FORTIETH ANNUAL REPORT

OF THE

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

1954

ADMINISTRATIVE REPORT
INCLUDING TECHNICAL REPORTS
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To the Congress of the United States:

In compliance with the provisions of the act of March 3, 1915, as amended, establishing the National Advisory Committee for Aeronautics, I transmit herewith the Fortieth Annual Report of the Committee covering the fiscal year 1954.

DWIGHT D. EISENHOWER.

THE WHITE HOUSE,
JANUARY 26, 1955.

Letter of Submittal

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON, D. C., *October 22, 1954.*

DEAR MR. PRESIDENT:

In compliance with the act of Congress approved March 3, 1915, as amended (U. S. C. title 50, sec. 151), I submit herewith the Fortieth Annual Report of the National Advisory Committee for Aeronautics for 1954.

The appropriation for operating the National Advisory Committee for Aeronautics for the present fiscal year is $52,000,000. The Committee has reviewed critically its scientific research programs and concluded that additional aeronautical research effort is necessary.

Briefly, the important facts are:

(1) The current trend toward leveling off expenditures for scientific research in aeronautics is forcing hard decisions to slow down or to defer indefinitely research projects essential to the timely development of new weapons.

(2) It is now wise to accelerate scientific progress. In the long run, scientific research is the best insurance that there will be "value received" from the country's whole aircraft program.

Attention is invited to the Committee's opening statement to the Congress regarding the necessity for maintaining our supremacy in the air.

Respectfully submitted.

JEROME C. HUNSAKER,
Chairman.

THE PRESIDENT,
The White House, Washington, D. C.

National Advisory Committee for Aeronautics

Headquarters, 1512 H Street NW., Washington 25, D. C.

Created by act of Congress approved March 3, 1915, for the supervision and direction of the scientific study of the problems of flight (U. S. Code, title 50, sec. 151). Its membership was increased from 12 to 15 by act approved March 2, 1929, and to 17 by act approved May 25, 1948. The members are appointed by the President. and serve as such without compensation.

FORTIETH ANNUAL REPORT

OF THE

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON, D. C., *October 22, 1954.*

To the Congress of the United States:

In accordance with the act of Congress, approved March 3, 1915, as amended (U. S. C. title 50, sec. 151), which established the National Advisory Committee for Aeronautics, the Committee submits its Fortieth Annual Report for the fiscal year 1954.

During the first half-century of powered flight the airplane was developed into a principal military weapon. Its use has altered the course of history and the destinies of nations.

Today, at the start of the second half-century of powered flight, the nuclear bomb carried now by the airplane, and ultimately by its unmanned counterpart, the guided missile, has become the most powerful military weapon of all time.

We are in a race to acquire the scientific knowledge necessary to create airplanes and missiles with the capabilities that permit military use at extreme altitudes across intercontinental distances, and with supersonic swiftness to penetrate enemy defenses. This race starts in the research laboratories. It may be decided there. The technical problems involved are complex, interrelated, and difficult. How rapidly we solve them will be determined mainly by the effort applied.

How close the race has become, we cannot know with certainty. We do know that Russian technical air progress is challenging. That fact was impressed upon our airmen in Korea. It was underlined in 1954 by Russia's open display of its new jet bombers.

Despite evidence of substantial acceleration of technology in Russia, we believe we still have a qualitative lead. Maintaining that lead demands more vigorous attacks upon the research problems before us. Continued effective teamwork by the military services and the aircraft industry can insure early exploitation of research results in practical applications leading to improved aircraft and missiles.

In these critical days we should be stockpiling research results and engineering data just as we are stockpiling strategic materials and weapons. During the past 3 years, however, there has been a "leveling off" of appropriations for scientific research in aeronautics. This leveling off has forced the NACA to make reluctant decisions to slow down, or to defer indefinitely, many important research opportunities.

To sum up, our national security requires that we be first in the air. Leadership in scientific research is the key element.

Respectfully submitted.

JEROME C. HUNSAKER,
Chairman.

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Part I—TECHNICAL ACTIVITIES

THE NACA—WHAT IT IS AND HOW IT OPERATES

One of the most important functions of the National Advisory Committee for Aeronautics is that of coordinating the aeronautical research carried on in the United States. The makeup of both the Main Committee and the 29 technical subcommittees embraces the several military and civil government agencies concerned with aeronautics, and includes members from scientific institutions, and the aviation manufacturing and operating industries. Thus wasteful and costly duplication of research and development effort is avoided.

In the conduct of its business, which is scientific laboratory research in aeronautics, the NACA, since its establishment in 1915 by the Congress, has functioned to serve the needs of all departments of the Government. The 17 members of the Main Committee are appointed by and report to the President. Serving without pay, they operate like a board of directors, establishing policy and planning the research programs to be followed by the 7,000 civil-service personnel who make up the technical and administrative staff of the NACA.

The Committee is assisted in the determination and coordination of research programs by 6 major and 23 subordinate technical committees, with a total membership of more than 400. These men are selected because of their technical ability, experience, and recognized leadership in a special field. They also serve without compensation, in a personal and professional capacity. They provide material assistance in the consideration of problems related to their technological fields, review research in progress both at NACA laboratories and in other organizations, recommend research projects to be undertaken, and assist in the coordination of research programs.

Membership on the technical committees and subcommittees, as well as the Industry Consulting Committee, is listed in part II of this report, beginning on page 67.

Coordination of research is also accomplished through frequent discussions by NACA technical staff personnel, with the research organizations of the aircraft industry, educational and scientific institutions, and other aeronautical agencies. The NACA maintains a West Coast office to further liaison with the aeronautical research and engineering staffs of that geographical area.

During the 39 years since its organization as an independent Federal agency, the NACA has sought to assess the current status of development of aircraft, both civil and military; to anticipate the research needs of aeronautics; to develop the scientific staff and special research facilities required, and to acquire the needed information as rapidly as may be consistent with the national interest.

The NACA's research programs have had both the long-range, all-inclusive objective of acquiring the new scientific knowledge essential to assure American leadership in aeronautics, and the immediate objective of solving, as quickly as possible, the most pressing problems, thus to give effective support to the Nation's current aircraft construction program.

Most of the problems to be studied are assigned to NACA's research centers: the Langley Aeronautical Laboratory in Virginia, where research is conducted on aerodynamic, structures, hydrodynamic, and other problems; the Ames Aeronautical Laboratory in California, which concentrates on aerodynamic research; the Lewis Flight Propulsion Laboratory in Ohio, which is concerned primarily with power-plant problems; and the High Speed Flight Station in California, where specially designed, specially instrumented research aircraft are used in full-scale research on transonic and supersonic problems. Aerodynamic problems in the transonic and supersonic speed ranges are studied, using rocket-powered models in free flight, at the NACA research installation located at Wallops Island, off the Virginia Coast.

The NACA also sponsors and finances a coordinated program of research at 27 nonprofit scientific and educational institutions, including the National Bureau of Standards and the Forest Products Laboratory. By this means, scientists and research engineers, whose skills and talents otherwise might not be available, contribute importantly to the Government's program of aeronautical research. Promising students also receive scientific training which makes them useful additions to the country's supply of technical manpower.

During the fiscal year 1954, the following institutions participated in the NACA's program of contract research:

National Bureau of Standards
Forest Products Laboratory

Battelle Memorial Institute
Polytechnic Institute of Brooklyn
California Institute of Technology
University of California
Carnegie Institute of Technology
Case Institute of Technology
University of Chicago (NORC)
University of Cincinnati
Columbia University
Cornell University
Georgia Institute of Technology
Iowa State College
Johns Hopkins University
Massachusetts Institute of Technology
University of Michigan
Syracuse University
University of Wisconsin
University of Alabama
Brown University
Stanford University
Stevens Institute of Technology
Agricultural & Mechanical College of Texas
Yale University
Armour Research Foundation
New York University

Proposals from such institutions are carefully screened to assure best use of the limited funds available to the NACA for sponsoring research outside its own facilities. Similarly, results from these projects are reviewed to maintain the quality of this part of the NACA program. Reports of the useful results are given the same wide distribution as other NACA publications.

During the fiscal year, most of the NACA technical subcommittees reviewed proposals for research projects from outside organizations, or gave attention to reports from completed contracts. Reports covering results of sponsored research totaled 42 during fiscal year 1954.

Research information, including that obtained in the Committee's laboratories and elsewhere under NACA sponsorship, is distributed in the form of Committee publications. Reports and Technical Notes, containing information that is not classified for reasons of military security, are available to the public in general. Translations of important foreign research information are published as Technical Memorandums.

The NACA also prepares a large number of reports containing information of classified nature. These, for reasons of national security, are closely controlled as to circulation. When it is found possible at a later date to declassify such information, these reports also may be given wider distribution.

Current announcement of NACA publications is contained in the NACA Research Abstracts. This service, in addition to telling of NACA publications, makes note of important research reports received from abroad.

In addition to other means of making research information readily available, the NACA each year holds a number of technical conferences with representatives of the aviation industry, universities, and the military services. Attendance at these meetings is restricted, because of the security classification of the material presented, and the subject material is focused upon a specific field of interest.

PROGRESS IN SUPERSONIC ERA REQUIRES INTENSE EFFORT

Today, fighter aircraft capable of supersonic speed in combat, in increasing number, are being delivered to our Air Force and Navy. Their faster-than-sound performance marks the beginning of a new era in aeronautics.

Today's fighters, the first to operate in the faster-than-sound range, are enormously improved over those of yesterday. But the time certainly will come when the best of the current fighter designs will have been outmoded.

How rapid will be America's progress in this supersonic era depends on many factors. The wings of future aircraft will be so designed as to provide greater *lift* at the same time their *drag* is being reduced. In fact, as more is learned about aerodynamic behavior of airplanes flying at the speeds envisioned, the matter of form or shape of the entire structure, the wings, the engine pods, the fuselage, becomes increasingly critical. New structures, new methods of fabrication may be required as speeds increase and the problems of aerodynamic heating become more urgent. And, of course, the engines to propel tomorrow's aircraft must be much more efficient and much lighter per pound of thrust than today's best.

What needs to be learned, what remains to be done, represents a very formidable task. The rate of accomplishment will depend, very largely, upon how skilled and how intensive is the attack on the total problem. Following are brief discussions of two facets of the total problem, which may bring into sharper focus the difficulties remaining, as well as the urgency with which solutions must be sought.

Consideration of the first of these facets is an outline of the very large effort made over a period of more than 10 years to learn how to fly in the transonic area—in that speed range where the laws, or rules, governing subsonic and supersonic airflow interact in such ways as to defy theoretical assessment. Here the emphasis is upon substantial progress made in a field where much further work will be required. This is an account of achievement in an area of aeronautical science where further research is required so that the mass of information gained experimentally will become adequate for the design of tomorrow's airplanes.

The second discussion has a different emphasis. It has to do with propulsion problems, but instead of reviewing past work, concentrates upon the engines for tomorrow's faster airplanes and missiles. More power, in quantities which even today seem fantastically large, appears to be requisite. The intercontinental character of the mission to which some of our airplanes and missiles may be assigned makes equally imperative the development of new fuels, and perhaps even radically new types of powerplants. In addition to providing sufficient power to propel airplanes and missiles at the desired velocities, the engine manufacturers must learn how to "build in" the fuel economy which will permit attainment of range as well.

DEVELOPMENT OF TRANSONIC RESEARCH TECHNIQUES

Early in World War II, successful development of the turbojet and rocket engines afforded power in sufficient quantities to enable aircraft speeds substantially greater than the 400-mph-plus performance of the best current fighters. The revolution in powerplants offered, for the first time, possibilities of supersonic flight.

There was, however, a very large requirement which had to be satisfied before the speed gains implicit in the new, more powerful engines could be attained. That requirement was acquisition of adequate aerodynamic information about the uncharted speed region, the transonic, through which planes powered by the new engines would have to pass to reach supersonic speeds.

In the United States, aeronautical research scientists recognized this vital need, and also that existing knowledge about transonic air flow was pitifully small. They were aware that earlier efforts to develop a body of useful transonic theory had failed—as, indeed, is largely the case today. Consequently, resort to experimentation would have to be made. But from their earlier work, they had learned that the usual experimental techniques would be inadequate for the task ahead. The principal tool of aerodynamic research, the wind tunnel, unfortunately was subject to "choking" phenomena at speeds near sonic velocity, and its use for transonic experimentation was impossible.

With the United States committed to the task of winning the war with types of already developed aircraft which could be manufactured most quickly, the talents and energies of aeronautical scientists were largely directed to work calculated to improve existing aircraft types. It was apparent, however, that a determined attack should be immediately launched on the problem of developing methods for conducting transonic research. As rapidly as could be managed, the NACA's effort in this direction was increased until the attack had been broadened to include all approaches which offered promise.

Two early proposals, after having been passed by, were successfully developed later, to provide the means for obtaining transonic aerodynamic information of great value. One of these called for the dropping of specially instrumented aerodynamic bodies, such as models of aircraft or wing-fuselage combinations, from aircraft flying at high altitude. Preliminary studies showed that, during a drop from about 35,000 feet, the speed of representative free-fall models could be expected to accelerate from Mach numbers of 0.5 to 1.3 ($M=1$ equals the speed of sound). The problem of devising adequate instrumentation, to record what happened as the model passed through the transonic range, was so difficult that this approach was abandoned until late in 1943 when the advances made in radar and radio-telemetering equipment warranted renewal of the falling-body work, with Great Britain's Royal Aeronautical Establishment joining forces with the NACA.

The spectacular performance accomplishments of the specially designed research airplanes, to speeds of 1,650 mph and heights of more than 80,000 feet, have obscured the fact that their prime justification was as tools to be used in developing necessary transonic information. As early as 1943, the idea of the research airplane was suggested. Propelled by the most powerful engine available and freighted with a mass of recording equipment, it could be flown at great altitudes where the density of air, and so the loads imposed on the structure, would be low.

It was late in 1944, however, when final decisions were made to undertake design and construction of the first two of a series of research aircraft. It was 1947 before the rocket-powered Bell X-1, sponsored by the Air Force, and the turbojet-powered Douglas D-558-I, sponsored by the Navy, completed demonstration of their minimum performance guarantees and were put to the task of investigating transonic problems.

The story of the research airplane has been recounted so completely elsewhere as to make unnecessary a detailed report here. Suffice it to say that from this continuing cooperative effort, in which the military services, the aircraft industry, and the NACA work as equal partners, have come returns in the form of transonic and supersonic information which have repaid many times the required investment.

In July 1944, NACA scientists first employed still another technique to gather transonic aerodynamic information. They used small, metal models installed on the wing of a fighter airplane to take advantage of the supersonic flow that occurs locally above a curved surface moving at high speed. (For example, the local Mach number increased smoothly over the curved upper surface of the wing, from about 0.40 to about 1.15 as the flight Mach number of an airplane was accelerated from about 0.30 to 0.76.) In this manner, lift, drag and other aerodynamic characteristics of straight-wing models at transonic speeds were determined. In May 1945, first tests of a sweptback wing model were made through the sonic range for correlation with theories on the effect of sweepback which R. T. Jones of the NACA had developed.

The wing-flow technique was used extensively, being refined by the improvement of local flow conditions in the test region, and by the improvement of data-recording equipment. The method was also employed in the study of stability and trim characteristics of airplane shapes in the transonic speed range.

The same principle, of supersonic flow occurring locally above a curved surface moving at high subsonic speeds, was transferred to high-speed subsonic wind tunnels by the Lockheed Aircraft Corporation. A "bump" was positioned in the tunnel test section to obtain the desired transonic-range air flows. This development gave somewhat more freedom in shaping the accelerating surface and had the obvious advantage of avoiding flight instrumentation problems.

Both the wing-flow and the "bump" techniques were limited to use of small models which was unfortunate because scale effects, especially at transonic speeds, makes difficult interpretation of the test results for use in the design of airplanes and missiles. The fact that with these techniques it was difficult to attain satisfactorily smooth transonic air flows when the test models were a highly swept wing, or an airplane or missile half-body model also spurred the continuing effort to develop more useful transonic research techniques.

Use of rocket-propelled models fired from the ground followed by about a year the first work with free-falling bodies. The instrumentation developed for the latter technique was available, so that it was possible to start productive experiments quickly with models built around the types of solid-fuel rockets which could be readily obtained. Emphasis was placed on studies of control effectiveness, as a prerequisite for automatic stabilization, as well as for general application to all aircraft designs that must fly in, or through, the transonic range. At the start of this work, emphasis was placed also on simple drag studies to help in defining the airplane shapes likely to be of greatest interest.

In the years since, the rocket-propelled model technique has been greatly improved and has become one of the most valuable tools for transonic research. By addition of powerful booster rockets, models of this kind have also been used in study of aerodynamic problems at supersonic speeds up to Mach numbers of 5, and beyond. The fact that high speeds are attained at relatively low altitude, where the air is dense, enables getting results of large-scale value which are readily usable for plane or missile design.

The urgency for development of a technique, whereby transonic experimentation could be carried on under the closely controlled conditions possible in the laboratory, was not lessened by development of the free-fall and rocket-methods. Rather, it became the greater as results from these other methods defined fundamental problems of fluid mechanics that would have to be studied in great detail for the design of satisfactory aircraft. Accordingly, considerable effort was continued in this direction.

In 1949, announcement was made by the NACA that an "annular throat" transonic wind tunnel had been put into use at its Langley Laboratory. In this device, small models were mounted on the rim of a rotor about five feet in diameter. The rotor was accelerated to very high speeds, while a fan moved the wake of the model downstream.

Although it was found that satisfactory two-dimensional airfoil data of limited scope could be obtained with this apparatus, it was obvious that as soon as a better and more versatile research technique could be devised, the annular throat tunnel would be discarded. It is to be doubted whether the NACA would have given the device the importance implied by the public announcement except that it served to explain away rumors that a successful transonic wind tunnel had been developed.

Such, in fact, was the case. The NACA had, for the first time, developed a wind tunnel design which would permit large-scale aerodynamic research to be conducted, in the laboratory, throughout the full transonic speed range. At the same time that announcement of the annular throat wind tunnel was being made, construction was being rushed towards completion of the first of the large transonic wind tunnels.

By late 1950, the Langley Laboratory placed its first transonic wind tunnel in useful operation, and within a few months, a second large transonic tunnel was completed. Since then, the NACA has begun intensive use of other transonic wind tunnels at the Langley and Ames Laboratories.

Indication of the importance of the successful development of the transonic wind tunnel may be found in the extraordinary efforts taken by the NACA to maintain, in a classified status, technical information forthcoming from use of the new research tool, as well as details about the new type of tunnel.

Except for a brief announcement, in mid-1951, that a "ventilated throat" type of transonic wind tunnel has been developed, the NACA maintained silence until the spring of 1954. By then, more than 3 years had passed, and it was learned that, during the intervening years, transonic wind tunnels had been devised elsewhere.

Attainment of transonic wind tunnels was no easy task, quickly accomplished. The NACA's work to develop a wind tunnel design suitable for transonic research was initiated prior to 1942. At that time it was felt sufficient progress had been made to warrant repowering existing high-speed subsonic tunnels with larger electric motors, against the future day when transonic test sections could be installed. By 1946, the intensive effort had succeeded to a point where design of a "transonic throat," for use in the first of the NACA's large high-speed subsonic wind tunnels, was commenced.

Millions of dollars, and the future value of one of the NACA's most valuable wind tunnels, were involved in this calculated gamble. The prize at stake was a vital time advantage for the United States in the world race to learn how best to fly in, and through, the transonic range.

How well this coordinated effort to develop adequate transonic research techniques has succeeded can best be determined by assessing the quality of today's military aircraft. Each incorporates information gathered from years of transonic experimentation. To an even greater degree, tomorrow's airplanes and missiles will benefit from this work.

Some of the technical aspects of the transonic wind tunnel are considered in this 1954 Annual Report of the NACA, on page 8. They will be found in the section on Fluid Mechanics.

POWER PLANTS

Hardly a decade ago, the prime goal in aeronautics was achievement of supersonic flight. What made the goal seem attainable then was the successful development of the turbojet engine, with its great improvement potential, which offered the possibility of providing the large amounts of power required for the faster-than-sound speeds desired.

In the years since, great progress has been made in learning how to fly supersonically, assuming sufficient power was provided. The straight-wing Bell X–1 was the first to reach faster-than-sound speeds, on October 14, 1947. The swept-wing Douglas D–558–II was the first to fly twice the speed of sound, on November 20, 1953. The Bell X–1–A reached 2½ times the speed of sound, 1,650 mph, on December 12, 1953.

These specially designed research airplanes all were powered by Reaction Motors' 6,000-lb. thrust rocket engines. (At 375 mph, 1 lb. of thrust is equal to 1 hp.; at 750 mph, 1 lb. of thrust equals 2 hp., and so on.) The amount of fuel these airplanes could carry provided full power for a matter of four minutes, or less. During each of the above-mentioned flights, speed was increasing when fuel was exhausted.

America's first production turbojet engine was the General Electric 1,600-lb. thrust I–16, based on the British Whittle design. The very great amounts of research and development effort concentrated upon turbojet improvement since then has resulted in an almost incredible technical advance; today's largest production engines provide six times the thrust of the I–16.

Spectacular as has been turbojet progress to date, equally large gains must yet be made to enable the deep penetration into the supersonic range by tactical aircraft required for military purposes. What are needed, of course, are engines to propel fighters at 2 or even 3 times the speed of sound, and to power bombers at supersonic speed throughout their long-range missions.

The progress made in transonic and supersonic aerodynamics in recent years has been very large. By incorporating recently acquired information in the design of new fighters, it has been possible to increase maximum performance from high subsonic to low supersonic speeds. And yet, it must be said that, on the basis of current aerodynamic information, the engine largely determines the kind of supersonic airplane performance that can be achieved.

Military demands for improved aircraft performance—speeds into the supersonic range, as well as operation at much higher altitudes over longer distances—have increased greatly the percentage of airplane total weight required for the engine and fuel weight. Where, in World War II, the percentage for a fighter could be as low as 20, today it may be as high as 60. Powerplant requirements vary, of course, as do airplane requirements, but the ratios remain remarkably constant.

Particularly difficult requirements are imposed on the propulsion system of a supersonic fighter. The fighter must attain the fastest speed possible; it must maneuver at high speed, and it must climb rapidly to altitude. The engine needed to provide such capabilities must produce a great amount of thrust and yet be small and light. In the case of the long-range supersonic bomber, the most critical requirement may be to obtain low specific fuel consumption, even if engine weight is greater as a consequence.

In simplest terms, tomorrow's turbojet engine may be expected to "handle" substantially more pounds of air, per square foot of frontal area, than current powerplants. Similarly, turbine-inlet temperatures will be markedly higher. The problems involved are extremely difficult and affect every part of the engine from inlet to tailpipe. The gains resulting from solution of these problems, expressed in terms of increased engine thrust per pound of engine weight, promise to rival the spectacular technical advances already made.

Because the engine of tomorrow must be capable of reasonably efficient operation at subsonic and transonic as well as supersonic speed, it may be necessary to design the inlet to enable varying the amount of air taken in. Inlet design, also, must be improved to minimize the pressure losses which occur when air is being taken in at supersonic speed.

The prime objectives of current compressor research are to increase not only the air flow capacity, but also the compression in each stage. In other words, it is hoped the compressor, which currently has as many as 13 stages, can be made shorter and lighter, while its capacity is being increased.

If tomorrow's turbojet is to assimilate a greater mass of air per square foot of frontal area, and if the compressor is to be capable of handling this greater volume of air, then the combustors (where the fuel is burned) must also be improved so that the flame will not blow out even though the velocity of the air passing through is as much as twice that found in currently used combustors. The burning process must produce higher gas temperatures than at present, to assure the desired higher turbine inlet temperatures.

The turbine in tomorrow's engine presents some of the most difficult problems. The higher gas flow desired will aggravate the already difficult problem of turbine stresses. Raising the turbine gas temperature by any significant amount may require blade, and perhaps also disk, cooling, as well as use of better materials.

The problems mentioned so far are but the most obvious of many. Tomorrow's engine most certainly will require the use of improved materials and of advanced design practice. Much work will be required to develop bearings which can withstand the heavier loads imposed under higher operating temperatures. New types of lubricants will have to be developed. The search will continue for more powerful fuels.

There are, of course, other powerplants which merit intensive study and development for possible application in satisfying requirements beyond the capabilities of the turbojet engine, even assuming fullest exploitation of the latter's potential. For example, it may be desirable to send a long-range missile along a ballistic-type trajectory. The course it traveled would quickly reach altitudes so high as to avoid the perils of aerodynamic heating. But at such heights, the air-breathing turbojet engine could not function. Instead, a rocket engine which carried an oxygen supply, as well as fuel, might be employed. The German V–2 missile used an alcohol-oxygen combination. Efforts to increase rocket range possibilities involve study of, among other possibilities, fuel-oxidant combinations with more than twice the specific impulse of the V–2 combination.

Beyond all this lies the possibility that we can successfully apply nuclear energy to supersonic aircraft propulsion. The power required to propel an airplane at supersonic speeds is very large, as much as five times the amount needed to sustain the same airplane at subsonic speeds. It has become increasingly apparent that if supersonic aircraft are to possess the ultimate in long-range capabilities, a way must be found to breach the fundamental limits inherent in engines using chemical fuels.

The performance capabilities to be realized from harnessing nuclear energy for aircraft propulsion would be nonstop supersonic flight to any point on the face of the earth, and return. Both experimental and analytical investigations of the many extremely difficult problems of nuclear aircraft engines are necessary Often problems are so complex as to require development of novel facilities which can be used to break them down into parts small enough for piecemeal study and solution.

With so large a gain the goal, industry, the Atomic Energy Commission, the military services, and the NACA are participating in vigorous, sustained attacks on the formidable technical problems that must be solved. Our national security requires that the research and development of nuclear powerplants for aircraft be carried forward with unceasing effort.

Some of the technical aspects of the NACA's coordinated research attack on powerplant problems are considered in this 1954 annual report, beginning on page 28.

AERODYNAMICS

During the past year the NACA Laboratories have continued to conduct theoretical and experimental research in the field of aerodynamics in order to arrive at satisfactory solutions to the fundamental problems of high-speed flight. Exploratory research on such basic fluid-mechanics problems as boundary-layer transition and heat transfer has been directed at establishing a foundation for the design of future aircraft of increased performance. In addition, important performance and stability and control problems arising in the course of the development of tactical military aircraft have necessitated applied research directed at supplying design data on problems of immediate interest. Studies of models of specific aircraft designs have also served to instigate general research studies by pointing out problems on which basic information is needed to obtain logical and practical solutions to future design problems.

Efforts have been continued to extend analytical and theoretical techniques for predicting aerodynamic characteristics of wings, bodies, and aircraft configurations and to verify recently derived theoretical techniques by comparing estimated and experimental results. Experimental investigations have been undertaken in the various NACA wind tunnels through the transsonic, supersonic, and into the hypersonic speed ranges. through the use of rocket-powered free-flight models and through flight research on full-scale aircraft. Because of the need for recording and analyzing large quantities of experimental data, increased use has been made of electronic digital computing equipment. Special analog computing equipment has also been employed to study the dynamics of simulated airplanes and missiles equipped with automatic control and guidance systems.

The Committee on Aerodynamics and its subcommittees on Fluid Mechanics, High-Speed Aerodynamics, Stability and Control, Internal Flow, Propellers for Aircraft, Seaplanes, and Helicopters have continued to give guidance to the NACA in planning and conducting its aerodynamic research programs. A technical conference on the aerodynamics of high-speed aircraft was held at the Ames Laboratory during the past year to assist in the early dissemination of technical information to the Armed Services and their contractors. Because of the evident effectiveness of technical conferences, such as this one, in the transmission of new research information, these conferences have become an established and important part of NACA reporting procedure.

The following paragraphs briefly describe many of the unclassified studies undertaken by the NACA during the past year in the field of aerodynamics.

FLUID MECHANICS

Boundary-Layer Transition

To predict accurately the drag of high-speed aircraft, it is necessary to know approximately where on the aircraft surfaces the laminar boundary layer will become unstable and undergo transition to the turbulent form. Recent experiments in NACA supersonic wind tunnels, reported in Technical Note 3020, show that the location of the transition point varies quite drastically from one tunnel to another. This result indicates that caution is required in the application of tunnel transition data to aircraft design and indicates that considerable further research is required to establish the factors affecting transition.

The transition-point location on models in wind tunnels has also been shown to fluctuate rapidly with time. In an attempt to understand the nature of the fluctuations, a statistical study was made on a 10° cone at Mach number 3.12. Results of this study, reported in Technical Note 3100, indicated that the fluctuations are of larger amplitude when the turbulence level in the tunnel air stream is high and that the fluctuations appear to have nearly a random distribution.

There exists little systematic information to show the effects of pressure gradient on transition at supersonic speeds. An investigation was conducted in the Langley 4- by 4-foot supersonic pressure tunnel at a Mach number of 1.6 on three bodies of revolution of fineness ratio 12.2. These bodies consisted of an ogive-cylinder, a cone-cylinder, and a parabolic body of revolution with correspondingly different pressure gradients. The results, which are presented in Technical Note 3193, indicated marked differences in the transition Reynolds numbers for these three bodies. For transition at the body base, the highest transition Reynolds number was obtained on a parabolic body with a long run of moderate, favorable pressure gradient. When transition occurred on the forward part of the body, the transition Reynolds number was highest for the ogive-cylinder and cone-cylinder bodies which have the strongest favorable gradients.

An experimental and theoretical study of the effects of heat transfer on boundary-layer transition has been made at the Langley Laboratory. Present theory indicates the importance of heat transfer on laminar boundary-layer stability but is inadequate for predicting quantitative values. Tests have been made at a Mach number of 1.6 of a parabolic body of revolution in the 4- by 4-foot supersonic pressure tunnel (which is unusually free of extraneous disturbances). For zero heat transfer, this body had a transition Reynolds number at the base of 11.5×10^6. Preliminary results of the investigation with heat transfer are reported in Technical Note 3165. By cooling the model an average of about 50° F, a transition Reynolds number of 20×10^6 was obtained. Heating the model an average of about 12° F decreased the transition Reynolds number from 11.5×10^6 to about 8×10^6. In Technical Note 3166 the investigation was extended to higher Reynolds numbers and greater amounts of heating. The highest transition Reynolds number obtained with cooling was 28.5×10^6. At this Reynolds number, boundary-layer transition was apparently caused by model surface-roughness effects.

