coefficients have been corrected for all wind-tunnel effects and are based on a wing area of 209 square feet. To aid in visualizing the condition of the airplane for each run, the letter accompanying the figure number is the same as that used to designate the various conditions of the airplane, shown on the chart of figure 13. The results of the force tests are summarized in figures 12 and 13.

It will be noted that the fairing over the exhaust stacks, the cover plates over the wheel wells, and the modifications to the engine cowling accounted for appreciable reductions in the drag of the airplane, with the latter accounting for by far the largest reduction. The summation of the drag reduction, which is obtained by adding the difference in the drag coefficients for conditions A and D and A and H, is equal to the drag-coefficient increment 0.0103, which constitutes about 27 percent of the high-speed drag coefficient of the normal airplane. It is also of interest to note that the modifications to the cockpit enclosure did not account for any reduction in drag.

A recommended type of exhaust stack to reduce the drag over the existing type is suggested by an investigation of air discharge openings conducted in the atmospheric wind tunnel. The stack would consist of a turn to the rear of 90° or preferably a slightly smaller angle to keep the exhaust gases off the fuselage with one or two guide vanes to reduce the bend loss.

The profile drag coefficients obtained by the momentum method for the five stations along the semispan of the wing are shown by the experimental points in figures 14 and 15. The profile drag coefficients for the two stations in back of aileron for the conditions of the normal aileron slot and sealed slot given in the table show that the aileron slot does not increase the wing drag.

Document 3-29(b), H. J. E. Reid, Engineer-in-Charge, LMAL, to NACA, "Drag-reduction investigation on XP-40 airplane," 7 Apr. 1939.

R.A. 637

Langley Field, Va.
April 7, 1939

From LMAL
To NACA

Subject: Drag-reduction investigation on XP-40 airplane.

Reference: NACA Let. Mar. 30, 1939, CW and Enc.

1. A conference was held in the full-scale wing tunnel to discuss the test program for investigating means of increasing the high-speed of the XP-40 airplane. Those in attendance were: Mr. Hovgard, of the Curtiss Company, and Messrs. Reid, DeFrance, Silverstein, and Dearborn, of the Laboratory staff. The items of the Curtiss letter to the Material Division forwarded with reference were reviewed and a number of them have been included in the test program. An investigation of the cooling system drag is of particular interest as it is believed that a substantial reduction in drag may be made by revising the present system.

2. It now appears that the tests will be under way not later than April 12.

3. The test program as drafted at the conclusion of the conference is enclosed.

H. J. E. Reid,
Engineer-in-Charge

CHD. AMD
Enc* 3 copies program

EWM

TEST PROGRAM ON XP-40 AIRPLANE IN FULL-SCALE WIND TUNNEL

1. Smooth fuselage conditions, power-off tests, gun-blast tubes and carburetor inlets removed, exhaust pipes off, Prestone radiator retracted into fuselage and radiator cowling removed, original windshield, propeller off, holes in propeller spinner sealed, oil radiator ducts sealed.
2. Same as 1, with exhaust stacks added.
3. Same as 2, with carburetor inlets added, blast tubes omitted.
4. Same as 3, with blast tubes added.
5. Same as 4, with oil-cooler duct inlets open.
6. Power-on, with airplane in same condition as 5. This is the same as normal condition of airplane except the Prestone cooling system is retracted into fuselage, and the lower fuselage line faired into a smooth line.
7. Modified radiator installations radiator raised above present position and proper opening allowed for cooling airflow. Force and airflow measurements will be made at high speed, and climbing conditions for both power-off and power-on conditions.
8. Original airplane conditions, with radiator in normal position. Force and airflow measurements for high-speed and climbing conditions, power-off and power-on.
9. Power-on tests to determine the effect of covering the gaps in the nose of the spinner and around the propeller blades.
10. Study of methods for improvement in carburetor inlets and blast tubes, in the event that previous tests indicate it to be desirable.
11. Measurements of drag and airflow for various modifications of the original radiator installation.
12. Modifications to fairings over the retracted landing gear.
13. Measurements of wing-profile drag by means of the momentum method for high-speed conditions.
14. Measurements of the drag change due to sealing the control surface slots by momentum method at high-speed flight condition.
15. Study of leading edge fillets at wing roots. Measurements at high-speed and landing conditions, flaps up and down.
16. Measurement of the critical compressibility velocities by means of pressure measurements along fuselage and wing.
17. Measurement of boundary-layer transition location in order to determine the possibilities for drag reductions due to smoothing the wing.

It is the opinion of the staff that no large improvement in the windshield can be made, so tests for this purpose will not be included.

Clinton H. Dearborn

Aeronautical Engineer

CHD:RL
8JD
EWM

Document 3-29(c), Don R. Berlin, Chief Engineer, Curtiss Aeroplane Division, Curtiss Wright Corp., to Major Carl F. Greene, Materiel Division, Liaison Officer, NACA Laboratory, Langley Field, VA, 4 Aug. 1939.

August 4, 1939

Major Carl F. Greene
Material Division
Liaison Officer
NACA Laboratory
Langley Field, Va.

Enclosure: (A) Three View Drawing P-2251.
(B) Power Plant Drawing P-2278.
(C) Alternate Power Plant Drawing P-2414.

Dear Carl:

We have your letter dated July 31st in which you inquire concerning the status of the Curtiss CP-39-13 full scale model to be tested in the NACA full scale tunnel. In answer to your questions which you have asked the following is submitted for your information:

(a) As you know, the program which involves testing at NACA was arranged and tentatively approved by conference at Wright Field on July 19, 1939. Although no written approval has been received here at this date, we believe that you can feel assured that the program is definitely approved by Wright Field. Mr. C.H. Dearborn of the NACA staff paid us a visit this week for the purpose of arranging details concerning the design of the model and the various tests which are desired. The points which were discussed and agreements which were reached are detailed as follows:

(1) The model will be complete and all details affecting aerodynamics will be represented as nearly as possible.

(2) It will not be necessary to incorporate a motor of propeller. Mr. Dearborn feels that the effect of power can be calculated and it would simplify the model and the test procedure to omit this refinement.

(3) The landing gear will be represented and it is desired to obtain tests with the landing gear in both the extended and retracted positions.

(4) The balance supports on the wing will be spaced somewhere between 16 and 18 feet. We will advise the exact spacing as soon as this can be determined. Supports for the model will be made in accordance with NACA drawing D-6947.

(5) Control surface gaps will be represented but control surfaces will not be made movable. The design of control surfaces will be such that they may be made movable should this appear to be desirable at a later date. The flap will be represented and will be made movable to determine its effect upon the lift characteristics of the airplane.

(6) No tail wheel will be represented.

(7) Jacking the hoisting provisions will be made. Leveling points will also be included.

(8) Two types of cooling systems will be completely simulated. Actual radiators installed within the wing contour together with suitable inlet and outlet ducts will be provided. The design of the ducts will conform to the latest NACA research on this subject. Another cooling system complete with radiators and ducts will be provided in a position below the motor. It is planned to conduct tests both ways to determine which of these cooling systems or combination of them will be most desirable.

(9) External ducts, exhaust manifolds, and gun blast tubes will be represented, but made removable with flush covers for the openings for the purpose of obtaining comparative tests.

(10) No tests will be required on the wing alone.

(11) Templates for the wing and tail surface profiles will be furnished for checking purposes.

(12) The construction of the entire model will be made rugged and a structural investigation will be made to insure that the strength requirements of NACA specifications will be met.

(13) In general, it is desired to secure lift and drag information only although this may be modified later if conditions make it advisable. In particular, we wish to make a thorough study of the drag characteristics of the design with the idea of securing every practicable refinement which can be realized.

(b) It is anticipated that the model will be available about August 25th for NACA to start their tests.

(c) Regarding your proposed inspection of the model we estimate that you could obtain a very good idea of its construction and appearance if you would come here between August 15 and 20. If you can arrange this we shall be very glad to discuss in further detail the outline of the testing procedure.

Enclosures (A), (B), and (C) will give you a preliminary idea of the important characteristics of the design as they now exist. It is requested that you convey the information contained herein to the proper NACA authorities including Mr. Dearborn as this letter constitutes a confirmation of the conference with Mr. Dearborn.

Sincerely yours,

Curtiss Wright Corporation
Curtiss Aeroplane Disivion

Don R. Berlin
Chief Engineer

Document 3-29(d), C. H. Dearborn, Aeronautical Engineer, LMAL, to Engineer-in-Charge, "Investigation for determining means of increasing the high speed of the XP-41 airplane," 8 June 1939.

L. M. A. L.
Langley Field, Va.
June 8, 1939.

MEMORANDUM For Engineer-in-Charge.

Subject: Investigation for determining means of increasing the high speed of the XP-41 airplane.

1. Attached is a test program of the XP-41 airplane. Two additional copies of this program have been prepared for transmittal to the Materiel Division Liaison Office.

C. H. Dearborn
Aeronautical Engineer.

CHD.P
Enc.

SJD.
EWM.

PROGRAM OF TESTS ON THE XP-41 AIRPLANE IN THE FULL-SCALE WIND TUNNEL

Test Description

1. Drag tests of the airplane in smooth condition with:
   oil-cooler scoop removed and duct openings closed
   ejector chute removed
   accessory exit closed
   landing gear faired
   intercoolers removed
   canopy rail faired
   carburetor scoop removed
   gaps in cowling at sheet-metal joints closed
   sanded walkway surface removed
   cockpit ventilator opening closed and faired
   blast tube fairing removed
   venturi tubes through cowling removed and openings closed
   aerial removed
   cowling exit closed.

   (a) with existing N. A. C. A. cowling.
   (b) with closed streamlined spinner.

2. Drag and air-flow quantity measurements of the ship as in 1 except with cowling exit open; flaps open and closed:

   (a) with existing N. A. C. A. cowling.
   (b) with faired spinner and three variations of inlet nose shape.

3. Momentum measurements behind the wing to determine the profile drag for:
   (a) normal wing surf ace.
   (b) a section of the wing with aerodynamically smooth leading edge.
4. Boundary-layer velocity surveys to determine the transition point on the wing:
   (a) normal wing surface.
   (b) on the section noted in 3 (b) above.

5. Drag and air-flow quantity measurements for new oil-cooler installation under the engine.

6. Determination of the drag of the airplane in normal condition as received and the individual drag of the following component parts of the airplane:

a. oil-cooler installation
b. ejector chute
c. cooling airflow
d. landing gear fairing
e. intercooler installation
f. canopy rail
g. carburetor scoop
h. gaps in cowling at sheet-metal joints
i. sanded walkway surface
j. cockpit bentilator
k. blast tube fairings
l. cowling venturi-tubes

7. Static pressure surveys on fuselage, canopy, cowling, and wing to determine the critical compressibility velocity.

8. Measurement of the engine-cooling air quantity for the original installation.

9. Measurement of the maximum lift in the normal service landing condition and tuft surveys to observe the progression of the stall.

10. Measurement of rolling moments and stick forces over the range of aileron deflections with flaps undeflected and deflected for service landing.

Document 3-30(a-c)

(a) General Henry H. Arnold to NACA, "Full-Scale Wind-Tunnel Tests of XP-39," 9 June 1939, copy in Research Authorization (RA) file 674, LHA.

(b) Smith J. DeFrance to Chief, Aerodynamics Division, Langley Aeronautical Laboratory, "Estimated High-Speed of the XP-39 Airplane," 25 Aug. 1939, copy in RA file 674, LHA.

(c) Larry Bell, President, Bell Aircraft, to George W. Lewis, NACA, 17 Jan. 1940, copy in RA file 674, LHA.

Although the NACA's drag cleanup work usually proceeded in a typical way, each aircraft that went through the process was different, posed its own problems, and required special attention. This was certainly true in the case of the Bell XP-39 Airacobra, the eleventh in the series of military planes subjected to the NACA cleanup operation. The following string of documents takes the reader through roughly eight months of NACA investigation into the design details of the XP-39, from August 1939 to April 1940. It is important to remember that, early in this stretch of time, Nazi Germany invaded Poland, World War II began, and President Franklin D. Roosevelt called for U.S. production of an incredible 50,000 planes a year. In August 1940, the air war known as the Battle of Britain would be fought. The development of advanced combat aircraft had grown extremely urgent.

Bell Aircraft designed the XP-39 as a 400-mph fighter. In the spring of 1939, at Wright Field in Ohio, the unarmed prototype flew to a maximum speed of 390 mph at 20,000 feet. The aircraft reached this speed, however, with a gross weight of only 5,500 pounds, about a ton less than a heavily armored production P-39. That meant that the existing aircraft, when normally loaded, would have a hard time exceeding 340 mph. Still, the test performance impressed the Air Corps enough for it to issue a contract three weeks later for 13 production model YP-39s. It should be clear from the first item in the string below that Gen. Henry H. "Hap" Arnold was desperate for a new fighter and hoped that the speed of the airplane could be increased considerably by cleaning up the drag.

The second item documents some of what NACA Langley was doing with the XP-39 in the late summer of 1939. Tests in the Full-Scale Tunnel indicated that the prototype in a completely faired condition had a drag value of only 0.0150 compared to 0.0316 in the original form. If everything possible were done to reduce drag, the airplane's maximum speed might be raised by as much as 26 percent. But as NACA realized, not all of the changes were feasible for the production aircraft. Bell made what changes it could and even considered the NACA's nonaerodynamic recommendation to try a more powerful engine, one with a geared supercharger. Even without that sort of engine, the refined aircraft, designated XP-39B, managed to reach a speed of 375 mph at 15,000 feet in its first trials. (This seemingly meant a *reduction* in speed of 15 miles per hour, but the XP-39B weighed about 300 pounds more than the original, due mostly to the addition of armament.) In the third item below, Lawrence Bell, the president of Bell Aircraft, wrote to the NACA in appreciation of what he called "extraordinarily satisfactory results."

At the request of Bell and the air corps, the NACA continued to pursue drag cleanup on the airplane. Unfortunately, the XP-39B's top speed never came anywhere near 400 mph—for that matter, no version of the Airacobra ever would. In September 1940, the first YP-39, having incorporated most of the suggestions called for by the NACA, notably the installation of propeller cuffs and wheel well covers, flew to a top speed of 368 mph at 13,300 feet. Deliveries of the first production model P-39s, which were very similar to the service-test YP-39, began four months later. In 1941, the United States sent nearly 700 Airacobras to Great Britain and the Soviet Union under Lend-Lease. After the Japanese attack at Pearl Harbor, the Air Corps rushed P-39 units into action in the South Pacific.

The production P-39s could never fly more than 368 mph. This speed ceiling seems to indicate that the NACA's drag cleanup program failed, as the first P-39 prototype in test flight had reached 390 mph. But the reason for this slightly slower speed was the hundreds of pounds of weight that had to be added in the form of a bigger power plant, guns, and heavy armor plate. In other words, if the P-39 had not gone through drag testing, it would have been slower than it ultimately was, maybe as much as 25 to 30 miles per hour. Although limited in its top speed, the plane possessed reasonable stability and roll rates and maneuvered well at low altitudes. These qualities made the airplane useful in combat ground support as a strafer and as a fighter-bomber.

Document 3-30(a), General Henry H. Arnold to NACA, "Full-Scale Wind-Tunnel Tests of XP-39," 9 June 1939.

Address reply to
Chief of the Air Corps
War Department
Washington D.C.

452.1

WAR DEPARTMENT
Office of the Chief of the Air Corps (4-e)
Washington

June 9, 1939.

SUBJECT: Full Scale Wind Tunnel Tests of XP-39.

TO: National Advisory Committee for Aeronautics,
Navy Building, Washington, D.C.
(Attention: Dr. G. W. Lewis,
Directory of Aeronautical Research)

It is requested that the XP-39 be placed in the full scale wind tunnel at Langley Field, Virginia, for the purpose of determining performance characteristics of this airplane and what improvements are recommended to increase aerodynamic and performance characteristics.

H. H. Arnold
Major General, Air Corps,
Chief of the Air Corps.

Document 3-30(b), Smith J. DeFrance to Chief, Aerodynamics Division, Langley Aeronautical Laboratory, "Estimated High-Speed of the XP-39 Airplane," 25 Aug. 1939.

Langley Field, Va.,
August 25, 1939.

MEMORANDUM For Chief Aerodynamics Division.
Subject: Estimated high speed of the XP-39 airplane.

1. In accordance with a telephone conversation with Dr. Lewis on August 24, a survey of the data obtained in the full-scale tunnel on the XP-39 airplane has been made to determine the high speed with a 1,350-horsepower engine and geared supercharger. The drag coefficient for the XP-39 airplane, as tested in the tunnel at 100 miles per hour without the turbo supercharger and with the original wing cooling ducts in both wings, was found to be 0.0210. From experience on other airplanes, this value has been corrected for Reynolds number of the tests to Reynolds number of flight speed, and the value is 0.0191, which would produce a high speed of 408 miles per hour. The wing cooling duct originally installed in this airplane is not of the best design, both as to the inlet and the structure within the duct. The properties of the duct can be improved, and if the structure can be removed from the duct, the corrected high-speed drag coefficient would be reduced to 0.0180, corresponding to a high speed of 416 miles per hour. The wheels at present on this airplane are too large to be completely housed within the wing and protrude below the lower surface. By decreasing the size of the wheels so that they can be completely faired within the wing, the corrected high-speed drag coefficient would be reduced to 0.0164, corresponding to a high speed of 429 miles per hour. The speeds quoted are based on 1,350 horsepower at 16,000 feet and a propulsive efficiency of 85 percent.
2. The above figures are based on the radiator size and quantity of air required for the engine which is now in the airplane. It is not known how much additional air would be required for cooling a 1,350 horsepower engine, and if this is increased by a large amount, the high-speed values given would be reduced, but it is not believed that there would be any difficulty in obtaining at least 410 miles per hour unless a compressibility shock wave is experienced on some part of the airplane. It appears that the recommended changes to obtain the higher speed, namely, installation of a 1,350-horesepower geared supercharger engine, improved wing ducts in both wings, and smaller wheels completely faired into the wing, could easily be made on the present airplane without much delay.

Smith J. DeFrance,
Principal Aeronautical Engineer.

Document 3-30(c), Larry Bell, President, Bell Aircraft, to George W. Lewis, NACA, 17 Jan. 1940.

BELL AIRCRAFT CORP.
2050 Elmwood Avenue
Buffalo, N. Y.

January 17, 1940.

Dr. George W. Lewis
National Advisory Committee for Aeronautics
Munitions Building
Washington, D. C.

