O.E. LANCASTER

Jet Propulsion Engines

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JET PROPULSION ENGINES

VOLUME XII

HIGH SPEED AERODYNAMICS

AND JET PROPULSION

JET PROPULSION ENGINES

EDITOR: O. E. LANCASTER

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FOREWORD

On behalf of the Editorial Board, I would like to make an acknowledgement to those branches of our military establishment whose interest and whose financial support were instrumental in the initiation of this publication program. It is noteworthy that this assistance has included all three branches of our Services. The Department of the Air Force through the Air Research and Development Command, the Department of the Army through the Office of the Chief of Ordnance, and the Department of the Navy through the Bureau of Aeronautics, Bureau of Ships, Bureau of Ordnance, and the Office of Naval Research made significant contributions. In particular, the Power Branch of the Office of Naval Research has carried the burden of responsibilities of the contractual administration and processing of all manuscripts from a security standpoint. The administration, operation, and editorial functions of the program have been centered at Princeton University. In addition, the University has contributed financially to the support of the undertaking. It is appropriate that special appreciation be expressed to Princeton University for its important over-all role in this effort.

The Editorial Board is confident that the present series which this support has made possible will have far-reaching beneficial effects on the further development of the aeronautical sciences.

Theodore von Kármán

PREFACE

Rapid advances made during the past decade on problems associated with speed flight have brought into ever sharper focus the need for a comprehensive and competent treatment of the fundamental aspects of the aerodynamic and propulsion problems of high speed flight, together with a survey of those aspects of the underlying basic sciences cognate to such problems. The need for a treatment of this type has been long felt in research institutions, universities, and private industry and its potential reflected importance in the advanced training of nascent aeronautical scientists has also been an important motivation in this undertaking.

The entire program is the cumulative work of over one hundred scientists and engineers, representing many different branches of engineering and fields of science both in this country and abroad.

The work consists of twelve volumes treating in sequence elements of the properties of gases, liquids, and solids; combustion processes and chemical kinetics; fundamentals of gas dynamics; viscous phenomena; turbulence; heat transfer; theoretical methods in high speed aerodynamics; applications to wings, bodies and complete aircraft; nonsteady aerodynamics; principles of physical measurements; experimental methods in high speed aerodynamics and combustion; aerodynamic problems of turbo machines; the combination of aerodynamic and combustion principles in combustor design; and finally, problems of complete power plants. The intent has been to emphasize the fundamental aspects of jet propulsion and high speed aerodynamics, to develop the theoretical tools for attack on these problems, and to seek to highlight the directions in which research may be potentially most fruitful.

Preliminary discussions, which ultimately led to the foundation of the present program, were held in 1947 and 1948 and, in large measure, by virtue of the enthusiasm, inspiration, and encouragement of Dr. Theodore von Kármán and later the invaluable assistance of Dr. Hugh L. Dryden and Dean Hugh Taylor as members of the Editorial Board, these discussions ultimately saw their fruition in the formal establishment of the Aeronautics Publication Program at Princeton University in the fall of 1949.

The contributing authors and, in particular, the volume editors, have sacrificed generously of their spare time under present-day emergency conditions where continuing demands on their energies have been great. The program is also indebted to the work of Dr. Martin Summerfield who guided the planning work as General Editor from 1949–1952. The cooperation and assistance of the personnel of Princeton University Press and of the staff of this office has been noteworthy. In particular, Mr. H. S. Bailey, Jr., the Director of the Press, and Mr. R. S. Snedeker,

who has supervised the project at the Press and drawn all the figures, have been of great help. Special mention is also due Mrs. E. W. Wetterau of this office who has handled the bulk of the detailed editorial work for the program.

Coleman duP. Donaldson
General Editor

PREFACE TO VOLUME XII

This volume considers those principles and problems encountered in combining components to form a complete engine. It relies heavily upon the other volumes which deal with basic principles or principles and problems related to components of an engine.

Section A gives a concise history of the development of rockets and air flow jet engines. Section B gives definitions of thrust and various efficiencies and derives relationships for the performance of the different jet propulsion systems. Section C gives the performance analysis of turbojets based on the internal solution of matching the compressor, combustor, turbine, and nozzle. It includes a discussion of off-design performance and describes the problems of control and testing which are unique to a complete unit. Section D treats the turboprop in a somewhat similar manner. It gives the logic for interest in a turboprop and discusses the additional complications. Section E is devoted to the ramjet, its performance, controls, and methods of testing. Section F discusses the wave engines in general, and in particular the pulse jet and the comprex. Section G treats the liquid rocket engine, from the consideration of appropriate fuels (both monopropellent and bipropellent) to the designing and testing of the motor. Section H gives a similar treatment for solid propelled rockets, with special stress on the stability and characteristics of burning. The possibility of a variety of hybrid engines, part rocket, part turbine, or more generally, part jet and part rotating machinery, is introduced in Sections I and J which treat two such cases—the ram-rocket and the jet rotor. Each section derives the possible performance and outlines the possible use of these engines. Section K deals with the problems in making a nuclear jet power plant suitable for aircraft. It gives the theory related to the shielding, heat transfer, and the production and control of a small lightweight reactor. The final section does not quite give a peek into the future, but it gives a systematic procedure for exploring the many possibilities of the types of jet engines.

At this point, I want to express my appreciation for the kind cooperation of the many authors who contributed to make the volume possible, and especially I want to give my heartfelt thanks to Dr. Coleman duP. Donaldson and his staff, whose spark and tireless efforts have brought our works to fruition.

O. E. Lancaster
Volume Editor

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JET PROPULSION ENGINES

SECTION A

HISTORICAL DEVELOPMENT OF JET PROPULSION

CHAPTER 1. A SHORT HISTORY OF ROCKET PROPULSION UP TO 1945

F. J. MALINA

A,1. Introduction. A thorough history of the development of jet propulsion engines in its major aspects cannot be written today,[1] especially for the period starting about 1935. During this period the industrially advanced nations of the world, including France, Germany, Italy, Japan, the United Kingdom, the United States, and the Soviet Union, initiated broad developments, the results of which have not been completely released to the scientific world. It was during this period that the practical developments of jet propulsion flowered to an unprecedented degree. A broad structure of knowledge was built giving jet propulsion a scientific discipline comparable to that which had been developed earlier in the twentieth century for other types of heat engines, such as steam and internal combustion engines.

A characteristic of this period was the restricted dissemination of information because of world political conditions, with the result that jet propulsion developments in many cases took place in national isolation. Therefore, priority of discovery of principles and the first application of methods from related fields cannot be easily determined, and loses much of its significance. Work on jet propulsion engines within each country advanced along parallel lines.

A second characteristic of the period was the use of teams of research workers on a scale seldom found previously for other technical developments. This further complicates the determination of individual priority of discovery, so that it frequently appears more equitable to credit a community of workers for many of the achievements of this period.

In this short chapter we will attempt to summarize technical advances up to the end of World War II. Because of the situation prevailing during the most vigorous period of the development of jet propulsion, this history

[1] This chapter was written in 1950.

will suffer in being partial both in the sense of incompleteness and of emphasis on developments in those countries where information has been more widely disseminated.

A,2. Classification of Jet Propulsion Engines. Before discussing the history of jet propulsion, it is necessary to have clear definitions of the terms to be used.[2] It is customarily said that jet propulsion is the method of propulsion based on Newton's third law of motion, i.e. the principle of reaction. However, this is not sufficiently specific, since all forms of propulsion in a fluid medium depend on a force of reaction caused by the momentum imparted to a portion of the fluid. For example, the action of the conventional propeller consists essentially in increasing the momentum of the air or water passing through the propeller disk; the thrust of the propeller can be considered as the reaction of the increase of momentum. The propulsion of a rowboat, a swimming person, or a flying bird is also based on the reaction principle, the propulsive force being furnished by the reaction of the increased momentum of the surrounding fluid.

In order to discriminate between jet and other forms of propulsion it can be said that, in the case of jet propulsion, matter is ejected from the propelled body in order to create momentum. This matter may either be wholly carried within the body or taken from the surrounding medium through which the propelled body moves. A jet propulsion device of the first kind is called a "rocket," a device of the second kind is called a fluid flow engine. In the latter the fluid refers to that taken in from the surrounding medium.

From the above definitions it is seen that a ducted propeller belongs to a class of jet propulsion devices, whereas a free propeller does not. The Stipa airplane built by Caproni and flown in 1932 should be classified as a jet-propelled aircraft. The Stipa airplane can be considered as a predecessor of the jet-propelled aircraft of today. However, it is more satisfactory to restrict the term "jet propulsion" to thermal jet propulsion systems. It is evident that mixed cases are possible where part of the energy is created by mechanical means, and part by thermal means.

An interesting example in nature of locomotion by jet propulsion is found in the squid (*Loligo*), a mollusk of the class of the Cephalopoda. The squid has a kind of siphon under its neck and is able to eject water to propel itself, especially when a short burst of speed is desired. The squid is thus equipped with a mechanical jet propulsion device.

The various systems of jet propulsion are classified below according to their method of operation [*1,2,3*]. As pointed out above, the two main

[2] This article is based upon a lecture by Th. von Kármán in 1943, given as the initial lecture of a course in jet propulsion for Army and Navy personnel at the California Institute of Technology.

classes are (1) rockets, which do not use the surrounding medium for propulsion, and (2) devices which do, whether the medium be atmospheric air or water. There are several types in each class as illustrated in Tables A,2a and A,2b.