In Technical Note 3103 the effect of pressure gradient on the amount of cooling required to stabilize the laminar boundary layer was investigated analytically. It was found that less cooling is required when the pressure decreases in the stream direction and more cooling is required if the pressure increases. This analysis was based on the results of Technical Note 3028, which develops an exact theory for laminar boundary layers with small pressure gradient at supersonic speeds.

The effect of cooling on boundary-layer transition in the entrance of a smooth, round tube was investigated experimentally at the Massachusetts Institute of Technology under the sponsorship of the NACA. Tests were made at diameter Reynolds numbers varying from 50,000 to 106,000. The levels of disturbance were such as to yield Reynolds numbers, based on the length to the start of transition, ranging from 500,000 to 1,800,000. Temperature differences between the wall and the free stream up to 270° F, were applied, but no significant effect of cooling on the point of transition was found, which is in contrast to the case of external flows about bodies of revolution. This work is presented in Technical Note 3048.

Skin Friction and Heat Transfer

The values of average laminar skin-friction coefficients measured in supersonic flows have in most investigations exceeded the theoretical predictions. This discrepancy has been indicated to be primarily the result of probe interference. An experimental investigation conducted in the Langley 9-inch supersonic tunnel at a Mach number of 2.41 and reported in Technical Note 3122 showed that, in the absence of probe effects, the experimental boundary-layer profiles, skin-friction coefficients, and growth of the laminar boundary layer were in good agreement with the theoretical predictions.

The subsonic effects of surface roughness on transition and skin-friction drag are relatively well known; however, little information exists on the effects of surface roughness at supersonic speeds. The problem is of importance in estimating airplane drag at supersonic speeds and in determining the allowable roughness of surfaces. An investigation was made in the Langley 4- by 4-foot supersonic pressure tunnel at a Mach number of 1.6 on a series of ogive-cylinders having a root-mean-square surface-roughness range from 23 to 480 microinches and over a Reynolds number range from 2.5×10^6 to 37×10^6. The results of this investigation are presented in Technical Note 3230. The investigation indicated that the effects of surface roughness at supersonic speeds are generally similar to those found at subsonic speeds. Both the allowable roughness height for a turbulent boundary layer and the variation with Reynolds number of the increment in skin-friction drag due to roughness are in good agreement with the low-speed data of Nikuradse. The allowable roughness height is nearly independent of model length and dependent primarily upon Reynolds number per foot. The allowable surface roughness, therefore, will decrease as the speed of aircraft is increased. There was found to be little effect of surface roughness on surface temperature recovery factors for laminar or turbulent boundary layers.

The well-known Reynolds analogy between heat transfer and skin friction in turbulent flow, as modified by Colburn for the effect of the laminar sublayer, has long been known to be a valid method of estimating the heat transfer in the case of subsonic turbulent pipe flow. Until very recently, however, there has not been evidence to show whether this analogy was also valid for supersonic turbulent boundary layers. A theoretical study of the problem has been given in Technical Note 2917 and led to the conclusion that in spite of the major differences between subsonic and supersonic boundary-layer flows, the modified Reynolds analogy would still apply. To see what the existing experimental data would show in this connection, the study reported in Technical Note 3284 was undertaken. It was observed that the heat-transfer data were sometimes confused by the presence of mixed laminar and turbulent flows and in some cases could not be used for this reason. In other cases, correction could be made for delayed transition to turbulent flow by estimating the effective origin of the turbulent part of the boundary layer. When corrected, the data from two free-flight and four wind-tunnel experiments were compared to corresponding but separately obtained skin-friction data, taking into account the effect of wall temperature, and were found to support the modified Reynolds analogy. It therefore appears that this is a proper basis for estimating the heat transfer of turbulent boundary layers in the moderate supersonic speed range.

A rapid and sufficiently accurate method, for most practical purposes, of determining laminar-boundary-layer characteristics in flow with a given free-stream Mach number and given velocity distribution at the edge of the boundary layer is presented in Technical Note 3157. The method can be applied to flow with zero pressure gradient for any (constant) Prandtl number of the order of unity and any given temperature distribution along the wall. For flow in an axial pressure gradient, the method can be applied for a Prandtl number of unity and any given uniform wall temperature. This research was carried out by the Polytechnic Institute of Brooklyn under the sponsorship of the NACA.

Local heat-transfer rates on the surface of a heated flat plate at zero incidence to an air stream flowing at Mach numbers of 1.69 and 2.27 were measured and are reported in Technical Note 3222. The Reynolds number range for both Mach numbers was 1 million to 10 million. Surface temperatures were maintained near recovery temperature. It was found that the variation of heat transfer with Mach number was in good agreement with previously reported variations of directly measured skin friction with Mach number on unheated bodies. The variation with Mach number of the average skin-friction coefficient, as determined from impact-pressure surveys, was in agreement with that from other momentum loss measurements but differed from the variation obtained from directly measured skin friction as reported by others.

The method of heat-transfer calculation given in Technical Note 3005 may be extended to calculation of heat transfer with equilibrium dissociation, provided the wall temperature is below dissociation temperatures. A numerical example for the heat transfer in the stagnation region of a blunt body shows that equilibrium dissociation probably has little effect on the actual heat-transfer rate.

Tests have been conducted to measure the heat-transfer characteristics of a hemispherical-nose body with and without drag-reduction spikes protruding from the nose. The tests were conducted over a Mach number range of from 0.2 to 5. It was found that the presence of drag-reduction spikes approximately doubled the heat transfer to the hemisphere and that almost the entire increase was confined to the forward half-area of the hemisphere. In the absence of spikes, the heat-transfer data at supersonic speed could be correlated over the entire Mach number range with the subsonic data by basing the fluid properties on conditions behind the normal shock wave. The data have been reported in Technical Note 3287.

An analysis has been made in Technical Note 3058 of the transient heat-conduction effects on three simple semi-infinite bodies with laminar boundary layers, a flat insulated plate, a conical shell, and a slender solid cone. The bodies were assumed to have constant initial temperatures and, at zero time, to begin to move at a constant speed and zero angle of attack through a homogeneous atmosphere. The heat input was taken as that through a laminar boundary layer and radiation heat transfers and transverse temperature gradients were assumed to be zero. The appropriate heat-conduction equations were solved by an iteration method, the zero-order terms describing the situation in the limit of small time.

Although NACA boundary-layer research has placed considerable emphasis on the supersonic range, many problems of the low-speed boundary layer remain unsolved. Among these are the factors that determine when the boundary layer will separate. In Technical Note 3031, which is a preliminary report of a series devoted to detailed examination of the low-speed, turbulent boundary layer, results on skin friction measured by a heat-transfer instrument in adverse pressure gradient are compared with results of probe surveys and with predictions of an empirical theory. Values obtained with the heat-transfer instrument agreed well with theory, whereas the probe-survey method showed large disagreement near the boundary-layer separation point. This disagreement indicates that the usual boundary-layer relations between skin friction and momentum loss are not valid near separation.

Shock-Wave Boundary-Layer Interaction

The number of investigations on the problem of shock-induced boundary-layer separation has increased to the point where it has become desirable to assess the results in an integrated fashion; thus, the present status of available information relative to the prediction of shock-induced boundary-layer separation is given in Technical Note 365. Experimental results showing the effects of Reynolds number and Mach number on the separation of both laminar and turbulent boundary layers are given and compared with results obtained by available methods for predicting separation. The flow phenomena associated with separation caused by forward-facing steps, wedges, and incident shock waves are discussed. Applications of the flat-plate data to problems of separation on spoilers, diffusers, and scoop inlets are indicated for turbulent boundary layers. Analysis of the various results indicates that, although no universal value of pressure-rise coefficient which causes incipient separation of the boundary layer has been found, there is a fairly narrow band of pressure coefficients from which predictions of turbulent separation can be made with an accuracy probably sufficient for engineering purposes. Further, it was indicated that there is a dependency of the pressure coefficient for separation on Reynolds number for the laminar boundary layer but little, if any, dependency on Reynolds number for the turbulent boundary layer. There is a dependency of this pressure coefficient on Mach number for both laminar and turbulent boundary layers. Caution should be exercised in attempting to predict the separation or loading on configurations which differ considerably from those for which experimental data are available.

A flight investigation of the behavior of laminar and turbulent boundary layers passing through shock waves on a wing at chord Reynolds numbers up to 26 x 10^6 is reported in Technical Note 3056. The results of the investigation indicated that at these Reynolds numbers the abrupt increase in momentum thickness and displacement thickness in the region of the pressure rise associated with the shock wave was significantly less with laminar than with turbulent flow ahead of the shock. In other words, there appeared to be a beneficial effect on the flow downstream of the shock wave, with laminar flow ahead of the shock as compared to the conditions with turbulent flow ahead of the shock, in that the downstream momentum thickness and shape parameter were about 50 percent less for the laminar than for the turbulent case. In other investigations made at low Reynolds numbers, the changes in the boundary layer passing through shock were found to be much greater for the laminar than for the turbulent case.

Turbulence

Hot-wire anemometry has been for many years the principal experimental tool in the study of turbulence. In research conducted at the Ames Laboratory, average heat transfer was measured on heated wires, such as are used in hot-wire anemometry, for Mach numbers covering the transonic range, extending from a subsonic value of 0.4 to a supersonic value of 1.4. Reynolds numbers based on wire diameter were varied independently from 18 to 144. The purpose of this investigation was to bridge the gap between existing data for subsonic and supersonic speeds. It was found that the dimensionless heat loss, or Nusselt number, for constant temperature loading could be correlated as a single-valued function of the Knudsen number throughout the range of the tests. This signifies that the average heat loss of hot wires is insensitive to velocity changes and responds, instead, to changes in state of the gas. The investigation is reported in the proceedings of the 1954 Heat Transfer and Fluid Mechanics Institute.[1]

An analytical investigation has been made of the effect of the rate of increase of turbulent kinetic energy in the stream direction on the development of turbulent boundary layers in adverse pressure gradients. A refinement of turbulent-boundary-layer theory is reported in Technical Note 3049 in which a general integral form of the boundary-layer equation which includes the Reynolds normal-stress term is derived. From this general form, two special equations—namely, the modified momentum equation and the modified kinetic-energy equation—are obtained. These modified equations include the effect of the Reynolds normal stress in the stream direction.

It is shown in Technical Note 3116 that the correlation of fluctuating static pressure in an incompressible and homogeneous turbulence with any fluctuating quantity in the flow field can be expressed in terms of the correlation of the same quantity with two or more components of the velocity. The correlations of pressure with itself and of pressure with two velocity components were investigated in detail for the case of isotropic turbulence. This investigation was carried out by the Johns Hopkins University under the sponsorship of the NACA.

An experimental and theoretical study has been made by the Johns Hopkins University under the sponsorship of the NACA of the instantaneously sharp and irregular front which separates turbulent fluid from contiguous "nonturbulent" fluid at a free-stream boundary. This study is reported in Technical Note 3133. The overall behavior of the front is described statistically, in terms of its wrinkle-amplitude growth and its lateral propagation relative to the fluid, as functions of downstream coordinate. It was proposed and justified that the front actually consists of a very thin fluid layer in which direct viscous forces transmit mean and fluctuating vorticity to previously nonturbulent fluid. Outside this "laminar superlayer" there is, presumably, a field of irrotational velocity fluctuations (the "nonturbulent" flow) with constant mean velocity. Theoretical analysis based on this physical picture gave results which are in plausible agreement with experimental results for three turbulent shear flows.

Methods of measuring the probability distributions and mean values of random functions as encountered in turbulence research were studied at the California Institute of Technology under the sponsorship of the NACA and are reported in Technical Note 3037. Applications to the measurement of probability distributions of the axial velocity fluctuation and its derivative in isotropic turbulence were shown. The assumption of independent probabilities of these parameters was investigated and the results indicate that the assumption is satisfied within a few percent.

Preliminary measurements have been made at the Johns Hopkins University under NACA sponsorship of velocity and temperature fluctuations in the flow behind a heated grid in a uniform air stream. Temperature correlation showed a reasonable degree of isotropy, and the temperature fluctuations died out at large distances more slowly than the turbulence, as has been predicted theoretically under some strongly simplifying postulates.

[1] See Stine paper listed on p. 66.

Gas Dynamics

Technical Note 3069 deals with iterative procedures as applied to problems in fluid flow. The purpose of this study was to show some of the possible mathematical troubles that may occur in the iterative procedures. First, it was shown that the example of incompressible flow past a sinusoidal wall of finite amplitude should be treated in the plane of the velocity potential and the stream function rather than in the physical flow plane. Then, two contrasting iterative procedures were utilized for the solution of this particular problem. One was the well-known small-disturbance method in which the physical-plane coordinates are determined in the form of Fourier series whose coefficients are analytically developed as series in ascending powers of the amplitude. In general, this method precludes any discussion of convergence, the tacit assumption being that no mathematical limitation intervenes before the solution ceases to be valid because of some physical reason. The other procedure was to state the problem in the form of an integral equation whose solution can be found by a process of successive approximations. The convergence of this method can usually be judged when the difference between any two successive approximations is deemed negligible. An included numerical example served to emphasize the superiority of the integral-equation approach over the small-disturbance method.

The equations of two-dimensional compressible flow were treated in Technical Note 3229 according to the Prandtl-Busemann small-disturbance method. In contrast to the usual procedure, the independent variables were the compressible velocity potential and stream function and the dependent variables were the rectangular Cartesian coordinates in the plane of flow. The six first-order differential equations corresponding to the first three iteration steps were put into complex-vector form. The particular integrals of the resulting set of three equations were then obtained directly. As an example, the general results of the analysis were applied to the case of subsonic compressible flow past a sinusoidal wall of small amplitude.

At very high flight speeds, temperatures high enough to cause ionization of air will be realized. The aerodynamic effects which arise are extremely difficult to calculate for the diatomic constituents of air; thus, it is necessary to carry out exploratory research on monatomic gases. The work of other investigators has indicated that simplified calculations may be used to determine the flow properties of strong shock waves in monatomic argon gas because the electronic excitation energy is small compared to the total internal energy. These investigators have used three simplified calculations for flow velocities up to a Mach number normal to shock of 20 without large error. In an investigation reported in Technical Note 3091, it was found, by comparison with more exact calculations, that the effect of electronic excitation is much more important for xenon than argon. The calculations indicated that the difference between excitation and nonexcitation increased with increased flow speed. The flow properties which were calculated by considering the effects of electronic excitation and ionization were compared with ideal-flow properties. Differences occurred between these two flows when ionization was present. The differences increased as the ionization increased at larger flow speeds. The calculations showed that the actual temperature and stream velocity behind the shock wave were less than in the ideal case, whereas the pressure ratio and specific-volume ratio across the shock are greater. The real stream velocity behind the shock reached a maximum value as the speed increased and then decreased. The results of the calculations were qualitatively compared with available experimental results. At shock speeds above a Mach number normal to the shock of approximately 8, the experimental results showed a visible glow in the shock. If the assumption is made that the intensity of the glow is approximately proportional to the percent of ionization, the experimental results appear to be in qualitative agreement with the results of calculations.

Simulation of flight conditions at extremely high speeds requires the use of air streams with high stagnation temperatures. One scheme for attaining the combination of high temperature and Mach number by means of shock tubes has been reported. This scheme consisted of a modification of existing shock tubes by using two diaphragms, rather than one, to produce very high Mach numbers at temperatures high enough to simulate flight conditions.

One of the important aspects in the aerodynamics of moving shock waves is the decrease in strength or attenuation of a wave as it moves along a surface. Much work has previously been done on waves moving over smooth surfaces, but little work has been done for the case of very rough surfaces. Accordingly, a study was undertaken to measure the attenuation of plane shock waves moving over rough walls in a shock tube. Measurements of the boundary-layer characteristics, including the thickness and velocity distribution behind the shock, have been made with the aid of new optical techniques which provide direct information on the local boundary-layer conditions at the rough walls. The shock speeds and shock pressure ratios were determined for the shock tube with both smooth walls and rough walls lining all four sides of the rectangular tube. A simplified theory based on Von Kármán's expression for skin-friction coefficient for flow over rough walls, along with a wave-model concept and extensions to include time effects, was derived. The results showed that, although agreement of boundary-layer measurements was good for all shock strengths and although agreement of shock-attenuation measurements with theory was good for all the shock strengths in the smooth-wall case, the agreement of attenuation measurements with theory for the stronger shocks was not so good for the rough-wall case.

Transonic Wind Tunnels

Earlier in this annual report, events which led to the development of transonic wind tunnels were discussed. As pointed out, the idea was conceived and the initial small-scale experiments were carried out at the Langley Laboratory about 8 years ago. This work was done by John Stack, assistant director of the Langley Laboratory, and his associates. Some of the theoretical work and some of the earlier experimental work done in small-scale facilities has now been declassified and is discussed briefly in the following paragraphs. In only a few cases have the available data been published in the form of Technical Notes.

The first account of the initial theoretical and experimental work was written in 1948 by Ray H. Wright and Vernon G. Ward of the Langley Laboratory staff. An approximate subsonic theory was developed for the blockage interference in circular wind tunnels with walls slotted in the direction of the flow. This theory indicated the possibility of obtaining zero blockage interference. Tests in a small-scale circular slotted tunnel based on the theory confirmed the theoretical predictions. Furthermore, the slotted tunnel could be operated at supersonic speeds merely by increasing the power input without any manipulations of the tunnel geometry and the supersonic Mach number produced could be varied in a continuous manner by varying the power. The phenomenon of choking, characteristic of closed tunnels, did not occur in the slotted tunnel. In these same experiments, a comparison of pressure measurements made on a practical-size nonlifting model, in the slotted tunnel, with measurements obtained on the same model, in a much larger closed tunnel, in which the interference effects were negligible, showed good agreement at subsonic Mach numbers not greatly above the critical speed and fair agreement over most of the model surface at stream Mach numbers up to 1.1.

Several later investigations have been conducted to determine the effects of tunnel-boundary interference due to slotted walls on models being tested in transonic tunnels.

A theoretical study was made for the lifting case. The results of calculating the interference for several typical slotted test sections showed that lift interference could be eliminated with some slot arrangements, that the lift interference was about 0.3 to 0.6 of that obtained in an open tunnel of similar dimensions, and that much smaller openings were required for zero lift interference than for zero blockage interference.

The problems of both lift and blockage interference have been treated mathematically from the standpoint of a homogeneous boundary, with the slot effect uniformly distributed over the surface of the boundary. With this approximation, it is possible to express the interference of multislotted tunnels as a function of a single wall-restriction parameter which combines the effects of two physical variables, the ratio of open to total slotted wall perimeter and the number of slots. A considerable simplification in the calculation and presentation of slotted-tunnel interference has resulted. Numerical results for lift and blockage interference for circular, rectangular, and two-dimensional slotted tunnels, as functions of the wall-restriction parameter, have been obtained.

Experiments with a nonlifting symmetrical wedge airfoil which blocked 8.9 percent of the tunnel cross section showed that the wedge choked a conventional, closed-throat tunnel at a Mach number of 0.7, but with choking eliminated by slots, whose free area was only one-eighth of the wall area, tests could be conducted throughout the transonic range. Not only was choking eliminated, but wall interference effects for most practical purposes were negligible. To investigate the effects of slotted-wall interference on lifting surfaces, straight and swept wings were tested in a 4½- by 6¼-inch slotted tunnel and in the Langley high-speed 7- by 10-foot tunnel (solid walls). In the former, the model spans were nearly equal to the tunnel height; whereas, in the latter, the spans were approximately 5 percent of the tunnel height. The lift, drag, and pitching-moment characteristics of these wings were generally similar even though the tunnels varied in area by the ratio of 350 to 1.

At supersonic Mach numbers, the wall interference problem becomes largely one of reducing or eliminating the reflection of incident disturbances. Analytical methods based upon known porosity characteristics of certain homogeneous boundaries indicate that substantial weakening of wall-reflected disturbances is possible through the use of porous boundaries. Tests at a Mach number of 1.62 showed large reductions in strength of shocks reflected from porous boundaries.

Aside from its advantage as a means of reducing jet-boundary interference effects, the slotted or porous wall has proven to be a satisfactory device for changing Mach number in the supersonic regime where complicated mechanical devices have frequently been used to increase the stream-tube area. In the slotted- or porous-wall tunnel, this increase in Mach number is accomplished by controlled removal of air through the boundary. Experimental results were found to be consistent with calculated results for Mach numbers up to 1.16 generated in a rectangular tunnel with two walls of sintered bronze. Higher Mach numbers (up to 1.45) were obtained with slotted walls, tested in the same facility. In both of these investigations, continuous Mach number control was possible throughout the range of the tests without changes in tunnel geometry.

In connection with an experimental program being conducted in a transonic test apparatus to develop a transonic tunnel having no limitations due to wave reflection, it was found necessary to establish the porosity characteristics of perforated materials in normal and parallel flow. The results, reported in Technical Note 3085, showed a pronounced reduction in effective porosity of the perforated wall in parallel flow as stream velocity increased. In normal flow the porosity was governed primarily by the open ratio. A new definition has been given which permits the concept of a discharge coefficient to be extended to the case of parallel flow.

Other Research Equipment and Techniques

A comprehensive analysis has been made of possible advantages that can be realized in high-speed wind-tunnel research by employing a heavy gas other than air. Heavy gases previously considered are either toxic, chemically active, or have a specific-heat ratio different from air. The study presented in Technical Note 3226 was based on the idea that, by properly mixing a heavy monatomic gas with a suitable heavy polyatomic gas, a heavy gas mixture can result which has the correct specific-heat ratio and which is nontoxic, nonflammable, thermally stable, chemically inert, and comprised of commercially available components. Some of the gas mixtures investigated showed promise in providing reduced power requirements for wind tunnels; extended Mach number and Reynolds number range of compressor research facilities, firing ranges, and wind tunnels; and more proper simulation, at low wind-tunnel temperatures, of certain desired flight characteristics than can be achieved through use of air at wind-tunnel temperatures.

Three different types of pressure indicators are discussed in Technical Note 3042. Each of these indicators had a unique feature, but all were designed with an attempt to combine both high-frequency response and high resolving power into one instrument. Of the mechanical-electrical-transducer type of pressure indicator, the wire strain gage led in simplicity. The capacitance type was more versatile because it permitted the use of very high frequency carrier systems and thereby cut down the effective interference in the electronic system. The system utilizing the stretching of a barium-titanate disk produced large signals and resulted in compact design, but it could only be used for dynamic measurements when temperature variations were slight. Five different types of pressure receivers were tested. The flat-diaphragm type led the others in simplicity, the spherical-diaphragm type exceeded in dynamic performance, and the catenary-diaphragm type was least affected by temperature change. This study was made at the Massachusetts Institute of Technology under the sponsorship of the NACA.

A study has been made of the aerodynamic problems associated with ion tracer velocity measurement techniques, with particular emphasis on those aspects unique to low-density gas dynamics. The results of this study made at the University of California under NACA sponsorship were reported in Technical Note 3177. A critical survey was made of the various techniques which have been employed and specific suggestions were offered relative to the successful use of ion tracer velocity measuring methods at low densities. A description was also included of some experimental work carried out with an ion-pulse airspeed indicator developed at the Ames Aeronautical Laboratory for use in low-density wind tunnels.

An experimental investigation of impact-pressure interpretation in supersonic and subsonic rarefied air streams has been conducted by the University of California under NACA sponsorship. Measurements were made at Mach numbers from 0.1 to 0.7 and 1.7 to 3.4 and in the Reynolds number range from 2 to 800 and the results reported in Technical Note 2995. A study of the effects of impact-probe size on the accuracy of pressure measurements indicated that corrections for viscous effects are less than 1 percent for probes in supersonic flows at Reynolds numbers above 200, where the Reynolds number is based on the velocity, density, and viscosity of the free stream, the reference dimension being the outer diameter of the probe. Viscous-effect corrections were presented for interpretation of pressure measurements at lower Reynolds numbers.

HIGH-SPEED AERODYNAMICS

Airfoils and Wings

A theoretical analysis has been completed in the Langley 4- by 4-foot supersonic pressure tunnel (Technical Note 3183) in which the airfoil section required for minimum wave drag on an arrow wing of constant thickness ratio was derived by linear theory. Optimum sections were derived for each of two conditions: either a fixed chordwise location of maximum thickness or a wing of fixed volume. In the case of a fixed position of maximum thickness, the optimum sections for a supersonic leading edge are very nearly double wedges. If the maximum-thickness line is swept behind the Mach line, a cusp occurs at maximum thickness and the nose is rounded. For the case of a wing of given volume, the optimum sections for a supersonic leading edge are very nearly circular arcs.

As a byproduct of the preceding analysis, simplified techniques for the evaluation of wing drag were evolved. In Technical Note 3185, tables are presented for the rapid computation of the wave drag of arrow wings of arbitrary airfoil section at supersonic speeds. The wave drag can be calculated with the aid of these tables in about one hour on a desk-type computing machine.

A study has been made and reported in Technical Note 3040 of the application of two-dimensional data and span-loading theory for estimating the local load characteristics on a swept wing with flaps. The estimated results have been compared with results measured in large-scale wind-tunnel tests. Two-dimensional pressure distributions, when corrected for sweep, were found to agree closely with the wing pressures for most local effects, either on or off the flap. This agreement continued to the higher lift coefficients and even improved near maximum lift where the flap-induced effects became minimized. The Weissinger method was found to provide reasonably accurate span loadings for this swept-wing configuration which had a relatively highly loaded type of flap. Two-dimensional lift data, together with span-loading theory, afforded quite accurate estimates of the local nonlinear lift characteristics, including maximum lift of sections outboard of the flap, but were inadequate for inboard sections of the wing where the three-dimensional boundary-layer control exercises a dominant effect.

Comparative tests utilizing unswept wings of aspect ratio 2.7, with NACA 65-009 and 9-percent-thick circular-arc sections, showed that the NACA 65-009 section had the lower drag in the speed range investigated (Mach numbers from 0.85 to 1.22). The drag difference was largest near a Mach number of 1.0.

Bodies

It is always desirable to minimize the drag of flight bodies, especially at supersonic speeds, and minimization theories for bodies have presented a challenge to the theoretical aerodynamicist for many years. The theoretical results, presented in Technical Note 3189, show the linearized-drag integral for bodies of revolution at supersonic speeds in a double-integral form. These results are not based on slender-body approximations but reduce to the usual slender-body expression in the proper limit. The drag integral is applicable to a larger class of bodies than the usual slender-body drag expression. Results for cones indicate that the drag integral with mass-flow continuity as the boundary condition gives good first-order results in the moderate supersonic speed range. The minimum-wave-drag problem for a transition section connecting two semi-infinite cylinders was solved with the aid of a suitably chosen auxiliary condition. The source distribution, the minimum drag, the slopes at the two ends of the section, and the radius at an intermediate point were obtained in terms of elementary functions. The entire shape was obtained in an integral form amenable to numerical evaluation. The minimum-drag shape obeyed the Gothert similarity rule. The minimum-drag shape and the minimum drag were unchanged to the first order when the flow direction was reversed.

In Technical Note 2550 (reported in the Thirty-eighth Annual Report, 1952) the shapes of a family of boattail bodies of revolution were derived within the limitations of linearized slender-body theory and proposed as the shapes of boattail bodies for minimum wave drag. The results of an experimental investigation of these bodies at Mach numbers of 1.62 and 2.41, presented in Technical Note 3054, showed that the experimental wave drag of the bodies defined in Technical Note 2550 was less than the theoretical and that the difference between experiment and theory increased with increasing Mach number. During the course of this work it was noticed that body shapes having lower theoretical drags could be found from the work of Lighthill and also by the method of characteristics. Thus, although the method of Technical Note 2550 was shown to be inadequate for determining the shapes of boattail bodies for minimum drag, the body shapes given by this method have relatively low experimental drag as compared with other boattail body shapes.

Nose shapes theoretically optimized for minimum pressure drag for given fineness ratio and Mach number have been derived. The derivation was accomplished by the use of an equation for calculating pressure distributions which is valid for shapes (not too dissimilar from circular-arc ogives) for which the Mach number divided by the fineness ratio is between about 0.5 and 1. Comparisons of the profiles and computed wave drags for the derived shapes with those for the Newtonian and Von Kármán optimum shapes showed that, for a fineness ratio of 3 and Mach numbers from 1.5 to 3, the new profiles are quite similar to the Newtonian but have less drag than either the Newtonian or Von Kármán shapes. However, the pressure-drag differences between the shapes at a given Mach number are small.

A study of the effectiveness of small conical windshields mounted ahead of bluff bodies was also made. The results showed that relatively small windshields could reduce the drag of the bluff body considerably. Increasing the size of the windshield afforded additional drag reductions but at a decreasing rate. The drag reductions were largest at the maximum test Mach number (1.4). Wedges mounted ahead of an unswept wing having a blunt leading edge gave no drag reduction.

Wing-Body Combinations

For the purpose of calculating the aerodynamic forces on the fuselage, the midwing wing-fuselage combination, with a fuselage of circular cross section, can be represented by a simple system of horseshoe vortices located on the wing with images located inside the fuselage. This work was reported in Technical Note 3057. By using this simplified mathematical model, or an extension of it for nonmidwing configurations with fuselages of arbitrary cross section, a method for calculating the lift and longitudinal center of pressure on the fuselage, in the presence of the wing at subsonic speeds, is presented. In addition, the report shows how the simplified mathematical model can be used for calculating the downwash behind the wing and for calculating ~~the spanwise lift distribution on the~~ wing for midwing configurations with axisymmetric fuselages.

In Technical Note 3105, a general method was developed for obtaining lift and moments on highly swept wings and wing-body combinations. The method is based on the two-dimensional cross-flow concept which greatly simplifies the prediction of forces acting on the type of slender aircraft considered most advantageous for supersonic flight.

As a part of a correlation program sponsored by the Advisory Group for Aeronautic Research and Development, a NATO organization, zero-lift drag measurements were made in flight, of a standardized wing-body configuration, over the Mach number range from 0.8 to 1.7 at corresponding body-length Reynolds numbers of 4×10^6 and 12×10^6. The measurements were made in free flight by using a gun type of launcher and are given in Technical Note 3081. For comparison, the total drag was estimated by calculating and summing the component drags. The body pressure drag was obtained by using the second-order theory of Van Dyke (Technical Note 2744, reported in the Thirty-eighth Annual Report, 1952). The base drag was calculated according to Chapman (NACA Report 1051, reported in the Thirty-eighth Annual Report, 1952). The wing pressure drag was obtained by the method of Beane (Journal of Aeronautical Sciences, vol. 18, no. 1, Jan. 1951, pp. 7–20). The friction drag was estimated according to Van Driest. The total drag so estimated agreed well with the measured drag.