My dear Dr. Lewis:

I wanted to tell you that after we had incorporated the changes in the XP-39B, as a result of the wind tunnel tests at Langley Field, we are getting extraordinary satisfactory results. From all indications the XP-39 will do over 400 m.p.h., with 1150 H.P. All of the changes were improvements and we have eliminated a million and one problems by the removal of the turbo supercharger.

The cooling system is the most efficient thing we have seen. The inlet ducts on the Prestone Radiator are closed up to 3% and the engine is still overcooling.

I want to convey to you personally and your entire organization, both at Washington and at the Laboratory, our very deep appreciation of your assistance in obtaining these very satisfactory results.

With kindest regards,

Sincerely yours,

Larry Bell
President

Document 3-31(a-c)

(a) Excerpts from Clinton H. Dearborn, "The Effect of Rivet Heads on the Characteristics of a 6 By 36 Foot Clark Y Metal Airfoil," NACA *Technical Note* 461, May 1933.

(b) Excerpts from Manley J. Hood, "The Effects of Some Common Surface Irregularities on Wing Drag," NACA *Technical Note* 695, March 1939.

(c) Charles Peyton Autry, Boeing Aircraft Company, "Drag of Riveted Wings," in *Aviation* (May 1941): 53-54.

In the 1930s, aeronautical engineers came to understand that low drag required more than a correct wing profile or overall effective aerodynamic shape. Without extremely smooth surface conditions, an airplane's design would be seriously compromised, especially as aircraft speed increased to well over 200 miles per hour. Wrinkles in metal, dents, even scratches and the grain embedded in paint upset the airflow. These surface irregularities caused various instabilities. They generated rising pressures in places where pressure otherwise would be falling, and in the critical boundary layer, they hastened transition from laminar to turbulent flow. Means had to be found to smooth out all these bumps and nicks if an airplane were ever to perform to the maximum potential foreseen by its designer.

One of the most upsetting surface features damaging the aerodynamic performance of an airplane once more and more aircraft came to be built out of stressed-skin metal were rivet heads. As indicated in the documents below, NACA researchers began looking into this problem in the early 1930s (although the first proposal for flush riveting seems to have been in a patent proposal made by Charles Ward Hall, president of the Hall-Aluminum Aircraft Corporation in Buffalo, New York, in 1926). They found that the dome shapes of rivet heads could increase a wing's parasite drag (i.e., the drag forces that are not formed in the production of lift) by as much as 25 to 30 percent over that of a smooth wing. If happening with a big transport plane, it could mean a drain on an engine amounting to about 150 horsepower. If the rivets were countersunk, the increase in drag would be considerably less, only 5 to 6 percent.

Both the NACA and the aircraft industry experimented throughout the late 1930s with various means by which to improve the surface finish of airplanes. They tried slicing off rivet heads and eventually moved to an advanced technique of flush riveting. (For a detailed analysis of this development, see Walter G. Vincenti, "Design and Production: The Innovation of Flush Riveting in American Airplanes, 1930-1950," in Vincenti's *What Engineers Know and How They Know It*, pp. 170-199.)

The three documents below, two from the NACA and one by a Boeing engineer, report on the problem of protruding rivet heads and other surface irregularities especially as it related to wing drag. It is noteworthy that none of the revolutionary American airplanes of the early 1930s, such the Northrop Alpha, Boeing Monomail, as well as the DC-1, DC-2, and DC-3, incorporated flush rivets to any significant extent. They all still had the problem of rivets protruding into the airstream. By the early 1940s, however, virtually all U.S.-built metal airplanes benefited from flush rivets. As with the NACA's drag reduction program, the development of flush riveting represents the final stage in the airplane design revolution, during which the last minor details of reinventing the airplane took form.

Document 3-31(a), Excerpts from Clinton H. Dearborn, "The Effect of Rivet Heads on the Characteristics of a 6 By 36 Foot Clark Y Metal Airfoil," NACA **Technical Note** *461, May 1933.*

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE NO. 461

THE EFFECT OF RIVET HEADS ON THE CHARACTERISTICS OF A 6 BY 36 FOOT Y METAL AIRFOIL

By Clinton H. Dearborn

SUMMARY

An investigation was conducted in the N.A.C.A. full-scale wind tunnel to determine the effects of exposed rivet heads on the aerodynamic characteristics of a metal-covered 6 by 36 foot Clark Y airfoil. Lead punching simulating 1/8 inch rivet heads were attached in full-span rows at a pitch of 1 inch at various chord positions. Tests were made at velocities varying from 40 to 120 miles per hour to investigate the scale effect.

Rivets at the 5 percent chord position on the upper surface of the airfoil produced the greatest increase in drag for a single row. Nine rows of rivets on both surfaces, simulating rivet spacing of multispar construction, increased the drag coefficient by a constant amount at velocities between 100 and 120 miles per hour. Extrapolation of the curves indicates that the same increase would be obtained at speeds over 120 miles per hour. Accordingly, if rivets spaced the same as those on

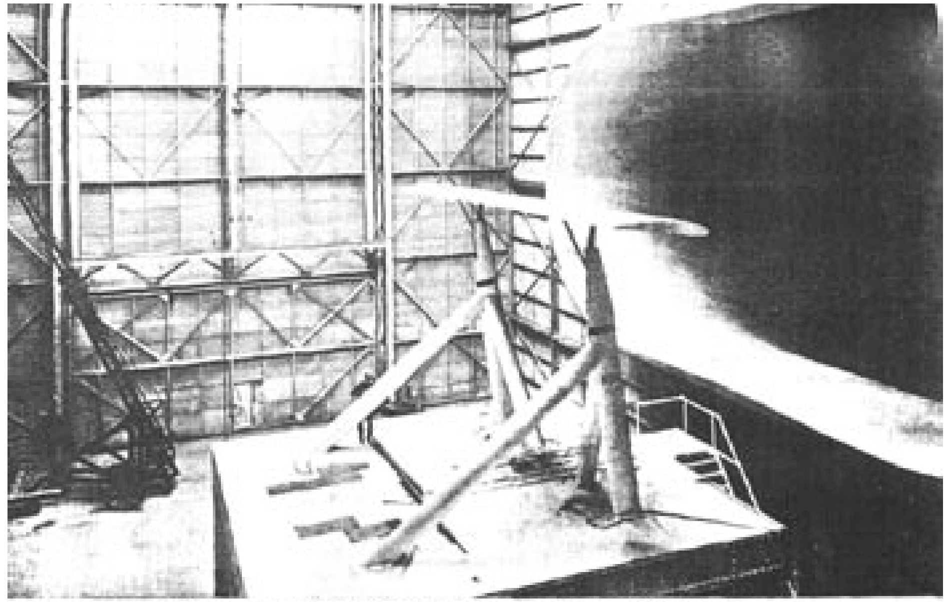

Figure 1.-The 6 by 36 foot Clark Y airfoil mounted on balance.

the test airfoil were used on a Clark Y wing of 300 square feet area and operated at 200 miles per hour the drag would be increased over that for the smooth wing by 55 pounds and the power required would be increased by 29 horsepower. The effect on the lift characteristics due to the rivets was found to be negligible.

INTRODUCTION

One of the most promising possibilities of improving the performance of airplanes lies in the reduction of drag. A recent airfoil investigation conducted in the N.A.C.A. variable-density wind tunnel on full-span protuberances (reference 1) and on short-span protuberances, including wing fittings (reference 2), showed that small protuberances have an important effect on the aerodynamic characteristics of an airfoil. This investigation was extended to include the determination of the effects caused by exposed rivet heads of a type common to metal airplane wing construction. The latter tests were conducted in the full-scale wind tunnel on a 6 by 36 foot airfoil.

Lead punchings formed to simulate rivet heads were attached to the airfoil first in single rows at various chord positions on the upper surface, then in nine rows on the upper surface, and finally in nine rows on both surfaces.

APPARATUS AND METHODS

The 6 by 36 foot Clark Y airfoil used in this investigation is shown mounted in the tunnel in Figure 1. Two structural steel H beams with steel-angle connecting members form the primary structure of the airfoil; the ribs and skin are of 1/16 inch

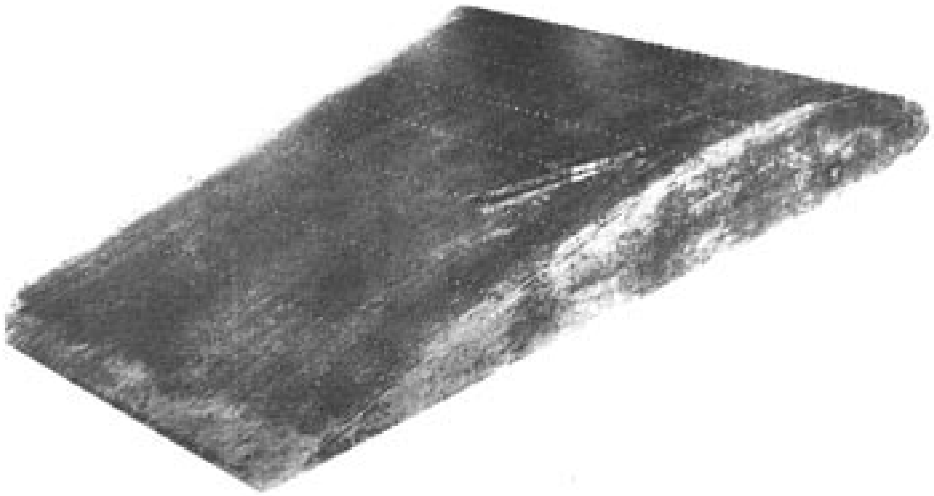

Figure 2.-Nine rows of rivet heads on upper surface of airfoil.

sheet aluminum. The outer surface of the skin was made as smooth as practicable by the use of butt joints and countersunk attaching screws. Rivet heads were simulated by gluing lead punchings to the surface of the airfoil as shown in Figure 2. These punchings were made from sheet lead with a die conforming in dimensions to the head of a 1/8 inch brazier head rivet. (Fig. 3.)

The airfoil was supported on the balance by two braced struts shown in Figure 1. All members were encased in fairings except the tops of the supports and the short struts for changing the angle of attack. The exposed members were made as small as practicable so that the tare drag would be a small percentage of the minimum drag of the airfoil. Tare-drag tests in which the airfoil was independently supported showed that the drag of the supports was only 4 percent of the minimum drag of the plain airfoils at 100 miles per hour. A description of the balance will be given with the description of the tunnel now being prepared as a Technical Report.

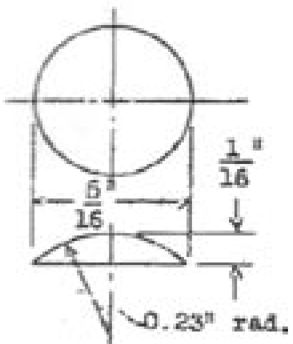

Figure 3.

TESTS

The effect on the drag of the airfoil of a single row of rivet heads at the leading edge, and at 5, 15, and 30 percent of the chord back of the leading edge on the upper surface was first investigated. The single rows, as well as the combination of rows at 10 percent chord intervals tested later, extended over the full span of the airfoil with the rivets spaced 1 inch apart.

Starting with the 5 percent chord position, nine rows were attached to the upper surface at increments of 10 percent of the chord and the drag measured. Nine

additional rows of rivet heads were later attached to the lower surface at the same chord positions as those on the upper surface and the drag again measured. The last condition of test is representative of the spacing of rivets on metal-covered wings of multispar construction. These tests were made at a dynamic pressure of 7.8 pounds per square foot, which corresponds to an indicated velocity of 55 miles per hour.

The plain airfoil and the airfoil with the nine rows of rivets on both the upper and lower surfaces were next tested at angles of attack in the region of minimum drag over a speed range from 40 to 120 miles per hour to investigate the magnitude of the scale effect. The effect of the rivets on lift was investigated by testing the airfoils from -8 degrees to 21 degrees angle of attack at a dynamic pressure of 16 pounds per square foot (79.2 miles per hour indicated velocity).

RESULTS AND DISCUSSION

Tunnel jet-boundary corrections have not been applied to the results presented in this report because the differences in lift were negligible and the differences in drag therefore would not be affected.

A comparison of the results obtained from the plain airfoil with these obtained with a single row of rivets at the various chord positions on the upper surface showed that the single row at the 5 percent chord position produced the greatest increase in minimum drag. This increase in drag amounted to 19 percent of the minimum drag of the plain airfoil. (Fig. 4.)

The nine rows of rivets on the upper surface of the airfoil at 10 percent chord intervals extending from the 5 to 85 percent chord positions caused a 21 percent

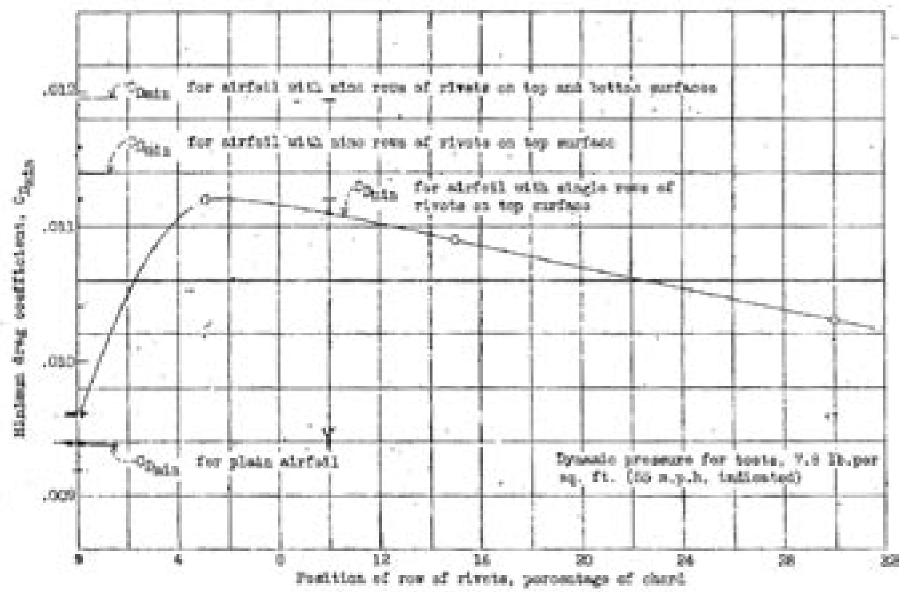

Figure 4.-Minimum drag coefficient for airfoil with rivet arrangements tested.

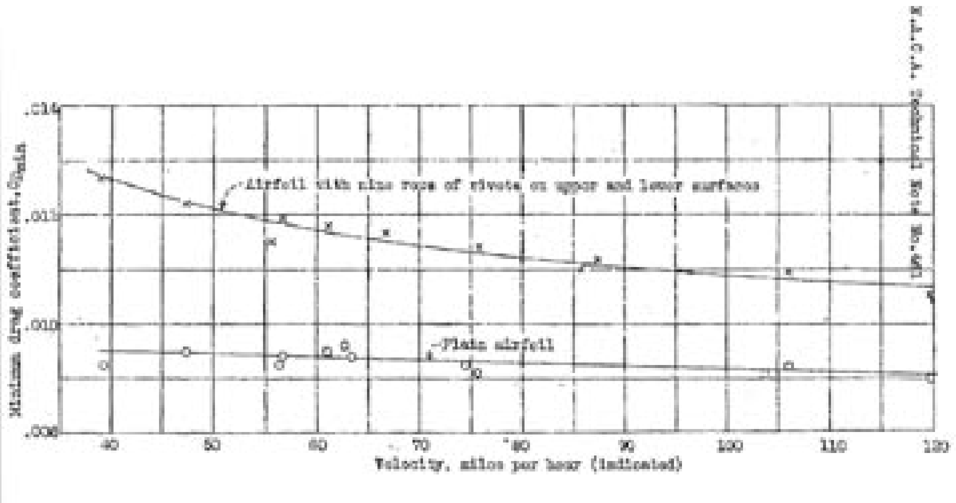

Figure 5.-Scale effect on $C_{D_{min}}$ for the plain airfoil and airfoil with 18 rows of rivets.

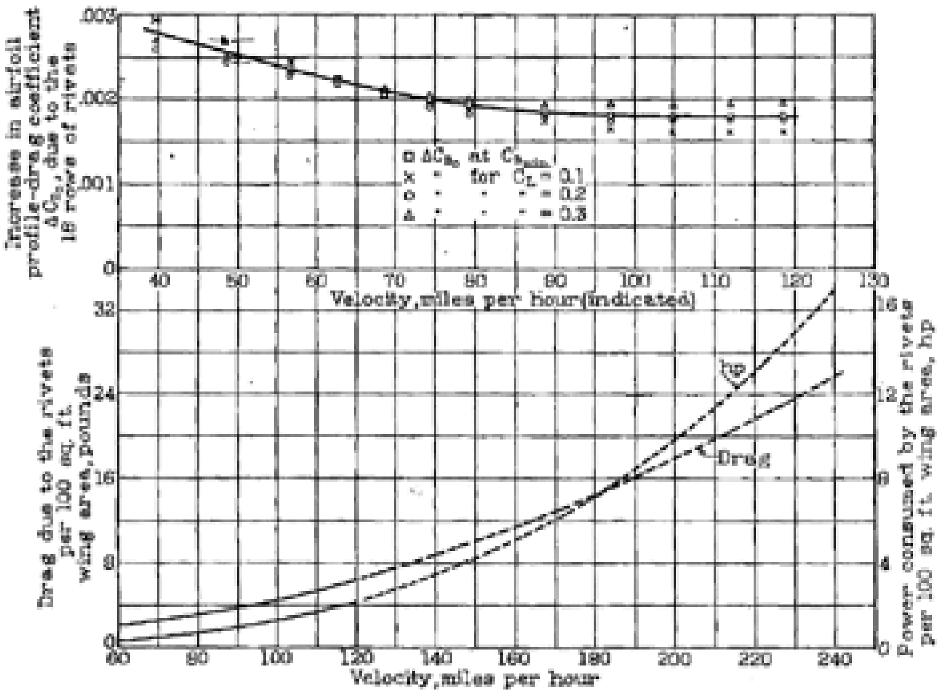

Figure 6.-Increase in profile-drag coefficient, drag, and power required due to the rivets.