Table A,2a. Rockets—jet propulsion systems not utilizing the surrounding medium.

1. Solid propellant rocket motor
   a. Constant volume combustion
   b. Constant pressure combustion
      (i) Restricted burning
      (ii) Unrestricted burning
2. Liquid propellant rocket engine
   a. Gas pressure feed system
   b. Turbopump feed system
3. Gas or vapor propellant rocket engine

Table A,2b. Jet propulsion systems utilizing the surrounding medium.

1. Jet reaction of engine exhaust
2. Campini system
3. Turbojet
4. Turboprop
5. Ducted turboprop
6. By-pass engine
7. Compressorless system
8. Jet-driven rotor
9. Ducted radiation

A more complex system of classification for jet engines has been proposed by Zwicky ([*4*] and Sec. L). He arranges the members of the class of jet engines in a so-called *morphological box* or manifold. The various dimensions of the box represent significant known chemical, mechanical, and aerodynamic or hydrodynamic characteristics of jet engines. Engines with different combinations of these characteristics can then be described in a systematic manner. On the basis of this scheme of classification he has proposed a new system of nomenclature.

A,3. From Antiquity to the Beginning of the Twentieth Century. Historical research has not so far been able to determine the first application of the jet propulsion principle. The earliest known account in this field tells of the construction of a flying dove of wood by Archytas, the founder of theoretical mechanics. Archytas was a Greek who lived in Tarentum in southern Italy around the fourth century B.C. Aulus Gellius in his *Noctes Atticae* gave origin to the idea that the dove flew by means of expanding vapor contained within it—perhaps the dove was jet propelled [*5,6*].

There seems no doubt that Heron of Alexandria invented a device which operated on the reaction principle. The exact date of his work is not definitely known, but it is probable that he lived in the beginning of the first century B.C. [*6*]. Among numerous inventions described by Heron in his treatises is the *aelopile*, which consisted of a hollow sphere that was rotated about an axis by steam issuing from two jets, which were arranged in opposite directions in a plane perpendicular to the axis of rotation. Centuries passed before the reaction principle was applied again, after which a continuous line of development can be traced up to the present time.

The reaction principle was applied in the solid propellant rocket, commonly known as the black powder rocket. The earliest reference to what appears to have been a black powder rocket was recorded in the *Wu Ching Tsung Yao*, an official publication dated 1040 A.D. which describes various kinds of weapons used by the army during the Sung Dynasty [*7,8*]. A fire arrow, the Huo Yao Pien Chien, is mentioned, and the description and name clearly indicate the use of black powder. It is stated that five ounces of powder were applied to the end of the arrow. Fire arrows of this type appear to have been put to use between 950 and 1000 A.D. Another type of fire arrow is mentioned, the so-called San Kung Ch'uang Tzu Nu, which was projected by a crossbow. It is stated that the arrow may be projected by the force of black powder, if the elasticity of the crossbow permits. The construction of black powder rockets spread rapidly to other parts of Asia, and it is believed that they were first introduced to Europe during the Mongol invasions.

The propellant of the black powder rocket was a mixture of potassium nitrate (saltpeter), charcoal, and sulfur of varying proportions. This mixture was used until the development of new types of explosives and propellants for guns at the end of the nineteenth century.

The discovery of black powder is still a lively subject of historical research. Recently, Needham [*8*] and Tsao Thien-Chin appear to have found the earliest mention of black powder in the seventh-century book of the Thang dynasty, *Chen Yuan Miao Tao Yao Lueh* (*Important Information about the True Origin of the Mysterious Tao*). A mixture of saltpeter, sulfur, arsenic, and honey, as the source of carbon, is described, with the warning that the mixture had caused serious accidents and should not be made.

A few formulas for black powder which have been used at various times have been tabulated on a percentage basis by Davis [*9*] as shown in Table A,3.

Little change in the formula arrived at in 1560 for guns and cannon has been made by the modern manufacturers of black powder. This mixture is "too fast" for rockets. A typical mixture used at present in

black powder rockets contains 60 per cent saltpeter, 30 per cent charcoal, and 10 per cent sulfur.[3]

It seems that John Bate, early in the seventeenth century, understood the individual functions of the three components of black powder when he wrote: "The Saltpeter is the Soule, the Sulphur the Life, and the Coales the Body of it." Saltpeter supplies the oxygen for the combustion of the charcoal, and the sulfur is called the *life* of the mixture because this element first catches fire and spreads it throughout the mass [*10*].

Table A,3. Black powder formulas.

	Saltpeter	*Charcoal*	*Sulfur*
8th century,[4] Marcus Graecus	66.66	22.22	11.11
	69.22	23.07	7.69
c.1252, Roger Bacon	37.50	31.25	31.25
1350, Arderne (laboratory receipt)	66.6	22.2	11.1
1360, Whitehorne	50.0	33.3	16.6
1560, Brusselle studies	75.0	15.62	9.38
1635, British Government contract	75.0	12.5	12.5
1781, Bishop Watson	75.0	15.0	10.0

In the sixteenth century, descriptions were printed of the method of fabricating powder rockets. Before that time, and even afterward, as far as special details were concerned, the manner of fabrication was a closely guarded trade secret, which was passed on from artisan to artisan. The rocket consisted of a paper tube constricted at one end to about two thirds of the tube diameter. The propellant was loaded into the tube by pounding (later pressing) successive layers around a tapered spindle. The conical cavity, extending from the constriction almost to the opposite end of the tube, provided a sufficiently large burning area for the charge to deliver a high enough thrust for propelling the rocket. This method of fabrication is still used for "sky rockets," with the exception that the exit hole for the gases is no longer provided by crimping the tube, but by inserting a perforated plug of clay.

The black powder rocket motor was brought to its highest state of development by Sir William Congreve [*11*], who was spurred into carrying out his work by the success achieved by Indian troops with rockets against the British during their campaigns toward the end of the eighteenth century. In 1817 Congreve listed seven types of rockets designed for the British army, one of which, for example, weighed 42 pounds and had a

[3] Lorrain Hanzelet (Jean Appier) has the following comments in his *La Pyrotechnie*, published in 1630: "It is necessary to have compositions according to the greatness or the littleness of the rockets, for that which is proper for the little ones is too violent for the large ones—because the fire, being lighted in a large tube, lights a composition of great amplitude and burns a great quantity of material, and no geometric proportionality applies."

[4] Date not certain. Modern scholars consider the thirteenth century to be closer to the truth.

range of 3000 yards. The principal difference between the Congreve rockets and the common fireworks rocket was the replacement of the paper tube by a cast iron tube. Congreve died in 1828 believing that rockets would replace artillery in warfare. After his death the use of rockets declined in spite of improvements made in their accuracy, principally by William Hale who developed at Woolwich around 1846 a rocket spun by tangential jets. At about this time the loading of rockets was greatly speeded up as manual loading by mallet and wedge was replaced by the hydraulic press and power drill [*12,13*].

Although empirical improvements were made in the construction of the powder rocket both in Asia and in Europe, the mode of operation of the device was not correctly understood until the seventeenth century. Vanoccio Biringuccio wrote the following in his *De la Pirotechnia*, which was published in Venice in 1540:

> "Now powder is earth, consisting of the four elementary principles, and when the sulfur conducts the fire into the dryest part of the powder, fire and air increase . . . the other elements also gird themselves for battle with each other and the rage of battle is changed by heat and moisture into a strong wind . . . "

The description of the reaction principle awaited the work of Galileo Galilei, and finally the formulation by Sir Isaac Newton of the laws of motion in his *Philosophiae naturalis principia mathematica*, which was published in 1687.

William's Gravesande, at the time of Newton, published a work entitled *Natural Philosophy* with the subtitle *An Introduction to Sir Isaac Newton's Philosophy*. In this book he illustrated Newton's third law of motion by means of a steam carriage consisting of a boiler topped by a converging nozzle, which erroneously came to be known as "Newton's steam carriage." It thus appears that s'Gravesande can be credited with being the first to describe a jet-propelled device with an understanding of the basic principle involved.

Jet propulsion engines from the time of the first application of the reaction principle, according to our system of classification, were all rocket devices using solid or gaseous propellants. Thermal jet propulsion engines did not make their appearance until the twentieth century. Jet propulsion for boats was suggested by John Fitch of the United States in 1790, and Rumsey in 1793 succeeded in devising a working system which propelled a boat on the Potomac at 4 miles per hour [*14*].

During the nineteenth century, gaseous propellant rockets were frequently described in patent literature; however, no important use of the proposals was made. Steam and compressed air were chosen as gases suitable, first, for ground locomotion and, later, also for propelling aircraft. Solid propellant rockets, beginning with their invention in the

10th century, were developed primarily for firework displays and as military ordnance. The development of military rockets was pushed into the background during the second half of the nineteenth century by the improvements achieved in gunnery. The spinning shell fired from guns was so much superior both in range and accuracy that most armies abandoned the use of rockets. Their use on a large scale as a military weapon did not return until World War II.

A,4. The Rocket from 1900 to 1945. Although the basic principle of operation of the rocket engine was known after Newton published the laws of motion in 1687, a further development in fundamental theory did not take place until this century.