Pressure-distribution data were obtained and reported in Technical Note 3128 for a wing-body combination at Mach numbers of 1.48 and 2.00 to investigate the effects of wing-body interference. The experimental pressure-distribution and span-loading results were compared with the linear, wing-body interference theory of Technical Note 2677. For small values of angle of attack and wing-incidence angle ($\pm 3°$) it was found that the experimental pressure-distribution results compared well with the linear theory. For larger angles, however, nonlinear effects of angle caused large differences from linear theory. The nonlinear effects of angle were fairly well predicted by shock-expansion theory for the case where the wing was at an angle of incidence relative to the body. In contrast with the pressure-distribution results, the lift loading was found to be very nearly linearly dependent on angle, at least to the maximum angle tested, 6°. In general, the effects of Reynolds number and Mach number were small.

Research Equipment and Techniques

An analysis of the transient temperatures in heat exchangers for supersonic blowdown wind tunnels was made by the Langley Pilotless Aircraft Research Division and reported in Technical Note 3078. Several air conditions at the entrance to the heat exchanger were evaluated, one in which the air temperatures was constant, another in which the temperature decreased exponentially, and a third in which the temperature of the fluid decreased linearly with time. For temperatures low enough to permit the neglect of radiation effects, the results give mathematical expressions for the air and heat-exchanger temperatures as functions of time and distance along the heat exchanger. To check the validity of the analytical results experimentally, a typical heat exchanger was simulated by a bundle of tubes preheated to a constant axial temperature of 202° F. Temperatures were measured when air at atmospheric pressure and constant temperature flowed through the tubes. Excellent agreement was obtained between measured and computed temperatures.

The relatively large amounts of power required to operate wind tunnels at Mach numbers of the order of 1.0 made it desirable to investigate the possibility of using some substance other than air as a testing medium. A number of studies relating to the use of Freon-12 as a substitute medium for air in aerodynamic testing have been made at the Langley Laboratory and are reported in Technical Note 3000. Replacing air in the Langley low-turbulence pressure tunnel with Freon-12 has increased the maximum attainable test-section Mach number from approximately 0.4 to 1.2 without necessitating any increase in the tunnel power or change to the tunnel fan. Because of the fact that the ratio of specific heats is approximately 1.13 for Freon-12 as compared with 1.4 for air, some differences exist between data obtained in Freon-12 and in air. Methods for predicting aerodynamic characteristics of bodies in air from data obtained in Freon-12 provide substantial agreement in all cases for which comparative data are available.

STABILITY AND CONTROL

Static Stability

In recent years the accent on high-speed flight has led to many changes in the design of the major components of airplanes. In order to provide general information which will aid the designer of present-day airplanes, a series of investigations has been conducted in the Langley stability tunnel on models having various interchangeable parts. As part of this general program, the investigation reported in Technical Note 3063 was made to determine the effects of wing position and fuselage size on the low-speed static- and rolling-stability characteristics of models having triangular wing and vertical tail surfaces. The results show that, although the high-wing—large-fuselage configuration provided a high value of maximum lift, it had less desirable directional-stability characteristics than did the various configurations having low wings.

A knowledge of wing spanwise load distributions is of importance from both aerodynamic and structural considerations, since it provides detailed information needed to determine aerodynamic characteristics of the wing and to insure structural integrity of the airframe. Calculations, based on analytical expressions derived in Technical Note 2898 (reported in the Thirty-ninth Annual Report, 1953), have been carried out by the Langley Stability Analysis Section to determine the effects of finite sideslip at supersonic speeds on the span loadings and rolling moments of thin, sweptback, tapered wings at an angle of attack. Several combinations of Mach number and planform were investigated, each being subject to the conditions that the wing tips were parallel to the axis of wing symmetry and that the trailing edge was supersonic. Results of the investigation, presented in Technical Note 3046, include charts which give the span load distribution for values of sideslip angles up to 10°. Variations of the rolling-moment coefficient with sideslip angle and of the effective dihedral with Mach number are also presented.

Prediction of the stability characteristics of complete airplane and missile configurations requires a knowledge of the aerodynamic forces and moments acting on all of the component surfaces of the airframe. A theoretical study of triangular tail configurations in sideslip has been conducted by the Langley Stability Analysis Section, using linearized supersonic-flow theory. A series of design charts which permit rapid estimation of the forces and moments of such configurations is presented in Technical Note 3071.

Analyses of present-day airplanes have indicated that interference effects between component parts of airplanes have a significant influence on aircraft loads and stability derivatives. The interference effects between fuselages and various size vertical tails in sideslip have recently been investigated in the Langley stability tunnel. The results, reported in Technical Note 3135, indicate that the mutual interference effects are primarily dependent on the ratio of vertical-tail span to fuselage diameter at the vertical-tail location. In general, the fuselage tended to increase the effectiveness of the vertical tail, the largest percentage increase being obtained for small values of the ratio of vertical-tail span to fuselage diameter.

Another theoretical investigation of interference effects has been made using a general method for predicting the interference flow field behind plane and cruciform wings of wing-body combinations at transonic or supersonic speeds. The method was applied to the calculation of the position of the vortex wake and the estimation of downwash at chosen tail locations behind triangular-wing and cylindrical-body combinations at a Mach number of 2.0. The effects of aspect ratio, angle of attack, angle of bank, wing incidence, and ratio of body radius to wing semispan on the vortex wake behind wings were studied. It was found, as reported in Technical Note 3227, that the two-dimensional line-vortex method permitted the calculation of vortex wake motion with reasonable facility and accuracy. A calculated sample wake shape agreed quantitatively with one observed experimentally, the sample results of the line-vortex method compared very well with an available exact solution. An empirical formula was derived to estimate the number of vortices required per panel to replace the circulation distribution and allow satisfactory estimation of downwash at tail locations. It was found that the shape of the vortex wake and the ultimate number of rolled-up vortices behind a wing depend on the circulation distribution along the wing trailing edge. For the low-aspect ratio plane-wing and cylindrical-body combinations considered, it appeared that downwash at horizontal-tail locations is largely determined, except near the tail-body juncture, by the wing vortices alone, for small ratios of body radius to wing semispan, and by the body upwash alone, for large values of that ratio.

Control

One of the effects of compressibility is a reduction in the effectiveness of conventional airplane control surfaces at velocities considerably above the critical Mach number for the airfoil. As part of a general program to study this problem and to evaluate various types of controls for this speed range, an analysis has been made of the effectiveness of a 20-percent-chord, plain trailing-edge flap, on the NACA 65–210 airfoil section, utilizing section lift-coefficient data obtained at Mach numbers from 0.30 to 0.875. The analysis includes a comparison of the effectiveness of this flap with that of a spoiler and a dive-recovery flap on the same airfoil section. The results reported in Technical Note 3127 indicated that the plain trailing-edge flap employed on the 10-percent-thick airfoil section at Mach numbers as high as 0.875 retained at least 50 percent of the effectiveness exhibited at low Mach numbers. The plain trailing-edge flap, as compared to the spoiler and the dive-recovery flap, appeared to afford the most favorable characteristics as a device for controlling lift continuously throughout the range of Mach numbers from 0.30 to 0.875. At Mach numbers above those for lift divergence of the airfoil section, either a plain flap or a dive-recovery flap was effective in providing auxiliary lift.

Methods for predicting the effects of aeroelasticity have, in the past, been based on the use of beam theory for structural analysis and strip theory for aerodynamic analysis. When applied to low-aspect-ratio wings, these simple theories sometimes become inadequate and more refined analyses are necessary. In Technical Note 3067, linearized lifting-surface theory is used in conjunction with structural influence coefficients to formulate a method for analyzing the aeroelastic behavior in roll at supersonic speeds of a rectangular wing mounted on a cylindrical body. Rolling effectiveness and aileron reversal speed were computed by using a numerical solution which incorporated matrices.

In connection with the use of all-movable lifting surfaces in aircraft design, there arises the practical problem of the effects upon the aerodynamic characteristics caused by the presence of a gap between the wing panels and the fuselage. Slender wing-body theory has been applied to the calculation of these effects. The analysis is applicable to an estimation of missile characteristics wherein the longitudinal control is obtained by variable incidence wings for which gaps exist at high deflections. The results of the investigation, reported in Technical Note 3224, showed that the lift produced by wing incidence and the lift due to combined wing-body angle of attack decreased very rapidly with gap size for small gaps and approached, as an asymptote, the value of lift attributed to isolated panels of the wing. For small gaps, the theory is somewhat in doubt since viscosity was not considered.

Damping Derivatives

The development of linearized supersonic-flow theory has allowed the evaluation of most of the important stability derivatives for a variety of isolated wing shapes. Recently attention has been focused on the thin, sweptback, tapered wing with streamwise tips. Stability derivatives resulting from "steady state" motions are already available for such wings. Two important derivatives resulting from time-dependent motion are $C_{L_{\dot{\alpha}}}$ and $C_{m_{\dot{\alpha}}}$, the lift and pitching-moment derivatives resulting from constant vertical acceleration (that is, linear variation of angle of attack with time). On the basis of a solution to the linearized time-dependent wave equation, these derivatives have been evaluated in the Langley Stability Analysis Section for a family of thin, swept, tapered wings with streamwise tips. The analysis is applicable, in general, at those speeds for which both the wing leading and trailing edges are supersonic. (A previously reported investigation treated the subsonic-leading-edge condition.) Results of the investigation, presented in Technical Note 3196, included a series of detailed charts from which fairly rapid estimates of the derivatives can be made for arbitrary values of aspect ratio, taper ratio, leading-edge sweepback, and Mach number.

The Langley Stability Analysis Section has conducted a theoretical investigation utilizing linearized supersonic-flow theory in which expressions have been derived for the contribution of the horizontal tail to the lift and pitching moment due to angle of attack, constant rate of pitch, and constant vertical acceleration. Numerical values of the aerodynamic coefficients associated with these motions are presented, in Technical Note 3072, for a number of two-dimensional wing-tail combinations, a triangular wing-tail combination, and a number of rectangular wing-tail combinations. Methods for calculating the flow fields behind wings with constant vertical acceleration were also developed. Calculated results are presented for the upwash behind two-dimensional wings and for certain regions behind triangular and rectangular wings for a constant rate of pitch and a constant vertical acceleration.

Two of the more important geometric variables which affect the lateral damping of an airplane are the aspect ratio of the vertical tail and the tail length. Although both of these variables have been investigated extensively for steady-flight conditions, almost no information is available to indicate the effects of variations in these parameters on the unsteady damping of the vertical tail, that is, the damping during a lateral oscillation. An experimental investigation has therefore been made at low speeds in the Langley stability tunnel to determine the effects of vertical tail aspect ratio and tail length on the tail contribution to the unsteady, lateral, damping and directional stability of a model oscillating continuously in yaw. The results, presented in Technical Note 3121, showed a reduction in the contribution of the tail to the lateral damping as the frequency of oscillation was reduced to low values. This reduction became more pronounced as the tail aspect ratio and the tail length were increased.

Dynamic Stability

Considerable interest has recently been shown in means of obtaining satisfactory stability of the Dutch-roll oscillation for modern high-performance airplanes without resorting to artificial stabilizing devices. In this regard, a theoretical analysis has been made to determine the design features that appeared to be most promising in providing adequate inherent stability. The results presented in Technical Note 3035 cover the case of fighter airplanes at subsonic speeds. Since the use of low-aspect-ratio sweptback wings is largely responsible for poor Dutch-roll stability, it is important to design the airplane so that the wing has the maximum aspect ratio and minimum sweep that will permit attainment of the desired performance. The airplane should have positive effective dihedral throughout the angle-of-attack range for satisfactory flying qualities. The radius of gyration in roll should be kept as low as possible and the nose-up inclination of the principal longitudinal axis of inertia should be made as great as practicable. The investigation indicated that by giving detailed consideration to these factors it may be possible to design fighter airplanes with acceptable inherent stability.

A method has been derived, in the Langley Stability Analysis Section, for estimating variations in the roots of the characteristic lateral-stability equation due to changes in mass and aerodynamic parameters of an airplane. Both the exact and simplified expressions indicating the rate of change of the Dutch-roll damping and frequency with respect to the stability derivatives and mass characteristics are presented in Technical Note 3134. The rates of change of the roots with respect to the parameters were shown to have a definite relationship to the amplitude coefficients and ratios of the lateral modes of motion of the airplane subsequent to applied forces or moments.

A study has also been made of the problem of determining nonlinear parameters of dynamic systems, such as aircraft or servomechanisms, whose measured responses are nonlinear. This problem cannot be solved in general but, for certain types of systems, adequate answers can be obtained. In the differential equations representing the motion of these particular systems, one or more coefficients will be functions of the nonlinear parameters. For the systems studied in Technical Note 2977, the nonlinear parameters are functions of either the amplitude of the dependent variables or their time derivatives. It was found that the nonconstant coefficients could be determined satisfactorily for first- or second-order systems, using one or a combination of the techniques developed. For higher-order systems the only practical means of obtaining results was found to be the derivative method.

The difficulty experienced by the human pilot in coping with stability and control problems of modern high-speed aircraft has led to emphasis on the automatic interceptor and guided missile, in which high-performance autopilots are utilized for precise maneuvering during combat operations. Conventional frequency-response servomechanism design methods can be applied to these systems, but unique problems are encountered in the attempt to predict or measure the necessary dynamic-response characteristics of the airplane, which can vary radically over the operating ranges of speed and altitude. Various methods developed by the NACA and others for evaluating frequency-response characteristics of aircraft have been summarized in a paper presented by the NACA at the June 1954, semiannual meeting of the American Society of Mechanical Engineers.[2] This paper deals mainly with the analysis of transient flight measurements. The basic airplane equations of motion were reviewed, and the theoretical transfer function which defines the "short-period" pitch response to an elevator deflection was derived. A numerical example is used to illustrate the evaluation of an analytical transfer function from graphical frequency response by a method which involves the use of graphical aids in conjunction with an analog computer. Also considered were the effects of variations in speed and altitude on the airplane dynamic response. While the examples in this paper were limited to aircraft, the basic analysis methods may be useful to workers in other fields.

The determination of stability derivatives and transfer functions from flight data has been the subject of a number of analytical studies in the past. A recent study has been made to develop a method for determining the lateral-stability derivatives, transfer-function coefficients, and the modes for lateral motion from frequency-response data for a rigid aircraft. The results are reported in Technical Note 3083. The vector technique was applied to the equations of lateral motion, so that the three equations of lateral motion could be separated into six equations. The method of least squares was then applied to the data for each of these equations to yield the coefficients of the equations of lateral motion from which the lateral-stability derivatives and lateral-motion transfer-function coefficients were computed.

The dynamic-stability and control characteristics of a cascade-wing vertically rising airplane in the takeoff, landing, and hovering phases of flight have been investigated, using a flying model in still air. The model had four propellers with thrust axes essentially parallel to the fuselage axis and distributed along the span so that the wings were completely immersed in the slipstream. The model had four wings arranged in a cascade relation to turn the slipstream downward approximately 90° to produce direct lift for hovering flight with the propeller thrust axis essentially horizontal. It was almost impossible for the pilot to fly the model without the use of artificial damping in pitch because of a violently unstable pitching oscillation. The rolling motion was slightly divergent but was easy to control without any artificial stabilizing device. The model apparently had considerable damping in yaw and the yawing motions could be controlled easily. Vertical take-offs and landings could be performed satisfactorily. The only unusual behavior noted when flying near the ground was a slight tendency to pitch nose down and move forward when the model was trimmed for hovering flight well above the ground. Some difficulty was experienced in controlling the vertical motions of the model, apparently because there was little damping of these motions. These results are reported in Technical Note 3198.

Flying Qualities

Although a large number of Douglas DC–3 airplanes have been built and widely used as transports for many years, only limited quantitative information is available

[2] See Smith and Triplett paper listed on p. 65.

on the stability and control characteristics of the airplane. Since the majority of transport pilots are familiar with the characteristics of this airplane, it was believed that quantitative information on its handling qualities would serve as a basis for comparison with the handling qualities of present and future transports. A series of tests have, therefore, been conducted on the DC–3 to determine these characteristics. Data are presented in Technical Note 3088 showing the longitudinal- and lateral-stability and control characteristics, the stalling behavior of the airplane, and the compliance of these flying qualities with the current Air Force-Navy specifications. Even though the DC–3 was designed and built more than two decades ago, the airplane satisfies most of the current specifications for its type. However, the airplane was found to have static-longitudinal instability for certain conditions of airspeed and center-of-gravity position in the power-on configurations; the specified maximum elevator control-force gradient in maneuvers was exceeded in most cases; the rudder forces required to overcome the adverse aileron yaw were excessive; and the rudder and aileron forces in steady sideslip tended to lighten at the higher sideslip angles. Typical frequency-response characteristics of the airplane were also presented.

Automatic Control and Stabilization

Through the use of automatic-control systems in high-speed aircraft, many possible methods of controlling aircraft motions, rates, and accelerations are afforded. As part of a general study of automatic control systems suitable for fighter-type aircraft, a normal-accelerometer control system has been analyzed and the results presented in Technical Note 3191. The effects of Mach number and altitude on the frequency and transient response of the airplane-autopilot combination are shown. The wind-tunnel determination of the airplane transfer functions combined with the experimental transfer function of an actual autopilot servocontrol is utilized to determine (1) the best gain adjustment and time constant of the systems at several combinations of Mach number and altitude, (2) the effects of varying these gain adjustments and time constants at constant Mach number and altitude, and (3) the effects of holding the best gain adjustments and time constants fixed while the Mach number and altitude are being varied. The performance of the normal-acceleration system is compared with that obtained for a pitch-attitude system previously analyzed and reported in Technical Note 2882.

Although autopilots and other automatic-control devices are usually designed on the basis of linearized theory, the limits of linear operation of these devices are frequently exceeded in practice. In some cases it may be desirable to utilize control systems in which the linear range is intentionally exceeded in order to reduce the power requirements of the controlling device. A method for analyzing these types of nonlinear systems through the use of a graphical technique is presented in Technical Note 3034. A one-degree-of-freedom system using control proportional to displacement and rate of displacement was analyzed for several limitations on maximum deflection and rate of deflection of the control. From the examples considered it was shown that, at sufficiently small amplitudes, the period and damping of the system corresponded to those provided by linear operation of the control; whereas, at very large amplitudes, the period and damping approached those of the uncontrolled system. It was also shown that limiting the rate of control movement can produce instability over a range of amplitudes. If the control produces primarily an increase in damping, the control remains effective in producing damping even at amplitudes several times that at which saturation effects are first encountered. This effect may be useful in reducing the power requirements of yaw dampers for airplanes.

High Lift and Stalling

The design of aircraft for high-speed flight has resulted in the use of swept and straight wings of low aspect ratio with wing thickness of the order of 10 percent chord or less. As a result of the inherently low values of the maximum lift coefficient for such wings, even when equipped with conventional trailing-edge flaps, increased attention is being given to the use of additional high-lift devices near the leading edge.

A two-dimensional investigation of the NACA 64A010 airfoil equipped with various combinations of a leading-edge slat, leading-edge flap, split flap, and double-slotted flap has been completed in the Ames 7- by 10-foot wind tunnels and the results are reported in Technical Note 3007. Optimum slat positions were determined for a Reynolds number of 6 million, and section lift and pitching-moment characteristics of the various model arrangements were ascertained for Reynolds numbers of 2, 4, 6, and 7 million. An empirical method was devised for determining, to a first approximation, the slat position which produces the highest maximum section lift coefficient for a given slat deflection angle. Pressure data for the leading-edge flap and slat were converted into coefficients of normal force, chord force, and moment, based on the geometry of the leading-edge device. A comparison of the normal-force coeffiicents for the leading-edge slat extended and for the leading-edge flap deflected 30°, presented in Technical Note 3220, showed that the load acting on the leading-edge flap was greater than the load acting on the leading-edge slat for the same trailing-edge arrangement and value of airfoil lift coefficient.

An investigation was also conducted in the Ames 7- by 10-foot wind tunnels of the low-speed, two-dimensional aerodynamic characteristics of a 10.5-percent-thick symmetrical airfoil, with area suction near the leading edge, to determine the effect of surfaces of different texture on the maximum lift. In addition, the effects of variations of the chordwise distribution of suction velocity were studied. The results of the investigation, reported in Technical Note 3093, indicated that the maximum lift coefficient and minimum suction quantity for a given lift were independent of the surface of the materials tested (perforated plates or sintered steel). The maximum section lift coefficient was increased from 1.3 to approximately 1.8 by means of area suction applied over the region from 0.3- to 3.0-percent chord on the upper surface. A minimum flow coefficient of 0.0012 at a free-stream velocity of 162 feet per second was attained with a permeability arrangement which gave a suction velocity at the trailing edge of the suction area equal to about 2 percent of the local velocity and no outflow at the leading edge.

In the course of the investigation of area suction for boundary-layer control, a comprehensive survey was made of the air-flow resistance characteristics of a variety of commercially available permeable materials. Three general types of porous materials were tested: granular (sintered metals), fibrous (felt cloths and filter papers), and perforated. The flow-resistance characteristics of the materials tested are published in Technical Note 3094 in a form intended to assist in the selection of materials for applications to boundary-layer control using area suction.

A flight investigation has been made of the practical problems associated with the use of porous-leading-edge suction. The wing leading edge of the test airplane was porous over approximately 83 percent of the span and the first 8 percent of the chord on the upper surface. Various other extents of suction area within these limits were also tested. Results of this investigation, described in Technical Note 3062, indicated that a wing equipped with porous-leading-edge suction can be constructed, which has sufficient strength and durability for use in flight, without adding excessive weight. For the type of porous material used in this investigation, clogging due to atmospheric dust did not appear to be a problem. For the light rain encountered in flight, the power required to produce a given flow coefficient was about 50 percent more than that required for the dry condition. Based on ground data, it was estimated that for flight in heavy rain the power required would be approximately twice that for the dry condition. At maximum blower speed the porous area became cleared within 3 to 4 minutes after water ceased to impinge on the surface. Under certain conditions, tests showed a severe vibration of the porous material induced by an "organ pipe" resonance of the air column within the ducts. As expected from wind-tunnel results obtained previous to this investigation, the use of leading-edge suction, with the small amount of power available, had little effect on the maximum lift coefficient developed with the airfoil section of this wing. In general, an appreciable drop occurred in maximum lift coefficient from the leading-edge-sealed configuration to the condition of zero suction with the porous-area configurations tested. Increments in lift coefficient due to the suction available generally brought the maximum lift coefficient back approximately to the value for the wing with the leading edge sealed. The maximum theoretical aerodynamic power, if duct losses are excluded, varied from 3.65 to 9.70 horsepower for the configurations tested.

Because of the thinness of modern wings, part of the wing often becomes completely stalled at only moderate lift coefficients, well below maximum lift. Thus, it has become important to study the characteristics of a stalled plate, since the wing will be influenced by these characteristics over part of its lift-coefficient range. Early experimental work has been supplemented and early analytical work extended to provide a semiempirical theory which will predict the characteristics of an inclined flat plate with stalled flow. The results of this study are reported in Technical Note 3038.

Spinning

During the spin demonstration of military airplanes in acceptance tests, the airplane must be equipped with an emergency spin-recovery device in the event that the normal recovery technique, using conventional controls, is not successful. In the past, a spin-recovery parachute attached to the wing tip or tail has been used as an emergency device. A serious disadvantage of parachutes is that the yawing moment opposing the spin varies during the spin and recovery; in addition, for some high-speed aircraft, heavy loads are imposed on the airplane structure by the large parachutes required for recovery. Apparently a device that would provide a definite and direct yawing, pitching, or rolling moment, such as a rocket, would be more desirable. An investigation has therefore been conducted in the Langley 20-foot free-spinning tunnel to determine the spin-recovery characteristics by use of rockets on a 1/19-scale model of an unswept-wing trainer airplane and to provide a comparison with available full-scale airplane results. A rocket was attached to each wing tip to fire in a direction to apply an anti-spin yawing moment. The rockets were fired individually and in combination. The results of the tests, presented in Technical Note 3068, indicated that the recoveries of the model were in good agreement with those of the corresponding airplane, and for the design tested, wing-tip rockets quickly terminated the spin.

Determination of spin and recovery motions for some current extreme airplane flight conditions cannot be made by the dynamic-model technique. For example, the wing loadings to dynamically simulate airplanes at extremely high altitudes are impracticably high for testing in the free-spinning tunnel; this technique also cannot be used to investigate spinning motions which may be encountered in flight at high Mach numbers. A study has therefore been made to investigate the possibility of analytically determining the motion of an airplane during spin and recovery. In Technical Note 3188, the results of the preliminary calculation of a spin-recovery motion are presented. The recovery was calculated step by step by using modified wind-tunnel rotary balance measurements, applicable equations of motion, and spin-geometry relationships. Certain inconsistencies were apparent which must be cleared up before the method can be accepted as adequate to give detailed spin-recovery motions for a specific airplane. The results of the recovery calculations indicated that initial small oscillations existed which increased gradually in amplitude with time, with nearly all the large significant changes occurring near the end of the time required for recovery.

Research Equipment and Techniques

It has been shown that the lateral dynamic-stability characteristics of high-speed airplanes are strongly dependent on the product of inertia of the airplane about its axes of rotation and on the direction of the principal longitudinal axis of inertia. A description of a simple method of experimentally determining these characteristics of an airplane is presented in Technical Note 3084. The results of the application of this method and a description of the associated equipment and techniques are given for both a simple model and a conventional airplane. Previously reported methods for the determination of the moments of inertia about the pitch and roll body axes are reviewed for use in the determination of the angle of inclination of the principal longitudinal axis.

In the process of analyzing the longitudinal frequency-response characteristics of aircraft, information on various methods of analysis has been obtained by the Langley Laboratory. In the investigation of these methods, the practical applications and limitations were stressed. In general, the methods considered may be classed as: (1) Analysis of sinusoidal response, (2) analysis of transient response, as to harmonic content, through determination of the Fourier integral by manual or machine methods, and (3) analysis of the transient response, through the use of least-squares solutions, of the coefficients of an assumed equation for either the transient time response or frequency response (sometimes referred to as curve-fitting methods). The investigation, described in Technical Note 2997, has led to the following observations: The curve-fitting methods appear to be less critical to inputs having regions of low harmonic content than Fourier methods and present the frequency response as analytical expressions (transfer functions); Fourier methods indicate characteristics of frequency response that may be missed in curve-fitting methods because of the limitations on the assumed form of the equations used in the curve-fitting methods.

In the determination of the dynamic stability of aircraft, it is often desirable to flight test the aircraft, or scale models, in order to determine transfer functions or frequency responses. Frequently, however, facilities for recording the data necessary for a general stability analysis are limited. Distribution of such items as fuel and propulsion units often preclude the possibility of locating recording instruments in the most desirable positions. A method is presented in Technical Note 3021 for deriving time-response and frequency-response data for angle of attack and normal acceleration at the center of gravity of an aircraft when these data are measured at locations on the aircraft other than the center of gravity and when the pitching velocity is not measured. The method involves the calculation of transfer functions for the measured quantities and operation on these transfer functions to derive the transfer function of the desired quantities at the center of gravity. Basic aerodynamic relationships and the forms usually assumed for the transfer functions are used in the derivation. The method appears to have particular application to missiles and models of aircraft where the number of telemetering channels available may limit the number of quantities that can be recorded, or where, because of space limitations, instruments cannot be placed in the most desirable locations.

Designers of aircraft are frequently confronted with the problem of predicting the behavior of aircraft having nonlinear pitching- and yawing-moment characteristics. The usual approach to these problems involves tedious calculations or use of elaborate simulators. A simpler approach suitable for some of the problems encountered, particularly those of missiles, is described in Technical Note 3125. The approach uses the analogy between a ball rolling in a suitably shaped bowl and a missile pitching and yawing in flight. Test results from a model representing a missile with linear moment characteristics are presented to verify the analogy. Several examples of the behavior of nonlinear systems are also given.

INTERNAL FLOW

Inlets

An investigation of a body of revolution with a circular nose inlet was conducted at low speeds in one of the Ames 7- by 10-foot wind tunnels to ascertain some of the effects of inlet-lip bluntness and profile on diffuser performance and body drag. A sharp inlet-lip profile was tested in addition to five circular-arc and two elliptical profiles for free-stream Mach numbers up to 0.330 with inlet flows from zero through choking.

The angle of attack and the angle of yaw were held constant at 0°. Results of the investigation are published in Technical Note 3170. As would be expected, the sharp lip provided the poorest pressure recovery for mass-flow ratios greater than 1.0 for all of the Mach numbers of the test. The improvement over the internal-flow characteristics of the diffuser with the sharp lip, caused by a slight bluntness of the lip of the inlet, depended to only a small extent on the shape of the lip profile. However, for moderate bluntness the effect of the shape of the profile assumed importance, with an elliptical profile providing better pressure recovery than a circular profile. Drag and surface-pressure measurements showed that, for mass-flow ratios less than 1.0, the change of the external drag of the body with mass-flow ratio was caused almost entirely by the change of the suction pressures in the vicinity of the inlet. In addition, the magnitude of the change was found to be equal, but of opposite sign, to the change of the calculated additive drag for the inlet, as long as the external flow was not separated from the lip.

A method is presented in Technical Note 3126 for calculating the profile coordinates for an inlet to be placed in the leading edge of an airfoil. The method includes an application of the principles of thin-airfoil theory which permits the change in velocity distribution caused by a variation in inlet profile to be calculated. Wind-tunnel tests of leading-edge inlets in an airfoil having an NACA 63_1–012 section were made to evaluate the effects of the inlets on the aerodynamic characteristics of the airfoil. The results indicated that the airfoil with an inlet devised by this design method had satisfactory aerodynamic characteristics.

Research on the NACA 1-series nose inlets was extended by tests of three representative configurations at transonic speeds in the Langley 8-foot high-speed tunnel. The results of this investigation showed that the critical Mach number was underestimated by a large amount when predicted from low-speed pressure distributions having a sharp local pressure peak on the inlet lip. In every case, an appreciable margin was found to exist between the critical Mach number and the force-break Mach number. Also, the force-break Mach number was found to be essentially unaffected by sharp local pressure peaks on the inlet lip such as are caused by operation below the design value of inlet-velocity ratio.