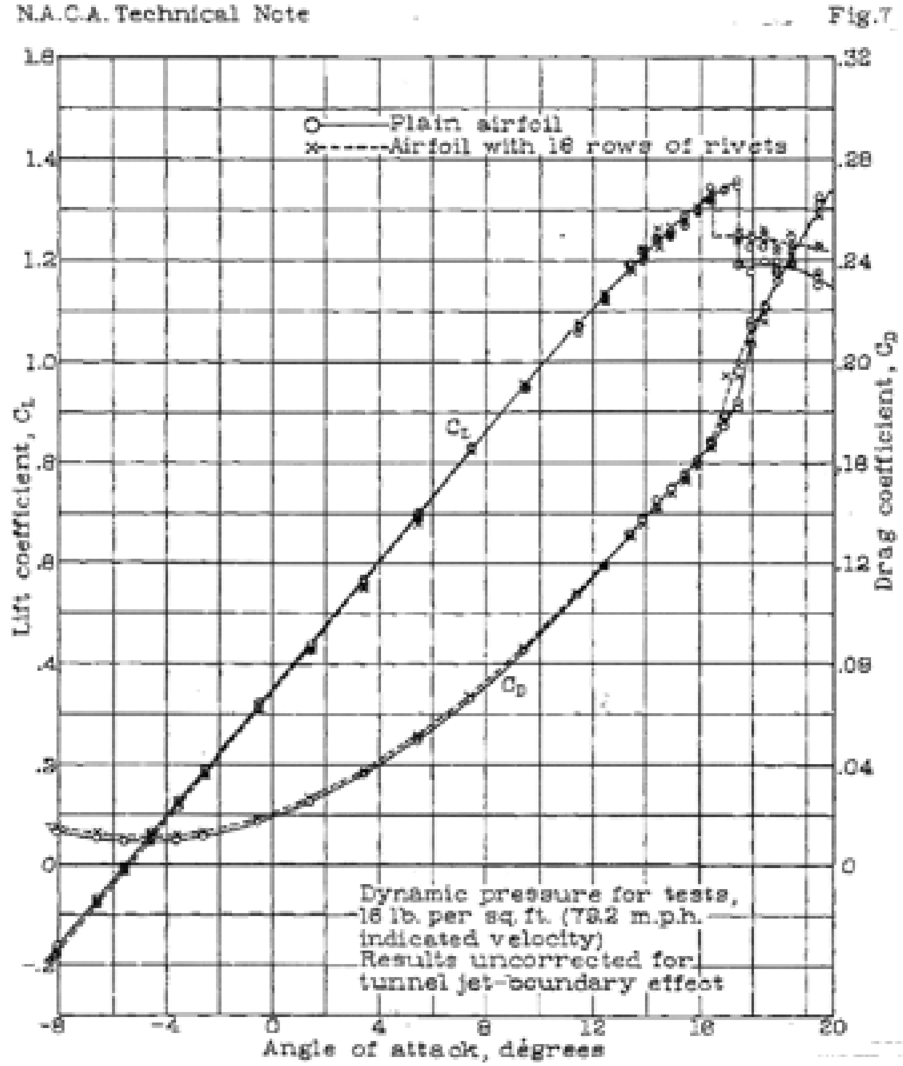

Figure 7.-Variation of lift and drag with angle of attack for plain airfoil and airfoil with rivets.

Fig.8

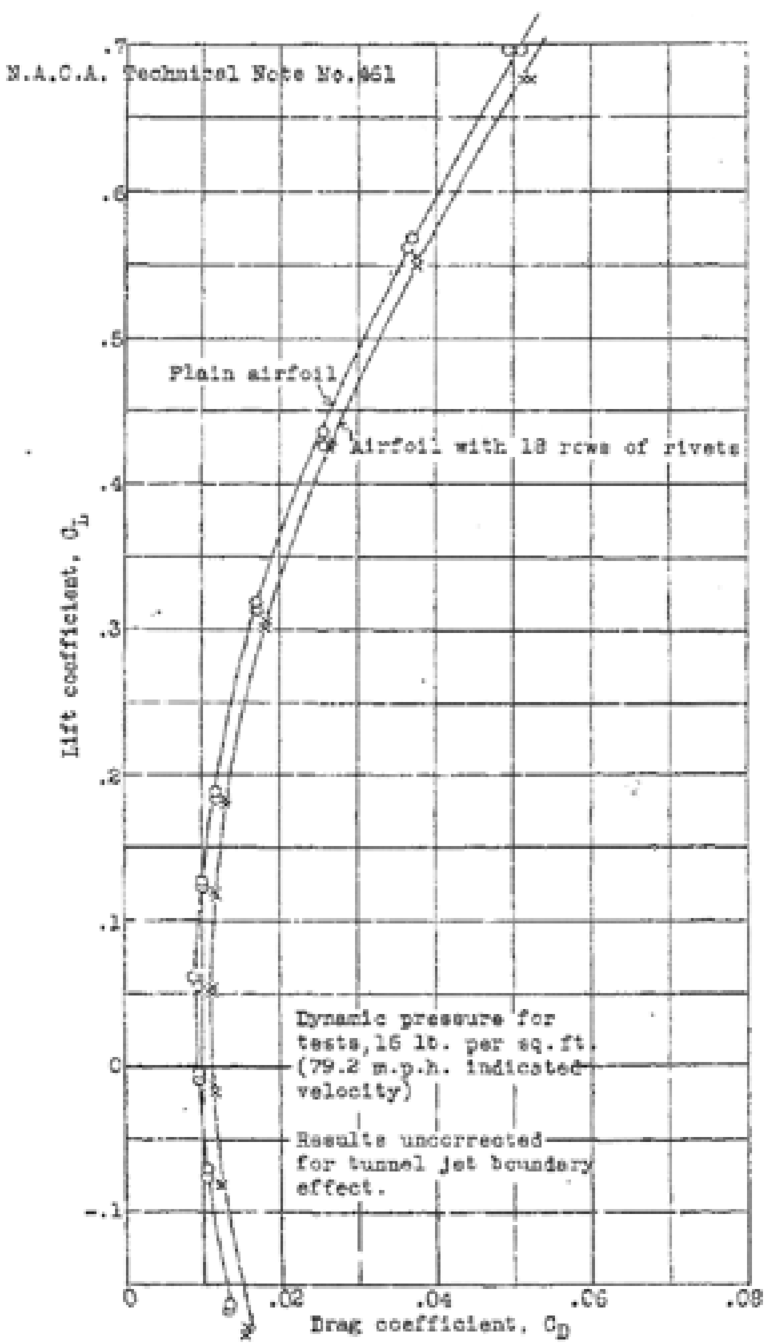

Figure 8.- Polars for plain airfoil and airfoil with rivets.

increase in minimum drag. This increase in drag is small compared with the increase of about 60 percent that would be obtained from the summation of increases in minimum drag for single rows shown in Figure 4. The fact that the increases in drag due to the single rows failed to become additive for a combination of the same rows was probably due to a serious disturbing effect in the boundary layer caused by the first row of rivets.

The nine rows of rivets on both surfaces produced an increase of 27 percent in drag. This is less than one-third more than the amount obtained with the rivets on the upper surface alone.

The preceding results were obtained from tests at 55 miles per hour. It will be noted in Figure 5 that the increase in minimum drag at 120 miles per hour for the airfoil with rivets on both surfaces is only 18 percent of the minimum drag of the plain airfoil. This difference in increase of minimum drag may be attributed to scale effect; it may be assumed that the same scale effect would be present with the single row of rivets at the 5 percent chord position and with the nine rows on the upper surface alone and that the percentage increase in minimum drag for these conditions would be proportionally reduced at the higher speed.

Figure 5 shows a greater scale effect for the riveted airfoil than for the plain airfoil at the lower test velocities. However, at the higher velocities this difference in the scale effect disappears, resulting in a constant difference in minimum drag. Differences of the minimum drag coefficients and drag coefficients corresponding to the lift coefficients of 0.1, 0.2, and 0.3 for the two airfoils throughout the speed range are plotted in Figure 6. The increase in the drag coefficient due to the rivets is, for practical purposes, due solely to an increase in the profile drag, as indicated by the parallelism of the polars in Figure 8. The difference in drag coefficients at velocities between 100 and 120 miles per hour is 0.0018. It appears reasonable to assume that this difference in drag coefficients would remain the same at velocities even higher than those employed for this investigation.

The effect of the rivets on lift is practically negligible, as shown in Figure 7. The burble angle occurs 1 degree earlier with a decrease of about 1 percent in the maximum lift coefficient.

The significance of the increase in profile drag may well be illustrated by estimating what effect it would have on the performance of an airplane. For this purpose an airplane with the following specifications was chosen and the assumption made that the wings were metal covered with exposed rivet heads on both surfaces in the same locations as those covered by the tests.

Wing area	300 sq. ft.
Wing section	Clark Y
Engine	500 b.hp
Fuel consumption	0.5 lb./b.hp-hr.
Propulsive efficiency	80 percent

High speed	200 m.p.h.
Cruising speed	170 m.p.h.

These specifications are representative of a modern high-speed transport or a military observation airplane.

The extrapolated drag curve in Figure 6 shows that the increase in drag caused by the rivets would be 40 pounds at the cruising speed of 170 miles per hour and 55 pounds at the high speed of 200 miles per hour. These drag forces, taking the propulsive efficiency into account, would consume 23 and 37 brake horsepower, respectively, at the cruising and high speeds. The increase in fuel consumption due to the rivets at the cruising speed, based on a weight of 6 pounds per gallon, would be 1.9 gallons per hour. This amount represents about 7 percent of the fuel consumption at the cruising speed. The high speed would be increased from 200 to 205 miles per hour by the elimination of the exposed rivet heads.

CONCLUSIONS

1. A single row of rivets located at the 5 percent chord position on the upper surface of the airfoil produced a greater increase in the minimum drag than any other position investigated.
2. Rivets added on the upper surface of the airfoil back of a single row at the 5 percent chord position had little effect on drag.
3. Nine rows of rivets on the lower surface increased the drag less than one-third of the amount that the same number of rows did on the upper surface.
4. The effect of rivets on maximum lift was negligible.
5. Exposed rivet heads of the type and spacing investigated would have an appreciable detrimental effect on the fuel consumption and high speed of an airplane.

Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
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THE EFFECTS OF SOME COMMON SURFACE IRREGULARITIES ON WING DRAG

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SUMMARY

The N.A.C.A. has conducted tests to provide more complete data than were previously available for estimating the effects of common surface irregularities on wing drag. The irregularities investigated included: brazier-head and countersunk rivets, spot welds, several types of sheet-metal joints, and surface roughness. Tests were also conducted to determine the overall effect of manufacturing irregularities incidental to riveted aluminum alloy and to spot-welded stainless-steel construction. The tests were made in the 8-foot high-speed wind tunnel at Reynolds numbers up to 18,000,000.

The results show that any of the surface irregularities investigated may increase wing drag enough to have important adverse effects on high-speed performance and economy.

A method of estimating increases in wing drag caused by brazier-head rivets and lapped joints under conditions outside the range of the tests is suggested. Estimated drag increases due to rivets and lapped joints on a wing of 20-foot chord flying at 250 miles per hour are shown.

INTRODUCTION

Improved streamlining has reduced form drag so much that on modern airplanes skin friction often constitutes more than half of the total drag. It is therefore important that skin friction be made as small as possible. It is obvious that protruding rivet heads, roughness, and other surface irregularities will increase skin friction. A knowledge of the magnitude of the drag increases is necessary before the designer can decide to what extent it is economical to eliminate these irregularities from the surfaces exposed to air flow.

Previous tests have shown that rivet heads (reference 1), certain arbitrary protuberances (references 2 and 3), and roughness (references 4 and 5) increase the drag of wings by important amounts. The N.A.C.A. has conducted additional tests to provide more complete data on the drag caused by irregularities of types commonly occurring on airplane wings so that their effects may be more accurately estimated. Some of the results of these tests are presented in this report. The irregularities for which data are presented include protruding rivet heads, spot welds, several types of lapped joints, imperfections in butted joints, surface roughness, and manufacturing irregularities. Most of the irregularities were tested in various systematic arrangements over different portions of the surface of an airfoil of N.A.C.A. 23012 section and 5-foot chord. The tests were made at lift coefficients of 0 to 0.30 and at Reynolds numbers up to 18,000,000.

A method of applying the results to predict the drag increases due to rivet heads and lapped joints on airplane wings under conditions outside of the range of these tests is presented.

APPARATUS

The tests were conducted in the N.A.C.A. 8-foot high-speed wind tunnel. The air flow in the closed circular test section of this wind tunnel is quite uniform and the turbulence of the air flow is so small that sphere tests have shown virtually the same critical Reynolds number as in free air (reference 6).

The airfoils used for the tests were of N.A.C.A. 23012 section and, except for a few supplementary tests of a 2-foot airfoil, they were of 5-foot chord. The noses of the airfoils were bare steel to reduce erosion; the rest of the surface was bare steel in some instances and painted in others. Except for the irregularities being tested, the surfaces were aerodynamically smooth; that is, further polishing would not reduce the drag.

The airfoils were mounted horizontally across the center of the test section as shown in figure 1. The tunnel-wall interference was reduced by enclosing the ends of the airfoils in shields that did not touch the airfoils or their supports but were supported independently of the balance. The span of each shield was 10 inches and the active span of the airfoils between the shields was 6 feet.

Actual rivets with their shanks pressed into holes drilled in the airfoil were used for most of the tests of protruding rivet heads but for some of the tests it was found more convenient to simulate the rivet heads with properly shaped lead punchings cemented to the surface with airplane dope. Countersunk rivets were simulated by annular cuts in the surface of the airfoil to represent the indentations around rivet heads that result from making countersinks by forming the sheet metal with punches instead of by cutting. Details of the rivets and the simulations are shown in figure 2. The rivet-head simulations tested on the 2-foot airfoil were two-fifths as large as the heads of the 3/32-inch brazier-head rivets.

Spot welds were simulated on the otherwise smooth model by depressions of the dimensions shown on figure 3. The cylindrical simulations were cut in the surface and the spherical simulations were formed by filling the cylindrical simulations with plasticine and forming the depressions with tools having spherical ends.

Most of the lapped joints were represented by cuts made in the surface of the airfoil but, in addition to this method, the plain lap facing aft was also simulated by a ridge built up on top of the normal airfoil surface with paper and lacquer-base glazing putty. Gaps such as occur between the edges of sheets in butt-jointed construction were simulated by square-edge grooves cut spanwise in the surface of the airfoil. Adjoining edges of the gaps were of equal height, representing construction in which the butted sheets are of exactly equal thickness. The dimensions of the simulations of lapped and butted joints are shown in figure 4. All the simulations represented sheets 0.018 inch (0.0003 chord) thick.

The chord positions at which sheet-metal joints and spanwise rows of rivets and spot welds were tested are shown in figure 5. The spanwise pitch of the rivets and spot welds was ¾ inch (0.0125 chord) except where otherwise noted.

Photomicrographs of samples of the different surface roughnesses tested, all to the same magnification, are shown in figure 6. The carborundum-covered surfaces were produced by spraying carborundum grains mixed with thin shellac onto the smooth airfoils. The common designations of the grain sizes are:

Average grain size k(in.)	Carborundum Company's designation
0.0037	180
.0013	FF
.0005	800-RA

From figure 6 it is apparent that the 0.0005-inch grains were piled on top of each other in such a manner that the degree of roughness was not equivalent to the grain size, as was the case with the larger grains. The photomicrographs also indicate that the shellac used to hold the grains was sufficiently thin that the effective size and shape of the grains were not appreciably changed. The sizes of the grains were determined from measurements made with a microscope and from measurements of the photomicrographs. The density (spacing) of the grains varied somewhat over the airfoils but the photomicrographs represent average conditions. The spray-painted surface was produced by spraying a lacquer-base primer surface onto the airfoil, probably a little rougher than is common practice. The sandpapering was done with No. 400 sandpaper lubricated with water. No attempt was made to limit the sandpapering strokes to any one direction but chordwise strokes predominated. The surface was polished by rubbing with a polish of the type used in polishing automobiles, waxing, and rubbing with a soft cloth.

Two airfoils, one of riveted aluminum-alloy construction and the other of spot-welded stainless steel, were tested to obtain a measure of the over-all effect of manufacturing irregularities incidental to conventional metal-wing construction. These "service wings" were both made by manufacturers accustomed to the respective types of construction involved. The manufacturers were instructed to employ conventional design, tolerances, and workmanship in order to make the models as nearly as possible representative of actual wings being produced at that time (1936). The riveted model employed the same rivet size and arrangement and the same lapped-joint positions as were tested in one instance on the more accurate wind-tunnel model but the thickness of the lapped sheets was 0.032 inch. The skin of the stainless steel model was 0.015 inch thick on the forward 45 percent and 0.0008 inch thick on the rear 55 percent. The average dimensions of the spot welds are shown in figure 3. The arrangement of spot welds and lapped joints on the stainless-steel model is shown in figure 7. Except for the discrepancy in the profile of the riveted service wing shown in figure 8, there were no departures from true profile large enough to have important effects. Figures 9 and 10 are photographs of the service wings arranged to reflect the image of a lattice so as to show the irregularities of the surfaces. The riveted service wing was furnished by the Bureau of Aeronautics, Navy Department, and the stainless-steel wing was furnished by the U. S. Army Air Corps.

METHOD

For each arrangement of surface irregularities tested, the lift, the drag, and the pitching moment were determined at left coefficients of approximately 0, 0.15, and 0.30, respectively. The tests at lift coefficients of 0.15 and 0.30 were made at speeds varying from 80 to 370 and 80 to 270 miles per hour, respectively, the upper limit in each case producing a wing loading of approximately 50 pounds per square foot. For the tests at zero lift, the speed was varied from 80 to about 430, and in some instances 500, was varied from 80 to about 430, and in some instances 500, miles per hour. At the highest speed compressibility effects were so large that the drag coefficient increased very rapidly as the speed was increased. The drag of the smooth airfoil was checked frequently during the tests.

Because of the high test speeds employed, the method used for determining dynamic pressure, air speed, and Reynolds number in the N.A.C.A. 8-foot high-speed wind tunnel must allow for compressibility effects. Bernoulli's equation for a compressible fluid, in a form given in reference 7, is

$$P_a = P_s + \frac{1}{2} \rho_s V_s^2 \left(1 + \frac{1}{4} M^2 + 1/40\, M^4 + 1/1600\, M^6 \ldots\right)$$

where P_a is the atmospheric pressure which, in the case discussed in reference 7, was virtually equal to the total pressure in the test section.

P_s, static pressure in the test section.

ρ_s, density of air in test section.

V_s, air speed in test section.

M, Mach number (the ratio of the air speed to the speed of sound in the air).