As pointed out in the previous chapter, the solid propellant rocket was built in accordance with the "tricks of the trade." No systematic study was made to improve rocket performance either by applying fundamental advances made in chemistry or by improving mechanical design by applying new knowledge of thermodynamics and gas dynamics. Furthermore, the powder-rocket motor did not contribute to the steady growth of these sciences as, for example, the steam engine did after James Watt patented the principle of the condenser in 1769. After the development of the rifled gun, there was neither the pressing need to design better rockets nor the curiosity to probe its potentialities.

Around the beginning of this century, the great progress achieved in the construction of a heavier-than-air flying machine stimulated a more serious consideration of the possibility of travel outside the earth's atmosphere by means of the rocket. Furthermore, basic research into the structure and behavior of the atmosphere and of extraterrestrial phenomena began to require a vehicle which could carry instruments to heights greater than could be reached by the balloon. By the 1920's it became evident that the power characteristics of the piston engine-propeller combination would limit the speed of the aircraft as well as the load that could be lifted from the ground or water within a reasonable take-off distance. The rocket engine promised to be of use for both of these problems. In addition, the demand for a tremendous increase in fire power for ground armies and the need for recoilless artillery for aircraft during World War II brought the relatively inaccurate rocket back into service as a military weapon.

Before reviewing the progress made in the design of rocket engines, it is worthwhile to mention some of the fundamental inventions and scientific advances of the latter part of the nineteenth century upon which this progress depended.

In 1884 Paul Vieille invented the first smokeless powder, colloided *poudre blanche* made by treating a mixture of soluble and insoluble nitro-cotton with ether-alcohol. In 1888 Alfred Nobel invented *ballistite*,

a colloidal mixture of nitroglycerin and soluble nitrocellulose. In 1895 Linde operated a plant producing liquid air in fairly large quantities, and in the same year Dewar successfully liquefied hydrogen.

The end of the nineteenth century saw the development of large scale steel-making processes and of steel alloys of high strength. In 1886 the process of electric welding was developed, and other welding techniques by the end of the century. Aluminum became a commercial metal with its production in the electric furnace from 1886. The plastics industry had its beginning in the latter part of the century [*14*]. By the beginning of the twentieth century the work of Carnot, Joule, Thomson (Lord Kelvin), and others had provided the essential basic theory required for understanding the thermodynamic aspects of heat engines, including the rocket motor.

Experience had been accumulated on the combustion of fuel with air in the internal combustion engine, and preliminary steps had been taken in the design of the gas turbine. Important contributions to the theory of the flow of compressible fluids had been made by Saint Venant, Wantzel, Reynolds, Riemann, Hugoniot, Mach, and others. The method of obtaining a high velocity jet by means of the convergent-divergent nozzle had been discovered by Carl de Laval.

In a British patent granted in 1889 [*15*] de Laval stated:

> "My invention relates to an improvement in turbines which are set in motion by means of a current of steam; and the object of the improvement is to increase, by complete expansion, the velocity of the steam current, thus producing the relatively largest quantity of *vis viva* of the steam.
>
> "I attain this object by the construction of the steam supply pipe in such a manner that the cross sections of the same are slowly increased near the turbine wheel and in the direction of the latter. The rate of increasing the cross sections is due to the proportion and distance between the smallest section and the largest one, in such a manner that in the steam passage between these two sections a permanent current of steam is produced under isoëntropical expansion."

A,5. Development of Rocket Engines. (From Literature Published up to 1940.) K. E. Ziolkowsky is generally considered the first to have proposed space travel on scientific grounds by means of the rocket. A paper, which he completed in 1898, entitled "Rockets in Cosmic Space," was published in the Russian journal, *Science Survey*, in 1903 [*16*]. He realized that if a rocket were to attain velocities necessary for space travel the rocket engine would have to be greatly superior to the black powder rocket, in both specific impulse and the absolute

magnitude of thrust. A summary of his work was recently published by Kosmodemjanski [*17*].

Ziolkowsky proposed the possible use of liquid propellants and also suggested the use of the step-rocket principle, which was demonstrated by the launching of the American-developed WAC Corporal sounding rocket [*18*] from the German-developed V-2 [*19*] on February 24, 1949. The WAC Corporal, in attaining a height of about 250 miles, entered air-free space and thus practically demonstrated Ziolkowsky's proposals almost exactly 50 years later. (The step-rocket principle was actually used in firework rockets as early as 1700, not to improve range but to evoke a better display, and in 1855 Colonel Boxer at the Royal Laboratory, Woolwich, had begun the development of a two-step life-line rocket [*20*].)[5]

An American contemporary of Ziolkowsky, R. H. Goddard, was speculating at the same time on the conditions of the upper atmosphere, and in a notebook dating back to 1899 he mentioned the possibility of employing rockets for this work [*21*]. In 1907 he prepared a paper suggesting that heat from radioactive materials could be used to expel substances at high velocity from an orifice, thus furnishing jet propulsion sufficient to navigate in interplanetary space. Publication of the paper was refused by several popular scientific journals.

The idea of using hydrogen and oxygen as a propellant for an interplanetary rocket, and the construction of such a rocket according to the step-rocket principle occurred to Goddard in 1907. Beginning in 1914 he took out several patents on the "rocket apparatus." Then in 1919 the Smithsonian Institution published his paper entitled "A Method of Reaching Extreme Altitudes" [*22*], which reported the pioneering experiments he had made to determine accurately the mechanical energy of a solid propellant rocket. The rocket motor was of the constant volume type, in that a charge of nitrocellulose propellant or smokeless powder was burned very rapidly with the nozzle of the rocket plugged at the time of ignition.

The following paragraphs are taken from Goddard's paper:

> "It was obviously desirable to perform certain experiments: First, with the object of finding just how inefficient an ordinary rocket is, secondly, to determine to what extent the efficiency could be increased in a rocket of new design. The term "efficiency" here means the ratio of the kinetic energy of the expelled gases to the heat energy of the powder, the kinetic energy being calculated from

[5] J. J. Driscoll has brought to my attention a book entitled "Artis Magnae Artilleriae" by Casimiro Siemienowics published in Amsterdam in 1650. It contains detailed information on the construction of black powder rockets in which the charges could be fired in parallel, and also arrangements for firing in series or a step-wise manner are shown.

the average velocity of ejection, which was obtained indirectly by observations on the *recoil* of the rocket.

"It was found that not only does the powder in an ordinary rocket constitute but a small fraction of the total mass ($\frac{1}{4}$ or $\frac{1}{5}$), but that, furthermore, the efficiency is only 2 per cent, the average velocity of ejection being about 1000 ft/sec. This was true even in the case of the Coston ship rocket, which was found to have a range of a quarter of a mile.

"Experiments were next performed with the object of increasing the average velocity of ejection of the gases. Charges of dense smokeless powder were fired in strong steel chambers, these chambers being provided with smooth tapered nozzles, the object of which was to obtain the work of expansion of the gases, much as is done in the de Laval steam turbine. The efficiencies and velocities obtained in this way were remarkably high, the highest efficiency, or rather "duty," being over 64 per cent, and the highest average velocity of ejection being slightly under 8000 ft/sec., which exceeds any velocity hitherto attained by matter in appreciable amounts.

"These velocities were proved to be real velocities, and not merely effects due to reaction against the air, by firing the same steel chambers *in vacuo*, and observing the recoil. The velocities obtained in this way were not much different from those obtained in air."

During World War I, experiments with this type of motor were made by Goddard and Hickman; however, at the end of the war in 1918 the development was dropped. Their work was resumed about twenty years later, and led to the development of the "bazooka" rocket weapon which was used in large numbers in World War II. The propulsion of a rocket by means of successive impulses was considered by Goddard in his paper of 1919 and further analyzed by Tsien and Malina in 1938 [*23*]. The use of the constant volume motor for long duration propulsion has not been found practical, mainly because of the difficulties in designing a mechanism for reloading the motor.

Interest in liquid propellant rocket motors received a great stimulus with the publication, by Oberth in 1923, of a book primarily concerned with theoretical aspects of interplanetary travel [*24*]. He suggested alcohol and liquefied hydrogen as likely fuels, with liquid oxygen as the oxidizer.

In 1928 Esnault-Pelterie published the lectures, which he had delivered the preceding year to the Astronomical Society of France, on the exploration of the upper atmosphere by means of rockets and on the possibilities of interplanetary flight [*25*]; and in 1930 he published an expanded version of his lectures, including a discussion of the thrust characteristics and the problem of combustion of the liquid propellant

rocket motor [*26*]. Beginning in 1928, Rinin began the publication of a series of nine volumes reviewing various aspects of rocket propulsion with special reference to interplanetary flight. His fifth volume entitled *Theory of Reaction Propulsion* was published in 1929 [*27*]. In 1929, Oberth published a second book dealing with interplanetary flight and general aspects of rocket motor design [*28*].

In Austria, France, Germany, the United Kingdom, the United States, and the Soviet Union [*29,30,31,32*] the publications of Oberth, Esnault-Pelterie, and Rinin led to the establishment of societies of enthusiasts for conquering space. Some of the societies carried out experimental programs on rocket engines, but the work was handicapped by adverse financial conditions, and the tendency to tackle problems of design in a highly empirical fashion without calling upon knowledge available in related technical fields.

Between 1930 and 1933, the Verein für Raumschiffahrt (Society for Space Travel) in Germany made experiments with liquid propellant engines using a propellant combination of liquid oxygen and gasoline or water-diluted alcohol. Cooling of the motors was attempted by surrounding the whole motor with a cooling jacket filled with water or fuel. The coolant was not circulated at velocities high enough to absorb the heat transferred to the walls in high specific impulse motors [*30*]. This society was the precursor of the German Army group which, starting in 1932 under the command of Colonel Walter Dornberger, developed the V-2 long range rocket missile.