Diffusers

A series of investigations was conducted to determine the effect of surface roughness on the performance of a conical diffuser with a 23° expansion angle and a 2:1 area ratio. In general, these investigations covered a range of Reynolds numbers extending through the usual flight range and inlet Mach numbers from about 0.25 to choking. The wide expansion angle resulted in very unstable, separated flow and inefficient performance for the case where the wall was in the smooth condition. Since the short-diffuser geometry was one which is frequently used in aircraft duct installations, because of space limitations, research was directed toward improving diffuser performance by using surface roughness to increase the turbulent exchange of momentum in the boundary layer. Two forms of roughness were investigated: graded cork particles cemented to the wall surface in bands of various widths (Technical Note 3066), and various numbers of both rough and smooth ledges of triangular cross section installed on the diffuser wall transversely to the flow (Technical Note 3123). The presence of a roughness strip near the diffuser inlet produced steady and reproducible flow regardless of whether flow separation existed in the diffuser. With almost the entire diffuser surface coated with cork particles, total-pressure-loss coefficients measured at the diffuser exit and at the tailpipe exit were 21 and 8 percent lower, respectively, than those for the smooth-surface diffuser. The static-pressure recovery and total-pressure loss were either unaffected or slightly impaired by the installation of ledges.

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During the course of these diffuser investigations, it was found that some of the total-pressure-loss coefficients obtained were appreciably in error because of the effects of turbulent velocity fluctuations on impact-tube measurements at the diffuser and tailpipe exits. A method was developed for correcting the results for these errors (Technical Note 3124) which is based principally on the assumption of flow continuity.

An investigation also was conducted to study the effects of the sharpness of curvature of the junction between a cylindrical intake pipe and a 10° conical diffuser of 2:1 area ratio on diffuser flow and performance characteristics. The investigation covered two widely different thicknesses of turbulent boundary layer at the diffuser inlet, Reynolds numbers based on inlet diameter from 1.2 x 10^6 to 3.2 x 10^6, and a mean inlet Mach number range of approximately 0.28 to choking. The static-pressure recovery of the diffuser was found to be entirely independent of the radius of curvature of the inlet junction. No important effects on exit velocity profile or choking mass-flow ratio were observed.

Ducts

One-dimensional, compressible, viscous-flow relations which permit the determination of flow conditions in a ducted helicopter blade were derived by the Langley Internal Flow Section (Technical Note 3089) for the purpose of estimating the performance of proposed helicopter jet-propulsion systems. The principal equation consists of the conventional one-dimensional expression with a centrifugal-force term added. The centrifugal force tends to raise the density and lower the Mach number and thus opposes the effects of friction. A limited number of calculations were made for the isothermal compression case over a wide range of helicopter operating conditions and relative duct sizes. The results of the calculations indicated that, with a constant-area duct 30 diameters in length, choking did not occur for duct inlet Mach numbers of less than 0.66, and a maximum stagnation-pressure ratio of 2.41 could be obtained across the duct.

Research Equipment

The use of afterburners with turbojet engines for thrust augmentation requires a supply of cooling air flow around the engine tailpipe. This large amount of cooling air for both airframe and engine has resulted in complex engine installations of the type wherein it is very difficult either to assess the actual performance of the installed engine or to investigate suspected performance losses caused by the cooling air flow without using detailed jet-exit, mass-flow, and momentum surveys. The very high exhaust temperatures (about 3,500° F) prevent the use of conventional, fixed, survey instruments without using special materials that are prohibitively expensive and difficult to fabricate. A power-operated moving probe has been developed and used successfully in flight tests to determine the thrust and air-flow characteristics of an afterburner-equipped turbojet engine. The swinging probe measures the radial variation of total pressure, static pressure, and temperature at the exit of the fuselage. Measuring these quantities makes it possible to evaluate the performance of the engine as installed. This method is being used to investigate the characteristics of cooling-air ejectors in flight. In view of the need in the aircraft industry for a method of measuring thrust in flight, a paper describing this research technique was presented at a meeting of the Institute of Aeronautical Sciences in Los Angeles.[3]

PROPELLERS FOR AIRCRAFT

Aerodynamic Problems

The increased interest in propeller-driven vertically rising airplanes and convertiplanes having tilting-axis propellers resulted in an investigation in the Langley full-scale tunnel to determine the aerodynamic characteristics of a propeller through a wide range of angle of attack. The results, published in Technical Note 3228, give a comprehensive amount of propeller performance data covering blade angles up to 67.5° and advance ratios up to 6.2 through angles of attack up to 87.5°. The preliminary inspection into the rate-of-descent regime (angles of attack of approximately 180°) indicated very severe propeller vibrations due to the mixing of the slipstream velocity with the free-stream flow. Calculations of the rate of change of normal-force coefficient with angle of attack, by using Ribner's method which was developed for relatively small angles of attack, do not adequately predict the measured characteristics above angles of attack of 15°. These empirical data are of immediate value to designers and provide basic information for extension of the theory to higher angle-of-attack operation.

As part of a general program to improve propeller performance at high speeds, an investigation of two two-bladed propellers of very high blade solidity (blade activity factors of 179 and 265) was made at high speeds in the Langley 8-foot high-speed tunnel through a range of blade angles from 20° to 70° for free-stream Mach numbers from 0.165 to 0.725. Although efficiencies of the order of 90 percent were obtained, the high-solidity blades were less efficient than blades of conventional solidity. The variations of power, thrust, and average lift coefficient with solidity, at a constant geometric angle of attack, were found to be analogous to the variations of wing lift coefficient with aspect ratio; this result indicated that high-solidity blades may be desirable at very high speeds. Because of the power limitations of the test equipment, conclusive evidence of the favorable effects of increased blade solidity at high speeds was not obtained.

As a result of the development of power plants which assured flight speeds considerably beyond Mach numbers of 0.725, an investigation was made of an NACA 4-(5)(08)-03 two-bladed propeller in the Langley 8-foot high-speed tunnel up to a forward Mach number of 0.913. These tests were the first to establish actual propeller efficiency in the transonic speed range. Although the efficiency loss was not as great as had been predicted by the more pessimistic estimates, the performance of this conventional propeller at transonic speeds was not acceptably high. The findings of this investigation gave impetus to propeller designs suitable for application at transonic and supersonic speeds.

Because of the interest in takeoff run calculations and propeller selection, outdoor static tests were made on four related NACA two-bladed propellers which had previously been investigated at high forward speeds in the Langley 8-foot high-speed tunnel. The propellers differed in camber and blade width. The blade angles tested ranged from 0° to 40° and the maximum tip Mach number was 0.93. The static-thrust measurements thus obtained provided basic information concerning the design compromises necessary in order to obtain satisfactory takeoff performance.

Incompressible-flow theories for propellers have been used to great advantage in the development of propellers designed to operate below the transonic speed range. The results of a study to extend the actuator-disk theory to compressible flow are presented and are

[3] See Rolls and Havill paper listed on p. 65.

an extension of a similar study presented in Technical Note 2164 (reported in the Thirty-seventh Annual Report, 1952). Comparison of incompressible- and compressible-flow results show large differences in the induced flow, especially at high-power loadings. The important findings indicated by this theoretical treatment are that a high level of efficiency can be obtained at high Mach numbers and that the installation of propellers in tandem is one means of delaying the large losses in efficiency associated with choked flow near the juncture of the propeller blades and the spinner.

One of the most serious problems facing the propeller industry has been that of predicting the aerodynamic excitation and resulting blade stresses arising from operation of a propeller with its thrust axis inclined to the air-stream. A study was made of the aerodynamic exciting forces and resulting stresses under carefully controlled wind-tunnel conditions with the 2,000-horsepower propeller dynamometer in the Langley 16-foot high-speed tunnel. It was indicated from this investigation that the aerodynamic exciting force of an inclined propeller may be computed accurately at low rotational speeds. As blade section velocities approach the speed of sound, however, the accuracy of prediction may not always be so satisfactory. A stress prediction, based on stresses at low rotational speeds which assumed a linear relation between first-order vibratory stress and the angle-of-inclination—dynamic-pressure product, proved to be conservatively high when the outer portions of the blade were in the transonic- and low supersonic-speed range.

The possibility of employing sweep in propeller blades to improve the propeller efficiency at high subsonic speeds has been investigated by using the 2,000-horsepower propeller dynamometer in the Langley 16-foot high-speed tunnel. The results indicated that, for the design investigated, there would be no benefit of sweep, at least up to the limiting Mach number of the investigation (0.6), which for these propellers was equivalent to a maximum tip Mach number of 1.05.

A fundamental study of subsonic compressible flow about a propeller has been made at the Langley 16-foot transonic tunnel and the results are reported in Technical Note 2983. Where other studies have used the simple momentum theory or modifications of the Glauert-Prandtl stretching technique, this study succeeded in developing the complete potential for the field and, in particular, the inflow distribution along the blade, by a theory that is exact within the limits of linearization.

SEAPLANES

Hydrodynamic Elements

The general program of research on hydrodynamic lifting elements has been extended to include the effects of vertical chine strips on the hydrodynamic forces and centers of pressure of planing surfaces having dead rise. Vertical chine strips are of particular interest because of their favorable effect on the spray characteristics and on the lift. Wetted length, resistance, and center-of-pressure location were determined at speed coefficients up to 25, load coefficients up to 80, and trims up to 30° for prismatic surfaces having basic angles of dead rise of 20° and 40° with vertical chine strips. These results are presented in Technical Note 3052. Comparisons of the more important planing characteristics are made with those for related surfaces, with and without horizontal chine flare, and for a flat plate. These comparisons show that vertical chine strips are a more effective means for increasing the lift of a given surface than is horizontal chine flare. This increase in lift, however, is accompanied by a substantial increase in drag, so that the lifting efficiency of a surface with vertical chine strips is approximately the same as that of a surface with chine flare.

The application of hydroskis to water-based airplanes has brought about a need for information on the characteristics of hydroskis when operating beneath the water surface. A theoretical and experimental investigation of the characteristics of simple flat plates having aspect ratios of 1.00 and 0.25 has, therefore, been made and the results are given in Technical Note 3079. The experimental investigation disclosed that two types of leading-edge separation can occur when lifting surfaces approach the water surface from below. One type, called white water and found only for the aspect-ratio-1.00 surface, caused a slight decrease in the lift and moment coefficients and a slight increase in the drag coefficient. The other type, called a planing bubble and found for both surfaces, caused a sharp drop in the lift, drag, and moment characteristics of the order of that to be expected in the transition from the submerged to the planing condition. The theoretical investigation was made to develop a method for the calculation of lift under conditions where the flow is not separated from the plate and where the water surface is far enough above the plate to have negligible influence on lift. The method of calculation was developed by modification of Falkner's vortex-lattice theory. The calculated lift was found to be in good agreement with the experimental results obtained in the tank and also with aerodynamic data obtained from a wind tunnel.

The present trend toward the use of underwater lifting surfaces on water-based aircraft and on surface vessels has emphasized the need for drag data on supporting struts which pierce the water surface. An investigation, therefore, has been made to determine the hydrodynamic drag of three surface-piercing struts at 0° angle of yaw at depths up to 6 chords for speeds up to 80 fps at various angles of rake. These results are presented in Technical Note 3092. Two of the struts had NACA 66_1–012 airfoil sections and the third strut had an NACA 66_4–021 airfoil section. Section drag coefficients, determined from plots of drag against depth, were in good agreement with available wind-tunnel results. Raking the struts changed the section drag coefficient as expected because of the change in effective thickness ratio with angle of rake. The drag coefficient corresponding to the drag at the surface intersection was approximately constant at Froude numbers above 8.0 and at subcavitation speeds. The inception of cavitation was noted at a speed higher than that predicted from two-dimensional-flow theory. This difference was due to the influence of the free-water surface on the pressure distribution.

Hydrodynamic Configurations

Results of wind-tunnel and tank investigations already are available for a related series of hull forms having a wide range of length-beam ratio. To supplement these results, the static properties and resistance characteristics of this family of hulls have been determined and are presented in Technical Note 3119. The static properties are presented as charts from which draft, trim, and upsetting moment for wide ranges of load, center-of-gravity location, and roll for any length-beam ratio in the series may be obtained. The resistance and trimming moments also are presented in the form of charts for models having length-beam ratios of 6 and 15.

HELICOPTERS

Loads

The correct prediction of the loads and stresses imposed on a helicopter in flight is essential to the reliability, availability, and utility of the helicopter. A fundamental aspect of the loads problem is a knowledge of the bending frequencies and mode shapes of the lifting rotor blades. A chart procedure for rapidly estimating these frequencies, for both rotating and nonrotating blades, has been worked out. Since the procedure was based on Southwell's equation, an evaluation of the method with regard to such parameters as higher modes, blade offset, and variable mass and stiffness distributions has been made. The evaluation shows that, when nonrotating-beam bending modes are used, Southwell's equation yields reasonably accurate bending frequencies for rotating helicopter blades. Several comparisons of frequencies estimated, by using the charts with values given by the manufacturer for several actual blades, show that the simplified procedure yields good practical results.

The designer must also know the extent to which gusty air affects rotor-blade stresses. An investigation of the effects of gusts was conducted at the Langley helicopter test tower and the results reported in Technical Note 3074. For the rotor conditions tested, in gusty winds up to 26 mph, the influence of gusts appeared to be secondary to the vibratory stress levels that resulted from the dissymmetry of the rotor downwash in forward flight.

Tests have also been conducted at the Langley helicopter test tower to determine the increase of rotor loading and induced velocity due to a rapid collective-blade-pitch increase during a jump takeoff or maneuver. The results (Technical Note 3044) showed that the rates of blade-pitch increase ranged from 6° to 200° per second and, in general, at the high pitch rates it was possible to develop over twice the normal thrust coefficient for about 0.2 second. The calculated thrust overshoot is shown to be in good agreement with experimental time histories.

Performance

The helicopter has now been developed to the point where future increases in speed and greater range dictate the necessity for reductions in parasite drag. Accordingly, a preliminary inspection of available literature dealing with airplane drag cleanup work, conducted in previous years in the Langley full-scale tunnel, has been made. The results were applied to a typical helicopter in Technical Note 3234. Substantial reductions in parasite drag may be realized by modifying the landing-gear installation as well as the rotor hub, air induction and exit systems, and exhaust stacks, and by eliminating air-leakage gaps and protuberances. For the typical helicopter examined, a 19-mph speed increase and a 25-percent increase in maximum range are indicated.

In an effort to assess the relative advantages of various helipcopter configurations for different applications, a general research program to determine the performance of multirotor configurations has been conducted at the Langley full-scale tunnel. A summary of the hovering and forward-flight tests of one coaxial and one tandem configuration is reported in Technical Note 3236. The results indicated that, although power requirements for the coaxial rotor in static thrust can be predicted with good accuracy from available single-rotor theory, more power is required in level flight than would be predicted for an equivalent single rotor. The tandem arrangement having zero rotor overlap and stagger indicated less power required for static thrust than predicted, but somewhat greater power for level flight than predicted from single-rotor theory.

Current design trends have resulted in increased interest in the use of blade twist for most rotor configurations. Theoretically derived charts for predicting the profile drag-lift ratio of a helicopter rotor having rectangular blades with −8° twist (blade pitch angle at tip 8° lower than at root, with linear variation between) have been prepared. Conditions for the onset of blade stalling are shown in the charts. A sample study is included to illustrate the theoretical effects of blade twist in forward flight, with reference to limiting forward speed, power required, power-off rate of descent, and blade motion. The sample study includes results for additional twist values to indicate the trends beyond the two values for which charts are available.

Stability and Control

One of the most important helicopter flying-qualities criteria utilized in current specifications deals with satisfactory maneuver stability, that is, no divergent tendency in pitch. It was found that the same criterion was generally applicable to both tandem- and single-rotor helicopters. A basis for designers and procurement agencies, to use in studying the maneuver stability of a prospective helicopter, is presented in Technical Note 3022. The report contains a chart from which combinations of pertinent stability derivatives that result in at least marginal stability can be conveniently determined. Methods for theoretically predicting the necessary derivatives are also discussed, as well as techniques for measuring the derivatives by means of flight tests.

Another important aspect of current flying-qualities specifications is the criteria for minimum helicopter directional stability and control. With the conventionally powered single-rotor helicopter, and with many jet-powered helicopters, these requirements must be met by an adequately designed tail rotor. As an aid in designing helicopter tail rotors to meet the directional criteria, theoretically derived charts and equations are presented in Technical Note 3156 by which tail-rotor design studies of directional trim and control response at low forward speeds can be conveniently made. The use of the charts and equations for tail-rotor design studies is illustrated, and comparisons between theoretical and experimental results are presented.

Helicopter fuselages in general, and tandem fuselages in particular, may exert a marked influence on helicopter directional stability and control. An experimental investigation was therefore made in the Langley stability tunnel to determine the directional stability of two tandem helicopter fuselages (Technical Note 3201). One fuselage represented a helicopter with overlapping rotors (overlap-type fuselage) and the other a helicopter with nonoverlapping rotors (non-overlap-type fuselage).. The overlap-type fuselage model was found to be directionally unstable for certain combinations of angle of attack and sideslip, but could be made directionally stable by blunting the vertical tail of the model or by using a thin tail in place of the original thick vertical tail. The non-overlap-type fuselage model was directionally unstable for positive angles of attack throughout the angle-of-slideslip range. Spoilers located around the fuselage nose were the only effective means found to make this fuselage stable without resorting to major design changes.

Airfoil Characteristics

Extension of the range of information on airfoil-section characteristics has been continued to meet the specialized requirements of rotating-wing aircraft. The increasing speed of helicopters, for example, has brought about the need for airfoil data at high subsonic Mach numbers at angles of attack as high as 30°. Data in this range were obtained in an investigation in the Langley low-turbulence pressure tunnel of four airfoil sections varying in thickness from 6 to 12 percent. Information which illustrates the effects of airfoil-section parameters and flow variables on the aerodynamic characteristics of symmetrical, two-dimensional airfoils at high angles of attack, obtained from the literature and recent investigations, is summarized in Technical Note 3241. Included in this summary are the results of an investigation of one section, through an angle-of-attack range from 0° to 360°, which show that the drag coefficient reaches a value of 2 at an angle of attack of 90°.

In an effort to develop a helicopter rotor having minimum profile-power losses, the NACA has derived a special series of helicopter airfoil sections. The most promising of such sections (NACA 8–H–12) is a laminar-flow airfoil which, when tested in the Langley two-dimensional low-turbulence tunnel showed low drag in the operating lift-coefficient range of most helicopter rotors without undue sacrifice in maximum section lift coefficient. The airfoil retained reasonable aerodynamic characteristics when tested in the rough condition. The practical aspects of blade construction, however, created doubt as to the achievement, in actual operation, of the low drag values obtained from the aerodynamically smooth, two-dimensional test specimens. A test rotor incorporating the NACA 8–H–12 section was therefore constructed and tested on the Langley helicopter test tower. The test results, reported in Technical Note 3237, indicated that controlling construction to tolerances of the order of 0.002 inch of true surface contour resulted in the realization of one-half of the theoretical profile-drag reduction, or a 6- to 7-percent reduction of the total torque coefficient.

Rotor Inflow

A knowledge of the inflow distribution through and about a lifting rotor is required in almost all fields of helicopter analysis. In view of the stimulation of interest in rotor-induced flow brought about by the current emphasis on loads, stability and control, and the expanded use of multirotor configurations, it was considered desirable to review the available information on the subject. Such a review is presented in Technical Note 3238. The available material is summarized in a table according to flight condition, type of information, source, and the reference papers in which the data can be found. Representative aspects of some of the reference material are discussed.

POWER PLANTS FOR AIRCRAFT

Now that research airplanes have reached speeds up to 2½ times the speed of sound and tactical aircraft capable of supersonic flights have become operational, the need is emphasized for still more powerful power plants having less weight and lower fuel consumption in order to provide higher supersonic speeds and greater range for tactical aircraft. The research effort of the NACA in the propulsion field, chiefly focused on the turbojet engine, has continued toward solving the problems of producing a tremendous amount of power in a small, lightweight engine with high efficiency to provide the desired speed and range capabilities. This means increased thrust and increased component and overall engine efficiency.

Considerable gains in thrust, for example, can be realized by increasing the airflow through the engine. Recently research compressors have been built and tested which not only have greater air capacity but, in addition, are lighter and have higher efficiency. Experimental combustors have also been developed which operate at high efficiency at the higher airflows. Thrust may also be increased by increasing turbine-inlet gas temperatures above the present limits imposed by material properties. Research in heat transfer and other fields related to the development of cooled turbines is providing the information necessary to permit engine operation at increased inlet temperatures.

Research is also being conducted on other problems related to increasing the supersonic capabilities and range of future turbojet-powered aircraft. Such research areas include improvement of fuel properties, improvement of lubricants and bearings to provide capability for withstanding the higher temperatures associated with supersonic flight, and improvement of engine inlet characteristics for supersonic applications.

Attention is also being given to the problems associated with rocket and ram-jet power plants. Especially as power plants for supersonic missiles, these engine types offer much promise.

The following sections present a discussion of recent unclassified research for aircraft power plants.

AIRCRAFT FUELS

Synthesis and Analysis

As part of a systematic study directed toward correlating hydrocarbon structure and physical properties, representative homologs in several series of dicyclic hydrocarbons have been synthesized and purified. In order to make this evaluation of structural effects more comprehensive, several hydrocarbons have been synthesized which contain one phenyl or one cyclohexyl group attached to a side chain of six or more carbon atoms. The study describes the synthesis, characterization, and purification of four aromatic hydrocarbons: 3-phenylhexane; 2-ethyl-1-phenylbutane; 2,4-dimethyl-3-phenylpentane; and 2,4-dimethyl-1-phenylpentane; and three cyclohexyl compounds: 3-cyclohexylhexane; 2-ethyl-1-cyclohexylbutane; and 2,4-dimethyl-3-cyclohexylpentane.[4] Although several of these hydrocarbons have been prepared previously, some of the physical properties for the correlation studies were not determined. It was also desirable in synthesizing these compounds to choose methods that would lead to an unequivocal structure for each hydrocarbon.

Another phase in the synthesis and purification of dicyclic hydrocarbons was conducted to correlate the physical properties with molecular structure. It was deemed desirable to extend this study to the fused dicyclic system. The study describes 1,2,3,4-tetrahydronaphthalene and four of its homologs, 1-methyl-, 1-ethyl-, 1-butyl-, and 1-pentyl-1,2,3,4,-tetrahydronaphthalene.[5] Of these, the 1-butyl and 1-pentyl compounds are reported for the first time. The other compounds in this series were synthesized in order to obtain a consistent set of precise physical constants on highly purified samples of these structurally related hydrocarbons.

As part of an investigation of hydrocarbons for possible components of aviation fuels, a series of alkylnaphthalenes and their hydrogenated derivatives were prepared. Physical properties (boiling points, refractive indices, densities, heats of combustion, and kinematic viscosities at four different temperatures) are presented for seven 1-alkylnaphthalenes and eight of their tetrahydro derivatives. Melting points and estimated purities are also given for the compounds that crystallize. The hydrocarbons were synthesized by well-known methods and were purified by precision fractional distillations to a purity in the order of 99 mole percent. Three of the compounds (5-butyltetralin, 1-isobutyl-tetralin, and 5-isobutyl-tetralin) are described for the first time.[6]

The identification of aromatic hydrocarbons has generally been possible only by physical methods such as examination of absorption spectra and comparison of

[4] See Lamberti and Wise paper listed on p. 65.
[5] See Karo, McLaughlin, and Hipsher paper listed on p. 54.
[6] See Hipsher and Wise paper listed on p. 64.

the physical properties of unknown with authentic compounds. Identification based on the melting points of solid derivatives offers several advantages, especially in those instances where sufficient quantities of the hydrocarbons or equipment necessary for physical examination are unavailable. Consequently, an investigation of phthalic anhydride derivatives of aromatic hydrocarbons was begun at the Lewis Laboratory in order to ascertain the usefulness of these derivatives in identifying the hydrocarbons. Phthalic anhydride derivatives of 25 mono-, di-, and tri-alkylbenzenes have been prepared, and the melting points of the derivatives have been compared to determine the usefulness of these compounds in distinguishing the hydrocarbons.[7] In general, these derivatives distinguish satisfactorily among the alkylbenzene hydrocarbons. However, it is not possible by means of the benzoic acids to identify 1,3-dimethylbenzene and 1,4-dimethylbenzene or *s*-butylbenzene and isobutylbenzene. No phthalic anhydride derivative was obtained from 1,4-dimethylbenzene and 1-methyl-4-ethylbenzene, and the derivative of isopropylbenzene could not be obtained in sufficient purity to report. Therefore, an investigation of the tetrachloropthalic anhydride derivatives was undertaken.[8] In the present study the tetrachlorophthalic anhydride derivatives of 28 mono-, di-, and tri-substituted alkylbenzenes are described. Of these, 22 are reported for the first time.

The methyl, ethyl, and propyl homologs of the 1,3-diphenyl- and 1,3-dicyclohexyl-2-alkylpropane series were among those hydrocarbons of interest at the Lewis Laboratory for an investigation of the effects of structure on combustion characteristics and physical properties. Although the methyl and ethyl hydrocarbons of both series have previously been described, the absence of some physical properties and lack of agreement among some of the published properties necessitated the work described.[9] The physical properties obtained were melting point, boiling point, density, refractive index, heat of combustion, and kinematic viscosity. Two new compounds, 1,3-diphenyl- and 1,3-dicyclohexyl-2-propylpropane, are reported.

As part of a study of the effect of structure on the properties of dicyclic hydrocarbons, the 2-, 3-, and 4-methyldiphenylmethanes and corresponding dicyclohexylmethanes were synthesized and purified.[10] The scale of the synthesis was planned so that approximately 500-milliliter quantities of each hydrocarbon would be made available in 99 mole percent purity for specific test purposes. None of the methyldicyclohexylmethanes has been mentioned in the literature and the preparation and physical properties of these hydrocarbons are thus described for the first time. The methyldiphenylmethanes have been prepared previously, but only boiling points, indices of refraction, and densities were reported and the data from the various sources are not in agreement. The physical constants for the methyldiphenylmethanes have been improved and extended.

In connection with the synthesis of several diphenylmethane hydrocarbons, the *o*- and *p*-monobromo derivatives of propyl-, isopropyl-, butyl-, isobutyl-, and *s*-butylbenzenes were prepared in 1- to 2-liter quantities. The mixtures of isomers were separated to determine the relative percentages of bromine substitution in the ortho and para positions. Each isomer was further purified to isolate samples on which to determine physical properties. Except for the *o*- and *p*-bromoisopropylbenzenes, the physical constants are reported more completely and on more highly purified samples than in previous chemical literature.[11]

Because of the increasing use of infrared spectra in analysis and identification, it was desirable to compile the infrared spectra of 47 hydrocarbons which were available in a high state of purity. This study is reported in Technical Note 3154. (The infrared spectra of 59 dicyclic hydrocarbons were previously obtained at the Lewis Laboratory and are described in Technical Note 2557.)

The dielectric constants of 26 alkylbenzenes were determinated at 20° and 30° C, and their molar polarizations and dipole moments calculated. Comparison was made of the present results with previous values available in the literature for 14 of the compounds, and some discrepancies among the values are discussed. The atomic polarizations of alkylbenzenes were calculated as a function of the number of side-chain carbon atoms. Dipole moments of these hydrocarbons, calculated from the Onsager equation, are discussed in relation to the moments found in the gaseous state.[12]

The preparation of pure hydrocarbons was conducted in order to obtain correlations of molecular structure with engine performance and with other physical and chemical properties of the compounds. In the purification process to obtain these hydrocarbons, distillation is extremely important. Distillation to obtain maximum purity of a compound often requires greater time than that required for the complete synthesis of the compound. The work was undertaken to determine, at atmospheric pressure, the efficiency of stainless-steel distillation columns with 30 feet of packed height and the efficiencies at atmospheric and reduced pressures of laboratory glass distillation columns with 6 and 7 feet of packed height.[13]

As a part of the program involving the investigation of organo-metallic compounds, a number of alkyl-

[7] See Lewenz and Serijan paper listed on p. 65.
[8] See Lewenz and Serijan paper listed on p. 65.
[9] See Caves, McLaughlin, and Wise paper listed on p. 63.
[10] See Lamneck and Wise paper listed on p. 65.
[11] See Lamneck paper listed on p. 65.
[12] See Altshuller paper listed on p. 63.
[13] See Walsh, Sugimura, and Reynolds paper listed on p. 66.

silanes have been synthesized and the following physical properties measured: boiling point, freezing point, index of refraction, molecular weight, and density. Phenylsilane and isobutylsilane are reported for the first time. Vapor pressures were measured using a static system, and the heats of vaporization were calculated from the data. By the use of a special sample-filling technique, it was possible to measure the heats of combustion in an oxygen-bomb calorimeter and then to calculate the heats of formation.[14]

There was considerable discrepancy in the reported values for the Si–C bond energy. In order to add to the information available, it was considered worthwhile to calculate the Si–C bond energy in some alkylsilanes recently obtained at the Lewis Laboratory.[15]

In connection with a program at the Lewis Laboratory involving the preparation and physical properties of alkylsilanes, 12 compounds of this type have been prepared in high purity and their infrared spectra have been determined on a Baird double-beam recording spectrophotometer employing sodium chloride optics. The compounds synthesized consisted of all the methyl- and ethylsilanes as well as *n*-butyl-, isobutyl-, vinyl-, and dimethyldi-*n*-propylsilane. The preparation of compounds with Si-H bonds was accomplished by the reduction of the appropriate alkylchlorosilane with lithium aluminum hydride in dioxane, except for triethylsilane, in which preparation, ether was the solvent. The tetra-substituted homologs were prepared by condensing the appropriate alkylchlorosilane with excess Grignard reagent.[16]

As part of a series of investigations on flame speeds, ignition, and other burning characteristics of high-energy fuels, several alkylsilanes were prepared in laboratory quantities and their chemical and physical properties reported. The results of preliminary combustion studies suggested that further investigation in a large-scale combustor might yield information useful for the evaluation of these compounds as fuels or fuel components for jet aircraft. A large quantity of mixed butylsilanes was synthesized by reducing butyltrichlorosilane with lithium aluminum hydride in dioxane solution. This was safely and successfully accomplished by operating in steel equipment under an atmosphere of oil-pumped nitrogen to minimize the hazards involved in conducting this operation on a large scale.