The quantity within the parentheses is the factor by which the impact pressure (q_c) shown by a pitot-static tube can be divided to give true dynamic pressure ($q = \frac{1}{2} \rho V^2$). This factor is called the "compressibility factor" and is often designated by $(1 + \eta)$. Accordingly,

$$(1 + \eta) = 1 + \frac{1}{4} M^2 + 1/40\, M^4 + 1/1600\, M^6 \ldots$$

Substituting first $\sqrt{1.4\, P_s/\rho_s}$ for the speed of sound, then 2q for $\rho_s V_s^2$, and finally, $q_c/(1 + \eta)$ for q gives,

$$(1 + \eta) = 1 + 0.357\, (q_c/P_s(1 + \eta) + 0.051\, [q_c/P_s(1 + \eta)]^2 + 0.0018\, [q_c/P_s(1 + \eta)]^2$$

From this relation, the compressibility factor $(1 + \eta)$ is computed in terms of q_c/P_s and plotted for use in computing results.

During tests, measurements are made of the pressure difference $P_1 - P_s$, where P_1 is the static pressure in the low-speed part of the return passage. From this pressure difference, qc for the model position is computed in accordance with a relation previously determined by pitot-static surveys of the air flow in the test section. The absolute value of P_s is computed from the barometric pressure, a previously determined value of $P_a - P_1$, and the measured pressure difference $P_1 - P_s$. The ratio q_c/P_s is then computed and, from this ratio and the curve described in the proceeding paragraph, $(1 + \eta)$ is determined. The true dynamic pressure on which force and moment coefficients are based is then computed from the relation

$$q = q_c/(1 + \eta)$$

The air temperature in the slow-speed part of the return passage, T_1, is measured with remote indicating thermometers. From T_1, P_1, and P_s, the temperature and the density of the air in the test section, T_s and ρ_s, are computed on the assumption that the flow is adiabatic. The air speed in the test section is then easily computed from ρ_s and q. The speed of sound in the air in the test section in miles per hour is $33.5\sqrt{T_s}$ where T_s is the absolute temperature in Fahrenheit degrees. The viscosity of the air follows from T_s and, since ρ_s has already been determined, the Reynolds number can be computed.

The assumption that the flow is adiabatic between the slow-speed part of the return passage and the test section is supported by tests which have shown that, except in the boundary layer near the tunnel walls, there is no appreciable difference between the total pressures at these two sections.

When the air in the wind tunnel is cool and its relative humidity is moderately high, fog condenses in the test section when the tunnel is operated at high speeds. Such condensation has appreciable effects on the thermodynamics of the air flow. When this condition is encountered, the tunnel is operated until the air becomes warm enough to dissipate the fog before test data are taken. Aside from this precaution, no allowance is made for the effects of humidity.

Air-flow measurements ahead of the model have indicated that blocking effects are unimportant under the conditions of these tests.

PRECISION

The only known systematic errors affecting the results herein presented are due to errors in the dynamic pressure resulting from constriction by the model of the flow through the test section. No correction for constriction has been applied because its magnitude is not yet accurately known. Preliminary tests have indicated, however, that it is not more than 6 percent at speeds up to 270 miles per hour or 9 percent up to 500 miles per hour. The drag increases herein presented may, therefore, be too high by 6 percent at the lower speeds and 9 percent at the higher speeds. Since most of the increases are small relative to the smooth-wing drag, these errors are generally unimportant.

The scatter of experimental points for separate determinations of the smooth-wing drag indicates that random errors in drag coefficients generally do not exceed ±0.0001, corresponding to ±1.4 percent of the smooth-wing drag; however, at speeds below 100 and above 400 miles per hour and at all speeds at a lift coefficient of 0.30, the variations are about twice this large.

RESULTS AND DISCUSSION

Method of Presentation

All results are presented in terms of increases in drag coefficient C_D, the amount by which the drag coefficient for any condition exceeded the drag coefficient of the smooth airfoil at the same speed and angle of attack. No corrections for tunnel-wall effects have been applied because this method of presentation makes corrections unnecessary except those due to constriction effects, which have been discussed under Precision.

Because of the rapid variation of drag coefficient with Mach number at the high test speeds, it was necessary to compute the drag differences at equal values of the Mach number. Equal Mach numbers correspond approximately, but not exactly, to equal Reynolds numbers for a given size model. This variation from test to test of the Reynolds number corresponding to a given Mach number is, however, small enough to be of little consequence and the results are therefore presented in terms of Reynolds numbers representing the averages for the various tests. For the 5-foot airfoils, an average Reynolds number of 10,300,000 corresponded to a Mach number of 0.3 and an average Reynolds number of 17,600,000 to a Mach number of 0.6.

That the effect of compressibility on the drag increments herein presented may be neglected is indicated by figure 11, which shows that rivets of geometrically similar size and arrangement increased the drag coefficients of the 2-foot and the 5-foot airfoils by substantially equal amounts at equal Reynolds numbers even though the Mach numbers differed widely. The results may therefore be applied solely on the basis of Reynolds number. In accordance with Reynolds' law, rivet size and arrangement must be considered in terms of wing chord when the results are used directly to predict the drag of rivets on wings of different chord.

RIVETS

Figure 12 shows the drag increments due to the various types and sizes of rivets in 13 spanwise rows on each surface. The spanwise pitch in each row was ¾ inch (0.0125 chord) and the most forward row on each surface was 4 percent of the chord behind the leading edge. The approximate percentage increments are spotted on the curves for a few representative points to aid in visualizing the magnitude of these drag increments. It is obvious that the drag increments are large enough to have important adverse effects on performance, being as much as 27 percent of the smooth-wing drag for the 3/32-inch brazier-head rivets. Even countersunk rivets may increase wing drag by amounts too large to be neglected.

Rivet heads increase the drag of a wing in two ways: first, each rivet head, being exposed to the air flow, has some drag in itself; and, second, rivets on the forward part of a wing cause the transition point to move forward and thereby cause an increase in skin friction. That this second factor may be responsible for a large part of the drag increase is indicated by figure 13. This figure shows that, when the front row of rivets was 4 percent of the chord from the leading edge, the rivet drag was reduced only slightly by increasing the spanwise pitch from 3/4 inch to 1½ inches. When the forward rows of rivets were 28 percent of the chord from the leading edge, they were behind the transition point and, as the pitch was varied, the rivet drag varied in proportion to the number of rivets on the wing.

Figures 14 and 15 show the variation of rivet drag with position of the forward rows. The position of the forward rows was varied by adding or removing rows at the front, rows behind the most forward ones always remaining in place at the chord position shown in figure 5. These figures illustrate again the importance of the shift of the transition point caused by rivets. As rows of rivets are added, starting at the back and progressing forward, the drag increases slowly until the region where transition occurs on the smooth wing (fig. 15) is reached, after which the drag increases much more rapidly. The shaded area in figure 15 indicates the excess of the rivet drag over what it would be if the original rate of increase were maintained forward of the transition point. This excess drag is plotted in figure 16 along with the computed difference between turbulent and laminar flat-plate skin friction for the Reynolds numbers involved. The agreement of the computed and the experimental curves indicated that the rapid rise of the curves of figures 14 and 15 forward of the 25-percent-chord position is largely due to forward movement of the transition point.

From the test results shown in figures 12 to 15, it can be concluded that the following measures are most effective in keeping rivet drag small:

(a) Rivets should be as far back on the wing surface as possible. It is especially important that there be no rivets forward of the point at which transition occurs on the smooth wing. With the rivet arrangement shown in figure 5, for example, 70 percent of the rivet drag was caused by the rivets on the forward 30 percent of the airfoil.

(b) Rivet heads should be as small as possible.

(c) There should be as few exposed rivet heads as possible; reducing the number by increasing the pitch in spanwise rows forward of the transition point, however, has little effect unless the pitch is larger than 0.025 chord.

SPOT WELDS

The drag increases caused by the three sizes of spot welds are shown in figure 17. These increases are smaller than those caused by any of the protruding rivet heads but are large enough so that an effort should be made to keep the depressions at spot welds as shallow as possible. The two points shown for the 0.012-inch-deep spot welds in figure 14 indicate that spot welds, as well as rivets, should be avoided especially forward of the smooth-wing transition point.

SHEET-METAL JOINTS

Increases in drag due to the presence of sheet-metal joints are shown in figure 18 for lapped joints and in figure 19 for butted joints with gaps between the edges of the sheets. The conventional plain laps facing aft are only slightly superior to the same type facing forward. Joggled laps cause only about one-half as much drag as either type of plain lap. It is important that the laps be accurately formed to lie inside the true airfoil profile because, if they rise outside the true profile (see fig. 4), appreciably more drag may be created.

Rounding the exposed edges of the sheets to a radius equal to the sheet thickness reduced the drag of the forward-facing laps to about the same magnitude as that of the square-edge laps facing aft. Rounding the edges of the plain laps facing aft to this radius reduced the drag very slightly but fairing the edges back to a width equal to three times the sheet thickness reduced the drag caused by this type of lap to about two-thirds of its previous magnitude.

The variation of lapped-joint drag with chord position of the forward laps was similar to that shown for rivets in figure 14.

From figure 18, it is seen that the drag increments caused by rivets and laps are not additive. Presumably, this condition is due to the fact that the rivets caused transition to take place at the most forward row. Adding laps back of this point therefore had no further effect on the transition point, and the additional drag due to the presence of the laps was only the direct drag of these laps in the turbulent boundary layer.

The 0.024-inch square-edge grooves caused only small increases in drag (fig. 19), indicating that only reasonably small tolerances need be maintained on the permissible gap between the edges of butted sheets.

ROUGHNESS

From figure 20, it is evident that even a small degree of surface roughness increases wing drag sufficiently to have serious adverse effects on high-speed performance and economy. Even the roughness due to spray painting may increase wing drag 10 to 14 percent in the high-speed and cruising range. Except at the lowest speeds, the 0.0013-inch roughness increased the drag considerably more than 3/32-inch brazier-head rivets. As in the case of other surface irregularities, it is especially important to keep the forward portions of wings smooth, but roughness may cause important increases in wing drag even when entirely behind the smooth-wing transition point. (See fig. 21.)

In the range of these tests, the drag increases caused by surface roughness vary considerably with scale (fig. 20). At the lower speeds, the drag due to roughness decreases rapidly as speed is reduced and the curves indicate that, for each degree of roughness, there is a speed or Reynolds number below which that roughness has no effect on drag. Conversely, it is indicated that for every speed or Reynolds number there is a limiting "permissible roughness," which will cause no increase in drag. The existence of such a permissible roughness has been shown by other tests and by the theory that roughness wholly submerged in the laminar sublayer will not increase the drag (reference 8 and 9). Estimating permissible roughness from the results herein reported involves questionable extrapolations, but nevertheless the results do indicate the same order of magnitude of permissible roughness as is tabulated for a flat plate with a wholly turbulent boundary layer in reference 8; even though, in the case of the airfoil, part of the boundary layer is laminar. This agreement suggests the conclusion that neither the 0.0005-inch grains nor the roughness due to spray painting had any great effect on the transition point. This conclusion is supported by figure 21 because the curves for the two smaller degrees of roughness do not rise so rapidly forward of the 25-percent-chord position as they would if the transition point moved forward with the roughness. There is need for further investigation of the degree of roughness a wing surface may have before premature transition is induced. The permissible roughness in the laminar boundary layer probably varies widely with dynamic scale, airfoil pressure gradient, and initial air-stream turbulence, so the indicated conclusion should not be applied where conditions differ from those of the present tests.

It is of practical interest to note that, within the limits of accuracy of the tests, the drag of the sandpapered airfoil was the same as that of the highly polished airfoil. For airplanes to have the smallest possible skin friction the surfaces must be smooth but need not be highly polished.

Because of the large scale effect on the drag of rough surface (fig. 20), it is essential that experimental investigations of the effects of surface roughness be made at large scale. Degrees of roughness large enough to have serious adverse effects under flight conditions may have no effect whatsoever under conditions of small-scale tests.

The effects of roughness on airplane wings can be estimated only approximately from the results of these tests because the effects depend on grain shape and grain spacing as well as grain size. The variation of drag with grain spacing is especially large (reference 10).

MANUFACTURING IRREGULARITIES

Figure 22, showing the results of tests of the two service wings (figs. 9 and 10), indicates that manufacturing irregularities may cause important drag increases over and above those due to the rivets and the lapped joints. For example, the drag of the riveted service wing was 42 percent greater than that of a smooth accurate wing (at a lift coefficient of 0.15 and a Reynolds number of 10,300,000), whereas the same arrangement of rivets and lapped joints caused only 29 percent excess drag on the more accurate wind-tunnel model. The rivets on the two models were identical but the lapped sheets were 0.032 inch thick on the service wing compared with 0.018 inch on the wind-tunnel model. It is estimated, by a method hereinafter explained, that the extra sheet thickness on the service wing was responsible for a difference

in drag coefficient of about 0.0004, or 5 percent of the smooth-wing drag. There remains a difference of 8 percent of the smooth-wing drag to be attributed to manufacturing irregularities such as sheet waviness, departures from true profile, and imperfections in the lapped joints. In the absence of protruding rivet heads on the forward part of the wing, equivalent manufacturing irregularities would probably have a much larger effect.

The drag of the riveted service wing at zero lift increased rapidly at Reynolds numbers above 14,000,000, corresponding to a Mach number of 0.43 or a speed of 330 miles per hour at 60° F. It is believed that this rapid rise in drag was due to a shock wave prematurely induced by a bulge in the lower portion of the nose of this model (fig. 8). The importance of making wings for high-speed airplanes accurately to true profile is evident.

The drag of the spot-welded stainless-steel service wing averaged about 20 percent greater than the drag of a smooth accurate wing. It is difficult to say how much of the excess drag of this model was due to the lapped joints because of uncertainty as to whether the spot welds and the sheet waviness would induce premature transition in the absence of the lapped joints.

APPLICATION OF RESULTS

It has been shown that the increase in wing drag caused by rivet heads can be divided into two parts: the increase in skin friction due to the fact that the rivets cause transition from laminar to turbulent boundary layer to occur abnormally far forward on the wing, and the direct drag of the rivet heads themselves. The separation of the drag increments into these two parts affords a basis for estimating the drag due to rivet heads and lapped joints under conditions of scale and of rivet and lap size and arrangement outside the range of these tests. The increase due to a forward shift of the transition point can be approximated by estimating the distance through which the transition point is moved and then applying the known difference between laminar and turbulent skin friction for the Reynolds numbers involved. The direct drag of the rivet heads can then be estimated from computed local velocities at the rivet heads and suitable rivet-drag coefficients.

In order to estimate the drag increase resulting from a forward shift of the transition point, it is necessary first to know where the transition point would be if there were no rivets on the wing. The results of recent tests by Becker, as yet unpublished, have indicated that, on smooth conventional airfoils, the transition point approaches the point of peak minimum pressure as the Reynolds number increases. When there is more than one peak of minimum pressure on a surface of the airfoil, transition will generally occur at the farthest forward peak even though it may be smaller than later peaks. Until a more complete understanding is available, transition at large Reynolds numbers may be assumed to occur at the farthest forward peak of minimum pressure on each side of the wing. It appears safe to assume that any protruding rivet heads large enough to be practicable will, if forward of the smooth-wing transition point, cause transition to occur at the rivets. Dust patterns on models in the 8-foot high-speed wind tunnel have indicated that the wake of turbulent boundary layer behind individual rivet heads in an otherwise laminar boundary layer spreads laterally with a total included angle of about 15°. This angle may be used in estimating the area over which premature transition is created by individual rivet heads.

Calculated skin friction based on free-stream dynamic pressure and flat-plate coefficients agreed with experimentally determined values on an N.A.C.A. 0012 airfoil. In the absence of further evidence, the increase in skin friction resulting from the estimated shift of the transition point can be estimated as the product of the area affected, the free-stream dynamic pressure, and the difference between the laminar and the turbulent skin-friction coefficients. The difference between local flat-plate skin-friction coefficients for turbulent and laminar boundary layers (reference 8) varies only slowly with Reynolds number. A constant value of 0.0026 is correct within 10 percent for Reynolds numbers between 1,000,000 and 10,000,000. This value of the coefficient is based on the total area involved instead of the wing area as usually defined, and the Reynolds number is based on the distance from the leading edge to the center of the area affected.

From the test results herein reported, a coefficient for the drag of brazier rivet heads in the turbulent boundary layer has been computed. This coefficient, based on the local dynamic pressure in the boundary layer at a distance from the surface equal to the height of the rivet heads and on the frontal area of the rivet heads, has the value of 0.32. A more convenient form of this coefficient for use with standard brazier-head rivets is that based on the square of the shank diameter in inches instead of on the frontal area of the head in square feet. In this form, the coefficient becomes 0.0020.

A coefficient computed in the same manner for plain lapped joints facing aft has the value 0.20 based on the frontal area in square feet but may be as high as 0.30 if the laps lie outside the true airfoil profile.

Since transition will occur at the most forward rivets, all the rest of the rivets will be in a turbulent boundary layer and may be treated as outlined in the preceding paragraph. The most forward rivets, if forward of the smooth-wing transition point, will be in a laminar boundary layer and this boundary layer will generally be thinner than the height of the rivet heads. These rivets, therefore, have a higher drag coefficient than those wholly submerged in the turbulent boundary layer. The coefficient expressing the drag of 3/32-inch brazier-head rivets in a laminar boundary layer about 0.004 inch thick was computed from the test results to be 1.3, based as before on the dynamic pressure at the top of the rivet head and the frontal area of the head. Based on the square of the shank diameter in inches, this coefficient becomes 0.0079.

In the application of these drag coefficients, the airfoil pressure distribution may be computed by Theodorsen's method (reference 11) and the boundary-layer thickness by the method of Dryden and Kuethe (reference 12). On the assumption that the one-seventh power law applies to the distribution of velocity in the boundary layer, the velocities and the dynamic pressures at the tops of the laps and the rivet heads can be computed for the different positions. The drags of the different laps and rivets can then be computed as the products of these dynamic pressures, the corresponding areas, and the applicable coefficients and be summed up along with the drag increment resulting from the shift of the transition point to obtain the total drag caused by the rivets and lapped joints on the wing.

This method of estimating rivet and lapped-joint drag has been applied to most of the arrangements of lapped joints and 1/16-inch and 3/32-inch brazier-head rivets tested and has yielded results in good agreement with the experimental values.

EXAMPLE

The magnitude of drag increments resulting from the presence of rivet heads and lapped joints on small airplane wings can be judged directly from the test results. As an example to illustrate the magnitude of the increments on large wings, the drag due to rivets and lapped joints on a wing having an average chord of 20 feet has been estimated. The chord positions of the spanwise rows of rivets and lapped joints assumed for this example are shown at the top of figure 23. It was assumed that the rivets were standard 3/32-inch brazier-head rivets, that the spanwise pitch was 1½ inches, and that the thickness of the lapped sheets was 0.040 inch. The flight speed was taken as 250 miles per hour at sea level, the corresponding Reynolds number being 45,000,000.