The American Rocket Society began experimentation on liquid oxygen-gasoline motors in 1932 and continued its activities for nearly a decade; a summary of the work of the society during this period was recently given by Wyld [*33*]. Experimental motor designs were tested, including a motor constructed of thick aluminum walls to absorb the heat transferred during short periods of operation, and a motor using regenerative cooling was tested in 1933. Several leading members of the American Rocket Society formed Reaction Motors, Inc., in 1941, which produced the liquid oxygen-alcohol engine for the Bell X-1 supersonic aircraft.

The possibility of augmenting the thrust of a rocket motor by allowing the jet to mix with the surrounding air in the presence of bodies which deflect the air set in motion by the jet was first tried experimentally by Mélot in 1924 [*34*]. The results of further studies of thrust augmentors were published in 1932 by Jacobs and Shoemaker [*35*], and in 1933 by Schubauer [*36*].

In 1933 Sänger published a review of the state of design theory of rocket motors in his book *Raketen-Flugtechnik* [*37*], and followed this with a paper in 1934 which gave some results of tests on a motor burning gaseous oxygen and diesel fuel oil. He reported obtaining an exhaust

velocity above 10,000 ft/sec at a chamber pressure of 1500 lb/in^2. The motor was cooled by circulating the cooling fluid through copper tubes which were wound to form the walls of the combustion chamber and exhaust nozzle [*38*]. In a paper published in 1936 he put forward the following design rules [*39*]:

(a) The volume of necessary combustion space (in cm^3) must bear to the area of the smallest cross section of the nozzle (in cm^2) the relation from 50 to 5000 cm.
(b) Expansion nozzles for the exhaust gases of rocket motors should have an average angle of flare over 25° and under 270°.
(c) The circulation of the coolant along the walls in contact with the flame must take place in canals and must be so positive that it maintains unfailingly at every section of the wall in contact with the flame a velocity of circulation of prescribed amount.

In 1935 Rinin read a paper at the Volta Congress in Rome, briefly reviewing some thermodynamic aspects of the rocket motor and of thrust augmentors [*40*]. In the same year Damblanc published the results of his experiments with black powder motors used in France in signal, life-saving, and "anti-hail" rockets. The largest motor tested was constructed in two steps; each charge gave a peak thrust of about 200 lb, and the duration of each charge, provided with a central conical hole, was about 1.5 seconds. A discussion of the mode of burning of the powder charge was also presented [*41*].

Goddard published his second Smithsonian report in 1936 [*42*], which contains a general presentation of his work in developing a liquid propellant rocket. He states the following in regard to static tests carried out on a liquid oxygen-gasoline engine during the period 1930 to 1932:

> "Although combustion chambers had been constructed at Clark University (from 1920 to 1922) which operated satisfactorily, it appeared desirable to conduct a series of thorough tests in which the operating conditions were varied, the lift (thrust) being recorded as a function of time. Various modifications in the manner of feeding the liquids under pressure to the combustion chamber were tested, as well as variations in the proportions of the liquids, and in the size and shape of the chambers. The chief conclusions reached were that satisfactory operation of the combustion chambers could be obtained with considerable variation of conditions, and that larger combustion chambers afforded better operation than those of smaller size."

Furthermore, he wrote,

> "The combustion chamber finally decided upon for use in flights was $5\frac{3}{4}$ inches in diameter and weighed 5 pounds. The maximum lift

(thrust) was 289 pounds, and the period of combustion usually exceeded 20 seconds.

"The masses of liquids used during the lifting period were the quantities most difficult to determine. Using the largest likely value of the total mass of liquids ejected and the integral of the lift-time curve obtained mechanically, the velocity of the ejected gases was estimated to be over 5000 feet per second."

In 1936 a rocket research group was formed at the Guggenheim Aeronautical Laboratory of the California Institute of Technology (GALCIT)[6] with the purpose of studying the design fundamentals of solid and liquid propellant motors. This group consisted of F. J. Malina, H. S. Tsien, A. M. O. Smith, J. W. Parsons, W. Arnold, and E. S. Forman. The *Journal of the American Rocket Society* carried a report by Malina in 1938 describing tests made with a motor burning gaseous oxygen and ethylene [*43*], and a report by Parsons and Forman in 1939 giving the results of experiments on solid propellant motors burning smokeless powder under conditions of constant volume combustion [*44*]. In 1940 Malina published a paper in which the characteristics of the liquid propellant rocket motor based upon the theory of perfect gases were summarized, and a universal ideal thrust diagram was presented, from which have been derived the charts showing the variation of thrust coefficient with pressure ratio and nozzle area ratio. Tsien's derivation of the λ factor which corrects the calculated value of *velocity thrust* for the divergence of gas flow in an expanding nozzle was also included in the paper [*45*].

Vogelpohl in 1937 published a study of the thrust characteristics of the rocket motor in which he considered the effect on the thrust of the ratio of specific heats of the exhaust gases [*46*].

Experimental studies of rocket motors were initiated in 1939 by Truax at the U.S. Naval Engineering Experiment Station, Annapolis, Maryland, with the compressed air and gasoline combination [*47,48*].

In 1938 Bartocci published the results of extensive experiments made with gaseous oxygen at room temperature [*49*]. He studied in particular the thrust delivered by the de Laval nozzle as a function of pressure ratio for various values of the nozzle area ratio. His tests showed clearly the loss in thrust caused by under- and overexpansion of the gas in the nozzle.

A,6. Development of Solid Propellant Rocket Motors. (From Literature Published after 1940.) The revival of interest in rocket propulsion during the first part of this century was strongly influenced by ideas of the conquest of space. Investigation was concentrated on liquid

[6] The Jet Propulsion Laboratory is an outgrowth of the work of this group.

propellant engines since they promised to give the highest possible performance, and therefore work on solid propellant rocket motors was not given sustained attention. This position was radically changed by developments in aeronautics and ordnance in the 1930's.

From a thermodynamic point of view two kinds of solid propellant motors have been developed. In the first kind, combustion takes place at essentially constant volume, and a propulsive impulse is provided as already indicated in the previous article. In the second kind, combustion takes place at essentially constant pressure. The constant pressure rocket motor can further be classified into restricted- and unrestricted-burning types, the former having been developed to deliver a thrust force for a period from about 2 seconds to more than 30 seconds, and the latter for a period of less than 2 seconds.

A partial history can be pieced together from publications released after World War II on the development of various types of solid propellant rocket motors during the period preceding the outbreak of war in 1939 and the end of hostilities in 1945.

Developments in the United Kingdom and the United States. In 1935 the Research Department at Woolwich Arsenal, England, initiated investigations on motors using cordite smokeless powder as a propellant. By 1936, encouraging advances in technique had been made, and Sir Alwyn D. Crow was entrusted with the direction of an enlarged program of research and development [*50*]. An unrestricted-burning cordite motor of a new type was developed under this program which went into service in 1940 as the propulsive unit for an antiaircraft rocket. The motor consisted of a steel tube sprayed inside with a suspension of finely ground alumina in a solution of sodium silicate which was baked to provide a temperature-insulating layer. The cordite propellant charge, in the form of a hollow cylinder, was supported in the tube so that both the inner and the outer surfaces could burn. Motors of this type using 2-in.- and 3-in.-diameter cordite charges were developed. A larger motor was also constructed with a propellant charge composed of 11 tubular sticks of cordite disposed in the space between the rocket body tube and an internal axial tube, perforated to allow free flow of gas but which prevented a loss of cordite [*20*]. A description of the propellant used and other details of the cordite motors has been published recently by Wheeler, Whittaker, and Pike [*51*].

In the United States the military services did not support research on armament rockets between the two World Wars; however, L. A. Skinner of the Army Ordnance Department began tests in 1932 on his own initiative, with limited funds and facilities at Aberdeen, Maryland, He was primarily interested in the possibility of accelerating a shell by means of a solid or liquid propellant rocket motor [*52*].

The National Defense Research Committee (NDRC) upon the urging

of C. N. Hickman, who had worked on armament rockets with Goddard in 1918, set up Section H in Division A of the NDRC in 1940. This section concentrated on the design of unrestricted-burning solid propellant motors for various armament purposes. Much work was done on the preparation of ballistite propellant charges formed by (1) solvent extrusion, (2) solventless extrusion, (3) casting, and (4) pressure molding. This work was done between 1940 and 1943 at Indian Head, Maryland, and then, under the direction of R. E. Gibson, at the Alleghany Ballistics Laboratory which was operated for Section H by George Washington University.

In 1941 Section H and the Army Ordnance Department began their cooperation on the project which developed the bazooka. The motor designed for the projectile used five tubular charges of solvent-extruded ballistite, each with a $\frac{3}{8}$-in. outside, and a $\frac{1}{16}$-in. inside diameter. The motor operated at a peak chamber pressure of about 500 atmospheres, and the burning time was about thirty thousandths of a second.

A great impetus to the NDRC rocket program was given by a memorandum submitted in 1941 to Vannevar Bush by C. C. Lauritsen upon his return from a visit to England, where he studied the work done by Sir Alwyn Crow's group. In the same year a development program was initiated at the California Institute of Technology with Lauritsen as Director of Research. The design of rocket motors was supervised by W. A. Fowler, and work on propellants and interior ballistics was carried out under B. H. Sage, D. S. Clark, and W. N. Lacey.