Fuels Performance Evaluation

Modern high-speed aircraft impose stringent requirements on aircraft fuel systems. As a result, designers have been faced with the problem of reliable systems to utilize jet fuels that may have physical property variations considerably greater than those encountered with aviation gasoline in conventional aircraft. In order to assist in the solution of this problem, the NACA has surveyed existing fuel literature and determined the ranges over which physical properties of jet fuel may vary. Since the release of this survey, additional information has become available on MIL–F–7914(Aer) grade JP–5 fuel and several of the current grades of fuel oils. This information has been prepared as a supplement to the survey.

Spontaneous ignition processes may be deleterious, as when they produce preignition in a reciprocating engine or create fires in the storing and handling of combustible materials, or they can be beneficial by serving as the ignition source in a Diesel engine or promoting smoother burning in a turbojet combustor. In any combustion process where self-ignition can occur, the time lapse before the flame appears is an important factor. Since the time delay of ignition is an inverse measure of the rate at which the reaction proceeds, a study of the factors influencing the delays may provide information on the kinetics and mechanism that prevail in the ignition process. The study was run with propane-oxygen-nitrogen mixtures at atmospheric pressure and at temperatures from 525° to 740° C. A flow system was chosen because it permitted control of temperature and pressure from the time of mixture preparation until ignition occurred.

Miscellaneous

Recent trends in aircraft development are toward operation at higher altitudes and increased flight speeds. The loss of aircraft fuel by evaporation from vented tanks can result in an appreciable reduction in flight range. An analysis was therefore conducted to estimate the liquid-fuel temperatures and the amount of fuel evaporation that would be expected in operation of long-range supersonic aircraft at high altitudes. Two types of fuel loss were considered: the loss due to adiabatic evaporation during climb and the loss due to aerodynamic heating effects. Some alleviation of the evaporation-loss problem has resulted from a reduction in jet-fuel volatility by changing from JP–3 to JP–4 specifications; however, a minimum volatility is still required to retain satisfactory engine starting characteristics and to assure adequate fuel availability. A flight plan was assumed, and heat-balance relations were employed to estimate the amount of heat transferred to fuel contained in a cylindrical fuselage. The influence of flight speed, flight altitude, fuel withdrawal rate, tank size, tank pressurization, insulation, and initial fuel temperature on the evaporation of MIL–F–5624A grade JP–4 fuel was predicted. The influence of fuel volatility was determined by comparing the evaporation obtained with JP–4 fuel and that obtained with a lower volatility fuel, MIL–F–7914 grade JP–5.

[14] See Tannebaum, Kaye, and Lewenz paper listed on p. 66.
[15] See Tannebaum paper listed on p. 66.
[16] See Kaye and Tannenbaum paper listed on p. 64.

COMBUSTION

Fundamentals of Combustion

The accelerated pace of combustion research in recent years has produced a large amount of data on the fundamental properties of combustion. Interpretation of these results, however, has been made more difficult by their very quantity and by the fact that, until recently, there had been little understanding of the possible relations of the combustion phenomena to one another. The aim of the work on pressure limits of flame propagation was to systematize part of the combustion data by making a quantitative connection between pressure limits and wall quenching.[17] The method of attack was to measure the pressure limits of flame propagation for propane-air mixtures in flame tubes of several different diameters. A rather limited study of this type has recently been carried out, but it was subject to difficulties in connection with ignition. The work avoided these difficulties, and the results are believed to be more precise. The investigation was limited to propane because of its ease of handling and because its combustion properties are representative of saturated hydrocarbons in general.

It has long been considered important to study catalysts for the combustion of fuels. Both positive and negative catalysts are of practical importance, the positive type because they may increase the heat-release rate or widen the range of stable burning and the negative type because they may act as fire-extinguishing agents. The effects of seven additives on the pressure limits of propane-air mixtures are reported. Each additive was chosen because it had been reported to have some effect on other combustion properties or because of general interest. The limits were measured in a flame tube of new design. Mixtures containing approximately 2 to 8 percent propane by volume were studied. The limit curves were without lobes on the rich side and were closely related to quenching-distance data measured by the flashback of a Bunsen flame.

The quenching of a flame by a channel of a given size and shape is an easily measured phenomenon which may supply much information relating to the many other associated flame phenomena. Thus, the distance of closest approach of a flame to a cold wall; the minimum ignition energy; the relative ability of a stable flame to generate a large amount of heat per unit volume per unit time; and the critical conditions of container geometry, pressure, and temperature under which this flame can or cannot exist may all be related to the quenching distance. The quenching distance itself varies with fuel type, oxidant type, fuel-oxidant ratio, quenching surface geometry, temperature, and total pressure. Flame quenching by a variable-width rectangular-slot burner as a function of pressure for various propane, oxygen, and nitrogen mixtures was investigated and reported.

Recent flame-quenching research has indicated that a set of simple relations should exist among the various channel geometries that are capable of just quenching a given flame at a given pressure. The effect of channel geometry on flame quenching, as calculated on the basis of average active particle chain lengths, is related among six different geometries: plane parallel plates of infinite extent, cylindrical tubes, rectangular slots, cylindrical annuli, and tubes of elliptical and equilaterally triangular shape. Experimental determination of the quenching behavior of propane-air flames over an equivalence-ratio range of 0.82 to 1.30 was made for a series of rectangular slots, cylindrical annuli, and cylindrical tubes in the pressure range 0.08 to 1.0 atmosphere. Generally good agreement between theory and experiment was found for both rich and lean flames. The average deviation of the predicted quenching distances from the observed ones was 4.3 percent for equivalence ratios less than or equal to unity and 8.6 percent for equivalence ratios greater than unity. These deviations are generally systematic, rather than random. It was also found that a relatively small cold surface may, when flame immersed, exhibit very large quenching effects.

Considerable effort has been made to determine the factors and mechanisms that govern the formation of smoke during the burning of fuels, so that, eventually, methods of controlling smoke formation can be devised. Particular emphasis has been placed on preventing smoke formation during the burning of fuels in combustion chambers. As the type of hydrocarbon present in a fuel is known to affect smoke formation, an investigation was conducted as part of the fundamental combustion program at the Lewis Laboratory to determine the maximum rate at which various pure hydrocarbons could be burned without producing smoke.[18]

Standard tests have been devised to determine the relative smoking properties of hydrocarbons. However, these tests give little information about the smoke-forming or smoke-burning capacities of flames subjected to outside influences such as motion or turbulence in the air surrounding the flame. The study presents the results of a systematic study of external variables which might be expected to influence the smoking tendencies of flames.[19] Included are studies of the smoke-burning capacities of ethylene-air flames and the effects of initial gas temperature, fuel-flow rate, flame length, and the secondary-air variation on the smoking tendencies of benzene-air flames. Experimentation was performed on a laboratory bench-scale apparatus utilizing glass equipment where feasible.

[17] See Belles, Simon, and Weast paper listed on p. 63.

[18] See Schalla and McDonald paper listed on p. 65.

[19] See Clark paper listed on p. 63.

Investigations of smoking tendencies at pressures other than atmospheric have been very limited. In order to obtain a more comprehensive understanding of the effect of pressure on smoke formation, an investigation was conducted to determine the variations in smoking tendency over a wide range of pressures for a variety of fuels. Six pure hydrocarbon compounds, a JP-4 fuel, and two blends of octane and toluene were investigated over a pressure range of about ½ to 4 atmospheres. For two of the hydrocarbons, namely, octene-1 and *n*-octane, the pressure range was extended to 9 and 12 atmospheres, respectively. Smoking tendencies were determined by burning the fuels as diffusion flames from a modified wick lamp in an enclosed chamber. The maximum relative rate at which the fuels could be burned without smoking was used as the criterion of smoking tendency.

The results indicated that over this pressure range the maximum smoke-free fuel flow is inversely proportional to the pressure. The purpose of the work was to obtain a more complete relation between smoking tendency and pressure by extending the pressure range to 20 atmospheres for the two fuel types ethane and ethylene.

The investigation was conducted to obtain a more comprehensive understanding and evaluation of the effectiveness of reducing smoke formation by varying the diffusion processes between the fuel and oxygen. The effect of diffusion processes on smoke formation was studied by determining the maximum relative rate at which eight pure gaseous hydrocarbon compounds could be burned from a 9-millimeter burner tube without smoking when air flow passed the flame at rates which were increased by gradual steps. In addition to air, mixtures of oxygen and nitrogen of increasing nitrogen enrichment were used to determine the variation in smoke formation as the oxygen concentration was changed. The effect of flame temperature was investigated by preheating the fuel and also by substituting argon-oxygen mixtures for the nitrogen-oxygen mixtures. The fuels used in this investigation were butene-1, cyclopropane, propene, pentene-1, neopentane, isobutane, ethylene, and *n*-butane. These fuels were selected because their smoking tendencies were in a range which was convenient to measure and their vapor pressures were sufficiently high to permit burning in the gas phase.

Combustion properties of hydrocarbon fuels (such as flame speed, flammability limit, ignition energy, and quenching distance) have been extensively investigated. For certain combustion processes, it is desirable to employ fuels which have more favorable combustion characteristics than the hydrocarbons but which possess similar physical properties. A class of compounds which might meet these requirements is the alkylsilanes. Consequently, several alkylsilanes were prepared; their physical properties (such as boiling point, melting point, heat of combustion, etc.) were reported. At the time these alkylsilanes were synthesized, the conditions under which they could be safely handled and stored, particularly in contact with air, were not known. A study directed toward establishing the conditions of temperature and concentration which will permit safe handling of this class of fuels is described.

Laminar burning velocity has been considered a fundamental property of combustible mixtures and, since it can be simplified as a one-dimensional steady-state problem, has been emphasized by theorists. In order to compare theory with experiment, however, it is desirable to show that burning velocity measurements are independent of the experimental technique. A nonaqueous soap-bubble method was used to measure the burning velocities of some ethylene-oxygen-nitrogen and methane-oxygen-nitrogen mixtures. Burning velocity calculations were based on high-speed schlieren motion-picture records of the flame growth and a theoretical expansion ratio. Soap-bubble burning velocity measurements were compared with measurements by other methods (Technical Note 3106).

The flame-velocity theory of Tanford and Pease, which relates flame velocity to the diffusion of free radicals from the flame zone, has led to considerable interest in flames of various hydrogen-atom concentrations. Although this flame-velocity theory of Tanford and Pease was originally developed for carbon monoxide, it was later extended to the hydrocarbon fuels. In other investigations, the flame velocities of hydrocarbons have been correlated with those changes in hydrogen-atom concentration which are produced by changes in initial temperature, hydrocarbon structure, and fuel concentration. However, it is advantageous to use carbon monoxide flames to determine the effect of hydrogen atoms on flame velocity, because the radical concentration can be easily changed by the addition of small amounts of water, without producing excessive changes in flame temperature or fuel concentration. In this research the effect on flame velocity of adding various amounts of water to a carbon monoxide-oxygen mixture was studied. The flame velocities were measured for 20 percent oxygen and 80 percent carbon monoxide mixtures containing either light water or heavy water The flame velocity increased from 34.5 centimeters per second with no added water to about 104 centimeters per second for a 1.8 percent addition of light water and to 84 centimeters per second for an equal addition of heavy water.

As a part of the overall program, it was necessary to obtain consistent minimum-spark-ignition-energy data for a number of pure fuels, some of which were to be investigated in engine combustion chambers. Minimum spark-ignition energies for 12 pure fuels were measured at reduced pressure, and the data obtained were extrapolated to 1 atmosphere. The fuels investigated included normal and cycloparaffins, olefins, carbon disulfide, and oxygenated compounds such as an alcohol, ether, propylene oxide, and tetrahydropyran; these fuels were ignited at reduced pressures by capacitance sparks of controlled duration. The minimum ignition energies obtained are related to the pressure, the quenching distance, and the maximum fundamental flame velocity of the fuel-air mixture.

Combustion-Chamber Research

The combustible content of exhaust gases is generally found by determining the concentrations of individual or classes of components and then summing the heats of combustion of each. This method is made difficult by the presence of a large number of compounds, which represent a variety of types, as well as the low concentration of individual compounds. Exhaust-gas analysis might find wider usage if it could be more easily applied, especially if an inexpensive device could be developed which would continuously indicate exhaust-gas composition in a flowing stream. The use of flow calorimetry to determine residual enthalpy appeared to meet these prerequisites. Calorimetry would not require a separate determination of the various types of combustible but would require only that these combustibles be oxidized and the resultant temperature rise be measured. Flow calorimetry was investigated as a means of determining combustion efficiency of turbojet and ram-jet combustors by measurement of the residual enthalpy of combustion of the exhaust gases. Development of a suitable calorimeter, its calibration, and its operation are described. Briefly, the calorimeter catalytically oxidizes the combustible constituents of the exhaust-gas samples, and the resultant temperature rise is measured. This temperature rise is related to the residual enthalpy of combustion of the sample by previous calibration of the calorimeter. Combustion efficiency was calculated from a knowledge of the residual enthalpy of combustion of the exhaust gas and the combustor input enthalpy.

In a preliminary attempt to gain insight into reasons for unusually high heat-transfer rates apparently encountered during unsteady or oscillating combustion, the aerodynamic heating of an oscillating surface was analyzed in Technical Note 3146. It was found that oscillations could appreciably alter the temperature of the fluid.

LUBRICATION AND WEAR

Fundamentals of Friction and Wear

The trend in jet-engine development is toward higher speeds by increasing operating temperature. Increased engine temperatures demand better lubricants and bearing materials. Molybdenum disulfide, MoS_2, is generally a very effective lubricant at high temperatures; however, very little information is available as to the effect of moisture and contaminants (such as silica and oil) on the effectiveness of MoS_2. Studies were therefore conducted with a low-speed kinetic-friction apparatus to clarify the role of these variables.

Studies of the effect of moisture (Technical Note 3055) showed that friction coefficients were high at high humidities. Wear increased as humidity was increased (probably as a result of the increase of both metallic contact and corrosion). Steel specimens were corroded by acids formed on contact of moisture with MoS_2. Other studies of the effects of contaminants (Technical Note 3111) showed that the contaminants present in commercial grades of MoS_2 do not increase friction but can adversely affect wear; for example 0.5 percent silica in MoS_2 can greatly increase wear. In room atmosphere, small amounts of oil (5 to 10 percent) in MoS_2 reduce friction below that obtained with either purified MoS_2 or with oil alone. At least 10 percent MoS_2 should be present in the oil to obtain the lowest friction.

Fretting Corrosion

Increased engine operating temperatures tend to increase fretting corrosion. Before preventive methods can be instituted, additional research is needed.

Studies were therefore made (Technical Note 3011) to measure the coefficient of friction and to determine the damage to the contact area during early stages of fretting of copper at a frequency of 5 cycles per minute. The results led to the conclusion that fretting of copper starts with the same mechanical damage that occurs during unidirectional sliding. During the early stages of fretting, high friction values accompanying the adhesion and metal transfer. After the initial high values of friction, a reduction reached a constant value approximately the same as that obtained with powdered metal contacts of either cuprous or cupric oxide. The presence of preformed cuprous or cupric oxide films on copper does not delay the occurrence of fretting but only lowers the coefficient of fretting.

The start of fretting and the cause of damage during the early stages of fretting of steel-steel combinations at low frequency were also investigated (Technical Note 3144). As with the copper specimens, fretting starts with severe adhesion between the surfaces. The adhesion varies with the material combinations as shown by the initial coefficient of friction, but is of primary importance because it precedes and initiates the other phenomena observed. In the early stages of fretting, several other wear phenomena, in addition to adhesion, occur. They include (a) plowing by protruding transferred material; (b) formation of debris (loose fragments); and (c) formation of films by compacting small particles. Fretting of powdered Fe_2O_3 compacts on each other showed that the friction coefficient was relatively constant after a few hundred cycles at a value of approximately 0.5. The fretting of steel on steel showed that the friction coefficient (0.6) was not too different from that of the Fe_2O_3 compacts. Studies of powdered Fe_3O_4 compacts on each other showed that the initial friction coefficient was low (0.3); the friction coefficient gradually increased until at 600 cycles the value of 0.5 was approximately the same as that for the Fe_2O_3 compacts. Examination and chemical analysis of the debris on the surface of the Fe_3O_4 compacts confirmed the presence of Fe_2O_3. Thus, the fretting in all three cases (1) steel on steel, (2) Fe_2O_3 on Fe_2O_3, and (3) Fe_3O_4 on Fe_3O_4, was, after a period of time, essentially that of Fe_2O_3 against Fe_2O_3 as suggested by the measured friction coefficient.

Bearing Research

The modern jet engine operates at very high speeds in order to realize its full potential. Consequently, bearing failures are increasing. Because of the importance of the cage failure problems in high-speed rolling-contact bearings, two experimental investigations were conducted: the first was to evaluate the high-speed performance of six different roller-bearing cage designs (four experimental and two conventional) and the second was to compare the merits of leaded brass and nodular iron as cage materials. Studies of the experimental cage and bearing designs (Technical Note 3001) showed that, for liquid lubricated bearings, considerable improvement in the operating characteristics, and in particular in the limiting speeds of a bearing, can be obtained by proper design. The combination of an outer race riding cage with inner race guided rollers gave the best overall performance. The better performance of this design over both the conventional inner race riding cage and the conventional outer race riding cage is the result of the relative ease of lubrication and cooling and of the adequate oil flow (inlet and exit) paths, which minimize oil churning and friction losses. The design principle emphasized in these results seems that of providing for easy flow of lubricant *into*, *through*, and *out of* the bearing. The studies showed that, for a given lubricant flow, bearing temperature could be appreciably reduced and higher limiting speeds could be attained. Comparisons of leaded brass and nodular iron as cage materials were made (Technical Note 3002) and showed that heavy wear accompanied cage slip. Nodular iron seemed to promote cage slip at DN values (product of bearing bore in mm and shaft speed in rpm) in excess of 1,200,000; in consequence, bearings with leaded brass cages showed less wear than did bearings with nodular iron cages at higher rotative speeds. Wear with the leaded brass cage could be increased markedly by inducing cage slip. These results suggest that cage materials may be a factor in reducing cage wear in a bearing operating under slip conditions. At DN values less than 1,200,000 (rotative speeds less than 16,000 rpm for 75-mm-bore bearings), wear in the bearings with nodular iron cages and with leaded brass cages was negligible.

Studies of deep-groove ball bearings under radial load at high speeds were also continued. The results showed (Technical Note 3003) that a previously developed cooling correlation for cylindrical-roller-bearing temperatures was applicable to ball bearings. A similar cooling correlation was developed for the power rejected to the oil. These correlations make it possible to predict either the inner- or outer-race bearing temperature, or the power rejected to the oil from single curves regardless of whether speed, load, oil flow, oil inlet temperature, oil inlet viscosity, or any combination of these parameters is varied.

A review of the trend of rolling contact bearings as applied to aircraft gas-turbine engines is included in Technical Note 3110. This review showed that the bearings for future higher output engines require extensive development because of the very severe requirements proposed under these conditions. The two major problems expected under extreme conditions of high speed and high temperature are: (a) cages, and (b) fatigue of materials for races and rolling elements of rolling-contact bearings. Some general approaches to the problem are discussed. The tool steels appear to show some promise for the temperature conditions as a material for races and rolling elements. Considerable research is, however, required on these materials in full-scale applications.

COMPRESSORS AND TURBINES

Compressor Research

If the blade-element profile is to be set at the desired angle of attack at each radius in any blade row of a turbomachine, it is first necessary to know the blade-element alinement for zero "effective" angle of attack, that is, for zero loading at the inlet. In many cases this alinement is considerably different from the upstream relative flow direction; and, if it is not known, improper angles of attack may result. A method was developed for estimating the effect of blade-thickness taper on the inlet axial-velocity distribution of an entrance rotor blade row with axial inlet, and the influence of this velocity distribution on the alinement of the rotor blade for zero effective angle of attack (i. e., zero blade loading at the nose). This alinement of the blade requires a deviation between the blade camber-line direction at the inlet and the upstream relative flow direction. The method was developed for compressible and incompressible nonviscous fluids, and results are presented for incompressible flow into a plane, two-dimensional cascade and for compressible flow into an entrance rotor blade row with tapered blades. It is concluded that, for the entrance rotor blade row investigated, blade taper has a large effect on the inlet deviation angle; whereas compressibility has a small effect, except perhaps at the hub, and the upstream relative flow direction also has a small effect (Technical Note 2986).

Recent trends toward high-pressure-ratio compressors have led to the expenditure of considerable effort toward the development of the supersonic compressor. Because some classes of these compressors involve a large reduction in annulus-flow area from entrance to exit, the calculation of the flow properties circumferentially from blade-to-blade cannot, in general, be considered on an ordinary two-dimensional basis. Annular-area reduction of the order of 2:1, or of higher orders, may occur, making it necessary to include the effect of radial variation in stream-filament height and curvature in the meridional plane in a blade-to-blade solution. An analysis of the circumferential blade-to-blade flow properties in a supersonic impeller has been made using the method of characteristics on an arbitrary stream surface of revolution. The method takes into account variable stream-filament thickness and curvature along the flow surface. Results, considering stream-filament-thickness variation alone, indicate an appreciable difference between the flow properties calculated with the plane-flow characteristic equations and those determined with the characteristics method demonstrated here. The effect of a stream-filament-thickness reduction from blade-passage inlet to exit was to reduce the relative velocity and the absolute value of the flow angle in the blade passage (Technical Note 2992).

Secondary flow is that motion of the fluid associated with the component of vorticity parallel to the direction of flow; or, for all practical purposes, it is the motion of the boundary-layer and other low-energy flow in directions different from the main flow. Flow-visualization studies, along with experimental and analytical investigations on both compressors and turbines, show that secondary flows are responsible for regions of low-energy air that cause flow blockage and deviation from design flow angles and thus reduce the efficiency and performance ratings of the turbomachines.

Secondary-flow tests were conducted on an accelerating elbow with 90° of turning designed for prescribed velocities that eliminate boundary-layer separation by avoiding local decelerations along the walls. Secondary flows were investigated for six boundary-layer thicknesses generated on the plane walls of the elbow by spoilers upstream of the elbow inlet. The passage vortex associated with secondary flows appears to be near the suction surface and away from the plane wall of the elbow at the exit and does not have appreciable spanwise motion as it moves downstream from the elbow exit. As the spoiler size increases, the boundary-layer form changes and a rather sudden difference in the secondary flow occurs, perhaps associated with the reduced importance of viscous effects in thick boundary layers. It is suggested that the strength of the secondary vortices is small and that the energy of secondary flows is small (Technical Note 3015).

Because secondary flows are an important source of loss, considerable analytical work has been done on this phenomena in stationary curved channels. The vorticity component has also been computed for an axial-flow rotating channel. In this investigation, which extends the work to rotating radial channels, the purpose was not to obtain exact data but rather to obtain some insight into the problems of secondary flows in these channels (as in centrifugal impellers). The results of this analysis are indications of the qualitative trends of variables that affect the secondary vorticity and therefore the secondary flow. The secondary vorticity decreases with decreased absolute angular velocity of the fluid, decreased inlet total-pressure gradient, decreased length of relative flow path, and increased relative velocity (Technical Note 3013).

The flow distribution throughout the passage of a rotating 48-inch radial-inlet centrifugal impeller has been studied in detail. It was indicated that secondary flows within the boundary layer and leakage through the blade-to-shroud clearance space resulted in a concentration of low-energy air at approximately 80 percent of the passage width from the pressure face at the shroud.

Comparison of the data obtained from the internal measurements made for the impeller of this investigation with hot-wire anemometer studies made at the impeller outlet of a similar impeller indicates that much can be learned about the internal flow picture with hot-wire surveys alone (Technical Note 3101).

Blade Vibration and Flutter

The destruction of compressors due to blade vibrations has led to investigations to determine what the aerodynamic forces are that act on blades when they vibrate and to determine if the aerodynamic forces are sufficient to cause the blades to vibrate or flutter. Preliminary measurements were therefore made of the oscillatory lift force acting on an airfoil vibrated in bending. Results were obtained for an isolated airfoil and for the same airfoil oscillated in a cascade, at low and high angles of attack. It was found that, at high angles of attack and at low values of the reduced frequency, the damping for the isolated airfoil can become negative. This would cause the airfoil to flutter. The oscillating lift force changes little, for the case considered, by placing this blade in a stationary cascade. It is indicated that for this case the effect of the cascade is generally to increase the damping by a slight amount.

Turbine Research

An investigation was made of losses and secondary flows in three different turbine-nozzle configurations in annular cascades. Appreciable outer-shroud loss cores (passage vortices) were found to exist at the discharge of blades which had thickened suction-surface boundary layers near the outer shroud. Blade designs having thinner boundary layers did not show such outer-shroud loss cores but indicated greater inward radial flow of low-momentum air in the blade wake and resulted in greater contribution to inner-shroud loss regions. The blade wake was a combination of profile loss and low-momentum air from the outer shroud and the magnitude of the wake loss is, to this extent, an indication of the presence or absence of radial flow. At a higher Mach number, shock-boundary-layer thickening on the blade suction surfaces provided an additional radial-flow path for low-momentum air and caused large inner-shroud loss regions accompanied by large deviations from design values of discharge angle (Technical Note 2989).

Turbine Cooling

It is necessary to cool the various parts of turbojet, ram jet, and rocket engines exposed to hot-gas flows to temperatures the materials can safely withstand. The use of both air and liquids for cooling these critical propulsion components is currently under consideration; the present discussion, however, is restricted to the use of air.

Calculations were made for the special case of a constant gas velocity along a cooled flat plate to determine the relative effectiveness of the three cooling methods considered (convection, transpiration, and film cooling). Air was used as the coolant as well as the outside flow medium. Calculations indicated the superiority of transpiration cooling for both laminar and turbulent flow. The superiority was reduced when the effects of radiation were included. For some cooling applications, however, there is evidence indicating that radiation may be neglected (Technical Note 3010).

An investigation was conducted to develop simple, inexpensive electric analogs for determining temperatures of cooled turbine blades. Analogs were made for an air-cooled 13-fin shell-supported blade, an air-cooled strut-supported blade, and a liquid-cooled blade. The accuracy of these analogs was determined by comparing the values of blade temperature obtained with values calculated by analytical methods. In general, good agreement was achieved (Technical Note 3060).

An approximate method was developed for the rapid determination of pressure and Mach number change for subsonic flow of a compressible fluid under the simultaneous action of heat transfer, friction, rotation, and area change. In the development of this method, the momentum equation was approximated and rearranged. Charts were prepared for convenience in a step-by-step integrated solution of the momentum equation. The solution converged rapidly and gave good accuracy with four steps. A linear and an exponential air-temperature variation along the duct length was considered with negligible effect on the calculated pressure change through the passage (Technical Note 3150).

In order to provide heat-transfer and friction coefficients for surfaces and flow conditions of interest to the design of present and future propulsion systems, tests and analyses must be made to predict the heat transfer and pressure drops at high surface temperatures and high heat flux rates. Research must be conducted for both conventional and unconventional fluids in smooth round tubes, roughened tubes, noncircular ducts, and entrance regions of ducts, parallel plates, and extended surfaces.

The effect of flow-passage shape on the heat-transfer and friction coefficients for air flowing through noncircular ducts at high heat flux conditions was studied. Measurements of average heat-transfer and friction coefficients were obtained with air flowing through electrically heated ducts having square, rectangular (aspect ratio, 5), and triangular cross sections for a range of surface temperature from 540° to 1,780° R and of a Reynolds number range from 1,000 to 330,000. The results indicate that, if Nusselt and Reynolds numbers are based on the hydraulic diameter of the duct, the data for the noncircular ducts could be represented by the same equations obtained in a previous investigation for circular tubes. Correlation of the average difference between the surface corner and midwall temperatures for the square duct was in agreement with predicted values from a previous analysis. However, for the rectangular and triangular ducts, the measured corner temperature was greater by approximately 20 and 35 percent, respectively, than the values predicted by analysis.

Heat-transfer and pressure-drop data for air flowing in short length-to-effective-diameter-ratio passages is needed to provide information for heat exchangers in which successive stacks of plates are used. The variables being studied are distance between parallel plates, gap spacing, and degree of misalinement between plates in successive stacks. Forced convection heat-transfer and pressure-drop data were obtained for plates of short length-to-effective-diameter ratio. Two such stacks were alined and misalined in the direction of air flow with gap spacings between stacks of $\frac{1}{32}$, $\frac{1}{8}$, and $\frac{1}{4}$ inch. Data were obtained with heat addition to the downstream stack only over a range of Reynolds numbers from 15,000 to 80,000 and average surface temperatures of about 680° R. The average and local heat-transfer coefficients were only slightly higher than predicted values from established round-tube data.

An analytical study of turbulent heat transfer and flow in the entrance region of smooth passages gave values of Nusselt number for air with a uniform wall temperature, uniform initial temperature, and velocity distributions that agreed closely with experimentally determined values (Technical Note 3016). A previous analysis for fully developed turbulent heat transfer and flow with variable fluid properties was extended and applied to the entrance regions of smooth tubes and parallel plates. Integral heat transfer and momentum equations were used for calculating the thicknesses of the thermal and flow boundary layers. The effect of variable properties was determined for the case of uniform heat flux, uniform initial temperature distribution, and fully developed velocity distribution.

An analysis was also made of turbulent heat transfer, mass transfer, and friction in smooth tubes for fluids with high Prandtl and Schmidt numbers (Technical Note 3145). The expression for eddy diffusivity from a previous analysis was modified in order to account for the effect of kinematic viscosity in reducing the turbulence in the region close to the wall. Use of the modified expression gave good agreement between predicted and experimental results for heat and mass transfer at Prandtl and Schmidt numbers between 0.5 and 3,000. The effects of length-to-diameter ratio and of variable viscosity were also investigated for a wide range of Prandtl numbers.

The results of experimental and analytical studies performed at the NACA to determine the heat-transfer and friction coefficients for the flow of air through tubes with a large difference in temperature between tube wall and air, for smooth tubes of circular cross section, for tubes of noncircular cross-sectional shapes, and for tubes with various degrees of surface roughness are summarized. The experiments for the smooth tubes of circular cross section covered a range of tube-wall temperatures from 535° to 3,050° R, inlet-air temperatures from 535° to 1,500° R, Reynolds numbers from 1,000 to 500,000, exit Mach numbers up to 1, and tube length-to-diameter ratios from 15 to 120. The tubes of noncircular cross-sectional shape were investigated at tube wall-to-air temperature differences up to 1,200° F and Reynolds numbers between 2,500 and 250,000. Three degrees of surface roughness, obtained by machining square threads into the inner surface of the tube, were investigated for temperature differences between the air and tube wall up to 1,500° F, and Reynolds numbers from 1,000 to 350,000.