One set of ordinates in figure 23 shows the drag caused by the rivets and the lapped joints remaining on the wing when all the irregularities forward of the chord positions indicated by the abscissas have been eliminated. For example, it is estimated that all the rivets and the lapped joints shown on the wing would increase the drag coefficient by 0.00115 but that, if the rivets and the lapped joints on the forward 30 percent of the wing were eliminated, the excess drag would be reduced to 0.00035.

The excess power required just to overcome the drag caused by the rivets and the lapped joints is shown by the second set of ordinates in figure 23. The additional assumptions that the wing area is 3,600 square feet and that the propulsive efficiency is 85 percent, have been used in computing this power. With all the rivets and the lapped joints shown on the wing, more than 500 horsepower would be required just to overcome their drag. If the forward 30 percent of the wing were made smooth, about 160 horsepower would be required to overcome the drag caused by the remaining rivets and lapped joints.

CONCLUSIONS

The most important conclusions drawn from the tests described in this report can be summarized as follows:

1. Rivets at 3/4 –inch pitch in 13 spanwise rows on each surface of an airfoil of 5-foot chord increased the drag from 6 percent for countersunk rivets of 27 percent for 3/32-inch brazier-head rivets. About 70 percent of these drag increments were due to the rivets on the forward 30 percent of the airfoil.

2. Lapped joints, arranged six on each surface, increased the drag of the airfoil from 4 percent for joggled laps to 9 percent for conventional laps.

3. Surface roughness may cause serious increases in drag; for example, the roughness due to spray painting increased the drag 14 percent and roughness of 0.0013-inch grain size increased the drag 42 percent.

4. Manufacturing irregularities increased the drag of a typical wing 8 percent of the smooth-wing drag over and above the increments due to the rivets and lapped joints.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., March 7, 1939.

Document 3-31(c), Charles Peyton Autry, Boeing Aircraft Company, "Drag of Riveted Wings," in* Aviation *(May 1941): 53-54.

DRAG OF RIVETED WINGS

With so much emphasis placed on speed in present designs even rivet drag is an important factor. The following is the result of some NACA tests.

By Charles Peyton Autry, *Boeing Aircraft Company*

This discussion is intended to provide a practical insight of the excess power required in the drag of riveted metal wings with surface laps, being particular to small areas at high speeds. For certain types of wing surface construction and disposition the excess power required is of such an amount that this construction is highly unsatisfactory. Charts for a range of wing areas, Reynold's numbers and rivet dispositions are plotted to extend a means of estimating the drag in terms of the power increments. These power values have been plotted with the inclusion of an

.85 propulsive efficiency.

The NACA tests conducted by Hood on wing surface irregularities included the effects of rivet heads and surface laps on wing drag. These tests were principally on an NACA 23012 airfoil of 5 ft. chord. The drag coefficient increments of the tests were employed in the power formula forming the curves as shown in Figs. 2, 3 and 4.

The rivets and laps considered herein are those shown in Fig. 1. The rivets are of 3/32 in. shank dia. The increments are based on these rivets for a 5-ft. chord. The tests also consisted of runs on a like airfoil and rivet arrangement, of which the chord was 2 ft. and rivets were two-fifths the size of the 3/32 in. rivets, drag coefficients of the two airfoils being increased by practically equal amounts at equal Reynold's numbers, although the Mach numbers were considerably apart. The tests are thereby applicable on the basis of Reynold's numbers alone.

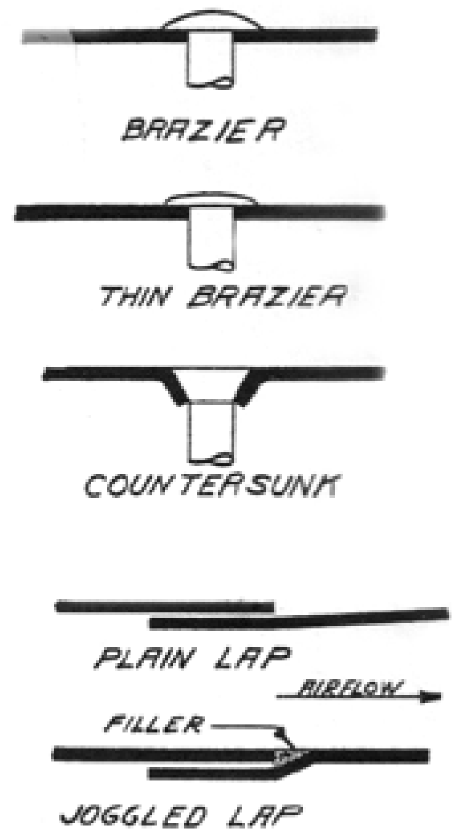


Fig. 1. Illustration of several types of rive and joints used in fastening sheet me for aircraft.

Chordwise locations of the rivet rows and laps, in these tests, were as given further for the power charts herein. Spanwise pitch of the rivets for the following power curves is .0125 of the chord or ¾ in. for the 5-ft. chord. The tests showed that rivet drag reduction was negligible up to the point of doubling the ¾ in. pitch with the forward rivet row at .04 of the chord from the leading edge. This was probably due to disturbance of the boundary layer, the drag increments being only slightly reduced by the increase in rivet pitch. With the forward row at .28 of the chord from the leading edge the rivet drag varied proportionately to the number of rivets as the pitch varied.

The tests also included the effect of various positions of the forward rows of rivets on the drag increments. Sheet thickness of laps as dealt with herein was .018 in. Drag increments as applied were taken at values when C_L = .15.

Power Increments of Two Wing Types at 400 m.p.h.

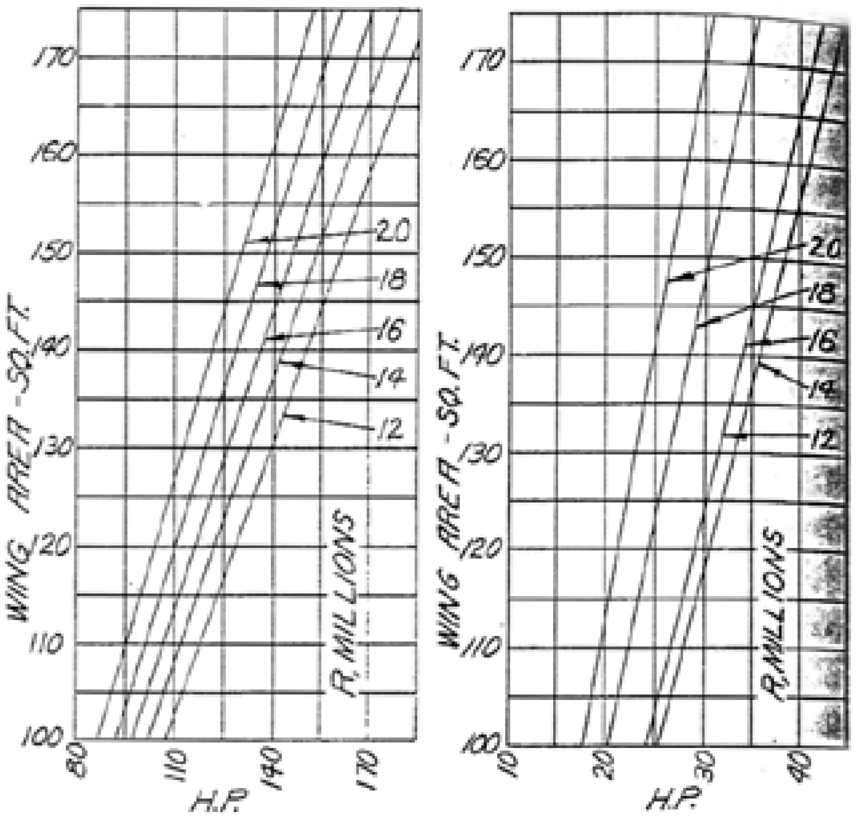


Fig. 2. Curves of horsepower plotted against wing area for 80 to 170 hp, range for various Reynold's Numbers.

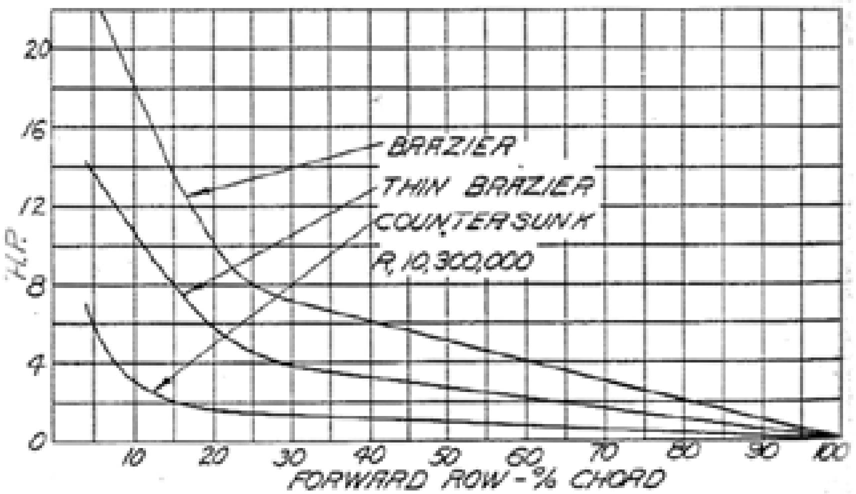


Fig. 4. The curves are for various types of rivets at a constant Reynold's Number of 10,300,000.

Fig. 2 and Fig. 3 present the values of these wings, Fig. 2 being with brazier-head rivets and plain laps. Fig. 3 gives the values for countersunk rivets and joggled laps with the bend fissure of the after sheet filled with a smooth surface filler. This lap is assumed to be representative of a lap having no drag, or that the lap drag has been successfully eliminated. The joggled laps at the plain lap positions without a filler in the after-sheet bend fissure has a zero drag increment at an R of 7,000,000 and its maximum increment at an R of 13,000,000. Such a wing of 175 sq.ft. at 400 m.p.h. would produce a power increment of 45 at an R of 13,000,000. The drag under consideration in this wing is then due to countersunk rivets, which is caused by an indention around the rivet head, in the sheet, the result of punching. Fig. 2 is not typical of a wing of excessive drag increment, probably being as low as any average wing having plain laps and brazier rivets, with the exception of any rearward difference of position of the forward rivet rows and laps or effective rivet pitch increase that might be made. Sheets conformed fairly close to profile without any excessive or needless variations.

The rivets are in 13 rows on each surface of the airfoil. The plain laps are 6 in number on each surface. The number or position of the joggled-filled laps has no bearing on the results, since they are assumed to offer no drag. The position of rivet rows on each surface of each wing are at .04, .08, .12, .20, .28, .36, .44, .52, .60, .68, .76, .84 and .92 of the chord from the leading edge. The position of the plain lap centers of each surface are .08, .20, .36, .52, .68, and .84 of the chord from the leading edge.

The charts of Fig. 1 and Fig. 2 are for various Reynold's numbers which will give usable power values for the range of wing areas given, the 400 m.p.h. speed, average chords for the areas given and an average value of ρ^s/μ.

The drag coefficient increment curves of the tests, plotted against Reynold's numbers were extended from the maximum of 18,000,000 to 20,000,000 to include the increment for the latter value.

In accordance with the power charts, Fig. 1 and Fig. 2, a wing with brazier-head rivets and plain aft-facing laps of the number and disposition given, the power increment may be reduced to about 22 percent by the use of countersunk rivets and the joggled-filled lap or its equivalent.

EFFECT OF FORWARD RIVETS ON POWER INCREMENT

Fig. 4 is illustrated of the power increments for a wing of 200 sq.ft. at 200 m.p.h., with three types of rivets. The abscissas of the chart is the most forward row of rivets. Rows aft of the first row are at the positions previously stated with the exception of the rows forward of the position in question, which are non-existent. The joggled-filled lap or its equivalent is assumed. Rivets forward of the smooth wing boundary layer transition point cause this point to move forward resulting in an increase of the turbulent boundary layer extension and a corresponding decrease in the laminar layer. Lapped joint drag varied similar to that of rivets. In Fig. 4 the rivets forward of the .30 chord point constitute about 75 percent of the total power value when the forward rivet row is at the .04 chord point.

Reduction of the power increment by increasing the spanwise pitch of the rivets forward of the smooth wing transition point may not be appreciably accomplished unless the pitch is of some marginal consequence above .025 of the chord, as previously stated from the referred tests.

CONCLUSIONS

A typical high-speed single-engine pursuit airplane incorporating the types of rivets and laps of Fig. 2 would be required to yield as much as 20 percent of its total power to the rivet and lap drag of the wing. This could be reduced to about 4 percent or less by the use of joggled-filled laps and countersunk rivets and elimination of rivets forward of the .30 chord point would further reduce the power increment to about 1 percent or less. The ultimate desired at high speeds would be complete elimination of rivet and lap protrusions or indentions. Preformed or smooth leading edges up to the .30 chord point and any practicable amount above are of much value.

Document 3-32

J. D. Akerman for the Boeing Aircraft Company, excerpts from *Manual for Design of Details Affecting Aerodynamical Characteristics of an Airplane*, Boeing Report No. D-2789, 12 Sept. 1940, Microfilm Roll No. 154a, Boeing Archives, Seattle, WA.

By the early 1940s, American aircraft manufacturers had come to understand the value of an extremely clean detail design. The problem was communicating this understanding to their engineering and production staffs. In the fall of 1940, a consultant for Boeing, aeronautical engineering professor J. D. Akerman of the University of Minnesota, prepared the following manual by which to emphasize to Boeing employees the importance of various meticulous details in the proper design and manufacture of airplanes. Akerman left nothing unexamined: rivets, seams, and joints; fasteners and screws; inspection cover plates; vents and relief drains; hinges; gaps or clearances between moving surfaces; spanwise gaps; chordwise gaps; cowlings and cowl flaps; fillets; air inlet openings; scoops; air outlet openings and leaks; streamlined covers on exposed parts; windows; doors and handles; and de-icers. All were carefully scrutinized as potential sources of disturbed airflow.

It is clear from Akerman's references at the end of his manual that NACA research played a formative role in defining the problem of the "numerous protuberances, crevices, or irregularities" affecting aerodynamic performance. Of his 14 references, 9 of them were NACA reports.

Akerman was an interesting and controversial character. For an analysis of his stormy career at the University of Minnesota, see Amy E. Foster, "Aeronautical Science 101: The Development of Engineering Science at the University of Minnesota," Master's thesis, University of Minnesota, 2000.

Document 3-32, J. D. Akerman for the Boeing Aircraft Company, excerpts from* Manual for Design of Details Affecting Aerodynamical Characteristics of an Airplane*, Boeing Report No. D-2789, 12 Sept. 1940.

BOEING AIRCRAFT COMPANY

MODEL A11
REPORT NO D-2789

MANUAL FOR DESIGN OF DETAILS AFFFECTING AERODYNAMIC CHARACTERISTICS OF AN AIRPLANE

Number of Pages: 30 Prepared by: J.D. Akerman

Date: 9-12-40 Approved by: Flight and Research Dept.

INDEX

INTRODUCTION

In order to meet the requirements for high top speed and economy of operations at cruising speeds, as specified by contractors for latest model airplanes, the aerodynamics efficiency and cleanness of design of every detail exposed to the wind has become of necessity one of the most important factors contributing towards the satisfactory performance of the airplane.

Low overall drag and aerodynamic efficiency, secured by well and tediously developed shape of major parts of an airplane, may be completely offset by numerous protuberances, crevices or irregularities appearing on the finished airplane. Very often such detrimental increases in total drag are due to a multitude of details designed, or simply installed on the airplane, without proper consideration of their individual or accumulative drags.

Extensive studies made by the N.A.C.A. and by the B.A.C. have definitely established the value and importance of clean detail design. On one actual service airplane, the combined total drag of excrescences alone constituted 45.4% of the total drag of that particular airplane, or 83% of the total drag of an equivalent ideal aerodynamically clean airplane (see B.A.C. report D-2487).

The following discussions and suggestions as to how to decrease the drag of each exposed part of an airplane by rational and careful design of the details may serve as a guide to the designers in order to eliminate most of the unnecessary drag on future airplanes. All given percentages of drag values are approximate averages for purposes of comparison.

FROM PRODUCTION AND COST STANDPOINT, IT IS OFTEN EASIER AND CHEAPER TO MAKE A SMOOTH, SIMPLE, WELL-DESIGNED PART THAN TO PRODUCE A RAGGED, HAPHAZARDLY designed part, assembly, or accessory, provided proper care is taken in design of details.

The accumulation of minute barnacles on a sea-going giant renders the ship's operation uneconomical and required periodical hull scraping. On airplanes we have reached a stage where small protuberances and irregularities are worse than barnacles and their removal is mandatory on modern high-speed aircraft.

It is essential to keep always in mind the following three basic rules:

1. AVOID MAKING ANY DETAIL WHICH WOULD CAUSE DEVIATION FROM ORIGINAL AERODYNAMIC CONTOUR – It can be done. There is always another way to do it. Somebody else probably did it on some other airplane.

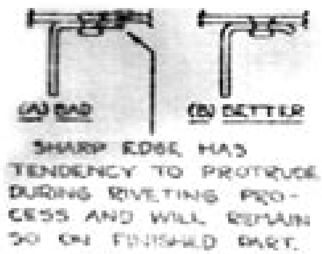
2. IF YOU CAN NOT AVOID MODIFICATION AND HAVE TO DESIGN A DETAIL PRODUCING SURFACE IRREGULARITY, MAKE THIS IRREGULARITY SMALL, SMOOTH AND WITH MINIMUM DRAG. Consult the following discussion for guidance and aerodynamic section for approval.

3. IF YOU HAVE TO MAKE "BARNACLES" MAKE THEM FEW, SMALL AND WITH MINIMUM DRAG. One "Barnacle" is negligible, but 300 designers may eliminate thousands of them and improve the performance, appearance and utility of the airplane.

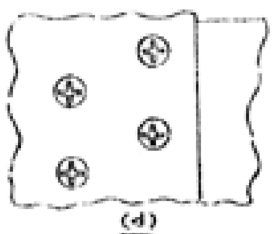
4. WHEN IN DOUBT CONSULT FLIGHT AND RESEARCH DEPARTMENT CONCERNING EFFECTS OF THE PROTRUSION ON THE DRAG OF THE AIRPLANE. At high speeds even small irregularities may be critical.