By the end of World War II this group had designed and produced several types of unrestricted-burning solid propellant motors using dry extruded ballistite. The basic features of the motors were similar to those developed in the United Kingdom. The largest motor was designed to propel the aircraft rocket called the "Tiny Tim" and the propellant charge consisted of four cruciform ballistite grains weighing a total of 146 pounds. The exhaust gases flowed through 24 nozzles spaced symmetrically around a central nozzle with a blowout disk which ruptured if the chamber pressure reached 2250 lb/in.^{2} [*52*].

The striking advance made during this period in the theory underlying the design of solid propellant motors of the unrestricted-burning type is clearly brought out in a book recently published by Wimpress [*53*]. Although the book lacks references to the builders of this body of knowledge, it must be recognized as a product of numerous research workers in the United Kingdom and the United States, who studied various types of solid propellant motors.

The development and large scale production of solventless double-base propellant used in American armament rockets has been described by Marsh in an article published in 1945 [*54*].

Active work on solid and liquid propellant rocket engines for assist-

ing the take-off of aircraft was initiated in the United States by General H. H. Arnold in 1938 when he asked the National Academy of Sciences to arrange for an investigation of these devices. Investigations were begun in 1939 at the California Institute of Technology (GALCIT) under the direction of von Kármán by Malina, Parsons, and Forman. After preliminary studies, it was decided to concentrate on the restricted-burning type of solid propellant unit since thrusts for periods up to 30 seconds were required. In 1940, in a fundamental study, von Kármán and Malina proved theoretically that stable burning in this type of unit was possible, and gave propellant and design criteria for stability [*55*].

In August, 1941, assisted take-off tests were made on the Ercoupe airplane piloted by H. A. Boushey, Jr. of the U.S. Army Air Corps. The units used had an amine-black powder propellant formulated by Parsons. The charge was pressed into the combustion chamber in 22 increments under a pressure of 18 tons, each unit delivering a thrust of about 25 lb for a period of 12 seconds. Further development of this type of unit was discontinued when it was found to have unsatisfactory temperature limits under storage conditions [*56*].

However a unit which met minimum temperature limits was developed in 1942, and arrangements for its manufacture by the Aerojet Engineering Corporation were made in 1943. The propellant charge, formulated by Parsons, consisted of 75 per cent pulverulent potassium perchlorate mixed with 25 per cent asphalt; the oxidizer was added to the heated fuel and poured while hot into the combustion chamber, and then allowed to cool into a tough mass. The rocket unit operated at a chamber pressure of 1800 lb/in.^{2} at a temperature between 3000 and 4000°F and developed an exhaust velocity of 5500 ft/sec. Copious quantities of smoke in the jet were produced by the oxidizer used in the propellant [*55,56*].

The theory and design of the restricted-burning unit was summarized in a recent paper by Seifert, Mills, and Summerfield [*55*].

Developments in other countries. In Italy, between 1927 and 1929, research was carried out on the constant pressure type of solid propellant motor under the direction of G. A. Crocco [*57*]. A motor using a tabular charge of colloidal solventless powder was developed to operate at chamber pressures between 50 and 100 atmospheres. The specific impulse obtained ranged from 150 to 170 lb-sec/lb. This motor was a precursor of the type developed to a high degree in the United Kingdom and the United States, as described above.

In Germany, work was carried out on solid propellant motors of various types mainly by the following concerns: Rheinmetall-Borsig, Wilhelm Schmidding, and Dynamit A.G. Before the end of World War II, Japan had put into service a piloted antiship missile propelled by three solid propellant rockets [*58*]. In the Soviet Union, development and production of solid propellant rockets was undertaken on a large scale

as evidenced by a variety of rocket weapons used during the war. Published information on developments in Germany, Japan, and the Soviet Union is not sufficient for a summary to be written with any degree of reliability at present.

A,7. Development of Liquid Propellant Rocket Engines. (From Literature Published after 1940.) The literature published up to 1940 shows that the basic problems of designing successful liquid propellant engines were correctly stated, at least in major outline. The properties of a number of propellants were fairly well known, especially those combinations using liquid oxygen as the oxidizer. In connection with the rocket motor the thermodynamics of the combustion process, and of the flow of the gaseous products through the exhaust nozzle to produce thrust, were understood; and a beginning had been made in solving the cooling problem, and the preliminary stages of development of suitable systems of propellant supply had also been reached. It may be noted that the greatest advance to be recorded in this article was the mechanical design of the complete engine, although a number of basic discoveries were made, especially in the domain of propellants, and the theory for guiding design was greatly expanded.

Developments in Germany. From about 1932 onward, work on liquid propellant engines was pursued intensively in Germany under a cloak of secrecy. Engines were developed for various applications around three types of oxidizer: liquid oxygen, hydrogen peroxide, and nitric acid. Work on liquid oxygen engines was concentrated at the army establishment in Peenemunde; on the hydrogen peroxide system at the H.W.K. (Hellmuth, Walter, Kommanditgesellschaft) firm in Kiel; and on the nitric acid system at the BMW (Bayerische Motoren Werke) firm in Munich and the Rheinmetall-Borsig firm in Berlin.

The principal center for the development of rocket engines based on liquid oxygen was established by the Army, under the leadership of Dornberger, at Peenemunde in 1937–1938, although work had been initiated in Berlin on the design of rocket missiles using this oxidizer in 1932. The research and technical development program at Peenemunde was directed by von Braun, Thiel, and Steinhoff. The work of the center culminated in the design and construction of the engine which propelled the A-4 rocket missile, better known as the V-2. Work on the engine started in about 1940 and incorporated the experience gained in testing smaller engines, the first designed in 1933 for the A-1 missile, which weighed 330 lb; the second designed in 1934 for the A-2 and delivering 660-lb thrust for 16 seconds; and the third designed in 1938 for the A-3, which weighed 1650 lb and had an engine delivering 3300-lb thrust for 45 seconds. The V-2 was first successfully launched in October, 1942, when it traversed a distance of about 120 miles.

The V-2 engine had a turbocentrifugal pump feed system which supplied the liquid oxygen and a fuel mixture of 75 per cent ethyl alcohol and 25 per cent water. The pumps were driven by an impulse turbine powered by steam from a hydrogen-peroxide generator developed by the Walter firm. The rocket motor, operating at a chamber pressure of 294 lb/in.^{2}, delivered 55,000-lb thrust at sea level for a period of 65 seconds. The average specific impulse of the motor was 223 lb-sec/lb. The motor was cooled regeneratively by the fuel, and in addition film cooling was resorted to by injecting a portion of the fuel through rows of small holes around the circumference of the chamber and exhaust nozzle. Expansion joints were provided in the outer wall of the motor to allow for the greater thermal expansion of the inner wall. The fuel and oxidizer were fed into the combustion chamber through eighteen separate injectors arranged at the end of the chamber. Ignition of the propellant mixture was accomplished either by a pyrotechnic or a liquid-operated torch that was inserted through the nozzle into the chamber [*19,33*].

In addition to the development of engines for missiles, the Peenemunde establishment constructed a liquid oxygen-alcohol assisted take-off engine delivering 2200-lb thrust for 20 seconds which went into service in 1942 [*59*].

The German development of rocket engines based on hydrogen peroxide has been carefully reviewed by Baxter [*60*]. The hydrogen peroxide system was developed almost exclusively by the firm of Walter at Kiel, which began work sometime after 1934 [*61,62*]. In 1936 the Deutsche Versuchsanstalt für Luftfahrt carried out flight tests on a Walter "cold" engine using a compressed air feed system, and a simple catalytic plate for producing decomposition of the peroxide. The following year, flight tests were made with an improved engine using liquid calcium permanganate as a catalyst and giving 650-lb thrust for 30 seconds.

In 1938 flight tests were made on the Heinkel 112 with an engine equipped with a turbopump, driven by decomposed peroxide, to replace the compressed air propellant feed system used previously. The turbopump system was adopted later for the engine used in the V-2. The thrust of the engine could be controlled by the pilot up to a maximum of 2200 lb.

The specific impulse obtained with 80 per cent hydrogen peroxide alone ("cold" engine) was about 155 lb-sec/lb. To obtain higher values, fuels were burned with the oxygen released by the decomposition of the peroxide. It was discovered that fuel mixtures based on hydrazine hydrate were self-igniting with liquid peroxide. A fuel mixture of 30 per cent hydrazine hydrate, 57 per cent methyl alcohol, and 13 per cent water when burned with the peroxide gave a specific impulse of about 210. To reduce hazards due to ignition delay, small amounts of potassium cuprocyanide were dissolved in the water making up the fuel mixture.

This fuel was used in the engine for the Me 163A interceptor aircraft. The rocket motor used in this engine, known as the Walter 109-509-A2 was regeneratively cooled by the fuel. The thrust delivered by the motor at sea level could be varied from about 500 to 3750 lb.

Work on engines utilizing nitric acid as an oxidizer was carried out by the Bayerische Motoren Werke (BMW). Hemesath, chief chemist at BMW, claims to be the inventor of fuels that are self-igniting (hypergol fuels) with nitric acid, and further states that at the BMW rocket station near Karlsfeld some 6000 fuel-nitric acid combinations were tested by 1945. An engine using a propellant combination with nitric acid as the oxidizer was developed for the Henschel 8-117 aircraft [*63*].