In order to predict the effective thermal conductivity of a powder from the properties of the solid and gas which make up the powder, analytical and experimental work has been undertaken. Previously an analysis and test results were reported for magnesium oxide powder in various gases. As a continuation of the general investigation of the effective thermal conductivities of powders, tests were conducted to determine the conductivities of magnesium oxide, stainless-steel, and uranium oxide powders in gases such as air, helium, and argon at temperatures between 120° and 1,455° F. Fair agreement was obtained between conductivities calculated from experimental data for fine magnesium oxide and stainless-steel powders and those calculated from a simplified analysis from a previous investigation, although the experimental values are somewhat higher. Tests were also made to determine the effect of gas pressure on effective thermal conductivity.

Heat Transfer

In such technically important problems as the cooling of turbine rotors and heat extraction from atomic piles, heat transfer is generally accomplished by a combination of forced and natural convection. Most theoretical treatments of this type of heat transfer have, in the past, been semiempirical in nature, or have neglected factors which, in modern application, can have sizable influence on the results. In Technical Note 3141, the case of combined forced and natural convection in a channel with linearly varying wall temperature is treated in detail without restrictions on the magnitude of convective velocities. Representative velocity and temperature profiles are presented and the effects of aerodynamic heating on the flow and heat transfer are discussed.

ENGINE PERFORMANCE AND OPERATION

Performance and Operating Characteristics

Ammonia Injection Into Air Stream.—Turbojet-engine thrust can be augmented by cooling the air entering or passing through the compressor. One of the basic requirements of an inlet-coolant injection system is that it distribute the coolant as evenly as possible in the inlet air stream in order to provide a uniform temperature at the compressor inlet. The penetration and cooling characteristics of the injected liquid must therefore be known before an adequate injection system can be designed. In order to provide the needed information, an investigation was conducted at the NACA Lewis Laboratory to measure and to correlate the isothermal contours formed by the penetration of a single jet of liquid ammonia directed normal to an air stream.

Data for this investigation were obtained at several distances downstream of the point of injection for several orifice diameters over ranges of ammonia-to-air velocity, density, and temperature ratio. A correlation of the temperature profile in a plane containing the axis of the ammonia jet was reported together with a correlation of the maximum width of the isothermal contour map formed by the penetration of the jet of liquid ammonia into the air stream.

Unconventional Power Plants.—A method of utilizing aerodynamic heating at high speeds to produce propulsive thrust is described in Technical Note 3140. This method consists of using coolants that will vaporize in circulating past the heated aircraft surfaces. The resulting gaseous coolant is then ejected rearward as in conventional rocket propulsion. Analysis showed that thrust greater than the friction drag of the cooled surfaces could be attained for some types of coolant. At hypersonic speeds, with hydrogen as the coolant, propulsion efficiencies comparable with those of conventional rocket engines are theoretically attainable. For use as an auxiliary power source, coolant vaporization can produce specific impulses comparable with those of current rocket propellants at all Mach numbers.

Propulsion Systems Analysis

Component Matching.—Optimum performance of turbojet-powered aircraft over the required range of flight speeds and altitudes can only be achieved if each component of the power-plant installation is properly matched with preceding and subsequent components. The turbojet engine and the inlet system supplying the engine air are two such components. A method of representing inlet air-flow capacities by the same parameter used to represent engine air-flow requirements is presented in Technical Note 3012. This method simplifies the problem of determining inlet operating conditions at the match point. The method has proved useful for the determination of inlet-geometry variations required to improve power-plant performance of supersonic aircraft.

Ducted-Airfoil Ram Jet.—An analysis of the ducted-airfoil ram jet for supersonic aircraft was conducted by the Langley Pilotless Aircraft Research Division several years ago. The practical advantage to be gained in a configuration where the ram jets are housed in the wing or tail rather than in the fuselage is that the fuselage space can be devoted exclusively to the housing of cargo, fuel, and controls. The possible range and acceleration performance were determined for aircraft with a fuselage consisting of a parabolic body of revolution with a fineness ratio of 10 and ducted wing or tail airfoils of various sizes relative to the fuselage size.

Power Systems

The practicability of powering helicopters with blade-tip jet units has been demonstrated in flight with ram-, pulse-, and pressure-jet systems. As part of a general program to evaluate in detail the overall rotor performance and burner characteristics as affected by centrifugal forces, tests have been conducted on the Langley helicopter test tower. The first series of completed tests on conventional circular-type ram-jet units were reported. The detrimental effect of high centrifugal loading on burner performance was reduced by increasing the blade radius from 9 feet to 18 feet (a reduction in gravitational units from 1,500 to 750 at a tip speed of 630 fps), and a 12.5-percent increase in maximum thrust and a 19-percent decrease in minimum specific fuel consumption resulted. The importance of blockage of internal air flow during power-off operation is shown, as well as a method for determining the net propulsive thrust of the ram-jet unit tested on a free-jet thrust stand and on the tip of a helicopter blade.

POWERPLANT CONTROLS

Control of Turbine Engines for Helicopters

The control of the speed of a gas turbine geared to a propeller involves controlling the speed of a single rotating body with damping forces from the engine and the propeller. When the propeller is replaced by a helicopter rotor, however, torsional flexibility between the gas turbine and the helicopter rotor may be sufficient to establish torsional oscillations. The investigation reported in Technical Note 3027 was conducted to determine whether the torsional oscillations in helicopter rotors have a detrimental effect on the control of gas-turbine engines to which they are geared. The effects of helicopter size and rotor weight on rotor dynamics were investigated, and the control characteristics of a gas-turbine engine in a large helicopter were studied. The control systems considered employed speed and torque control by fuel flow.

Analysis of Control Systems

A basic characteristic of turbojet engines is that a change in fuel flow or exhaust-nozzle area causes both speed and temperature to change. Whenever these engine parameters are used in a double-loop control configuration, a disturbance in one loop will introduce an error signal into the other loop, which is referred to as interaction between the individual control loops. The investigation reported in Technical Note 3112 was made to determine some of the practical aspects of noninteracting systems and to compare these with an interacting configuration. Stability limits and response characteristics were obtained for one basic double-loop system and also for several modifications of the system. An analog computer was used to simulate a current turbojet engine with variable exhaust nozzles along with the necessary sensor and servo components of the engine control.

The recent application of the high-speed, high-output hydraulic servomotor in both the aircraft and the industrial fields has created a need for an analysis of the dynamics of the pressure-generating equipment. The transient response of the pressure-regulating relief valve in a hydraulic circuit was analyzed by means of an electrical analogy of the hydraulic circuit and is presented in Technical Note 3102. Measurements of the transient response of a hydraulic relief valve are presented and compared with responses calculated from the differential equation of the equivalent electrical network. The comparison of experimental and analytical responses shows that the response of the relief valve can be adequately predicted by means of this network.

HEAT-RESISTING MATERIALS

High-Temperature Materials

The turbine blade continues to limit the power output and effective life of jet engines. Current work in progress is concerned with (a) improving the reliability of the turbine blade, thereby increasing its effective life; and (b) extending the maximum operating temperature of the blade, thereby increasing the power output of the engine.

The alloys currently used or considered for use in turbine blades have existed for several years, and it appears that the empirical approaches to their improvement (from both the maximum-use temperature or reliability standpoints) are rapidly being exhausted. However, it is felt that further improvement is possible if a greater insight is gained into the fundamental behavior of these alloys.

Alloys used in jet engines are complex mixtures of chemical compounds and solid solutions that are reacting with each other at high operating temperatures. The life of the alloy depends on the specific compounds formed and the rates at which the constituents react with each other. Heat treatment of the blades is desirable in order to put these constituents into their most stable form, so that reactions during operation will be minimized. Before this can be accomplished, the chemical structure and the physical form of the constituents should be identified. This has been done (Technical Note 3109) for eleven cobalt-base alloys. Six different carbides as well as oxides, nitrides, and the sigma phase were identified. Some of these microconstituents exist as a lamellar or platelike precipitate. The effect of certain alloying elements on the quantity and stability of this phase was also studied.

The structural constituents having been identified, the effect of various heat treatments on alloys was studied using a typical cobalt-base alloy (Technical Note 3107). The microconstituents could be dissolved at a high temperature and then reprecipitated in a controlled manner by either isothermal or aging heat treatments. The relation of the microstructures developed to the long-time elevated temperature strength at jet-engine stress conditions was also investigated (Technical Note 3108). It was found that the optimum high-temperature properties are associated with a dispersion of fine precipitate scattered throughout the grains of the microstructure. Such a structure is obtained by (a) a heat-treating cycle consisting of a solution treatment to produce a homogeneous solid solution, (b) aging at a temperature low enough to produce scattered nucleation sites without permitting the growth of large particles, and (c) aging a second time at a temperature slightly above that of the first aging in order to complete the precipitation at the scattered nucleation sites, thereby stabilizing the alloy.

In addition to understanding the chemical behavior of the alloy, it is important to understand the behavior of the alloy when subjected to complex stresses. Such stresses are created by surface conditions imposed by finishing operations or notches which may be inherent in the design or result from scratches caused by machining. These stresses are superimposed on the normal tensile and vibratory loading to which the turbine blades are subjected.

In one study (Technical Note 3142) of the behavior of the alloys N-155 and S-816 it was found that the stress imposed by finishing operations may increase the fatigue strength of these alloys. However, if these stresses were removed, surface roughness lowered the fatigue strength at temperatures of 1,500° F by as much as 10 percent.

In conjunction with the Gas Turbine Panel of the joint A. S. T. M.–A. S. M. E. Committee on Effects of Temperature on the Properties of Metals, the effects of notches on the fatigue behavior of notched bars of N-155 were studied in the temperature range of 1,350° to 1,500° F. The alloy was sensitive to the presence of notches with reductions in strength ranging from 34 to 40 percent at 1,350° F and 30 to 37 percent at 1,500° F.

An alternate method of improving jet-engine reliability and increasing maximum operating temperature is to protect marginal materials from the deteriorating effect of the highly oxidizing atmosphere and to insulate them from the high temperature through the use of ceramic coatings. The status of the development of such protective coatings and the components to which they are applicable was reviewed.[20]

Research data published in the last several years on the properties of molybdenum show that this metal has an excellent potentiality for high-temperature applications. Molybdenum, however, has two undesirable characteristics, namely, poor oxidation resistance and difficult joining properties. Joining by welding has to date been none too successful although much work has been done in this field. Since little work had been done on joining of molybdenum by brazing, a program was initiated to develop a satisfactory brazing alloy for molybdenum for elevated-temperature applications. The brazing characteristics of 28 alloys with liquidus temperatures in the range of 2,000° to 2,500° F were established in vacuum. The tensile strengths of molybdenum joints bonded with two of these alloys—one of 84 percent nickel plus 16 percent titanium, and another of 52 percent niobium plus 48 percent nickel—gave excellent results (Technical Note 3148). Therefore, these two alloys may be potentially useful for brazing molybdenum for service application.

Cermets continue to remain promising as a turbine-blade material for high operating temperatures. A method of fabrication which has been termed "freeze-

[20] See Francisco and Ault paper listed on p. 64.

casting" has been developed. Refractory powders can be cast to shape, thereby eliminating the machining operations which are present in the conventional cold-pressing and sintering methods. The method consists in preparing an extremely rich slip of such materials as titanium carbide with a small amount of binder, casting the slip into a mold, and freezing to retain the shape of the casting. The casting is then dried by sublimation and subsequently sintered by conventional methods.

One factor which greatly affects the reliability of turbine blades is their ability to withstand the impact of foreign particles which may pass through the turbine. This is a particularly severe problem for cermets because of their brittleness and the susceptibility of adjacent blades to damage from a failed blade. A method of measuring the impact resistance of cermets was developed which yields results which are more reproducible and fundamentally more significant than those obtained with the more conventional metallurgical methods. The energy to cause fracture is compared directly with the measurement of kinetic energies of the test pieces.

A metal-oxide system was evaluated and found to have adequate thermal shock resistance for turbine operation. The strength, however, was marginal, possibly because fabrication techniques had not been perfected.

Because of its excellent resistance to oxidation and its outstanding strength at elevated temperatures, molybdenum disilicide, $MoSi_2$, has promise as a material for high-temperature application. The poor resistance of $MoSi_2$ to thermal shock limits its application. On the premise that thermal-shock resistance can be increased by the introduction of a ductile metal binder, a series of evaluations was conducted to determine the effect of metal additions on the properties of $MoSi_2$. The addition of 6 percent nickel, cobalt, or platinum resulted in a lowering of the modulus-of-rupture strength and the high-temperature oxidation resistance without any significant change in thermal-shock resistance. The metal additions reacted with the disilicide to form complex silicides so that sufficient ductile binder was not provided.

In the search for satisfactory container materials for molten sodium hydroxide, the lack of agreement between the results obtained by various investigators has been a source of considerable annoyance. It was felt that the difficulty was due to lack of recognition of the factors involved. Therefore, an investigation was conducted on nickel to determine the effect of temperature level, temperature gradient, and test duration on corrosion and mass transfer by molten sodium hydroxide under free-convection conditions. A base temperature range from 1,000° to 1,600° F with temperature differences to 500° F was studied. The rate of mass transfer was strongly dependent on both temperature level and gradient. The rate showed little tendency to decrease for test durations up to 200 hours, although the concentration of nickel in the melt approached a limiting value after 100 hours.

Stresses Research

Thermal stresses in the steady states have acquired increasing interest for aircraft: in ductile material applications, as in turbine disks and blades, and in brittle material applications, as in nozzle diaphragms, combustion-chamber linings, high-temperature coatings, and heating elements.

To better understand the heating element problem, an analytical investigation was conducted to determine the effect of temperatures, thermal stress, and shock plates of constant conductivity and of conductivity that varies linearly with temperature. As a result, working formulas were derived that give an insight into steady and transient temperature and stress mechanisms involved (Technical Note 2988).

The problem of correlating and extrapolating stress-rupture data of use for engine components has recently received considerable attention. Because of the large number of materials currently of interest for gas-turbine applications, and the difficulty of obtaining long-time data for all these materials, an extrapolation procedure is of great value.

Several parameters have been proposed to permit correlation and extrapolation of creep and stress-rupture data. On the basis of analysis of published isothermal data it was concluded that a linear parameter yielded the best correlation of the available data. In order to more directly evaluate this parameter and to compare its results with those obtained by other methods, an experimental investigation was made in which rupture data at constant nominal stress were obtained for five commercial high-temperature alloys. The results of this investigation confirmed the previous conclusions regarding the suitability of the linear parameter and also permitted a formulation of a more general parameter in terms of time, temperature, and stress.[21]

The investigation of the influence of stress concentrations was expanded to include a determination of the effects of strain-hardening characteristics and notch sharpness on notch tensile properties at room temperature. As the notch sharpness increased, the true stress-strain curve was elevated and the ductility decreased. It was also determined that a higher strain-hardening rate results in (a) a lower rate of triaxiality increase with sharpness, (b) a lower amount of triaxiality developed, and (c) a higher sharpness at which the maximum triaxiality is developed.[22]

The recent interest in the effects of stress concentrations on high-temperature alloys has brought about a number of publications, both in this country and in

[21] See Manson and Brown paper listed on p. 65.
[22] See Schwartzbart and Brown paper listed on p. 65.

Europe, covering the many variables associated with notch-rupture testing. A critical review of the published results was undertaken to correlate the effects of time and temperature in the test, the notch geometry, alloy composition, and heat treatment.[23]

Physics of Solids

By means of fundamental studies of the properties and nature of solids, an insight may be obtained that will lead to practical improvements in their use. One of such fundamental studies concerned the nature of the surface on the creep characteristics of zinc. The electromotive force produced by a given metal electrode is known to be extremely sensitive to the surface state of the metal and to the amount of cold work which it has undergone. Experiments carried out indicated that the creep rate of a zinc single crystal is increased if the surface is slowly electropolished during creep, and that a reversal of the current will cause the creep rate to decrease. Studies in which the creeping zinc crystal was one electrode in a cell showed that a sharp rise in electrode potential results when elongation begins. The sharp rise is followed by a gradual decay to a value which was different from that observed before loading.

An experimental investigation of the energy state of pure copper which has been fatigued almost to fracture at room temperature was also carried out. Calorimetric measurements were made of the heat necessary to raise the temperature of the fatigued copper samples from room temperature to 450° C. About 0.4 more calorie per gram was required to bring a fatigued sample from 250° to 400° C than was necessary for the same sample after it had been subjected to a temperature of 450° C. These results are in sharp contrast to those obtained for cold-worked copper where about 0.3 calorie per gram is released between 150° and 250° C, a temperature range within which the fatigued sample apparently neither released nor absorbed energy.[24]

Work on color centers in alkali halide crystals has been continued. Previous studies had been conducted to determine the free sodium content of NaCl crystals containing color centers. As a further step toward the understanding of the mechanism involved in establishing color centers in NaCl, measurements have been made of the free-chlorine content of NaCl crystals subjected to electrolysis before and after exposure to X-rays, as well as the free-chlorine content of normal NaCl crystals after irradiation. Crystals which have not been irradiated contain no free chlorine, whereas those exposed to X-rays contain between 0.1 and 0.6 free-chlorine atom per vacancy pair.[25]

Preliminary studies of the effect of radiations on the properties of materials were also continued. Several subjects were of interest. One of these was a method developed for studying multiple scattering of light particles by the use of a magnetic cloud chamber. This method has been applied to both electrons and positrons in the momentum range between 2,000 and 9,000 Gauss-cm in argon in 1 and 2 atmospheres pressure. The root-mean-square angle of scattering for positrons has been found to be a few percent smaller than that for electrons. The energy dependence established experimentally was smaller than that predicted by current theories of multiple scattering. The accuracy of the method when applied to the determination of the energy and momentum of a set of identical particles, the nature of which is unknown, is also discussed.[26]

In addition, an experimental study was conducted to determine the suitability for measuring ranges and cross section of a low-pressure cloud chamber with an atmosphere consisting of hydrogen, ethyl alcohol, and water. Sharp tracks were obtained in the time interval of 0.014 second between 90 and 99 percent completion of expansion. Calculations indicated that in this time interval the changes that could take place in the composition and density of the gas in the cloud chamber would introduce only negligible errors into the measurements and, hence, the low-pressure cloud chamber tested could be used without modification to obtain accurate values of ranges and cross section.

A knowledge of the energy band structure for the motion of an electron in a crystal yields considerable information about such properties of solids as electrical conductivity and cohesive energy. However, the determination of this band structure cannot be obtained by exact methods. The correctness of the assumptions involved in a particular method may be estimated by applying the method to a one-dimensional model which can be solved exactly and comparing the results of the approximate method with those of the exact method. In accordance with these considerations, an exact evaluation of the energy band structure for a one-dimensional, periodic, square-well potential has been made in terms of the well depth for the entire range of possible ratios of well width to hill depth. The results indicate that, as the potential depth was varied for a fixed ratio of well width to hill width, the curves bounding distinct bands cross. The location of these crossings was derived and the number of times that a given pair of boundary curves can cross was considered.[27]

ROCKET ENGINES

Propellants

Concentrated solutions of nitric acid containing oxides of nitrogen are commonly used in many propellant and chemical applications. Certain properties, such as vapor pressures and boiling points, are of considerable interest. A recent literature survey indicated that

[23] See Sachs, Brown, and Newman paper listed on p. 65.
[24] See Welber and Webeler paper listed on p. 66.
[25] See Hacskaylo, Otterson, and Schwed paper listed on p. 64.
[26] See Groetzinger paper listed on p. 64.
[27] See Allen paper listed on p. 63.

data are lacking on vapor pressures of ternary systems of nitric acid, water, and nitrogen dioxide at temperatures greater than 25° C. To obtain more complete vapor-pressure data, an investigation was made to determine the total vapor pressures of the ternary systems of nitric acid, water, and nitrogen dioxide within the temperature range of 20° to 80° C and within the composition range of 83 to 97 percent nitric acid, 0 to 15 percent water, and 0 to 6 percent nitrogen dioxide. The total vapor pressures of 28 acid mixtures were reported. The data were obtained by use of an isoteno-scope. The experiments were planned with a view toward establishing a relation between total vapor pressure and composition of the acid solution.

The data reported in the literature on the vapor pressures of the system nitric acid, nitrogen dioxide, and water are incomplete with respect to the composition range of interest. A study was undertaken to provide the necessary information. Total vapor pressures were measured for 16 acid mixtures of the ternary systems of nitric acid, nitrogen dioxide, and water within the temperature range 10° to 60° C and within the composition range 71 to 87 weight percent nitric acid, 7 to 20 weight percent nitrogen dioxide, and 1 to 10 weight percent water. Heats of vaporization were calculated from the vapor-pressure measurements for each sample for temperatures of 25°, 40°, and 60° C.

Rocket Combustion

Methods for the instantaneous measurement of combustion parameters to assist in studies of oscillatory and transient combustion conditions in rocket engines have been under investigation. Technical Note 3033 describes an experimental investigation of the use of a two-color electro-optical pyrometer as a means for measuring rapid changes in combustion temperatures. The development and application of an electro-optical two-color pyrometer for the measurement of temperature and emissivity in the exhaust gases from an open-tube combustor using liquid oxygen and a hydrocarbon fuel are described. Measurements were made during both normal and oscillatory combustion. Satisfactory correlation was obtained between temperatures measured simultaneously by the two-color method and by a microwave absorption technique. Agreement between temperature and sound intensity change was also found to exist. During many combustion experiments, radiation intensity varied inversely with temperature and directly with emissivity, indicating a need for caution in the interpretation of combustion photographs on the basis of light intensity. However, on the basis of response to temperature changes, the review of existing techniques for the measurement of temperature indicated that this method was the most desirable of those applicable to rocket engines.

AIRCRAFT CONSTRUCTION

The need for increased research in the field of aircraft construction continues. High-speed flight of modern aircraft and missiles, made possible by advances in aerodynamics and propulsion, has brought the designer to the realization that aerodynamic heating, which was believed to be a problem of the future, needs solving today. Aerodynamic heating has complicated many problems, such as aeroelastic deformation, vibration and flutter, air loads, and choice of construction materials. The NACA is attempting to re-solve these problems in their more complicated forms through increased research in the elevated temperature field. While much effort is being expended on these old but newly complicated problems, the NACA continues its research on other equally important problems, such as fatigue, structural design methods, gust loads, buffeting, and landing loads.

To complement the work under way at the NACA laboratories, a portion of this research was and is being performed under contract at universities and other nonprofit organizations. A description of the Committee's recent unclassified research in the field of Aircraft Construction is given on the following pages and is divided into four sections: (1) Aircraft Structures, (2) Aircraft Loads, (3) Vibration and Flutter, and (4) Aircraft Structural Materials.

AIRCRAFT STRUCTURES

Static Properties

For aerodynamic reasons, wings of supersonic aircraft should be thin. This requirement confronts the aircraft-structures designer with the very difficult problem of providing such wings with adequate strength and stiffness and yet maintaining high structural efficiency. Multiweb wing structures with relatively thick skins and light internal structure are more efficient than the more conventional skin-stiffener arrangement for highly loaded thin wings. In Technical Note 3082, the results of tests of 53 multiweb beams of various proportions are presented. The beams, fabricated from 75S–T6 aluminum-alloy sheet material, had channel-type webs which had been cold formed with bend radii of four times the web thickness. Local and wrinkling modes of buckling were observed prior to failure. All failures occurred with the formation of a trough in the compression skin extending across the web attachment flanges. The stress levels achieved at buckling and failure are discussed in terms of existing theory. Based upon the failure stresses, design charts are presented which permit rapid selection of the most efficient proportions for given values of an appropriate structural index.

Using an electrical analog computer, the California Institute of Technology has analyzed a number of sweptback wings, straight multicell wings, and multicell delta wings. Deflections and all internal forces were calculated and the vibration modes obtained. The effects of shearing strains in the ribs and spars were also determined. A description of the analyses of the sweptback wings is presented in Technical Note 3115, of the straight multicell wings in Technical Note 3113, and of the multicell delta wings in Technical Note 3114.

A variation of the multiweb structure is that in which the continuous webs are replaced by posts along spanwise lines. Such multipost structures are considered as a means of simplifying fabrication problems and of providing an open interior in which internal stores (such as arms or fuel) are more readily carried. In Technical Note 3118, the results of a computational program are presented which gave numerical values of the stiffnesses required of the various components of a multipost-stiffened wing to achieve desired buckling-stress values under bending loads. Two arrangements of the posts were considered, upright posts and posts used as diagonals of a Warren truss. This work extends and summarizes similar calculations previously published by the NACA.

Cutouts produce highly distorted stress distributions and thus make structures very vulnerable to failure by fatigue; moreover, if the cutouts are large, they produce marked reductions in the stiffness of the structure, especially in wings. In Technical Note 3061, a method is presented for calculating the effects produced by rectangular cutouts of any size in torsion boxes such as wings. Key stresses are obtained by a simple theory; detailed stress distributions are then estimated with the aid of simple rules or empirical curves. Comparisons are made with the results from three series of tests in which the dimensions of the cutouts varied over a wide range.

In order to furnish experimental verification for theories being developed for cutouts in fuselages, strain measurements have been made on a systematic series of cutouts (with openings from 30° to 130°) in stiffened circular cylinders. The results are presented in Technical Note 3039 for pure torsion, Technical

Note 3073 for pure bending, and Technical Note 3192 for shear load.

Because of their high strength-weight ratios, sandwich panels are finding increased use in aircraft structures. A number of bonded sandwich panels composed of two aluminum-alloy sheets separated by aluminum-foil honeycomb cores were tested under combined axial load and lateral pressure by the National Bureau of Standards. The results of these tests are presented in Technical Note 3090. Values of lateral deflections and axial strain computed from extensions of an existing sandwich-column theory were compared with those obtained experimentally. In most cases the theory was conservative in predicting larger strains or deflections than those measured. This discrepancy is attributed to the fact that the theory does not take into account the anticlastic bending which was observed in the panel tests.

Integral construction is that in which stiffening elements are machined, pressed, or rolled on a sheet or plate structural element. By contrast, in nonintegral construction, stiffening elements are riveted, welded, or otherwise joined mechanically to the plate. Integral construction offers a substantial increase in structural efficiency and a decrease in assembly time. In an investigation of the buckling strength of plates with wafflelike stiffening under combined stresses, theory and experiment were compared and were found to be in good agreement. The results of the investigation are presented in Technical Note 3059. For the variety of configurations of wafflelike stiffening investigated, the 45° stiffening was found to be the most effective over a wide range of combinations of compression and shear loading.

An investigation has been made by the National Bureau of Standards of the compressive strengths of stiffened flat-sheet panels to include the range where ultimate strengths approach the compressive yield strengths of the materials. The results of these tests indicated that the ultimate strengths of the panels may have been limited by the strengths of the rivets. The results of this study are presented in Technical Note 3023.

The materials from which aircraft structures are made are subject to change as new alloys are developed and as design conditions require new and stronger materials. In order to make allowance in design for the changes in material properties, correlation must be effected between material properties and structural strength. Heretofore, such correlation has been restricted to relatively simple structural shapes or to small changes in properties for more complex assemblies such as stiffened panels. In Technical Note 3064, results are presented of a study of the effect of large variations in material properties on the compressive strengths of stiffened panels. Flat skin-stringer compression panels of stainless steel, mild steel, titanium, copper, four aluminum alloys, and a magnesium alloy were tested. The results show the effect of variations in yield stress, Young's modulus, and both yield stress and Young's modulus for constant yield strain on the buckling and load-shortening characteristics of the panels.

In an attempt to compare bonded construction with riveted construction, the National Bureau of Standards performed a number of static tests on riveted and bonded sheet-stringer panels at room temperature. The test results, which are presented in Technical Note 3215, indicated that static-strength properties of both types of construction are comparable and that the choice in any given case would depend upon the particular designs being compared. The tests also showed that the scatter of the results obtainable with bonded construction was not significantly greater than that obtainable with riveted construction and that cleavage was not always the governing factor in the strength of bonded panels.

The basic problem of buckling (elastic and plastic) of flat-plate structural elements under combined shear and compression has been treated theoretically, but, because of the difficulty of making such tests, experimental results are meager, particularly for the plastic range. In the Technical Note 3184, the results of buckling tests of long square tubes loaded in compression, torsion, and combined compression and torsion are compared with theoretical compression- and shear-buckling curves and with theoretical interaction curves for the buckling of simply-supported flat plates. A compression-buckling curve, previously compared with experiment, was again shown to be in good agreement with experimental results; the shear-buckling curves derived from compressive stress-strain data by the secant-modulus method were in good agreement with experimental results; and the theoretical interaction curves previously presented were in good agreement with the results of the combined-load buckling tests. The direction of the loading path was shown to have little or no effect on the shape of the interaction curve.

Very short columns, for which column buckling does not occur, can show definite postbuckling strength, that is, excess strength beyond that indicated by the critical plate-buckling stress. This postbuckling strength decreases with increasing slenderness ratio and column buckling becomes the governing influence except for extremely small slenderness ratios. Cornell University has studied this effect and developed a general theory for predicting the ultimate strength in the postbuckling of columns of any shape and slenderness. Specific expressions were derived for columns with H and square-box sections and numerous tests have been made on columns of these two shapes. The results of this study, which are presented in Technical Note 2994, show consistent agreeement between the tests and the developed theory.

For many aeroelastic problems it is desirable to have a fairly simple relation between the loads and deformation (particularly at angles of attack) of a wing. For swept wings and wings having discontinuities in plan-form such relations are inherently more complex than those for straight wings. An approximate method of calculating the deformations of such wings of uniform thickness has been developed and is presented in Technical Note 2978. The method employs an adjustment to the elementary beam theory to account for the effect of the triangular portions of the wings on the deformations of the remaining straight portions.

Dynamic Properties

The ever increasing importance of flutter as a design criterion has resulted in demands that the accuracy of computing vibration modes and frequencies be assessed and improved where necessary. In order to answer this demand, several investigations were initiated.

In Technical Note 3158, the use of the substitute-stringer approach for including shear lag in the calculation of transverse modes and frequencies of box beams is discussed. Various thin-walled hollow rectangular beams of uniform wall thickness were idealized by means of the substitute-stringer approach. The resulting frequencies of the idealized structures were compared with those of the original beams. The results indicated how the substitute-stringer idealization could be made in order to yield accurate representation of the shear-lag effect in dynamic analysis.