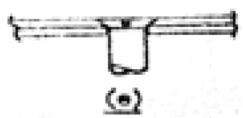
RIVETS, SKAMS AND JOINTS

Tests conducted by NACA indicated that a smooth flush rivet construction service wing had 17% extra drag due to: "(1) sheet waviness, (2) departure from true profile, and (3) imperfect laps." All three causes may be traced back, in most cases, to the original design of the wing parts. A design should be such that fast manufacturing practices could be applied and still the product would be uniform, regular and within specified tolerances without special hand adjustment of "fussing" by the craftsman. Even if the tolerances are indicated, the product can not be expected to be uniform and smooth if the original design is not fool-proof against subsequent variations and distortions of the part during the process of manufacturing.

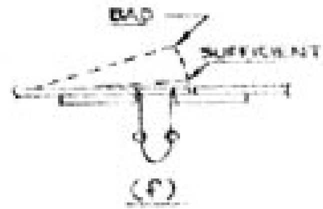


Since joints and fastenings on airplanes at the present time are made by means of rivets, and since riveting constitutes 60% of manufacturing cost of assembly of an airplane, IT IS JUSTIFIABLE FROM THE PRODUCTION AND FINAL COST STANDPOINT TO EXERCISE PARTICULAR CARE IN DESIGNING EACH JOINT. Some general rules will always help:





1. Air hammered rivets will give a worse surface than squeezed or "Kuck" pulled rivets. Therefore, make the design so that squeeze or Kuck riveting processes are applicable in subassemblies or assembly. It will be faster, cheaper and smoother.
2. Air hammer used on surfaces, and bucking bar on the inside, will produce large indentation, distortion of sheet, and rolling up of the edge.
3. Exercise care in designing joint "back up" parts. Inclined approach of riveting tool will produce bump.
   Theoretical joint as in (d) will result in distortion as shown in (e).
   A flat and sharp edge "backing part" will never produce smooth surfaces. (f)
4. In making seams the following principles should be kept in mind:
   (1) Any seam perpendicular to air stream will have approximately 70% more drag than one parallel to the air stream.
   (2) Design part so that it is possible to produce a gap of not more than 1/32 of an inch.
   (3) SPECIFY on drawing that gap should not be over 1/32 in.
   (4) Specify [ ] 1:2 as in (b).
   (5) DO NOT MAKE JOINTS ON FIRST 33% of streamlined body or curvature. It may produce compressibility wave.
   (6) AVOID MAKING joints in next 30%.



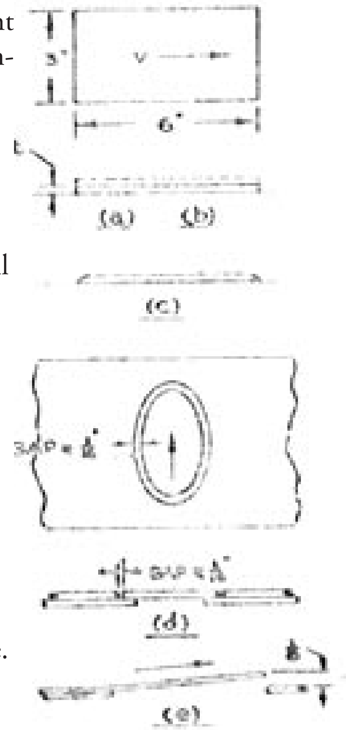
FASTENERS AND SCREWS

With the introduction of smooth wing and fuselage construction, particular care should be given to provide smooth surface where a row of permanent fasteners or screws are used in place of rivets. Some places to mention: gas tank covers, fillets, door hinge line, etc.

1. A row of fasteners just behind the front spar, as shown on Fig. (a) or (b), would destroy all the benefits secured by smooth riveting behind the row.
2. A Fastener, as in (e) and screw, as in (c), are preferred to the others.
3. Recommended use of Reed and Prince screws wherever possible in countersunk holes (d).
4. Use Camlock Fastener's, precision type (e) where Reed and Prince screws cannot be used.
5. AVOID SCREWS OR FASTENERS IN FIRST 33% WING OR STREAMLINED BODY.
6. A single, double, or triple row of Reed and Prince screws or fasteners would leave a very detrimental effect on the wing or fuselage, and therefore, special attention should be made to [ ] a COVERING STRIP of thin adhesive transparent weather resistant tape with proper water proofing. This type should be required for two reasons:
   a. To provide aerodynamically smooth surface.
   b. TO ASSURE INSTANTANEOUS INSPECTION THAT ALL SCREWS AND FASTENERS ARE AS ORIGINALLY INSPECTED, SINCE THEY CARRY STRESS AND PROVIDE NECESSARY AIR, GAS OR WATER TIGHTNESS.
7. No Finger Lifts (f) on covers should be used. They provide dangerous air leaks, place to collect water and ice, increase drag, are useless when mechanics are working with gloves.
8. "Snap On" fasteners (f) should not be used. Securely closed air and water tight inspection doors and covers should be provided with inside spring to open cover automatically if locking device is not closed.

INSPECTION COVER PLATES

Do not make external cover plates. Consult Flight and Research Department if outside cover plate is contemplated.



1. Best installation:
   a. Depressed smooth plate
   b. Groove not more than 1/16
   c. Short and perpendicular to airstreams
2. Drag on same plate 6” X 3” fig. (a) and (b) will be decreased if short side is perpendicular to the airstream instead of long side
   For t = 1/16” by 33%
   t = 1/4” by 53%
3. Drag will be decreased if
   t = 1/16” is substituted for t - 1/8 by 69%
   t = 1/16” is substituted for t - 3/16 by 83%
   t = 1/16” is substituted for t - 1/4 by 88%
4. [ ] edges will decrease plate drag 20 to 30%.
5. Make covers to close flush. A gap 1/8 inch produces drag equal to drag of a 1/8 [ ] plate.
6. Covers should close tight to prevent air and water leaks.
7. No “Finger Lifts” on outside covers. (See notes on fasteners)

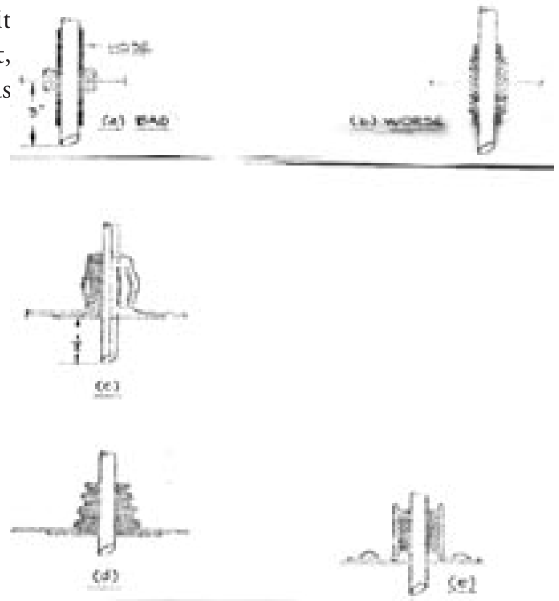
GENERAL CONSIDERATIONS

1. Do not make inspection plate line in leading edge of wing, control and tail surfaces.
2. No joint lines in forward 10% of chord; it may destroy [ ] flow at high speeds.
3. A joint line in first 10% of chord will have 50% more drag than one at 30% of chord.
4. Use flush rivets on cover plates. They are small parts and can be riveted on “Brco” rivets with flush rivets just as easy as with round head rivets.
5. The plate adds some drag; do not add extra rivet drag.

VENTS AND RELIEF DRAINS

On a recent model airplane there were counted 44 relief drains or vents. The drag of those drains constituted 1.5% of the total drag of a smooth airplane. In terms of N.P. it required 76 H.P. to drag those extensions through the air, using 50 gals. of gas during a service flight.

Present drain installation is intolerable as shown on Fig. (a) or (b).

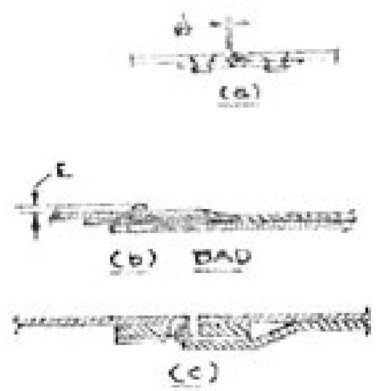


In order to get some benefit of the 50 gals. of gas per flight, the drag of the vent or drain ends should be eliminated by:

1. COMBINING SEVERAL LINES IN ONE AND DRAINING OUT THROUGH TRAILING EDGE OF WING.
   Care should be taken in doing this that NO LOW BENDS are in the line WHERE GASOLINE MAY BE TRAPPED, and that at the trailing edge, tubing does not stick out as a sharp point WHERE STATIC ELECTRICITY DISCHARGES would be produced to IGNITE GAS VAPOR or interfere with radio. Use static proof preventive coating.
2. Designing a special [ ] and short pipe and, as suggested in Fig. (c), (d) or (e) may be allowable on cowling or other places where draining through the trailing edge is impossible. Consult B.A.C. Dug. 1-17524 and 1-17525. COMBINING SEVERAL LINES IN ONE OUTLET STILL SHOULD BE [ ].

HINGES

If for some reason a hinge is unavoidable on one side of plate:

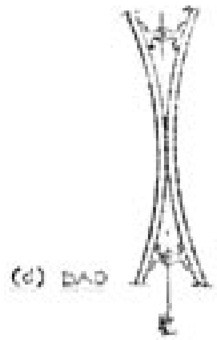


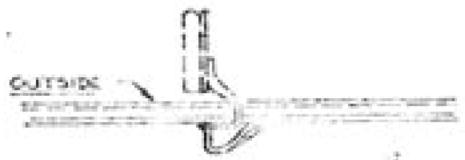
1. Make hinge line flush.
2. Use flush rivets – hinge lines are bad; do not make them worse with rivet heads.
3. Hinge line parallel to air stream will have 85% less drag per linear foot than one perpendicular to air stream.
4. No hinge line in first 30% of chord of wing, control surface, tail surfaces or any curvature less than 9° radius.
5. Faulty installation, as fig. (b), is very common. To avoid protrusion “E” installation (c) is suggested IF hinge line IS unavoidable and

(1) Perpendicular to airstream,

(2) More than 2 inches long,
(3) Located on forward portion of fuselage or cowling where local air velocities are high.

6. On many former models on latches, doors or covers, hinges are of the external type as in Fig. (d). Such external hinges are not found on latest high speed aircraft models (not even on modern automobiles) and will not be tolerated on new designs. A suggested alternate inside hinge for curved surfaces is shown in Fig. (c).





GAPS OR CLEARANCE BETWEEN MOVING SURFACES

Gaps between fixed and moving parts of surfaces fall into two types:

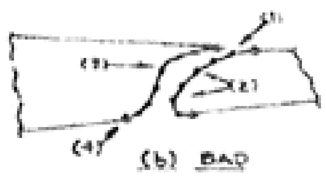
(1) Spanwise
(2) Chordwise

SPANWISE GAPS

(1) Spanwise gaps along front part (nose) of control surfaces play a very important part in the effectiveness of the control surfaces. By a slight change of shape or size of the gap, effectiveness may be decreased easily from 10 to 20%, and in some cases up to 30%. SOME SLIGHT CHANGES MAY MAKE THE GAP VERY DANGEROUS FROM THE STANDPOINT OF ICE FORMATION. Therefore, extreme care should be exercised to design the gap according to the specifications from the Flight and Research Dept. Proper dimensioning and design of details should be made so that the production departments could produce and duplicate the same gap on all airplanes of the same model and still make parts interchangeable. Very often all efforts to produce good control surfaces and aerodynamically efficient gaps are rendered ineffective by poor design of parts and surfaces inside of the gap.


(a)

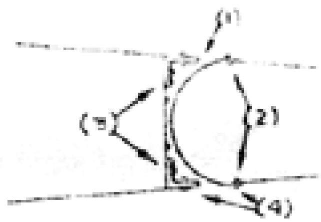
To illustrate the case, Fig. (a) was specified by aerodynamics section. When completed it had some or all protuberances as shown on Fig. (b) causing:



(1) Change of effectiveness of slot and controls; extreme drag of lap joint in high velocity air


(a)

stream: change in hinge moment of surface.
(2) Extreme drag in high velocity air stream capable of producing turbulent flow and compressibility wave. Reduction of air flow through gap and reduction of effectiveness of gap.
(3) Pocket producing whirling flow reducing effectiveness of gap and causing extra large drag.
(4) Changes of flow through gap produce excessive drag. Laps and rivets produce areas for excessive ice formation.
(5) Brackets, supports or partitions inside of spanwise gaps and slots should be just as smooth and free from rivets, bolt heads, and sharp corners as wing surface since the air velocity in the gap may be even higher than on the outside of the wing.

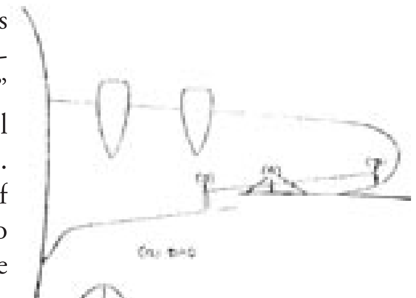


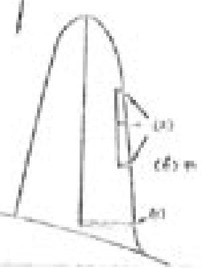
Any turbulence of air INSIDE OF GAP (or outside) does not "just happen." It is created by force and causes drag using power. The only bank from which continuous withdrawals are made to pay for those forces or drags is the power plant. Like tax taken—they amount to a large total.

EVEN IF YOU CANNOT SEE THE IRREGULARITIES FROM OUTSIDE, THEY ARE EXPOSED TO HIGH VELOCITY AIR STREAMS AND CAUSE LARGE DRAGS AND QUITE OFTEN ARE DANGEROUS FROM THE STANDPOINT OF CHARGES IN EFFECTIVENESS OF CONTROLS AND ICE FORMATIONS.

CHORDWISE GAPS

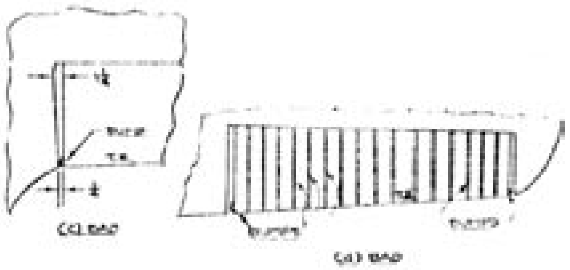
On one recently manufactured airplane gaps (1), (2), (3), and (4) fig. (a) and (b) were measured and the clearance space varied from 1/4" to 1-1/4." Such variations of gap could not all be attributed to poor manufacturing processes. Most of the variations are due to inconsistency of design and lack of specific detail dimensions to provide interchangeability of parts, and give the same clearances between parts.



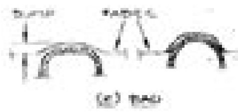


There is no reason at all why any clearance gap should be more than 1/4" on a metal airplane when 1/8" would provide ample room to take care of discrepancies in overall dimensions of sub-assemblies.

On one end of a moving part a gap width measured on top of gap 1-1/4" and on T.B. 1/4", fig. (c). Such discrepancy is not permissible on future aircraft.



A very common cause for excessive gap on control surface is the bump of cloth on the trailing edge corner. Such bumps are due to the poor practice of sewing on fabric and finishing the corner. Care should be taken to specify that corners should not protrude (it is done on other airplanes) and gap should be no help to the production because it requires adjustments and fits in assembly. Design specifications and detail dimensioning will help to avoid misfit of sub-assemblies.



FABRIC COVERINGS

On fabric covered parts of control surfaces on present designs, the ribs, rib tape, and stitching protrude from 3/32 to 3/16 of an inch above the surface of the covering. Such protrusions are bad from the standpoint of drag and amount to a large drag item since they are on all horizontal and vertical control surfaces.

Such irregularities of surface are also bad from the standpoint of control surface effectiveness since their protrusion tends to increase the thickness of the turbulent air layer over the surfaces.

Suggest a rib stitching as in Fig. (f) or better.

COWLING

The general outlines of cowling shape specified by the aerodynamics division should be adhered to in every detail.

The front part of the cowling is the critical place where high velocities are encountered. It has been determined by tests and calculations that even on present models, air velocities approaching the velocities of sound are encountered in several places on cowlings. If the front part of the cowling is continuously exposed to such high air velocities, it is obvious that certain practices in use now should be discontinued.

sq. in. gauge pressure difference between inside and outside of cowl.

6. Design cowling joints parallel to the axis in places of circular joints by designing cowling which would open as orange peels. Provision should be made to have joints smooth and tight. Fig. (a)

COWL FLAPS

On many present day airplanes cowl flaps stay open when they should be closed or tend to close when they should be open, see fig. (a) and (b). In the design of cowl flap particular care should be given to the following:

1. Opening and closing mechanics should be so designed that the pilot could definitely set cowl flaps at any desired position to secure necessary cylinder temperature control.
2. Deflections in operating mechanism and in flaps should be reduced to minimum so that flap position would not vary due to outside or inside air loads. See Fig. (a) and (b)
3. When flaps are closed longitudinal and circumferential joints should be airtight to prevent air leaks, fig. (c). Such air leaks reduce the efficiency of air outlet opening and may affect cooling of the motor. Such air leaks also produce very undesirable high drag.
4. There should be enough control to pull flaps in closed position, fig. (b).
5. Inside of flaps and flap mechanism are exposed to airstream and should be made as smooth as possible.
6. All seams and edges should be smooth and flush.

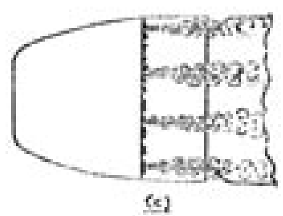
FILLETS

Filleting should be provided wherever there are two surfaces intersecting at an angle over 45%. See (a). Such filleting should be installed according to the specifications of the Flight and Research Dept. and if such specifications are missing, it should be provided in detail design, see (b). Moreover, the Flight and Research Dept. should be consulted, because there are many cases where special filleting would not be justified unless the curvature is incorporated in the parts where there are rivets, bolts or screws to cover up.

A very common mistake is shown in Fig. (c). The drag of exposed edges, rivets, or bolts may be greater than the gain provided by the fillet.

Air leaks between fillets and body are extremely undesirable. See note on air leaks.

A gap between body and fillet edge, or protruding edge is very objectionable.