In 1936, Sänger was called from Vienna to set up a research institute at Trauen, Germany, to study the fundamentals of liquid propellant rocket engines leading to the construction of an engine delivering a thrust of 100 tons. In 1939, experimental work was undertaken on a number of problems, including studies of fuels containing powdered light metals, of liquid ozone as a possible oxidizer, and of motor cooling. Research on rocket engines was terminated in 1942 and efforts were concentrated on ramjet research [*64,65*].

Developments in the United States. The principal groups that worked on liquid propellant engines were the following: the Jet Propulsion Laboratory at the California Institute of Technology; the Aerojet Engineering Corporation; Reaction Motors, Inc; and Project TED ESS 3401 of the Navy Bureau of Aeronautics.

At the California Institute of Technology, under von Kármán and Malina, among various activities that were initiated in 1939 for the Army Air Forces (cf. Art. 6) and in 1944 for the Army Ordnance Department, was the development of the nitric acid-aniline engine in which the propellant was fed to the combustion chamber by gas pressure. Assisted take-off flight tests were carried out for the Army Air Forces on the A-20A airplane piloted by Dane in April of 1942 with two engines, each delivering 1000-lb thrust for 25 seconds [*56*]. The rocket motor was uncooled, and the spontaneously igniting propellant combination was fed into the combustion chamber through an impinging-jet injector by gas pressure from compressed nitrogen cylinders. By the end of 1942 a service-type engine based on this prototype was designed under M. Summerfield at the Aerojet Engineering Corporation, which was founded in that year by von Karman, Malina, Haley, Parsons, Forman, and Summerfield [*66,67*]. Haley was president of the corporation during its first formative years. By the end of the war the Aerojet Engineering Corporation had also delivered a service type assisted take-off engine of the acid-aniline type which delivered 1500-lb thrust for 38 seconds [*68*].

From the 1942 prototype, an engine for the WAC Corporal sounding rocket was developed in 1945 under Malina for the Army Ordnance

Department at the Jet Propulsion Laboratory. This engine incorporated an aniline-cooled motor built by the Aerojet Engineering Corporation to deliver 1500-lb thrust for 45 seconds. A compressed air propellant feed system was used [*18,56*].

Work on a monopropellant engine was initiated at the Jet Propulsion Laboratory and the Aerojet Engineering Corporation in 1944, and considerable progress had been achieved at the latter organization by the end of 1945 under the supervision of Zwicky [*69*].

Reaction Motors, Inc., with L. Lawrence, Jr. as first president, carried out developments primarily on engines using the liquid oxygen-alcohol propellant under contracts from the Navy Department. It constructed the 1500 N4C engine which powered the Air Force experimental supersonic airplane X-1. The engine consisted of four regeneratively cooled motors, each delivering a maximum thrust of 1500 lb. The total thrust of the engine could be varied between 1000 and 6000 lb in four steps. The propellant was supplied to the motors by means of centrifugal pumps driven by a gas turbine [*70,33*].

The Navy Department in 1941, under the leadership of Bolster of the Bureau of Aeronautics, initiated project TED EES 3401 at the U.S. Naval Engineering Experiment Station to carry out research on liquid propellant rocket engines for assisting the take-off of large flying boats [*71*]. Truax was assigned as officer in charge of the project [*72*]. During studies of chemical gas generation methods for pressurizing propellants, Stiff found that aniline and nitric acid ignited spontaneously. On the basis of this information Malina at the California Institute of Technology initiated development of the nitric acid-aniline engine, and later the Truax group also adopted this combination of the development of a prototype engine delivering 1500-lb thrust for 35 seconds. This engine, in which a compressed gas feed system was used, was incorporated in droppable assisted take-off units which were flight tested on the Navy flying boat PBY-2 by Gore in the spring of 1943.

The project also developed a prototype engine for the Gorgon, a radio-controlled winged missile. The propellant combination used was mixed nitric-sulfuric acids and monoethylaniline. The engine delivered a thrust of 350 lb for 2 minutes.

Goddard, beginning in 1942, worked at the U.S. Naval Engineering Experimental Station on a prototype liquid oxygen-gasoline engine.

The progress made during this period in the theory and design of liquid propellant engines, arising primarily from work in the United States, has been presented in books by Zucrow [*73*] and by Sutton [*74*].

Development in other countries. In 1929, in Italy, work was initiated on liquid propellant rocket motors by G. A. Crocco. Studies were made with the gasoline-nitrogen dioxide combination in a motor that was regeneratively cooled—the chamber by the oxidizer and the nozzle

by the fuel. The motor was operated at a chamber pressure of 140 lb/in.^{2} for durations up to 10 minutes. Further development on the motor was discontinued in 1930 because of lack of funds [*57*].

In 1932, work was resumed on propellants of the monopropellant type for possible use in torpedo and rocket engines. Studies by R. M. Corelli and L. Crocco were made of the decomposition of stabilized liquid explosives such as dinitroglycerin. In 1933 Crocco suggested experiments on nitromethane. In addition to studies of decomposition, tests were also made of its stability with respect to shock and temperature [*57,75*].

There is at present insufficient published information on work undertaken in other countries for a summary to be made with any degree of reliability.

CHAPTER 2. ROCKET DEVELOPMENT SINCE 1945

R. C. TRUAX

A,8. Liquid Propellant Rockets. The end of World War II brought about an abrupt change in the course of rocket development in all major nations of the world. In Germany, where technical progress in rocketry overshadowed that of any other country, the development of rocket-propelled vehicles came to a dead stop. The United States, taking a cue from the German progress in rocket-propelled missiles and piloted aircraft, abandoned its preoccupation with assisted take-off and began many ambitious programs in the missile and aircraft fields. In other western countries, rocket development followed a somewhat similar course, but on a considerably lower scale of effort.

Available information on rocket development trends in the Soviet Union is very meagre. During World War II, the Soviets pioneered in the large scale use of solid propellant rocket artillery. There is some evidence to indicate that they have built on this base and expanded it by information acquired from Germany at the close of the war. It appears that the Russians picked up the development of many German guided missiles in a very direct fashion, utilizing prior developments to a maximum extent, and continuing at a high and more constant level of support.

Technical progress since 1945 in the field of liquid propellant rockets has been characterized by a host of minor improvements, rather than by major advances.

The turbopump system for feeding propellants to the combustion chamber has gained much wider acceptance. Numerous new power plants embodying pump-feed have been developed for applications where the weight-saving justifies the additional complexity. To avoid the necessity

of supplying a third propellant to drive the pump turbine, gas generators which utilize the primary propellants for this purpose have been brought to a high level of perfection. The use of hydrogen peroxide for turbine drive is still preferred, however, in some cases. Single-stage pumps are now universally used for all discharge pressures. Simpler and more reliable shaft seals have been developed. Perhaps the most spectacular improvement has been the increase in allowable suction specific speed. This improvement permits, alternatively, lower suction pressures or smaller pump size and weight for a given flow rate and discharge head.

Improvement in combustion chamber design has been considerable. Very important reductions in size and weight over, say, the V-2 propulsion system have been brought about through better mechanical design and the use of higher combustion pressures.

In the direction of increased specific impulse, gains made have resulted primarily from the use of increased combustion pressures and from the use of improved propellants. The combination liquid-oxygen-gasoline, used since the earliest experiments of R. H. Goddard because of its cheapness and potentially high specific impulse has, in a sense, come into its own through cooling developments which permit it to be used at maximum performance. No marked improvement in specific impulse has been effected through better understanding of the process of mixing and combustion, although numerous high performance injection systems have been developed by empirical methods.

The years since the end of World War II have been a progressive increase in the maximum thrust of rocket power plants. From the fifty-five thousand pounds of the V-2 engine, thrust has increased to the order of hundreds of thousands of pounds. These giant engines consume more than a thousand pounds of propellant every second and in some applications generate millions of horsepower.

A great many propellant combinations have been investigated, both experimentally and theoretically. The three oxidizers in use prior to 1945, namely, liquid oxygen, nitric acid, and hydrogen peroxide, are still the only ones used for other than experimental purposes. Hydrocarbons have become very nearly standard for use with liquid oxygen. Aniline and various amines, much used with nitric acid during the war, are being replaced by other higher performance fuels such as hydrazine and its derivatives which, like aniline, ignite upon contact with nitric acid.

A moderate amount of work has been done with high performance combinations such as liquid hydrogen, liquid oxygen, fluorine-ammonia, fluorine-hydrogen, etc. Undesirable physical or logistic qualities have thus far prevented acceptance of these combinations for any practical applications.

The most spectacular advances in the field of liquid propellant rockets that have taken place since 1945 have been not in improvements in pro-

pulsion engines themselves, but in the application of rocket propulsion to various types of vehicles. The most important of these vehicles are piloted aircraft and guided missiles. In the United States only research aircraft have been rocket-propelled. A series of these craft have been produced. The first of these was the X-1, designed by the Bell Aircraft Corporation. The X-1 used a Reaction Motors liquid oxygen-alcohol engine developing 6000 lb of thrust. This airplane was the first to exceed the speed of sound in level flight. A second airplane, the Douglas D558-2 "Skyrocket," exceeded 1200 mph and ascended to an altitude of nearly 80,000 ft. An improved version of the X-1, designated X-1A, attained a speed of some 1600 miles per hour and an altitude substantially in excess of 80,000 ft. The X-1A and the D558-2 both use an improved version of the Reaction Motors engine used in the X-1. The X-2, which began its flight tests in 1955, uses a Curtiss-Wright rocket engine of about 15,000-lb thrust.