In Technical Note 3070, the effects of local panel oscillations on bending and torsional vibrations of box beams with flexible covers and webs are reported. Theoretical analyses of simplified models are made in order to shed light on the mechanism of coupling between local and overall vibrations and to derive results that can be used to estimate the coupling effects in box beams.

The usual static methods for calculating the maximum load which a structure can support are not applicable in principle when the load is applied rapidly. As a contribution to the solution of the problem, the effect of dynamic loading on the strength of an inelastic column was investigated. The results are presented in Technical Note 3077. The maximum loads of idealized inelastic H-section columns whose pinned ends approach each other at a constant rate are presented. The solutions indicate that, as the rate of end displacement decreases, the dynamic buckling solutions approach the static solution as a lower limit. For all rates of end displacement investigated, the static maximum load may be employed as a conservative estimate of the maximum column load.

Thermal Properties

Aircraft structures, when subjected to aerodynamic or other forms of heating for extended periods of time, may creep; that is, deform progressively and fail under relatively low loadings. The Polytechnic Institute of Brooklyn has investigated the creep buckling of columns whose material can exhibit instantaneous and retarded elasticity as well as pure flow. A general theory for the creep deflection of such columns is presented in Technical Note 3136. Equations for the deflection of an idealized H-section beam column, whose material flows nonlinearly with time, were also derived and presented in Technical Note 3137. The critical time required for infinite deflections of such columns to develop were then established by these equations over a wide range of the parameters involved. The results of these calculations are presented in Technical Note 3138 and enable the designer to determine the critical time once the parameters appearing in the basic uniaxial tensile- or compressive-creep law are determined experimentally at design temperature. A theoretical study was also made of the time-temperature-dependent buckling of an initially curved column heated uniformly at a prescribed time rate. The results of this study are presented in Technical Note 3139 and indicate that the deflected column retains the shape of its initial curvature and that the additional deflection becomes very large when the temperature-dependent Euler load approaches the applied constant load.

Aircraft flown at very high speeds are heated aerodynamically and must be designed to withstand the resulting thermal stresses. Before calculating these stresses the temperature distributions must be determined. The temperature distributions depend on, among other things, the thermal bond between adjacent structural parts. Syracuse University has studied factors affecting the thermal bond of a number of interface joints. The results, which are presented in Technical Note 3167, indicate the degree to which these factors influence heat flow through the joints.

Fatigue Properties

The occurrence of a fatigue failure in modern aircraft structures may vary in severity from a simple nuisance crack to the catastrophic destruction of the entire airplane. Full-scale research on this problem has been conducted on wings of the Curtiss C-46 airplane and was previously reported. Further tests, reported in Technical Note 3190, were of the constant-level type run at four different levels to establish the load-lifetime relationship for a built-up aircraft structure constructed of 24S-T material. In addition to results on the load-lifetime relationship, the scatter in lifetime, and the effective stress-concentration factors for an airplane wing, information was also obtained on the rate of crack propagation and the manner in which cracks develop in a composite structure under constant alternating load. Appendixes of this report explain the use of bonded wires as fatigue crack detectors and the use of laminated fiber glass for the repair of fatigue cracks.

AIRCRAFT LOADS

Basic Air-Load Distribution

The two-dimensional aerodynamic characteristics and the loadings on an NACA 64A010 airfoil-slat combination have been studied for a Mach number range from 0.25 to 0.85, with a corresponding Reynolds number range from 3.4 million to 8.1 million. As reported in Technical Note 3129, the changes in loadings on the airfoil and slat were found to be directly related to the changes in camber and airfoil-slat chord line associated with each of the several slat positions investigated. Displacing the slat nose below the extended chord line of the airfoil produced the highest gains in maximum lift at low Mach numbers, but resulted in adverse effects at supercritical Mach numbers, such as those which occur with cambered airfoils. The most favorable results for section lift coefficients above 0.80 and for the widest range of test Mach numbers were obtained with the nose of the slat on the extended chord line of the airfoil. The energizing effect on the boundary layer on the upper surface of the airfoil, which is often attributed to the stream of air flowing through the gap, was found to be of secondary importance in determining slat performance.

Pressure-distribution tests on wings ranging in sweepback from 40° to 50° have yielded loads data useful for the structural and aerodynamic design of aircraft. Force and moment data, obtained by pressure measurements on a 45°-sweptback wing of aspect-ratio 8, taper-ratio 0.45, and NACA 63_1A012 airfoil sections, have been compared with the calculated loadings obtained by the standard methods proposed by Weissinger, Falkner, and Multhopp, as well as by several variations of these methods. The most accurate shape of the span-load distribution was predicted by the standard Multhopp solution. All methods that predicted a fairly accurate loading shape indicated a lift-curve slope about 8 percent low. Since the calculation methods are based on thin-wing theory, the underestimation of the lift-curve slope is probably attributable to the finite thickness of the wing. The Multhopp solution was shown to be in good agreement with experimental results for a twisted and cambered wing.

The applicability of a well-known finite-step method to the calculation of subsonic spanwise-load distribution, lift-curve slope, lateral center of pressure, and aerodynamic center of unusual plan forms has also been investigated. Computing forms have been developed to simplify calculation of span loadings for conventional swept, M-plan-form, and W-plan-form wings. Tables of the downwash in the plane of a yawed vortex are presented. Comparison of loading results, by using 20 steps, with lifting-surface results indicated that the 20-step method generally overestimated the amount of loading at the wing tip. However, values of lift-curve slope, lateral center of pressure, aerodynamic center, and loading shape across the inboard three-quarter semispan obtained by using the 20-step method were generally in satisfactory agreement with lifting-surface results. For a representative W plan form, it was found that the use of 20 steps provided a span loading that was essentially in agreement with 40-step results.

On the basis of linearized supersonic-flow theory, equations have been derived for the span-load distribution resulting from constant vertical acceleration (that is, linear variation of angle of attack with time) for a series of thin sweptback tapered wings with streamwise tips. The analysis is applicable, in general, at those speeds for which both the wing leading and trailing edges are supersonic. Results of the investigation, which are presented in Technical Note 3120, include detailed charts from which the span loadings may be obtained for arbitrary values of the aspect ratio, taper ratio, leading-edge sweepback, and Mach number.

Since wing flexibility serves to modify the span loading, considerable effort is required to obtain the load distribution when flexibility is present. In order to reduce the work for the designer, calculations are being made of the span loading for a wide variety of wing plan forms covering sweep angles, aspect ratio, and wing taper. Various types of twists were covered for the case where the wing stiffness varied in a prescribed manner. To account for other types of wing stiffness distribution, aerodynamic influence coefficients were computed for the same plan forms. A part of this investigation, which covers the case for unswept wings at subsonic speeds, is reported in Technical Note 3014.

The need for structural loads data for the various conditions of flight has required the investigation of sweptback wings with trailing-edge flaps. In order to supply loads data and to evaluate existing methods for estimating loads, a 45°-sweptback wing of aspect-ratio 8, with flaps of various spans and spanwise positions, was studied. Experimental values of incremental span loading, bending moment, pitching moment, and center of pressure (when compared with values calculated by available methods) showed only fair agreement. However, modifications which increase the accuracy of the analyses have been developed and additional data have been published showing the effects of selected spans of flaps on loadings for a large range of angle of attack.

The incorporation of various stall-control devices to improve stability on sweptback wings alters their load distribution and emphasizes the need for load information on the wing and the devices themselves. Spanwise and chordwise loadings on a 45°-sweptback wing of aspect-ratio 8 with leading-edge flaps and fences have been measured, and additional data have been obtained on loads over leading-edge stall-control devices when mounted on wings having an aspect ratio of 4 or less.

Maneuver Loads

The magnitude and frequency of occurrence of maneuver accelerations experienced by five types of commercial transport airplanes during routine operations have been obtained from time-history (VGH) records and are presented in Technical Note 3086. The results are compared with available gust-acceleration data for the operations considered. It was shown that maneuver accelerations may contribute substantially to the total-load histories of transport airplanes, particularly in the case of medium-altitude operations.

Other Dynamic Loads

As part of an investigation to develop a technique for using rocket-powered models to study gust loads in continuous rough air and to study the influence of stability and configuration on gust loads, tests were made of a rocket-powered, tailless swept-wing model at Mach numbers from 0.8 to 1.00. The model showed a pronounced pitching motion at the short-period frequency during rough-air flight. The load intensities showed a rapid increase of roughly 85 percent as the Mach number increased from 0.8 to 1.0. Of this 85-percent increase, 55 percent appeared to be associated with the rapid decrease of the short-period damping in this Mach number range. The results of this investigation, reported in Technical Note 3161, indicated that the use of rocket-powered models for gust-loads studies is feasible and practical. The limitations of the technique are discussed and several suggestions are given for improving the precision of the experimental results. The data obtained were in substantial agreement with the results derived from theoretical calculations based on the use of power-spectral methods of generalized harmonic analysis.

A study of the buffeting loads measured on the wing and tail of a fighter-type airplane was reported in Technical Note 3080. It was found that the loads, obtained on the assumption that buffeting is the linear response of an aerodynamically damped elastic system to an aerodynamic excitation which is a stationary random process, gave sufficiently good agreement with measured loads to suggest the examination of buffeting of other airplanes on the same basis. Least-squares analysis of the data indicated that, in the stall regime, the square root of the dynamic pressure was found to be a better measure of the load than was the first power. The loads measured in maneuvers of longer duration were, on the average, larger than those measured in maneuvers of short duration. In the flight regime, where shock waves appear on the wing or tail surfaces, the magnitude of the load at a given speed and altitude was determined by the extent of the penetration beyond the buffet boundary. For a modification of the basic airplane, in which the wing natural frequency in fundamental bending was reduced from 11.7 to 9.3 cps by the addition of internal weights near the wing tip, a 15-percent decrease in wing loads and a similar percentage increase in tail loads resulted.

Studies made to evaluate the influence of wing bending flexibility on the structural response to gusts of two twin-engine transports and one four-engine bomber are summarized in Technical Note 3006. The studies encompass some previously reported and some new flight studies, some calculation studies based on discrete- or single-gust encounters, and some new calculation studies for continuous-turbulence encounters, based on the methods of generalized harmonic analysis. It was shown that the discrete-gust approach reveals the general nature of the flexibility effects and leads to qualitative correlation with flight results. The studies, based on the harmonic-analysis approach, showed good quantitative correlation with flight results and may allow for a much greater degree of resolution of the flexibility effects.

An investigation was conducted on a 45°-sweptback-wing model, having interchangeable round and sharp leading edges, in order to determine the effect of leading-edge separation on the loads experienced by the model in gusts. Leading-edge separation served to increase the gust load. The amount of increase apparently depends upon the gust-gradient distance and velocity but cannot be predicted from the results of these few tests. Attempts at correlating the increased load values with the lift curve obtained in steady flow yielded no consistent results. Thus, it appears that the load increase was due to participation of the leading-edge vortex in the unsteady-lift phenomena associated with the gust and was not directly related to the lift curve obtained in steady flow. The results of tuft studies showed a hysteresis effect under unsteady conditions in the flow change between the "no vortex" and "vortex" regimes and further indicated that the rate of change in angle of attack due to the gust and the extent of penetration into the gust are both important in determining whether separation occurs.

About 50,000 hours of V–G data obtained from one type of four-engine civil transport airplane, operated over three different commercial airline routes of the United States from 1949 to 1953, are analyzed in Technical Note 3051 to determine the magnitude and frequency of occurrence of gust loads and gusts. The normal-acceleration increments for each of the three operations equaled or exceeded the limit gust-load factor, on the average, twice (once positive and once negative) in about 1.0×10^6 flight miles. A derived gust velocity of 50 fps (evaluated on the basis of Technical Note 2964 which was reported in the Thirty-ninth Annual Report, 1953) was equaled or exceeded twice in about 0.5×10^6 flight miles for each of the three operations. The frequency of occurrence of the gust loads and the gusts for the present operations is in general agreement with those from similar operations of other civil transports recently investigated.

Technical Note 3041 summarizes gust-velocity data obtained by reevaluating the normal accelerations and airspeeds from V–G records taken on civil transport airplanes from 1933 to 1950. The reevaluation was made on the basis of a "derived" gust velocity U_{de} which is related to the "effective" gust velocity U_e by a conversion factor that is a function of the type of airplane and operating altitude. Although the value of the conversion factor varies from about 1.6 to 2.0 for the data presented, the conclusions drawn from the previously presented data based on U_e (in particular, the relative levels of turbulence indicated between different routes) remain essentially unchanged.

Landing Loads

Technical Note 3246 presents some information on landing-gear applied drag loads and on the nature of the wheel-spin phenomenon in landing, based on studies under controlled conditions. In particular, a study has been made of the nature and variation of the coefficient of friction between the tire and the runway during the wheel spin-up process. Also, comparisons have been made of the various results obtained in impacts with forward speed, impacts with forward speed and reverse wheel rotation, spin-up drop tests, and impacts with forward speed and wheel prerotation.

The statistical measurements of landing contact conditions of transport airplanes at Washington National Airport, discussed in the Thirty-ninth Annual Report, 1953, were extended to include a total of 478 landings of all types of current transports in clear-air daylight conditions. It was found (Technical Note 3194) that gusty wind conditions generally had a substantial effect in increasing the value of sinking speed, bank angle, and rolling velocities at contact but had little effect on the airspeed at contact. The results indicated that the values of sinking speed, bank angle, and rolling velocity likely to be equaled or exceeded once in 1,000 landings under non-gusty conditions are 3.5 fps, 4.8°, and 4.4 deg/sec, respectively. Under gusty conditions, however, the values are increased to 4.7 fps, 6.6°, and 5.5 deg/sec. In general, the transport airplanes landing at Washington National Airport touch down with a considerable speed margin above the stall; as a matter of fact, the results indicated that in one out of a thousand landings the airspeed at contact would equal or exceed 160 percent of the stalling speed. The photographic method employed for obtaining statistical data on vertical velocities of aircraft, immediately prior to landing contact, developed for use with land-based airplanes is described in Technical Note 3050. It requires no instrument installation on the aircraft or interference with airport operation and employs relatively simple data reduction. Vertical velocities can be obtained by this method within a probable maximum error of ±0.31 fps.

VIBRATION AND FLUTTER

Flutter

The ability of the newer military airplanes to attain transonic and supersonic speeds plus the use of thinner wings has caused flutter to become an increasingly important problem. This increased importance is exemplified by the increasing number of flutter models of specific airplanes which are being tested in wind tunnels or in flight by the rocket-model technique. A corresponding increase is under way in the experimental research which has general application.

From an investigation using rocket-powered models, it was found that for swept wings a type of flutter occurred that was of a different nature from the classical bending-torsion flutter. This investigation showed that bending stiffness may be the major parameter affecting flutter of swept wings.

Flight tests and a mathematical analysis were carried out to demonstrate and confirm a type of subsonic flutter, involving rigid-body motions and wing deformations, which included at least four degrees of freedom. The period of the oscillation was well within the range of period found in dynamic-stability work on rigid aircraft with free controls. The calculated values of the flutter speed were conservative. It was found that wing bending stiffness is the important parameter for preventing such flutter.

In the mathematical treatment of the flutter phenomena for subsonic speeds, it has been customary to simplify the problem by employing two-dimensional theoretical coefficients to represent the aerodynamic forces on the airfoil. This simplification leads to inaccuracies at high subsonic speeds and for low-aspect-ratio lifting surfaces. Research is under way to develop improved methods of treating the aerodynamic forces.

Technical Note 3131 describes a study of the kernel function of the integral equation that relates the downwash distribution to the lift distribution for a harmonically oscillating finite-span wing in compressible flow. It was found that this kernel function could be reduced to a form that could be accurately calculated by separating the kernel function into two parts; a part in which the singularities are isolated and analytically expressed and a nonsingular part which may be tabulated. The form of the kernel function for the sonic case is treated separately. Results for the special cases of Mach number of zero and frequency of zero are also given.

Theoretical studies are also being carried out on the aerodynamic forces on airfoils in supersonic flow. The supersonic aerodynamic coefficients for an oscillating surface, needed by flutter analysts, were developed from an expansion of the velocity potential as expressed by a power series in terms of the frequency of oscillation.

The air forces and moments, obtained in this way, for a rigid triangular wing with subsonic leading edges oscillating in pitch and translation were previously reported (Report 1099). An extension of this work to the case of a triangular wing with subsonic leading edges experiencing harmonic deformations in supersonic flow has been made in Technical Note 3009. The oscillations considered were such that the amplitude of distortion of the wing can be represented by a general quadratic expression for a surface. Although only terms appropriate for expressing the potential to the third power of the frequency are presented, additional terms may be obtained if they are desired.

The aerodynamic lift and moment for a harmonically oscillating rectangular (unswept) wing moving at supersonic speed has also been derived from an expansion of the velocity potential represented by a power series in terms of the frequency of oscillation. The lift and moment coefficients for a wing oscillating in pitch and translation, derived to the third power of the frequency, have previously been reported in Report 1028. In order to obtain greater accuracy for wings oscillating at the higher reduced frequencies and at low supersonic Mach numbers, the expansion of the potential has been carried out to the seventh power of frequency and is reported in Technical Note 3076. Comparison of flutter speed for a moderate-aspect-ratio wing, as calculated using two-dimensional coefficients and also coefficients based on the theory of Technical Note 3076, showed significant differences for supersonic Mach numbers near unity.

Airfoil Thickness Effects

In general, the expressions for the aerodynamic forces at supersonic speeds for lifting surfaces in unsteady motion are based on the assumption of linearized potential flow. This is of questionable accuracy inasmuch as the effects of thickness may have an important effect on pitching moments.

A study has been made and is reported in Technical Note 2982 in which a solution to the second order in thickness was derived for harmonically oscillating two-dimensional airfoils in supersonic flow. For slow oscillations of an arbitrary profile, the result was found as a series including the third power of frequency. For arbitrary frequencies, the method of solution for any specific profile is indicated, and the explicit solution derived for a single wedge.

Nonlinear thickness effects were generally found to reduce the torsional damping and so enlarge the range of Mach numbers within which torsional instability is possible. This destabilizing effect varied only slightly with frequency in the range involved in dynamic-stability analysis but may reverse to a stabilizing effect at high flutter frequencies. Comparison with a previous solution, exact in thickness, suggested that nonlinear effects of higher than second order are practically negligible.

AIRCRAFT STRUCTURAL MATERIALS

Research Techniques

The use of silver chloride as a material for photoelastic stress analysis offers the possibilities of studying both elastic and plastic states of stress in a crystalline metallike material on either a microscale or macroscale. In order to realize this possibility, however, it is necessary to relate the stress state quantitatively with the observed relative retardation and extinction angle. In Technical Note 3043, these relationships were developed from a general theory of stress birefringence, according to a stress-dependent hypothesis. This hypothesis and the resulting analytical relationships have been experimentally vindicated by measurements made on a variety of single-crystal specimens of silver chloride tested in simple tension in the elastic and plastic stress ranges.

Adhesives

Present-day structural adhesives are often complex mixtures of several resin components. Some of these components may become soft and elastic at elevated temperatures or very rigid and brittle at low temperatures. These changes can be expected to affect the initial or subsequent characteristics of joints in aircraft. Some time ago, in order to help stimulate development of new or modified adhesives whose resistance to both high- and low-temperature degradation would allow use in bonded aircraft structural joints, the NACA instituted a program at the Forest Products Laboratory to survey existing adhesives to determine the causes of the physical property changes. As a result of this program, the Forest Products Laboratory developed a metal-bonding adhesive that has both greater resistance to low- and elevated-temperature degradation up to 600° F and is also easier to use than present adhesives.

Metallurgy

The seriousness of the problem of hydrogen embrittlement in metals, especially in the high-strength heat-treated steels, has increased with the increased use of these steels in aircraft structures. In an effort to help solve this problem, the NACA contracted with the Battelle Memorial Institute to study the factors affecting hydrogen entry into metals, during chemical or electrochemical processing, and subsequent embrittlement. In Technical Note 2696, a theory was postulated which accounted for hydrogen entry, and, in Technical Note 3164, further experimental evidence is provided which supports the work of the previous paper. Factors which are effective in the control of hydrogen embrittlement can be inferred from the results of these reports.

Plastics

Realizing the importance of transparent glazing materials in aircraft, the NACA instituted a program at the National Bureau of Standards to determine means and methods of improving the strength characteristics of existing glazing materials. Results of this program have been reported previously in the Thirty-sixth, 1950, Thirty-seventh, 1951, Thirty-eighth, 1952 and Thirty-ninth, 1953, volumes of the NACA Annual Report. Recently published results of this program have shown a correlation between strength characteristics and craze resistance and chemical composition of cast polymethylmethacrylate plastic.

Fatigue

Fatigue of aircraft structural materials remains as one of the most important problems facing the aircraft designer. Structural integrity must be maintained in an aircraft structure throughout its usable life. The materials research being carried on by the NACA is necessary to place enough information regarding this problem in the hands of the designer so that he can properly design a structure.

In Technical Note 3019, an investigation was reported which utilized the statistical methods developed in previous research to study the scatter found in aluminimum alloys and the effect of the morphology of the carbide phase on the scatter in a eutectoid steel. The scatter in fatigue life for 24S and 75S aluminum was found to be comparable with that reported previously for steel. As was found in previous investigations, the scatter increased with decreasing stress level. A pronounced effect of microstructure was found when a coarse pearlitic structure was compared with a coarse spheroidized structure in the same steel with the same tensile strength. The scatter from the pearlitic structure was significantly less than that in the spheroidized, this result being attributed to the stress concentration produced by the sharp carbide lamellae.

The recent use of high-strength steels in many aircraft applications made necessary the research to provide fatigue data on some of these materials. The effect of overstressing on the fatigue properties of SAE 4340 steel has been studied statistically. The effect of microstructure on the susceptibility to reduction in fatigue life due to cycles of overstress was investigated. When tested at an equivalent percent stress, the quenched and spheroidized structure was found to be more susceptible to fatigue damage.

The effect of overstress on the endurance-limit statistics was studied for the quenched and spheroidized structure. Enough specimens were tested to determine the endurance-limit statistics of damaged specimens by the probit method. The decrease in the mean endurance limit due to cycles of overstressing was much greater than would be expected from nonstatistical investigations which are reported in the literature. The effect is interpreted as support for the belief that the bulk of the fatigue damage takes place before the first 30 percent of the total fatigue life.

In a recently published report, the results of additional static and fatigue tests were made on aluminum-alloy 355-T6 sand-cast specimens. Direct-stress fatigue tests were made on plate-type specimens with a single 1-inch-diameter as-cast cored hole and on specimens in which the cored hole was reamed to 1$\frac{1}{16}$-inch diameter. In addition, direct-stress fatigue tests were made on 0.300-inch-diameter specimens, with various degrees of porosity, machined from the butt ends of plate-type specimens.

Within the range of stresses used, there were no significant differences in the fatigue strengths of sand-cast specimens with a 1-inch-diameter cored hole when tested with the hole in the as-cast condition or with the hole enlarged $\frac{1}{16}$ inch in diameter by reaming. When the results of tests on the specimens with a 1-inch-diameter cored hole were compared with the results, from a previous investigation, for specimens in which a small cast boss was removed and a 1-inch-diameter hole drilled and reamed in the center of the plate-type specimen, there was found to be no significant difference in the results except for a slight difference in the static strengths.

The direct-stress fatigue test results on 0.300-inch-diameter round polished specimens indicate no correlation between the fatigue strengths developed and visual porosity ratings.

Recent experiences, in widely separated areas in aeronautics, have shown an urgent need for fatigue information in an area hitherto almost universally neglected; namely, the high stress—low cycle range of loading (from 2 to 10,000 cycles). Several investigations in this field have been undertaken. In Technical Note 3017, there are presented results of axial-load fatigue tests, at a stress ratio of 0, performed on notched and unnotched sheet specimens of 61S-T6 aluminum alloy and 347 and 403 stainless steels. Special emphasis was placed on tests at high stress levels producing failures in small numbers of cycles. It was found that the stress-concentration factors effective in fatigue of notched specimens were somewhat less than the theoretical elastic values at low stresses and were approximately equal to 1 at the ultimate strength. The minimum life to failure at stresses near the ultimate strength was drastically reduced with increasing stress-concentration factor.

In Technical Note 3132, there are presented results obtained on notched specimens made of 24S-T3 and 75S-T6 aluminum-alloy sheet material, with theoretical stress-concentration factors equal to 4.0, subjected to completely reversed axial loads. Failures occurred in less than 50 cycles at two-thirds of the static tensile strength and in as few as 2 cycles when the applied load was near the static strength of the specimen. The S–N curves were found to be concave upward for almost the complete range of fatigue lives; a reversal of curvature occurred at about 10 cycles of load. The fatigue strengths were equivalent for specimens, made of each of the two materials, tested at stresses below 25 ksi; above that stress the 75S–T6 specimens had the greater fatigue strength. Compared on the basis of percent of ultimate tensile strength, the 24S–T3 specimens were stronger at all stress levels. Test techniques and special test apparatus are described.

Elevated-Temperature Materials Research

With the advent of transonic and supersonic speeds, the problem of aerodynamic heating and its effects on material strength properties has required much research in the field of elevated-temperature characteristics of materials. In order to shorten the time required to obtain useful data, attempts have been made to extrapolate and generalize results by the use of suitable time-temperature parameters. In Technical Note 3195, a parameter based upon rate-process theory was successfully applied to rupture and creep data for aluminum and various aluminum alloys. The optimum value of the constant in the parameter, which provided the best correlation of the data, was determined for each material and application. Master curves of stress versus parameter, summarizing extensive data on the aluminum alloys, are presented for rupture, minimum creep rate, and time to reach 1- or 2-percent strain. Predictions of long-time life from short-time data are shown to be possible.

Occasionally, metals intended for elevated-temperature use appear to be promising for special room-temperature uses. In Technical Note 3197, room-temperature stress-strain curves are presented for compression, tension, and shear loadings on four compositions of titanium carbide with nickel binder. Values of ultimate strengths, moduli of elasticity, moduli of rigidity, Poisson's ratio in the elastic region, density, and hardness for the four materials are tabulated.

OPERATING PROBLEMS

The safety, efficiency, and economy of all-weather flight form the basis for NACA research on pertinent operating problems. Turbulent air, icing clouds, and other meteorological manifestations of the atmosphere affect all types of aircraft. The mechanisms of aircraft fire ignition and suppression still require study as do the factors involved in aircraft crash survival on land or water. Aircraft noise continues to be a problem which tends to increase in scope with the use of more powerful turbojet and turboprop engines in routine aircraft operations and with the increased use of helicopters.

A successful conference on jet-engine noise was sponsored by the NACA in order that foreign and domestic scientists in the noise field could exchange views and determine possible avenues for research in this field. Conclusions indicated that more research effort was necessary to understand the fundamental mechanisms of aircraft-noise generation, propagation, and alleviation.

In furtherance of the development of an internationally accepted standard aeronautical atmosphere, the tables of the properties of the International Civil Aviation Organization (ICAO) standard atmosphere have been computed, published, and adopted by the NACA (Technical Note 3182). The new ICAO standard atmosphere supersedes the tables and figures of the NACA standard atmosphere (Report 218) and provides an international standard for the calibration of altimeters and other navigational and flight instruments necessary for air-traffic safety on international air routes. It forms a common basis for, and necessary first step toward, development of internationally accepted airworthiness specifications not only for commercial aircraft for use in international air traffic but also for aircraft manufactured in one nation for use within another nation.

In addition to the parent Committee on Operating Problems, the Subcommittee on Meteorological Problems, the Subcommittee on Icing Problems, the Subcommittee on Aircraft Fire Prevention, and the Special Subcommittee on Aircraft Noise have aided in guiding the research on specific operating problems which have been accomplished at the NACA laboratories and under contract with nonprofit research organizations.

In the following paragraphs, most of the unclassified investigations on Operating Problems have been summarized.

Ditching

Hydroskis have been considered as a positive means of eliminating the hazardous motions and structural damage associated with the ditching of land planes. Model investigations have therefore been made to determine the ditching characteristics of three typical multiengine airplanes equipped with possible arrangements of hydroski ditching gear. The behavior of the models was determined from visual observations, acceleration records, and motion pictures of the landings. The results indicated that a ditching gear of one or more hydroskis would afford very satisfactory water landings, as compared with landings without skis. It is possible that critical damage could be eliminated from ditching by using a hydroski ditching gear; thus, the chances of survival and rescue would be greatly increased.

Airspeed Calibration

A flight investigation has been made to determine whether temperature and pressure conditions in the lower stratosphere (35,000 to 45,000 feet) would meet the requirements of the temperature method of calibrating airspeed installations. Measurements were taken with instruments installed in a swept-wing fighter airplane at altitudes up to 45,000 feet on 4 clear days in the spring of 1953 (Technical Note 3075). The results indicated that, although the temperature lapse rate of the atmosphere was generally favorable to the use of the method, large and erratic variations of temperature with time and distance precluded an accurate calibration.

The accuracy of an aircraft radio altimeter has been evaluated for use in a method of airspeed calibration differing from the radar-phototheodolite method only in that the geometric altitude is obtained from an aircraft radio altimeter (Technical Note 3186). Comparison of geometric altitudes measured during simulated calibration runs showed that the accuracy of the radio altimeter was of the same order as that of the radar-phototheodolite unit; thus, calibrations of comparable accuracy could be expected. In the proposed method, all the necessary equipment is contained within the airplane; the desired quantity is measured directly and requires a minimum of calculations for the data reduction; the tests are not restricted to a small area or to excellent visibility conditions; and the equipment required is readily available. The method provides, therefore, a rapid and convenient means for calibrating the airspeed system of high-performance aircraft, the only disadvantage being that tests must be performed over a large body of water.

Static-Pressure Measurement

Flight tests have been conducted to determine the variation of static-pressure error with angle of attack of two similar service pitot-static tubes and of one of these tubes with three modified orifice arrangements (Technical Note 3159). To increase the range of insensitivity to airstream inclination, the orifice configuration was modified. Results of the tests of the best of the orifice arrangements showed that, for Mach numbers between 0.20 and 0.68 and for Reynolds numbers between 0.9×10^5 and 1.4×10^5, the error remains within 2 percent of the impact pressure over an angle-of-attack range of $-10°$ to $30°$. At higher angles of attack, the static pressure increased rapidly and reached values as high as 15 percent of the impact pressure at an angle of attack of 45°; thus, tube usefulness is limited to angles of attack below 30°.