AIR INLET OPENINGS

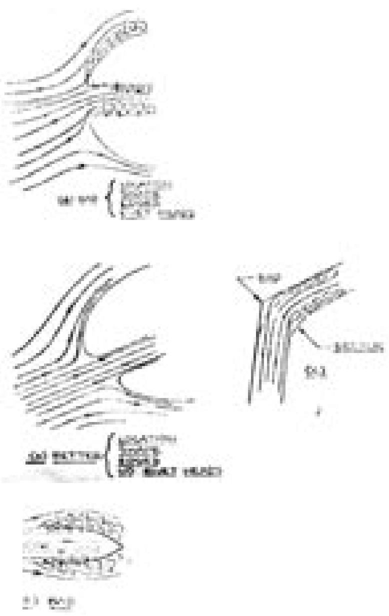
(1) FOR LOCATION OF AIR INLET HOLES, CONSULT FLIGHT AND RESEARCH DEPT.

(2) Edges and inside corners should be rounded by proper size radii (b).

(3) Edges, radii and inside surface of ducts should be just as smooth as the outside of an airplane. Flow in ducts produces drag, turbulent flow and compressibility [wave] same as outside air.

(4) Bends in ducts should be smooth and gradual in order to avoid turbulence, drag, and resistance to flow of air in duct (c).

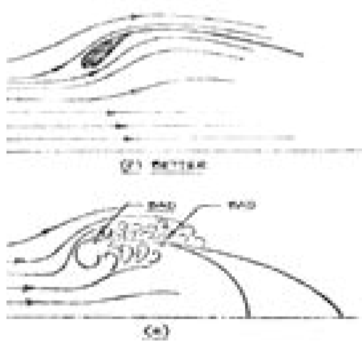
SCOOPS

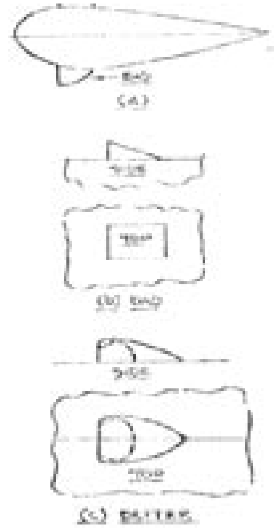


A modern airplane should not have individual scoops for each accessory hanging over all parts of the airplane. All cooling air for individual use should be taken from one or two main air inlet openings as motor cowl, inter-cooler air ducts or main ventilating air shaft. If, with permission of the Flight and Research Dept., a scoop has to be installed, the following principles should be observed:

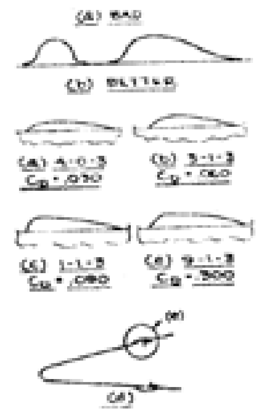
(1) Do not install scoop in region of high velocity air stress. On a recent small high speed airplane, tests disclosed that a large scoop was installed in the location of the maximum air velocity on the airplane. The designer could not have selected a worse place if he had been instructed so.
Flight and Research Dept. should be consulted for locations of each scoop.

(2) On a recent airplane were counted 14 scoops, as shown on figures (a) and (b), not counting several large cooling scoops.
A streamlined shape with tail and rounded edges figure (c) and (f) would have cut the drag at least 50%.



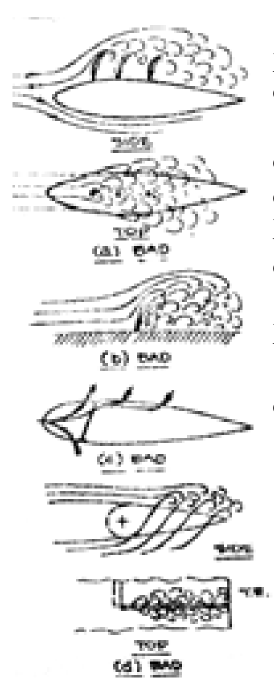


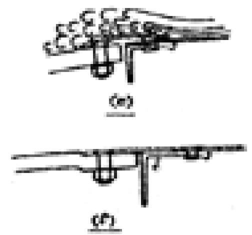
(3) In designing a scoop, particular care should be taken in the design of the forward edge.
A very misleading conception is that a sharp edge, as figure (d), would have small drag.
The point is that all scoops are designed to admit enough air at low speed in climb. At high speed or cruising only part of the air goes through the scoop and outside streamlines are deflected as shown.



(4) Seams and ridges as shown in figure (e) are very bad, since they produce turbulent flow around the whole body, resulting in very large drag.
Do not make scoops without consulting Flight and Research Dept. or trying to eliminate them completely.

AIR OUTLET OPENINGS AND LEAKS



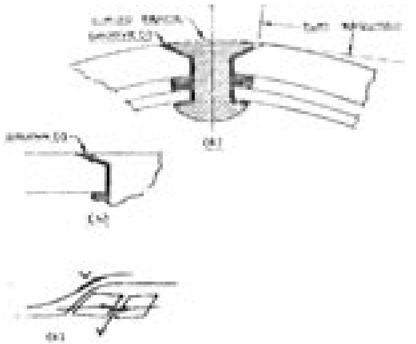


(1) The first and most important principle in designing air or gas outlets is:
(a) Air streams discharges as shown on (a) disturbs smooth laminar flow over the body behind the discharge hole and over a large area causing considerable increase in body resistance.
(b) The discharged air stream itself constitutes a body which has to break up and deflect the streamlines of the passing air (b). Such drag is just as bad, and combined with distortion of flow behind, even worse than the drag of a solid body.

(c) AIR DISCHARGE SHOULD BE REARWARD

(2) The real bad hidden culprits are the air discharge holes which are not supposed to be there at all, but which are there as a result of unintentional openings, holes, slits, gaps, etc., air discharges into low pressure areas breaking up smooth streamlined flow and causing accumulative drag of considerable magnitude.

It should be realized that each air jet resembles a "whisk broom" or a "floor mop" sticking out in the air stream. THE AIRPLANE WOULD LOOK PRETTY BAD with all these many "brooms" and "mops" sticking out and particularly if they were painted red, Fig. (f). The air jets are just as bad even if they are invisible and are not painted red.



The airplane of yesterday had too many air leaks on the wing, body, and nacelles but the airplanes of tomorrow should be airtight.

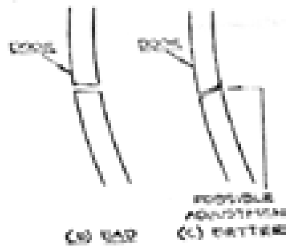
Recent studies of the effect of small air jets on the drag of a streamlined body (Ref. 6) show that a small air jet on a 10 inch dia. and 30 inch long streamlined body increases the drag of the body from 3 to 6 times depending on the location of the hole.

It is obvious that all efforts to produce streamlined low drag parts of an airplane may be useless if unnecessary air leaks are allowed.

STREAMLINED COVERS ON EXPOSED PARTS

A common malpractice will produce the cover as shown in Fig. (a).

A much better streamlined cover will be one which is even easier to manufacture as shown in Fig. (b).



Whenever there arises a question as to whether to make a short and wide cover or a long and narrow one, decrease the cross-sectional area. The forward part should not be blunt and should not have too sharp a break in its curvature.

NACA report (RIM 154) on windshield resistance gives valuable information for design of windshields, domes, covers and fairings over protruded parts. Shape 4-0-3 fig. (a) although very similar to shape 3-1-3 fig. (b) has only half of the drag and only 30% of the drag of form 1-1-3 fig. (c), and 10% of drag from 9-1-3 fig. (e).

All screws, rivets and fasteners should be the flush

type and should provide an air tight joint.

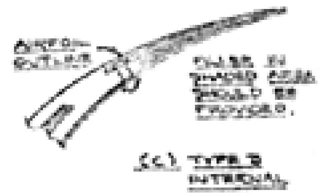
When transparent fairing is installed as in Fig. (d) and (e) a joint similar to the one in (e) is often designed, destroying most of the advantages gained by the installation of the fairing.

Joint as in (f) or better is suggested.

WINDOWS

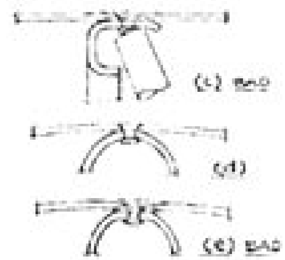
All windows should be flush and conform to the curvature of the airplane, except on places where a flat surface is specifically called for by the contractor.

In case a flat surface is required, care should be exercised that this flat surface is not increased by flat frames around this flat surface. This can be avoided by providing slight curvature in frame surfaces. Fig. A.



In case a frame is on a spherical surface, bending the frame in one plane only would produce a flat ring in a plane perpendicular to the plane of bend. This flat can be eliminated by giving very slight curvature to the frame, as in figure (a). Unless the frame is curved a flat circular ring would be produced on curved surface.

Provisions should be made that the groove (1), fig. (a) and (b) is filled with suitable filler which is water, gas, and crack proof. Such fillers are available.



MAKE ALL WINDOWS FLUSH

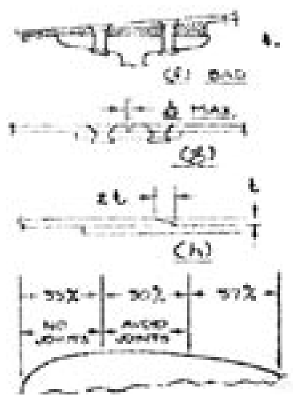
In design of windshields, it should be remembered that at points the air velocity is very high. Tests indicated that the velocity of sound is reached there at normal flying speeds.

Therefore,

(1) Make corners round and not less than [9 M].

(2) Make smooth without ridges, grooves, or rivet heads. They will cause tremendous drag and destroy laminar flow.

(3) The shape of corners on fuselage are of such importance that they should not be designed without consulting Flight and Research Department for data secured by wind tunnel and flight tests.



DOORS AND HANDLES

One most common fault of door design is the poor fit as far as flush surface is concerned on the side of the smooth body. There is always a protrusion of the door from 1/8 to 3/8 of an inch above the contour of the body.

Such protrusions are very often the result of improper design of joints as in (b), where inside corner of door requires a large gap in order to clear the frame and does not provide the solid stop of flush installation.

Joint (b) also does not make provision to prevent air leaks.

HANDLES

Handles formerly used, as shown in (d) should be eliminated and handles such as on the Yale-Clipper lock (e) or Krug disappearing lock (f) should be used.

There is a possibility of providing a very simple flush handle as in (g). Care should be taken that the spring flap comes flush and the gap is small. The dependency of action, rugged construction should not be neglected in choice of handles and locks. It is important from the standpoint of operation.

A very poor handle is commonly used as in (h). If such a handle is mandatory it may be modified as shown in Fig. (3). A step as in (k) may also be very easily modified as shown in Fig. (1).

It seems that on each airplane in the field, an after-thought has made it necessary to provide eyes to fasten the motor covers. Such an eye is very crude and could be improved by using a flush eye belt as in (c).

DEICERS

Present deicer installation is very bad and is installed on planes as an after-thought producing six spanwise ridges along the L edge of the wing.

The new type of deicer, C-313, type 3 and 4, is smoother and better adapted for modern aircraft, since this new deicer is on continuous thickness with beveled edge. Provisions should be made in the contour of the wing to accommodate the deicer. However, the surface of the part under the deicer should be smooth and the recess made gradual in order to have a smooth wing if the deicer is taken off.

The question of designing a wing with a recess for deicer or installing a deicer above the smooth L.E. may be answered from the following considerations:

- (a) Most flying is done with the deicer on.
- (b) For real service flight, deicers will be on, and then best airplane performance (airfoil characteristics) are needed.
- (c) To meet (a) and (b) a recessed wing construction may be justified. Fig. (c) or exceptionally smooth installation should be provided by:
    - (1) [Rivets] should be flush and smooth.
    - (2) Joint must be smooth since the top joint is in a place where ridge and roughness may increase the wing drag from 20 to 30%.

Document 3-33(a-b)

(a) Laurence K. Loftin, Jr. Chap. 6, "Design Maturity, 1945-1980," in *Quest for Performance: The Evolution of Modern Aircraft* (Washington, DC: NASA SP-468, 1985), pp. 137-161.

(b) Laurence K. Loftin, Jr. Chap. 6, "Design Trends," in *Quest for Performance: The Evolution of Modern Aircraft* (Washington, DC: NASA SP-468, 1985), pp. 151-162.

The following documents from Laurence K. Loftin's classic 1985 book *Quest for Performance* serve well as an abbreviated review of the aerodynamic development of propeller-driven aircraft since 1945.

Although a historical survey, it also represents a "primary source" document in its own right, in that it reflects the experience and perspective of a veteran NACA/NASA aeronautical engineer looking back over a period of 35 years in which he was internationally active as a research aerodynamicist. Loftin's views on what has happened to propeller-driven aircraft since the end of World War II are thus both timeless and a reflection of their time.

As readers will clearly see, Loftin believed that propeller-driven aircraft reached a plateau during World War II above which it may not ever rise. Too many factors inherent to the propeller limit how fast a propeller-driven aircraft can travel.

Furthermore, basic physics interfers with improving maximum lift/drag ratio, a value that has not improved much since 1945. In Loftin's view, it may still be possible to acquire small increases in lift/drag through improved structural materials, by enabling minor reductions in drag and/or increases in wing aspect ratio. But other than that, little further improvement can be anticipated. Breakthroughs relevant to smoothing out turbulent flow and controlling laminar flow in the boundary layer — complicated matters to be discussed in the next chapter on airfoil development — represented perhaps the last best hope for another round of major aerodynamic progress. But Loftin does not mention these topics here, as he was skeptical that these developments would ever materialize.

Document 3-33(a), Laurence K. Loftin, Jr. Chap. 6, "Design Maturity, 1945-1980," in* Quest for Performance: The Evolution of Modern Aircraft *(Washington, DC: NASA SP-468, 1985), pp. 137-161.

CHAPTER 6: DESIGN MATURITY, 1945-80

BACKGROUND

In the years since the end of World War II, turbojet- and turbofan-powered aircraft have come to dominate an increasingly large segment of aeronautical activity. The propeller-driven aircraft, however, remains an important part of aviation, both in this country and in various other parts of the world. The new propeller-driven aircraft that have appeared since 1945 differ little in configuration from those seen in the years immediately before and during World War II, nor has the level of aerodynamic refinement exceeded that of the earlier aircraft. The turboprop propulsion system is probably the most significant technical advancement to be incorporated in propeller-driven aircraft. In the realm of reciprocating engines, the supercharger has come into widespread use, both in commercial transport aircraft and in contemporary general aviation aircraft. The supercharger, together with the advent of cabin pressurization, has resulted in highly efficient cruising flight at high altitudes. High-altitude operation also offers the passengers freedom from the discomfort of rough air to a degree that was not possible in unpressurized aircraft.

A few examples of propeller-driven transports of the post-World War II period are described and discussed here, as are a number of contemporary general aviation aircraft.

Two families of large, long-range, propeller-driven transports dominated U.S. airlines, as well as many foreign airlines, until the jet transport began to appear in significant numbers toward the end of the 1950's. These families of aircraft, which served on both long-range domestic and international routes, were the Douglas DC-6 and DC-7 series and the Lockheed Constellation series. Both were derived from aircraft developed during World War II; they had four supercharged engines and pressurized cabins, and both series underwent large increases in size, power, and weight during their development history.

Representative of the long-range, four-engine transport is the Lockheed L. 1049G Super Constellation illustrated in figure 6. 1. The prototype Constellation, known by its USAAF designation of C-69, first flew on January 9, 1943, and the model L. 1049G first flew on December 12, 1954. The total number of all models of the Constellation constructed was 856.

The Lockheed L.1049G was powered by four Wright turbocompound engines of about 3250 horsepower each. The Wright 3350 turbocompound engine employed a two-speed gear-driven supercharger and, in addition, was equipped with three exhaust-driven turbines. The three turbines were geared to a single shaft that in turn was hydraulically coupled to the engine crankshaft. Each turbine was driven by

the exhaust of six cylinders. About 15 percent of the total power of the engine was obtained from reclamation of exhaust gas energy. The specific fuel consumption was probably the lowest ever achieved in a reciprocating aircraft engine.

Figure 6.1 - Lockheed 1049G Super Constellation 91-passenger four-engine airliner; 1954. [mfr]

The gross weight of the aircraft was 133,000 pounds, which was more than twice that of the Boeing B-17 "heavy" bomber of World War II fame. The wing loading was 80.6 pounds per square foot, and.the corresponding stalling speed was 100 miles per hour. The wings employed very powerful Fowler-type extensible slotted flaps to maintain the landing speed within acceptable limits. The landing gear was of the tricycle type that was standard on most post-World War II transports. The maximum speed of the aircraft was 352 miles per hour, and the normal cruising speed was 331 miles per hour at 23 000 feet. The pressurized cabin was capable of seating 71 first-class passengers or 91 coach passengers. Some versions of the aircraft were capable of carrying an acceptable payload nonstop from the east coast of the United States to the west coast. The zero-lift drag coefficient of 0.0211 and the maximum lift-drag ratio of 16 indicate a highly refined and efficient aerodynamic design.

Today, many Constellations and their Douglas counterparts are in operation in nonscheduled activities in different parts of the world. The use of these aircraft in long-range scheduled operations, however, terminated in this country during the 1960's following the introduction of the high-performance jet-powered transport.

The turbopropeller, or turboprop engine, is basically derived by gearing a conventional propeller to the shaft of a gas generator composed of a compressor, burner, and turbine. The turboprop engine may therefore be thought of as a turbojet engine that transmits power to the air by means of a propeller rather than through the jet exhaust. The turboprop engine is light and relatively simple as compared with the large high-power reciprocating engines. For example, a modern turboprop engine may develop between 2 and 3 horsepower per pound of weight, as compared with a maximum of about 1 horsepower per pound for a reciprocating engine, and has been made in sizes of up to 15,000 horsepower. The specific fuel consumption of the turboprop engine, however, is somewhat higher than that of the best reciprocating engines. The turboprop engine has been used in a number of highly successful transport aircraft and is still in fairly widespread use, particularly for short-haul, commuter-type transports.