A fourth rocket airplane of the research type has been reported to be under development in the United States. This airplane is designated the X-15. Altitudes in excess of 100 miles and speeds of well over four thousand miles per hour are expected.

Rocket-propelled, man-carrying airplanes have also appeared in countries other than the United States. One of the most notable of these is the French "Trident." This airplane uses a large rocket engine in the fuselage as a primary source of propulsion and two small, wingtip-mounted turbojet engines for landing.

Numerous auxiliary rocket power plants for short-time performance improvement of manned aircraft have appeared, including both liquid and solid propellant engines for take-off assistance.

At least three research rockets have been developed since 1945 which should be noted. A rather small rocket, known as the Aerobee, was developed as an enlarged version of the WAC Corporal, the first rocket developed solely for atmospheric sounding. The Aerobee has been fired in considerable numbers. An improved version known as the Aerobee-Hi has also been produced and fired successfully to a height of 163 miles. The Aerobee is essentially a two-stage rocket having a low impulse solid propellant first stage, and a high impulse liquid propellant second stage.

A somewhat similar altitude research rocket, known as "Veronique" has been produced by the French. This rocket is notable for its unique launching system. Although the Aerobee and the WAC Corporal used a high acceleration launch coupled with a high launching tower to attain stability, the Veronique uses a system of cables which maintain the rocket in a vertical altitude until aerodynamic stability is attained.

The largest altitude rocket designed primarily for research is the U.S. Navy's "Viking." This is a single-stage rocket of ten to fourteen thousand pounds gross weight. Twelve of these rockets were fired in the period 1945

to 1955. Altitudes up to 158 miles were attained with payloads in the range of 500 to 1000 pounds.

The Viking and Aerobee programs led to the most ambitious scientific project in the rocket field—the creation of a small artificial satellite of the earth, a program which was announced in mid-1955.

The satellite itself, weighing some 20–30 pounds, is to be placed in an orbit by a three-stage rocket. The first stage has evolved from the Viking, the second stage from the Aerobee, and the third stage is expected to be a new solid propellant design. The all-up weight of the vehicle is expected to be in the neighborhood of 22,000 pounds. This rocket, known as the Vanguard, exemplifies modern rocket practice. The first stage uses a turbopump liquid oxygen-hydrocarbon engine, the second stage a pressure-fed nitric acid-hypergolic fuel system, and the third stage an internal-burning solid propellant power plant of high performance.

The period from 1945 to 1955 has seen many military applications of the liquid propellant rocket engine. Development of the Corporal, a late wartime project, has been completed and the missile placed in operational use. This is a short range surface-to-surface missile developed for the U.S. Army. The Nike, a solid-liquid combination rocket for antiaircraft use, is also in service.

Several much more ambitious projects have recently been initiated. Two intermediate range ballistic rockets, Thor and Jupiter, are under development for the Air Force and Navy, respectively. The latter is being developed by the Army primarily for Naval employment.

The most spectacular application of liquid propellant rocket techniques is, in many respects, the intercontinental ballistic missile. Two separate missiles in this range category are at present under active development for the U.S. Air Force. In these applications, the liquid propellant rocket reaches the ultimate in size, performance, and complexity.

The future of the liquid propellant rocket appears directed towards higher performance missiles, satellites and, eventually, interplanetary space ships. From a technical standpoint, only minor improvements in weight, reliability, and performance appear possible until means are found for applying effectively the energy latent in fission or fusion of atomic nuclei.

A,9. Solid Propellant Rockets. The end of World War II saw solid propellant rockets in large scale use. Hundreds of thousands were used by all participants for barrage purposes. Although their accuracy was not equal to that of rifled artillery, they could be fired very rapidly in great numbers from lightweight equipment. In the air, rockets were mounted on smaller airplanes for ground bombardment. Solid propellant rockets were used to assist take-off of heavily loaded aircraft. In Germany, some solid propellant rockets were used in experimental guided missiles.

The solid rockets of 1945 were largely short burning time devices. For unguided projectiles, this short burning time was required to give accurate trajectories. Propellant burning rates were quite high. In order to get long durations, end-burning types of charges were necessary. The necessity for inhibiting burning on the outer surfaces caused considerable difficulty, for any increase in burning area led to greater pressures and possible case rupture. Longer durations also led to increased case and nozzle weights, because all metal parts exposed to the flame had to have sufficient thermal capacity to avoid overheating.

Two basic types of solid propellants were in use by the end of the war, double-base and composite. The former was a composition of nitrocellulose and nitroglycerin. The charges were usually formed by solventless extrusion in large presses. The press size requirements put an upper limit on economical charge size.

Composite propellants were mechanical mixtures of fuel and oxidizer. These propellants were frequently castable, and thus amenable to production in large size grains. The most common propellant of this type was composed of a mixture of potassium perchlorate in asphalt. The consistency was controlled by the addition of petroleum oil. This particular propellant was widely used for JATO (jet-assisted take-off) units. It suffered from severe temperature limitations because of the change in viscosity of the asphalt with temperature. It also burned to give potassium chloride as one of the products of combustion. This material appeared as a dense white smoke which caused many operational difficulties.

Many improvements have been made in solid propellant rockets during the last ten years. These improvements have widened the field of application, reduced the cost, and improved the reliability. One of the most important improvements has been the development of the internal burning grain. This configuration causes the case to be insulated from the propellant gases by the unburned propellant for all but the last fraction of a second. This insulation permits a very considerable reduction in case weight. Nozzle weights have been reduced through careful design and the use of ceramics.

New propellants have been developed which give considerably higher performance. Ammonium perchlorate and ammonium nitrate have largely replaced potassium perchlorate in mixed propellants. This substitution not only has improved the specific impulse, but has eliminated the smoke problem. Propellant formulations are now available to cover a wide range of burning rates. Many propellants burn stably at much lower pressures than have been possible in the past. Variation of burning rate with charge temperature has been greatly reduced.

One of the drawbacks of solid propellant rockets stems from the fact that fuel and oxidizer are premixed in the rocket case. The propellant therefore is basically an explosive. The problem of safety is one of formu-

lating the propellant so as to have a low sensitivity to those sources of ignition or detonation that might reasonably be encountered. All propellants are of course potentially reactive, so that the problem of stability in storage is an important one. Considerable progress was made in these areas in the years between 1945 and 1955. One company demonstrates the safety of a particular propellant by using it for ash trays!

The many improvements in solid propellant rockets have led to a marked widening of their sphere of application. With size and burning time restrictions largely eliminated, the solid rocket is competing with the liquid in the guided missile field. The solid rocket is still heavier, in general, than the turbopump type of liquid propellant rocket. It is also inferior in specific impulse to the liquid oxygen-gasoline rocket. In many cases, however, the greater simplicity of the solid propellant rocket permits the use of multiple stages to overcome the weight and specific impulse disadvantages.

Unguided solid propellant rockets are being provided as fighter armament, replacing machine guns and cannons. The most popular of these, known as "Mighty Mouse" has a diameter of only 2.3 inches. Many of these rockets are fired in salvo, permitting a very high rate of fire in the short periods of contact encountered in sonic and supersonic combat.

The unguided barrage rocket is being replaced by large rockets carrying nuclear warheads. The currently operational variety is known as "Honest John." It carries a 1500-lb warhead a distance of some ten miles.

The solid propellant rocket has a monopoly in the field of air-to-air guided missiles. Three missiles in this category have been disclosed by the United States Government. These are known as Sparrow, Sidewinder, and Falcon. The first two are Navy weapons, the last an Air Force development.

In the air-to-surface category, a missile known as "Bullpup" is under development. This missile also uses a solid propellant rocket motor.

All surface-to-air missiles in production use either a solid propellant booster or sustainer, or both. Terrier, a Navy Missile, is a two-stage rocket; it uses a short duration, high thrust booster and a lower thrust, longer duration sustainer. Both rockets use solid propellants. The Navy's Talos missile uses a solid booster and a ramjet sustainer. The Army Nike uses a solid booster and a liquid sustainer.

Surface-to-surface missiles less frequently have solid propellant rocket power plants, but even here the solid rocket is extremely useful. The turbojet-powered Regulus, Matador, and Snark all use solid rockets to obtain essentially zero-length launching. A comparatively short range ballistic missile, known as Sergeant, is also under development. This missile uses one of the largest solid propellant grains ever manufactured.

For experimental use, many applications of solid propellant rockets have been made. A very high performance rocket called "Deacon"

has been used for atmospheric sounding. Since this rocket is small, and hence loses much of its impulse in overcoming aerodynamic drag, many have been launched from balloons, from altitudes of 80,000 ft or more. The Aerobee research rocket uses a solid propellant booster giving 18,000 pounds of thrust for 21 seconds. In the United Kingdom, an all-solid propellant atmospheric research rocket comparable to the Aerobee has been produced. A solid propellant rocket is scheduled to provide the final stage of propulsion for the satellite to be put in orbit during the International Geophysical Year.

Many other experimental devices, such as high speed sleds, torpedoes, aerial targets, etc., have used solid rockets.