AERONAUTICAL METEOROLOGY

Atmospheric Turbulence

Information on turbulence at high altitudes in the vicinity of jet streams has been obtained in a flight investigation undertaken by the Navy Bureau of Aeronautics with the assistance of the NACA. Flying at about 30,000 feet, turbojet fighter-type aircraft, equipped with instruments including NACA VGH flight recorders, made traverses of the jet stream. An evaluation of the records for the locations of the turbulent areas and the frequency and intensity of the gusts in the turbulence was performed. The analysis indicated light turbulence on approximately one-half the VGH records evaluated from 39 successful flights.

Turbulence information was also obtained in a flight investigation to evaluate an experimental C band (5.5 cm) airborne weather radar for avoidance of severe turbulence in thunderstorm areas. NACA cooperation in this project was to provide a VGH recorder and evaluate the gust records. In general, the flights did not penetrate the thunderstorms, and thus the gust velocities encountered were lower than might be expected within the more active portions of thunderstorms.

Physics of the Icing Cloud

The NACA has nearly completed the low-altitude portion of the previously reported NACA-Air Force-Airline Statistical Icing Program utilizing approximately 100 NACA pressure-type icing-rate instruments on various types of United States aircraft flying in many parts of the world. This mass of statistical data has been put on punch cards for rapid analysis. The program is now being extended to altitudes above 25,000 feet. Instrumentation has been installed on about half of the 30 high-altitude Air Force aircraft that are participating in this program.

ICING PROBLEMS

Droplet-Impingement Studies

In the problem of design of systems to protect aircraft from ice formations, a knowledge is required of the areas to be protected and the rate of impingement of icing-cloud droplets on these areas. The analytical study is continuing with the use of the NACA droplet-trajectory analog to determine the amount of water in droplet form impinging, the extent of impingement, and the rate of water impingement per unit area of affected surface on low-drag airfoils, bodies of revolution, internal ducts with turns, and other aerodynamic bodies of fundamental geometric shape. The impingement characteristics on a thin low-drag NACA 65A004 section at an angle of attack of 4° have been calculated (Technical Note 3047). In order to present the effect of a change in airfoil thickness from 12 to 4 percent, the results for the NACA 65A004 airfoil were compared with those for NACA airfoils 65_1–208 and 65_1–212 previously reported. Results showed that the rearward limit of impingement on the upper surface decreased as the airfoil thickness decreased, that the rearward limit of impingement on the lower surface increased with a decrease in airfoil thickness, and that the total water intercepted decreased as the airfoil thickness was decreased.

All-weather aircraft and missiles frequently employ instrumentation located in the nose section of the fuselage that is sensitive to impinging atmospheric water droplets and ice accretion. For example, it has been found that the operation of an aircraft radar system located in a nose or wing radome is affected by a layer of ice or water distributed over the radome surface. Therefore, it is necessary to evaluate, for given flight conditions, the expected distribution of various sizes of impinging water droplets over the nose section of the aircraft or missile. In addition, problems, such as those encountered in the performance of external armament during flight in icing conditions, require the evaluation of droplet impingement on bodies of revolution in order to determine where ice will form.

Although a large variety of body shapes are used for radomes, rocket pods, and bombs, the impingement calculations may be made for a body selected to approximate a large group of these practical shapes. A prolate ellipsoid of revolution is a good approximation for many of these bodies. Calculations of the impingement characteristics have been made for an ellipsoid with a fineness ratio of 5 (Technical Note 3099) and for an ellipsoid with a fineness ratio of 10 (Technical Note 3147). The characteristics, total rate of water impingement, extent of droplet-impingement zone, and local rate of water impingement are given in terms of dimensionless parameters.

An aircraft moving through a cloud alters the concentration of cloud droplets in the immediate vicinity of the aircraft. A knowledge of this spatial variation of local droplet concentration (about an aircraft or missile during flight through clouds, drizzle, or rain) is important when choosing the location of devices which protrude into the stream or when determining the heat required to protect the devices from ice. An evaluation of the variation of local liquid-water concentration about an ellipsoid of fineness ratio 5 moving in a droplet field has been reported (Technical Note 3153).

The question has often been presented as to whether the equations of motion of droplets in a flow field should account for any droplet evaporation that may exist. An analysis of the maximum evaporation rates of water droplets approaching obstacles in the atmosphere has been made in order to show that the change in droplet size is negligible for droplet sizes of interest in the design of protective systems (Technical Note 3024).

Ice Protection and Heat Transfer

The work of the airplane designer can frequently be aided by the preparation of summary or design-data reports on a particular subject. Such a report has been prepared to provide a procedure for the design of ice-prevention systems in which heated air is employed (Technical Note 3130). Design equations are included along with methods of selecting appropriate meteorological conditions for the specified flight region. In order to facilitate the design, a simple electrical analog was devised which solves the complex heat-transfer relationships existing in the thermal-system analysis.

An electrical analogy has also been useful in determining the heating requirements for the cyclic de-icing of hollow steel propellers fitted with two types of internal electric heaters. Solutions were obtained to the transient heat-flow equations depicting the cyclic de-icing of propellers (Technical Note 3025). The results showed the impracticability of utilizing an internal tubular heater and indicated the advantages of using an internal shoe-type heater, a type which distributes the heat more evenly to the blade surface.

The effects of primary and runback ice formations on the section drag coefficient of a 36° swept NACA 63A–009 airfoil section equipped with a partial-span leading-edge slat have been determined. In general, the studies showed that icing on a thin swept airfoil will result in more detrimental aerodynamic-drag characteristics than on a thick unswept airfoil.

The icing characteristics, the effects of icing on radar operation, and the requirements for a fluid-icing-protection system for two radomes were determined. The fluid-protection system, which used ethylene glycol for both anti-icing and de-icing, gave adequate protection within requirements of practical magnitude.

An experimental investigation to determine the mass transfer by sublimation, heat transfer, and skin friction from an iced surface has been made (Technical Note 3104). The results indicate that sublimation as a means of removing ice formations of appreciable thickness is usually too slow to be of much value in the de-icing of aircraft at high altitudes.

An analytical study was made of the combined heat and mass transfer from a flat plate in terms of Prandtl's simplified physical concept of the turbulent boundary layer (Technical Note 3045). For conditions of reasonably small heat and mass transfer, the ratio of the mass- and heat-transfer coefficients is dependent on the Reynolds number of the boundary layer, the Prandtl number of the medium of diffusion, and the Schmidt number of the diffusing fluid in the medium of diffusion. For the particular case of water evaporating into air, the ratio of mass-transfer coefficient to heat-transfer coefficient was found to be slightly greater than unity.

AIRCRAFT FIRE PREVENTION

As a part of the overall study of crash fires of reciprocating and turbojet-engine aircraft, the hazard of igniting airplane fires by electrostatic sparks generated when detached airplane parts fly through clouds of dust and fuel mist was investigated (Technical Note 3026). Within the limits of the variables studied, the rates with which airplane wreckage collected a charge were directly proportional to the rate with which clay dust or fuel mist was intercepted. Maximum rates of experimental electrification were used to relate energy accumulation to wreckage sizes and trajectories and to estimate minimum hazardous wreckage sizes and trajectories. Comparison of sizes and trajectories of wreckage shown in motion pictures of airplane crashes with these estimated sizes and trajectories indicated that the hazard is small. Of the remedial measures considered, polyethylene coatings were found to offer promise of protection against electrostatic spark ignition.

AIRCRAFT ACCIDENT SURVIVAL

The factors which affect human survival in airplane accidents followed by fire have been studied by conducting several full-scale crashes (Technical Note 2996) and related tests. For the severe crash fires investigated, the results indicated that the survival times, as limited by pain or skin burning, ranged from 50 to 300 seconds. If a significant increase in escape time is to be made, protection must be provided against the hazards of skin burning, respiratory injury, and toxic gases. The studies also showed that fuel spillage must be taken into account. In addition, the propeller fragments launched by impact of rotating propellers with the ground would have a reduced probability of entering the airplane if the rotating blades moved away from the fuselage in their travel below the axis of rotation. This survival program is continuing in an effort to obtain aircraft crash load data, which can be related to dynamic seat load requirements to insure maximum protection to passengers and crew during a survivable accident.

AIRCRAFT NOISE

The increasing power and speed of aircraft have accentuated the noise problem. This problem is of concern to people living near airports and test facilities and to crews and passengers in the aircraft. In addition aircraft structures and equipment are likely to fail because of fatigue as a result of the very intense noise fields near high-powered propellers and jets. The major efforts on noise research during the past year have been devoted to the intense noise fields near the source and their possible alteration to reduce the noise.

Propeller Noise

A theoretical analysis of the sound-pressure field of a rotating propeller in forward flight in free space was made by replacing the normal-pressure distribution over the propeller, associated with thrusts and torque, by a distribution of acoustic pressure doublets acting at the propeller disk (Technical Note 3018). The basic element used to synthesize the field is the pressure field of a concentrated force moving uniformly at subsonic speeds. The effect of forward speed on the propeller-noise pressure field may be calculated and evaluated by application of this approach.

Measurements of free-space oscillating pressures near a static pusher propeller, in the region where a wing might be located, for the tip Mach number range of 0.50 to 1.20 have been made and the results compared with available theory (Technical Note 3202). The results of these and other measurements showed that the maximum pressures occurred at a greater radial distance as the axial distance was increased and as the order of the harmonic was increased. The direction of rotation was found to be significant with regard to the magnitudes of wing surface pressures or, more precisely, pressures were higher on surfaces which the propeller blade approached.

Some aspects of the helicopter-noise problems have been studied (Technical Note 3239). Conclusions indicate that, for helicopters powered by reciprocating engines, the engine and accessories, such as gearing, are primary sources of noise. For comparable-sized helicopters utilizing tip-jet propulsion, the noise levels will be considerably higher and the rotor system may then be one of the primary sources of noise.

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Model and Full-Scale Jet Noise

Experimental studies of the pressure fluctuations near jet exhaust streams were made during unchoked operation of a turbojet engine and a 1-inch-diameter high-temperature model jet and during choked operation of various sizes of model jets with unheated air (Technical Note 3187). The tests for unchoked operation indicated a random spectrum of rather narrow band width which varied in frequency content with axial position along the jet. Pressure surveys from the model tests along the lines parallel to the 15° jet boundary indicated that the station of greatest pressure fluctuations is determined by the jet velocity and radial distance, with a tendency of the maximum fluctuation to shift downstream as either parameter is increased. From model tests, the magnitude of the fluctuations appears to increase (at about the second power of jet velocity) at points just outside the jet boundary and to increase (at increasingly higher powers of jet velocity) as the distance from the boundary was increased.

It was found that the use of grids in the jet exhaust stream was beneficial in reducing the magnitudes of the low-frequency noise components. These devices may have some application to the problem of ground muffling of jet engines.

Choked operation of model jets with unheated air indicated the appearance of a discrete-frequency component of very large magnitude. Shadowgraph records of the flow showed that this condition is associated with the appearance of flow formations, suggestive of partly-formed toroidal vortices, in the vicinity of the shocks. Elimination of these formations in the laboratory by the use of "teeth" or auxiliary air jets was found to eliminate the discrete-frequency component and thereby to reduce the overall noise level.

A study of two jet-noise-suppression devices consisting of teeth projecting into the jet stream was conducted on a current full-scale axial-flow turbojet engine. The sound fields obtained with the toothed devices showed a slight reduction in maximum sound pressure level (2 db) compared with the sound field from a standard nozzle. The sound fields of the toothed devices were very similar and (when compared with a standard nozzle) showed a reduction of sound pressure level downstream of the jet with increased levels on the front and side. The total radiated power from the toothed and standard nozzles was very nearly the same (±1 db). Because of the small reduction in maximum sound pressure level and because the total radiated power in all cases was nearly the same, it was concluded that the toothed devices investigated do not represent a satisfactory solution to the jet-noise problem.

RESEARCH PUBLICATIONS

REPORTS

No.

1111. An Analysis of Laminar Free-Convection Flow and Heat Transfer About a Flat Plate Parallel to the Direction of the Generating Body Force. By Simon Ostrach.
1112. Hydrocarbon and Nonhydrocarbon Derivatives of Cyclopropane. By Vernon A. Slabey, Paul H. Wise, and Louis C. Gibbons.
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No.

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1157. Analytical Derivation and Experimental Evaluation of Short-Bearing Approximation for Full Journal Bearings. By George B. DuBois and Fred W. Ocvirk.

TECHNICAL NOTES [1]

2975. Structural Efficiencies of Various Aluminum, Titanium, and Steel Alloys at Elevated Temperatures. By George J. Heimerl and Philip J. Hughes.

2977. Techniques for Calculating Parameters of Nonlinear Dynamic Systems From Response Data. By Benjamin R. Briggs and Arthur L. Jones.

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2986. Effect of Blade-Thickness Taper on Axial-Velocity Distribution at the Leading Edge of an Entrance Rotor-Blade Row With Axial Inlet, and the Influence of This Distribution on Alinement of the Rotor Blade for Zero Angle of Attack. By John D. Stanitz.

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2994. Column Strength of H-Sections and Square Tubes in Postbuckling Range of Component Plates. By P. P. Bijlaard and G. P. Fisher.

[1] The missing numbers in the series of Technical Notes were released before or after the period covered by this report.

No.

2995. New Experiments on Impact-Pressure Interpretation in Supersonic and Subsonic Rarefied Air Streams. By F. S. Sherman.

2996. Appraisal of Hazards to Human Survival in Airplane Crash Fires. By Gerard J. Pesman.

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2998. The Effects of Camber on the Variation With Mach Number of the Aerodynamic Characteristics of a 10-Percent-Thick Modified NACA Four-Digit-Series Airfoil Section. By Albert D. Hemenover.

2999. Impingement of Droplets in 90° Elbows With Potential Flow. By Paul T. Hacker, Rinaldo J. Brun, and Bemrose Boyd.

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3001. Comparison of Operating Characteristics of Four Experimental and Two Conventional 75-Millimeter-Bore Cylindrical-Roller Bearings at High Speeds. By William J. Anderson, E. Fred Macks, and Zolton N. Nemeth.

3002. Effects of Bronze and Nodular Iron Cage Materials on Cage Slip and Other Performance Characteristics of 75-Millimeter-Bore Cylindrical-Roller Bearings at DN Values to 2×10^6. By William J. Anderson, E. Fred Macks, and Zolton N. Nemeth.

3003. Investigation of 75-Millimeter-Bore Deep-Groove Ball Bearings Under Radial Load at High Speeds. II—Oil Inlet Temperature, Viscosity, and Generalized Cooling Correlation. By Zolton N. Nemeth, E. Fred Macks, and William J. Anderson.

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3007. Lift and Pitching Moment at Low Speeds of the NACA 64A010 Airfoil Section Equipped With Various Combinations of a Leading-Edge Slat, Leading-Edge Flap, Split Flap, and Double-Slotted Flap. By John A. Kelly and Nora-Lee F. Hayter.

3008. Effects of Finite Span on the Section Characteristics of Two 45° Swept-back Wings of Aspect Ratio 6. By Lynn W. Hunton.

3009. Velocity Potential and Air Forces Associated With a Triangular Wing in Supersonic Flow, With Subsonic Leading Edges, and Deforming Harmonically According to a General Quadratic Equation. By Charles E. Watkins and Julian H. Berman.

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No.

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3017. Axial-Load Fatigue Tests on Notched and Unnotched Sheet Specimens of 61S–T6 Aluminum Alloy, Annealed 347 Stainless Steel, and Heat-Treated 403 Stainless Steel. By Herbert F. Hardrath, Charles B. Landers, and Elmer C. Utley, Jr.

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3039. Experimental Stress Analysis of Stiffened Cylinders With Cutouts. Pure Torsion. By Floyd R. Schlechte and Richard Rosecrans.

3040. Use of Two-Dimensional Data in Estimating Loads on a 45° Sweptback Wing With Slats and Partial-Span Flaps. By Lynn W. Hunton and Harry A. James.

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3045. Analogy Between Mass and Heat Transfer With Turbulent Flow. By Edmund E. Callaghan.

3046. Theoretical Calculations of the Effects of Finite Sideslip at Supersonic Speeds on the Span Loading and Rolling Moment for Families of Thin Sweptback Tapered Wings at an Angle of Attack. By Windsor L. Sherman and Kenneth Margolis.

3047. Impingement of Water Droplets on NACA 65A004 Airfoil and Effect of Change in Airfoil Thickness From 12 to 4 Percent at 4° Angle of Attack. By Rinaldo J. Brun, Helen M. Gallagher, and Dorothea E. Vogt.

3048. Experimental Investigation of the Effects of Cooling on Friction and on Boundary-Layer Transition for Low-Speed Gas Flow at the Entry of a Tube. By Stephen J. Kline and Ascher H. Shapiro.

3049. An Analytical Investigation of the Effect of the Rate of Increase of Turbulent Kinetic Energy in the Stream Direction on the Development of Turbulent Boundary Layers in Adverse Pressure Gradients. By Bernard Rashis.

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3054. Investigation at Supersonic Speeds of the Wave Drag of Seven Boattail Bodies of Revolution Designed for Minimum Wave Drag. By August F. Bromm, Jr. and Julia M. Goodwin.

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3055. Friction and Wear Investigation of Molybdenum Disulfide. I—Effect of Moisture. By Marshall B. Peterson and Robert L. Johnson.

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Part II—COMMITTEE ORGANIZATION AND MEMBERSHIP

The National Advisory Committee for Aeronautics was established by Act of Congress approved March 3, 1915 (U. S. Code, title 50, sec. 151). During the past year the Committee's enabling legislation as amended from time to time was further amended by Public Law 384, 83d Congress, approved June 3, 1954, which provides that the Committee shall include "one Department of Defense representative who is acquainted with the needs of aeronautical research and development." This representation is in place of the provision of law formerly existing that one of the Committee's members should be the chairman of the Research and Development Board, which had been abolished by Reorganization Plan No. 6, effective June 30, 1953. The NACA enabling legislation provides also that the members of the Committee, all of whom are appointed by the President, shall include two representatives each of the Department of the Air Force, the Department of the Navy, and the Civil Aeronautics Authority, and one representative each of the Smithsonian Institution, the United States Weather Bureau, and the National Bureau of Standards. In addition 7 members are appointed for 5-year terms from persons "acquainted with the needs of aeronautical science, either civil or military, or skilled in aeronautical engineering or its allied sciences." The representatives of the Government organizations serve for indefinite periods, and all members serve as such without compensation.

Pending the enactment of the legislation which became Public Law 384, President Eisenhower on March 2, 1954, appointed Hon. Donald A. Quarles, Assistant Secretary of Defense (Research and Development) a member of the Committee. Mr. Quarles thus succeeded Hon. Walter G. Whitman, who had been serving as a member of the NACA while chairman of the Research and Development Board prior to its abolition.

The following other changes in membership of the Committee have occurred during the past year:

On December 3, 1953, the President appointed Mr. Preston R. Bassett, president of the Sperry Gyroscope Co., and Mr. Ralph S. Damon, President of Trans-World Airlines, members of the NACA for terms expiring December 1, 1958. They succeeded Dr. Theodore P. Wright, vice president for Research of Cornell University, and Mr. William Littlewood, vice president of American Airlines, whose terms of service expired December 1, 1953.

Honorable Oswald Ryan, member of the Civil Aeronautics Board, was appointed to the NACA on January 27, 1954, succeeding Hon. Robert B. Murray, Jr., Under Secretary of Commerce for Transportation, who resigned from the Committee because of the number of his other duties and responsibilities.

General Nathan F. Twining, USAF, Chief of Staff of the Air Force, was appointed a member of the Committee April 19, 1954, to succeed Lt. Gen. Laurence C. Craigie, USAF, upon the latter's transfer from duty at Air Force Headquarters as Deputy Chief of Staff, Development.

In accordance with the regulations of the Committee as approved by the President, the chairman and vice chairman and the chairman and vice chairman of the executive committee are elected annually.

On October 22, 1954, Dr. Jerome C. Hunsaker was reelected chairman of the NACA and of the executive committee, Dr. Detlev W. Bronk vice chairman of the NACA, and Dr. Francis W. Reichelderfer vice chairman of the executive committee.

The Committee membership is as follows:

Dr. Jerome C. Hunsaker, Massachusetts Institute of Technology, Chairman.
Dr. Detlev W. Bronk, President, Rockefeller Institute for Medical Research, Vice Chairman.
Honorable Joseph P. Adams, member, Civil Aeronautics Board.
Dr. Allen V. Astin, Director, National Bureau of Standards.
Mr. Preston R. Bassett, President, Sperry Gyroscope Co., Inc.
Dr. Leonard Carmichael, Secretary, Smithsonian Institution.
Mr. Ralph S. Damon, President, Trans World Airlines, Inc.
Dr. James H. Doolittle, Vice President, Shell Oil Co.
Rear Admiral Lloyd Harrison, USN, Deputy and Assistant Chief of the Bureau of Aeronautics.
Mr. Ronald M. Hazen, Director of Engineering, Allison Division, General Motors Corp.
Vice Adm. Ralph A. Ofstie, USN, Deputy Chief of Naval Operations (Air).
Lt. Gen. Donald L. Putt, USAF, Deputy Chief of Staff, Development.
Hon. Donald A. Quarles, Assistant Secretary of Defense (Research and Development).
Dr. Arthur E. Raymond, Vice President, Engineering, Douglas Aircraft Co., Inc.
Dr. Francis W. Reichelderfer, Chief, U. S. Weather Bureau.
Hon. Oswald Ryan, member, Civil Aeronautics Board.

Assisting the Committee in its coordination of aeronautical research and the formulation of its research programs are four technical committees: Aerodynamics, Power Plants for Aircraft, Aircraft Construction, and Operating Problems. Each of these committees is aided by from 4 to 8 technical subcommittees. The Committee is advised on matters of policy affecting the aircraft industry by an Industry Consulting Committee.

Membership of the committees and their subcommittees is as follows:

COMMITTEE ON AERODYNAMICS

Mr. Preston R. Bassett, Sperry Gyroscope Co., Inc., Chairman.
Dr. Theodore P. Wright, Cornell University, Vice Chairman.
Dr. Albert E. Lombard, Jr., Directorate of Research and Development, U. S. Air Force.
Col. Daniel D. McKee, U. S. A. F., Wright Air Development Center.
Rear Adm. R. S. Hatcher, U. S. N., Assistant Chief of the Bureau of Aeronautics for Research and Development.
Mr. F. A. Louden, Bureau of Aeronautics, Department of the Navy.
Capt. C. K. Bergin, U. S. N., Assistant Chief of the Bureau of Ordnance for Research.
Maj. Gen. Leslie E. Simon, U. S. A., Chief, Ordnance Research and Development Division.
Mr. Harold D. Hoekstra, Civil Aeronautics Administration.
Dr. Hugh L. Dryden (ex officio).
Mr. Floyd L. Thompson, NACA Langley Aeronautical Laboratory.
Mr. Russell G. Robinson, NACA Ames Aeronautical Laboratory.
Capt. W. S. Diehl, U. S. N. (Ret.).
Mr. Clarence L. Johnson, Lockheed Aircraft Corp.
Dr. A. Kartveli, Republic Aviation Corp.
Mr. Bartram Kelley, Bell Aircraft Corp.
Dr. Clark B. Millikan, California Institute of Technology.
Dr. William J. O'Donnell, Republic Aviation Corp.
Dr. W. Bailey Oswald, Douglas Aircraft Co., Inc.
Mr. Kendall Perkins, McDonnell Aircraft Corp.
Dr. Allen E. Puckett, Hughes Aircraft Co.
Mr. George Snyder, Boeing Airplane Co.
Prof. E. S. Taylor, Massachusetts Institute of Technology.
Mr. Charles Tilgner, Jr., Grumman Aircraft Engineering Corp.
Mr. R. H. Widmer, Convair, Division of General Dynamics Corp.
Mr. Robert J. Woods, Bell Aircraft Corp.

Mr. Milton B. Ames, Jr., Secretary

Subcommittee on Fluid Mechanics

Dr. Clark B. Millikan, California Institute of Technology, Chairman.
Dr. Theodore Theodorsen, Air Research and Development Command, U. S. Air Force.
Mr. E. Haynes, Air Research and Development Command, U. S. Air Force.
Mr. Phillip Eisenberg, Office of Naval Research, Department of the Navy.
Mr. John D. Nicolaides, Bureau of Ordnance, Department of the Navy.
Mr. Joseph Sternberg, Ballistic Research Laboratories, Aberdeen Proving Ground.
Dr. G. B. Schubauer, National Bureau of Standards.
Dr. Carl Kaplan, NACA Langley Aeronautical Laboratory.
Mr. John Stack, NACA Langley Aeronautical Laboratory.
Dr. D. R. Chapman, NACA Ames Aeronautical Laboratory.
Mr. Robert T. Jones, NACA Ames Aeronautical Laboratory.
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Dr. Antonio Ferri, Polytechnic Institute of Brooklyn.
Dr. Hans W. Liepmann, California Institute of Technology.
Dr. C. C. Lin, Massachusetts Institute of Technology.
Dr. William R. Sears, Cornell University.

Mr. E. O. Pearson, Jr., Secretary

Subcommittee on High-Speed Aerodynamics

Dr. Allen E. Puckett, Hughes Aircraft Co., Chairman.
Mr. David W. Lueck, Air Research and Development Command, U. S. Air Force.
Mr. H. L. Anderson, Wright Air Development Center, U. S. Air Force.
Mr. F. A. Louden, Bureau of Aeronautics, Department of the Navy.
Dr. H. H. Kurzweg, Naval Ordnance Laboratory.
Mr. C. L. Poor, III, Ballistic Research Laboratories, Aberdeen Proving Ground.
Mr. John Stack, NACA Langley Aeronautical Laboratory.
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Mr. Macon C. Ellis, Jr., NACA Langley Aeronautical Laboratory.
Mr. Arthur A. Regier, NACA Langley Aeronautical Laboratory.
Mr. Newell D. Sanders, NACA Lewis Flight Propulsion Laboratory.
Dr. Richard H. Bolt, Massachusetts Institute of Technology.
Mr. Allen W. Dallas, Air Transport Association of America.
Dr. Robert O. Fehr, General Electric Co.
Dr. Stacy R. Guild, Johns Hopkins Hospital.
Dr. Hans W. Liepmann, California Institute of Technology.
Dr. Ross A. McFarland, Harvard School of Public Health.
Mr. M. M. Miller, Douglas Aircraft Co., Inc.
Mr. John M. Tyler, Pratt and Whitney Aircraft Division, United Aircraft Corp.
Dr. P. J. Westervelt, Brown University.
Mr. H. W. Withington, Boeing Airplane Co.

Mr. Mason T. Charak, Secretary.

Industry Consulting Committee

Mr. Mundy I. Peale, Republic Aviation Corp., Chairman.
Mr. C. W. LaPierre, General Electric Co., Vice Chairman.
Mr. William M. Allen, Boeing Airplane Co.
Mr. J. H. Carmichael, Capital Airlines, Inc.
Mr. M. P. Ferguson, Bendix Aviation Corp.
Mr. L. R. Grumman, Grumman Aircraft Engineering Corp.
Mr. Roy T. Hurley, Curtiss-Wright Corp.
Mr. William Littlewood, American Airlines, Inc.
Dr. F. N. Piasecki, Piasecki Helicopter Corp.
Mr. Dwane L. Wallace, Cessna Aircraft Co.

Dr. T. L. K. Smull, Secretary.

Special Committee on Procedures for Unitary Facilities

Dr. Hugh L. Dryden, National Advisory Committee for Aeronautics, Chairman.
Col. F. H. Richardson, U. S. A. F., Arnold Engineering Development Center.
Mr. F. A. Louden, Bureau of Aeronautics, Department of the Navy.
Mr. W. H. Weeks, Civil Aeronautics Administration.
Mr. A. T. Colwell, Thompson Products, Inc.
Maj. Gen. Edward M. Powers, U. S. A. F. (Ret.), Wright Aeronautical Division, Curtiss-Wright Corp.
Mr. Edward C. Wells, Boeing Airplane Co.

Part III—FINANCIAL REPORT

Funds appropriated for the Committee for the fiscal years 1954 and 1955 and obligations against the fiscal year 1954 appropriations are as follows;

	Fiscal year 1954		Fiscal year 1955
	Allotments	Obligations	Allotments
SALARIES AND EXPENSES APPROPRIATION			
NACA headquarters	$1, 463, 490	$1, 448, 568	$1, 461, 900
Langley Aeronautical Laboratory	19, 472, 300	19, 433, 956	19, 884, 200
Ames Aeronautical Laboratory	7, 995, 264	7, 966, 112	8, 679, 930
Lewis Flight Propulsion Laboratory	17, 602, 707	17, 576, 208	17, 916, 515
High-Speed Flight Station	1, 462, 900	1, 436, 927	1, 679, 160
Pilotless Aircraft Station	770, 800	756, 025	696, 430
Western Coordination Office	17, 924	17, 702	18, 470
Wright-Patterson Coordination Office	12, 729	12, 580	13, 395
Research contracts with educational institutions	746, 000	743, 944	750, 000
Research contracts with other Government agencies	196, 400	196, 400	200, 000
Reserve reappropriated for the fiscal year 1955	1, 000, 000	----------	−1, 000, 000
Other reserves	259, 486	----------	700, 000
Unobligated balance	----------	1, 411, 578	----------
Total	[1] 51, 000, 000	51, 000, 000	[2] 51, 000, 000
CONSTRUCTION AND EQUIPMENT APPROPRIATION			
Langley Aeronautical Laboratory	7, 938, 290	3, 195, 767	1, 220, 000
Ames Aeronautical Laboratory	4, 640, 700	3, 933, 340	349, 000
Lewis Flight Propulsion Laboratory	10	10	3, 330, 000
Reserve for transfer to later years	310, 000	----------	101, 000
Reserve transferred from prior years	−1, 450, 000	−1, 450, 000	−380, 000
Unobligated balance	----------	[3] 5, 759, 883	----------
Total	[1] 11, 439, 000	11, 439, 000	[2] 4, 620, 000

[1] Appropriated in the First Independent Offices Appropriation Act, 1954, approved July 31, 1953; the Construction and Equipment appropriation includes $4,200,000 to complete the financing of the fiscal year 1951 program.

[2] Appropriated in the Independent Offices Appropriation Act, 1955, approved June 24, 1954.

[3] $5,717,631 of this balance remains available for obligation until expended.