Figure 6.2 - Vickers Viscount 810 40-passenger turboprop airliner,- 1948. [Peter C. Boisseau]

The first civil airliner to be equipped with turboprop engines was the Vickers Viscount depicted in figure 6.2. The aircraft employed four Rolls-Royce Dart engines of 1600 horsepower each and had a gross weight of about 60 000 pounds. Depending upon the configuration, 40 to 59 passengers could be carried in the pressurized cabin. The cruising speed of the Viscount was 334 miles per hour at 25 000 feet. The aircraft employed double-slotted flaps and was equipped with a tricycle landing gear. The Viscount made its first flight in July 1948 and subsequently was used by airlines all over the world. A total of 441 Viscounts were built and many are still in use.

Two turboprop aircraft of much larger size were constructed in the United Kingdom. These were the Vickers Vanguard, with a gross weight of 146 500 pounds, and the Bristol Britannia, with a gross weight of 185 000 pounds. Many types of turboprop transport aircraft have been designed and built in Russia, as well. The largest passenger carrying turboprop ever built was the Tupolev Tu-114. This aircraft has a gross weight of 377 000 pounds and is equipped with four 14 795 equivalent shaft horsepower turboprop engines. Each of these engines drives two counterrotating propellers. The wings are sweptback, which is unusual for propeller-driven aircraft; the amount of sweep is 34°. The aircraft carries 220 passengers and cruises at a speed of 478 miles per hour at an altitude of 29 500 feet. The Tu-114 is no longer in airline use, but a version known as the Bear is employed by the Soviet military forces as a reconnaissance aircraft. The Lockheed Electra is the only large turboprop airliner to be developed in the United States. Although the Electra was an efficient high-performance aircraft, it was never produced in large numbers because it was introduced at about the same time as the Boeing 707 jet airliner and could not compete with this aircraft. A few Electras are still in service with the scheduled airlines, and a number are employed in nonscheduled activities. The naval version of the aircraft, known as the P-3 Orion, is employed by the U.S. Navy for antisubmarine patrol work.

Figure 6.3 - Lockheed C-130 turboprop cargo transport; 1955. [mfr]

A number of highly successful turboprop aircraft have been developed for use as cargo carriers. The largest of these aircraft is the Russian Antonov AN-22, which weighs over 550 000 pounds and is equipped with four 15 000-horsepower engines. The Lockheed C-130 is perhaps the best known of the turbo prop-powered cargo aircraft and the one that has been produced in the greatest numbers. The C-130 is used by all branches of the United States military forces and by the military forces of over 20 foreign governments. A commercial cargo version of the aircraft is also available. The first production contract for the aircraft was placed in 1952; over 1500 models of the C-130 have been built, and the aircraft is still in production.

A Lockheed C-130 is shown in figure 6.3. Many variations of the C-130 have been produced, and engines of slightly different power ratings have been employed. The aircraft has an unswept wing mounted in the high position at the top of the fuselage and is equipped with four Allison T-56 turboprop engines of 4910 equivalent shaft horsepower each at takeoff. In order to minimize weight and complexity, the landing gear is retracted into blisters located on either side of the fuselage, rather than into the wing or engine nacelles. The high wing position is advantageous for a cargo aircraft because it allows trucks and other types of equipment to move beneath.the wing, and the fuselage can be brought close to the ground without causing interference with the engines and propellers. A rear loading door may be deployed from the bottom of the upswept, aft portion of the underside of the fuselage. The proximity of the forward portion of the fuselage to the ground results in an aft-loading ramp with only a small inclination to the ground so that vehicles can be readily driven or pushed into the aircraft. The Lockheed C-130 has a gross weight of 155 000 pounds and cruises at a speed of 386 miles per hour at 20 000 feet. The wing loading is 88 pounds per square foot, and the landing speed is 115 miles per hour.

A great variety of twin-engine airliners has been developed both in the United States and abroad during the postwar years. These aircraft are smaller than the large, long-range, four-engine aircraft and are employed on short-haul types of operations. Twin-engine airliners have been developed with both reciprocating and turboprop engines. The twin-engine Martin 404 and Convair 440 aircraft and earlier versions of these machines were perhaps the most-used postwar twin-engine transports powered with reciprocating engines. These aircraft are similar in configuration to the Douglas DC-3 but are larger, faster, and are equipped with pressurized cabins; in addition, they both employ the tricycle type of landing gear. The Fairchild F-27 (a Dutch Fokker design built under license by Fairchild in this country) and the Japanese YS-IIA are probably the best known turboprop twins in the United States. The British Hawker Siddeley 748 turboprop-powered twin-engine airliner is widely used in many countries of the world.

Although the long-range propeller-driven transport is essentially a thing of the past, smaller, short-range aircraft of this type are becoming more numerous. Since the advent of airline deregulation in the United States in the latter part of the 1970's, there has been a large growth in short-haul, commuter-type airline operations. Many aircraft employed in this type of service are foreign built, are of high-wing configuration, and are equipped with two turboprop engines. Passenger capacity varies between 20 and 30, and at least one four-engine aircraft of this type carries 50 passengers. Generally speaking, these aircraft have straight wings and employ no new configuration concepts. In fact, some of them have fixed landing gears and strut-braced wings. Since high speed is unimportant and low initial and maintenance costs are critical, these retrogressive technical features are justified on a cost-effectiveness basis. The final forms of the commuter-type transport, however, are yet to emerge.

GENERAL AVIATION AIRCRAFT

The term "general aviation" covers all types of flying except military and commercial airline operations. Only contemporary aircraft designed for business and pleasure are considered here. General aviation aircraft designed for business and pleasure are available in both single-engine and twin-engine models; most models are equipped with horizontally opposed reciprocating engines. However, several high-performance turboprop types are offered. Single-engine types may be had with high- or low-wing location, retractable or fixed landing gear, controllable-pitch or fixed-pitch propeller, and in sizes varying from two place to seven place. The twin-engine aircraft usually employ the low wing location and have retractable landing gear and controllable-pitch propellers. The twins may be had with or without turbosupercharging, with or without pressurized cabins, and with varying seating capacities. The modern aircraft designed for business or pleasure is almost invariably of all-metal construction, as contrasted with the metal, wood, and fabric construction typical of the pre-World War II general aviation aircraft. Reliability of the internal systems employed in the aircraft and the precision of the radio and navigational equipment have greatly improved as compared with pre-World War II standards. The general aviation aircraft of today are almost universally equipped with an electrical system to power the radios and other types of equipment installed in the aircraft and to operate the self-starter. Hand starting of production aircraft is a thing of the past. The cabins of these aircraft are generally relatively comfortable, are equipped with heaters for wintertime and high-altitude use, and are sometimes equipped with air conditioning for use on the ground and at low altitudes in the summer. The open cockpit is a thing of the past in production aircraft, except for special sport and aerobatic aircraft. Many aircraft employ complete instrumentation and communication equipment for flight under IFR conditions. Most contemporary aircraft employ a tricycle gear that greatly eases the problem of aircraft handling on the ground. The basic aerodynamic configuration of contemporary general aviation aircraft, however, differs little from those in use in 1939.

CONTEMPORARY TYPES, 1970-80

General aviation aircraft are manufactured in a number of different countries; however, the majority of these aircraft are produced in the United States. The major U.S. producers are the Cessna Aircraft Company, the Piper Aircraft Corporation, and the Beech Aircraft Corporation. Each offers a wide variety of aircraft designed for various needs and markets. Six aircraft of different levels of performance, size, and price produced by these manufacturers for different segments of the market are briefly described here.

Two single-engine aircraft representative of the lower performance and price spectrum are shown in figures 6.4 and 6.5. The Piper Cherokee 180 shown in figure 6.4 is an all-metal aircraft with an internally braced, cantilever wing mounted in the low position. The aircraft shown has four seats and is equipped with a 180-

Figure 6.4 - Piper Cherokee 180 contemporary general aviation aircraft. [NASA]

horsepower, four-cylinder Lycoming engine of the opposed type. The engine drives a fixed-pitch propeller. The landing gear on the aircraft is fixed, and although not visible in the photograph, the horizontal tail employed on the Cherokee is of the all-moving type equipped with a geared tab. The Cherokee 180 has a maximum speed of 148 miles per hour at sea level and cruises at 141 miles per hour at 7000 feet. The stalling speed with the split flaps deflected is 61 miles per hour. The gross weight of the aircraft was 2450 pounds. The Cherokee 180 is representative of one of the lower cost members of a complete family of Piper aircraft that carry the Cherokee name. Some of these aircraft have six or seven seats and more powerful engines that drive controllable-pitch propellers. Other versions of the Cherokee employ a retractable landing gear. The flight of the first production aircraft was made in February 1961, and well over 25 000 Cherokees of all types have now been produced.

The Cessna Skyhawk shown in figure 6.5 is one of the lower cost members of an entire series of Cessna aircraft of the same basic configuration. The Skyhawk, like the Cherokee 180, is equipped with a fixed tricycle landing gear and has a four-cylinder, horizontally opposed engine driving a fixed-pitch propeller. Unlike the Cherokee 180, however, the Cessna Skyhawk is a high-wing configuration with a single wing strut on either side of the fuselage to brace the wing. The Skyhawk has a maximum speed of 144 miles per hour and cruises at 138 miles per hour at 8000 feet. The stalling speed with the flaps deflected is 49 miles per hour. The gross weight of the Cessna Skyhawk is 2300 pounds, and the wing loading and power loading are 13.1 pounds per square foot and 15.3 pounds per horsepower, respectively. These values are in the same order as for the Piper Cherokee. The zero-lift drag coefficient of the Skyhawk is 0.0319 as compared with 0.0358 for the Cherokee, and the maximum lift-drag ratios for the two aircraft are 11.6 and 10.0, respectively.

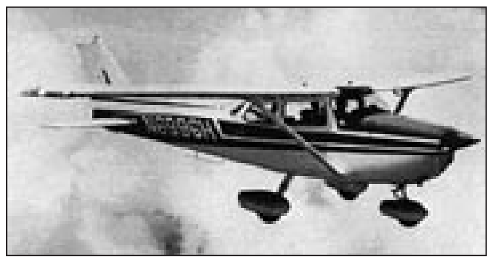


Figure 6.5 - Cessna Skyhawk contemporary general aviation aircraft. [mfr]

Two representative high-performance single-engine general aviation aircraft are shown in figures 6.6 and 6.7. The Beech Bonanza V-35B shown in figure 6.6 is of all-metal construction, has an internally braced wing mounted in the low position, has single-slotted flaps, and is equipped with a fully retractable tricycle landing gear. The aircraft is equipped with a six-cylinder, horizontally opposed Continental engine of 285 horsepower that drives a controllable-pitch propeller. The aircraft can be configured for four, five, or six seats. The unique Butterfly tail combines the stability and control functions of both the conventional vertical and horizontal tails. The gross weight of the aircraft is 3400 pounds. The aircraft has a maximum speed of 210 miles per hour at sea level, cruises at 203 miles per hour at 6500 feet, and has a stalling speed of 63 miles per hour. The zero-lift drag coefficient is a very low 0.0192, and the corresponding maximum lift-drag ratio is 13.8. The prototype of the Bonanza first flew in December 1945, and the aircraft has been continuously in production since 1947. Approximately 10 000 Beech Bonanzas have been built.

Figure 6 6 - Beech Bonanza V-35B contemporary general aviation aircraft. [mfr]

The Cessna Cardinal RG II shown in figure 6.7 is a high-performance aircraft with an internally braced wing mounted in the high position. The aircraft is equipped with a fully retractable tricycle landing gear and is equipped with a four-cylinder, horizontally opposed, Lycoming engine of 200 horsepower that drives a controllable-pitch propeller. The Cardinal is of all-metal construction and is equipped with trailing-edge flaps and an all-moving horizontal tail employing a geared trim tab. The aircraft has a maximum speed of 180 miles per hour at sea level, cruises at 171 miles per hour at 7000 feet, and has a stalling speed of 57 miles per hour. The aircraft weighs 2800 pounds. The.zero-lift drag coefficient of the Cardinal is 0.0223, and the corresponding maximum lift-drag ratio is 14.2.

Figure 6 7 - Cessna Cardinal RG II contemporary general aviation aircraft. [mfr]

The first twin-engine aircraft designed specifically for business use was probably the Beech Model D-18, first produced in 1937. This aircraft was similar to the Douglas DC-3 in general appearance, although much smaller, and was in continuous production from 1937 until the early 1970's. A wide variety of twin-engine aircraft of various sizes and with different levels of performance are now offered for business use. Two contemporary twin-engine aircraft are shown in figures 6.8 and 6.9.

The Cessna 310 shown in figure 6.8 is representative of one of the smaller contemporary twin-engine aircraft offered for business use. The aircraft is a low-wing configuration with an engine mounted in each wing on either side of the fuselage. The aircraft can be had with both normally aspirated engines or with turbosuperchargers. The engines are six-cylinder, horizontally opposed, Continental engines

of 285 horsepower each that drive controllable-pitch, full-feathering propellers. The aircraft normally has a seating capacity of five but can be configured for six. Maximum speed is 238 miles per hour at sea level, and cruising speed is 223 miles per hour at 7500 feet. The wings are equipped with split flaps which with a wing loading of 30.7 pounds per square foot result in a stalling speed of 77 miles per hour. The gross weight of the aircraft is 5500 pounds. The. Cessna 310 has a zero-lift drag coefficient of 0.0267 and a maximum lift-drag ratio of 13. The Cessna 310 was first flown in January 1953 and has been in continuous production ever since. The aircraft is unpressurized and may be thought of as the smallest of a whole line of Cessna twins, both pressurized and unpressurized.

Figure 6.8 - Cessna 310 contemporary twin-engine general aviation aircraft. [mfr]

The Beech Super King Air 200 shown in figure 6.9 is an example of a new, relatively large, high-performance twin-engine business aircraft. Provision is provided for 2 pilots and 6 to 13 passengers, depending on the configuration. The cabin is pressurized to permit comfortable cruising flight at high altitudes. Power is provided by two Pratt & Whitney PT6A-41 turboprop engines of 850 shaft horsepower each. The engines drive controllable-pitch, full-feathering, reversible propellers. The low-wing configuration of the aircraft is conventional although the use of a T-tail on a straight-wing propeller-driven aircraft is somewhat unusual. The use of this tail arrangement is said to reduce both vibration resulting from the slipstream of the engines and trim changes with flap deflection. The aspect ratio of the wing is 9.8, which must be considered as relatively high for any aircraft. The King Air 200 has a maximum speed of 333 miles per hour at 15,000 feet and a maximum cruising speed of 320 miles per hour at 25,000 feet. The aircraft is equipped with single-slotted flaps that together with a wing loading of 41.3 pounds per square foot give a stalling speed of 92 miles per hour. The gross weight of the aircraft is 12,500 pounds. The Beech Super King Air 200 was certified in December 1973 and is now in series production.

Figure 6.9 - Beech Super King Air 200 contemporary twin-engine turboprop general aviation aircraft. [mfr]

OTHER TYPES OF GENERAL AVIATION AIRCRAFT

The six aircraft just described may be considered as representative of generic classes of aircraft designed for business and pleasure use. In order to gain a true appreciation of the wide variety of such aircraft offered today, the reader is referred to the current year's issue of Jane's All The World's Aircraft. Other types of aircraft of interest and not described here are specially designed agricultural aircraft intended for spraying and dusting crops. These aircraft will also be found in Jane's, as will many types of sport and aerobatic aircraft. Another segment of general aviation aircraft is made up of the so-called home builts. These aircraft, which are built by individuals or clubs at home, are gaining in popularity and are flown in relatively large numbers in this country. They are usually not certified under any of the pertinent federal air regulations but, rather, operate in an experimental category. Many of the more popular types of home-built designs are also described in Jane's All The World's Aircraft.

Document 3-33(b), Laurence K. Loftin, Jr. Chap. 6, "Design Trends," in* Quest for Performance: The Evolution of Modern Aircraft *(Washington, DC: NASA SP-468, 1985).

CHAPTER 7: DESIGN TRENDS

INTRODUCTION

This chapter briefly summarizes the progress in design of propeller-driven aircraft since the end of World War I by showing how a number of important design and performance parameters varied over the years 1920 to 1980. The following parameters are discussed:

(1) Maximum speed, V_{max}
(2) Stalling speed, V_s
(3) Wing loading, W/S
(4) Maximum lift coefficient, $C_{L',max}$
(5) Power loading, W/P
(6) Zero-lift drag coefficient, C_D,O
(7) Skin friction parameter, C_F {Ptr, in number 7, make a line over the "C" as shown.}
(8) Maximum lift-drag ratio, $(L/D)_{max}$

The values of each of these parameters are plotted against the appropriate year in figures 7.2 to 7.9. All of the parameters could not be obtained for some of the aircraft; in particular, the zero-lift drag coefficient and the maximum lift-drag ratio could not be determined for a number of the aircraft because of insufficient performance data from which to make the desired calculations. The symbols identifying each aircraft are given in figure 7.1 and have been used throughout the subsequent figures. At the left side of each figure (figs. 7.2 to 7.9), bars have been drawn to indicate the spread of each parameter during World War I. The year for which the

Figure 7.1 - Symbols used in figures 7.2 to 7.9.

characteristics of a given aircraft are plotted is in some degree arbitrary. For example, most of the World War II aircraft characteristics are plotted for the year 1942. In other cases, aircraft that were used for a number of years are shown at a year corresponding to the first year of production, or after the aircraft had achieved a fully developed status. The points for the different aircraft show a large spread in the different figures; hence, lines representing an upper and lower bound are shown on each figure. (The shape of these bound lines may be varied according to the manner in which the data are interpreted. The lines shown are only suggested fairings of the data points presented.) One of these bounds corresponds to aircraft developed with the highest technology available at a particular time, and the other is for aircraft of a relatively low and slow-changing level of technology. Neither of these bounds represents boundaries of maximum and minimum values but, rather, corresponds to higher and lower levels of technology for operational aircraft of a particular time period. No data for racing or special performance aircraft are given in the figures.

MAXIMUM SPEED

Trends in maximum speed of propeller-driven aircraft are shown as a function of time in figure 7.2. The maximum speeds of high-technology operational aircraft are seen to increase steadily from about 125 miles per hour in 1920 to nearly 450 miles per hour in the World War II years. The highest maximum speed shown is for the P-51D aircraft, which had a speed of 437 miles per hour at 25 000 feet. Late in the war, a Republic P-47J achieved a speed in level flight of 507 miles per hour at 34 000 feet. The upper bound through the years closely follows the advancement of fighter-type aircraft. The large increases in maximum speed that occurred between World War I and World War II resulted from increases in engine power and reductions in drag area through improved aerodynamic efficiency. For example, the 10 000-pound P-51 fighter of World War II had a drag area of only 3.8 square feet (this corresponds to a circular disc 2.20 feet in diameter) and was equipped with an