In the field of long range ballistic missiles, the use of solid propellants appears to offer certain advantages in terms of launching manpower, readiness time, and general simplicity. If the difficulties of transporting, handling, and storing such large containers of explosive can be adequately solved, and if the inherent performance disadvantage and lack of precise controllability of the solid rocket can be overcome, at least some application in this field seems sure to follow.

CHAPTER 3. AIR FLOW JET ENGINES

A. D. BAXTER

A,10. Introduction. At the beginning of the twentieth century heavier-than-air machines for flight were making their first feeble hops a few feet from the ground. The principles of aerodynamics were sufficiently well understood for this to have occurred much earlier if a suitable propulsion plant had been available. The primary requisite for such a unit was a high power/weight ratio. However, the steam engine which had held undisputed sway in the power plant field for nearly 150 years was not easily adaptable, with its indirect heat supply, to this role. Consequently it was only the advent of a new form of motor which brought the aeroplane into the realm of practical achievement. The new plant was the internal combustion engine which converted fuel energy directly into heat energy of the working fluid.

This engine, in its early days, still far from satisfied the demands of aircraft designers and, even then, air flow jet propulsion in many of its forms was visualized [*76*] as a more advanced stage in power plant development. As with many inventions, the time was not ripe for its development and for some forty years the piston engine was improved and flourished, until it brought aircraft performance to a point where jet propulsion became quite attractive. At this stage the efficient conversion of heat energy directly into kinetic energy produced as great an advance as the original development of the internal combustion engine. The gas turbine

was the medium for introducing this revolution which changed a barely considered idea into a practical form of propulsion almost overnight. Much of this very rapid change can be attributed to the comparative ease with which the main engine components can be developed and tested independently of one another, together with their amenability to analytical treatment.

This transformation in power plants came only toward the end of World War II, so that the real repercussions on aircraft design and high speed flight were not apparent until the postwar period. What is clear, however, is that a field of very high speed flight is opening up which invites the application of jet propulsion motors of even more simple design than the gas turbine. Ramjets are in this class, but their development, within the period covered by this survey, is almost negligible compared with what the turbojet has produced already and what the ramjet may be expected to produce in the future.

The history of this tremendous development in propulsion is very much confused in its early years by the interweaving of new ideas, so that attempts to produce a logical sequence of progress in one basic type may become disjointed if it is not realized that intermediate steps in an alternative type may have influenced development. As an example, the turbojet engine was not completely segregated from the turboprop in its development nor the ramjet from the ducted fan with afterburning. In the following articles some account of this will be taken while following the broad divisions of the subject as indicated in Table A,2b.

A,11. Piston Engine Jet Development. There are three elements common to all air flow jet propulsion systems, namely, air compression, combustion, and expansion. Two of the main distinguishing features are the method of compression and, where this is achieved mechanically, the form of the compressor and its drive. In modern developments the compressor is generally of the radial or axial continuous flow type driven almost exclusively by a turbine, from which we derive the classification of turbojet. It has not always been so and, perhaps naturally, the piston engine played a large part in early inventions.

Lorin, the French engineer, is regarded as the pioneer of jet propulsion and he wrote many papers advocating "propulsion by direct reaction" from 1908 onward. In that year he proposed a system using a reciprocating engine and exhaust jet [77]. Each cylinder, after performing its normal operations of suction, compression, and ignition, allowed the combustion gases to exhaust down a funnel to the atmosphere. All the gas energy, except that necessary for providing compression of the new charge, was directed into the jet. No power was taken from the crankshaft and the scheme represents the first suggestion for using part of the combustion gas energy of the reaction system for the purpose of

compression. Its chief disadvantage was the hopelessly inadequate mass flow that the engine could deal with.

In 1909 another Frenchman, Marconnet, gave examples of jet reaction [*78*] in which compression was effected by a Rootes blower. Both of these pioneers and others concentrated much of their attention thereafter on the simple ram compression engine (Art. 13), but were forced to give some thought to mechanical compression in order to overcome the problem of static operation. In 1917 a British inventor, Harris, reverted to the mechanically driven compressor. His invention [*79*] used a centrifugal blower with forward-facing intake, driven by a small twin cylinder piston engine. Among its novel features was a proposal to burn powdered coal in the compressed air and to augment the thrust by ejector action of the resultant high temperature jet.

Later, Mélot propounded a double-acting free piston engine somewhat akin to the earlier arrangement of Lorin, but without the crank drive [*76*]. Combustion pressure on one side of the piston forced it to compress mixture on the other side. As the piston moved, it opened ports through which the burned gas was ejected and the major portion of its energy utilized in propulsion. In 1928, in Budapest, Fono made various jet propulsion claims [*80*] similar to those of Mélot and Lorin, but included one to interpose a single-stage axial compressor in what was essentially a Lorin duct.[7] The compressor was to be driven by wind vanes exposed to the slipstream, and for operation without forward motion a special driving motor was provided.

G. G. Smith has described various schemes of this period [*81*, pp. 45–47], including those of the Swiss, Schurter, and the German Junkers Company. The latter was probably the most advanced form of piston-jet combination proposed. It consisted of a multistage axial compressor driven by a multirow radial two-stroke engine. Some of the compressed air was used for scavenging and charging the cylinders, and the remainder was passed round them to provide cooling before entering a duct with the engine exhaust and flowing to a propulsive nozzle. Fuel was to be burned in the duct to increase the propulsive effort. Campini, an Italian jet propulsion inventor, thought along much the same lines, using reciprocating engines to drive centrifugal blowers or ducted fans [*82*]. His scheme was actually built and installed in a Caproni monoplane, but the airplane was not regarded as a success, having a maximum speed of only 230 mph even with afterburning [*83*].

A,12. Turbojet Development. Piston engine compressor drives were never extremely successful, being relatively heavy and complicated, and it remained for the purely rotary drive of the turbine to justify jet propulsion. The first appearance of a turbine was in 1921

[7] See Art. 13.

when Guillaume combined an axial compressor and a multistage turbine [*84*]. The patent drawings show obvious influences of piston engine techniques, but nevertheless include ideas which have been used in more modern designs. The next turbine proposal came from Whittle in a British patent in 1930 [*85*]. This contained all the main features of his later designs, but was allowed to lapse through lack of financial support.

From the early thirties there were small but growing bodies of engineers in various countries, who were beginning to appreciate the potentialities of the gas turbine as an aircraft power plant. In a short while, national interests were involved and most of the work became shrouded in the veil of secrecy which was lifted only at the end of World War II. Much parallel but completely isolated work was proceeding simultaneously in the belligerent countries and has since been well described by Schlaifer [*86*]. In the present article the sequence of events will be considered separately for each country, but first it is worth recording the milestones of the earliest jet propulsion flights as some measure of each nation's progress in the field.

The first flight in the world of a turbojet-powered aircraft was made by the Heinkel He178 on August 27, 1939 [*87*], four days before the outbreak of war between Germany and Poland. On the other hand, Japan was so far behind that she purchased German licenses and copied the Messerschmitt Me262 twin-jet fighter. The result was neither a successful aircraft nor engine, the combination making its first flight a few days before the end of the war in Asia in August, 1945. The Italian Caproni-Campini C.C.2 was reported to have flown in 1940 [*83*], but there was no reliable information concerning it until a widely publicized flight to the Italian Experimental Establishment at Guidonia in November, 1941. The aircraft did not use a gas turbine but was propelled by the Campini system already mentioned (Art. 11).

The first British flight was on May 14, 1941, when the Gloster aircraft E28/39, powered by the Whittle W.1 engine brought Whittle's long and arduous efforts to fruition [*88*]. A little under four and one-half years later, the direct descendants of this initial combination, the Gloster Meteor IV with two Rolls-Royce Derwent V engines, raised the world speed record from 469 mph to 606 mph [*89*]. In America the first jet flight came on October 1, 1942, when the Bell XP-59A, fitted with two General Electric Company type I.A engines, took the air [*90*, p. 108].

United Kingdom. Developments in Britain followed two distinct lines, one originating in the work of Air Commodore Sir Frank Whittle, utilizing the centrifugal air compressor, the other emanating from A. A. Griffith and associated with the axial compressor.

As long ago as 1926, Griffith, working at the Royal Aircraft Establishment, propounded an aerodynamic theory of turbine design based upon flow past airfoils as distinct from flow through passages. The subsequent

experiments and reports have been described by Cox in a Wright Brothers lecture [*91*]. They include, in November, 1929, an official paper by Griffith on the prospects of the gas turbine driving an airscrew. He concluded that it would be lighter, less bulky, and more efficient than contemporary piston engines. His scheme was for a turbocompressor unit consisting of a number of mechanically independent wheels, each carrying a set of compressor blades surrounding a set of turbine blades. Such a contraflow contrarotating unit was built and tested in 1939, almost ten years later [*92*], but it is difficult to know why work was allowed to lapse for that period. In fact, no interest was taken in the gas turbine and axial compressor fields from the time of these early Griffith proposals until 1936.

In that year, the Royal Aircraft Establishment obtained authority to build an axial flow compressor and in the following year another paper on the same lines as the earlier one received more favorable treatment. Work began on a turbine-airscrew engine design and arrangements were made to obtain the collaboration of the Metropolitan Vickers Electrical Co., Ltd. The early history of this association has been described by Constant [*93*, p. 411] and the later developments by D. M. Smith [*94*]. At first, effort was concentrated on the turboprop engine and it was not until July, 1940, when the work of Whittle had clearly demonstrated the importance of first achieving an axial compressor turbojet engine, that design of such an engine commenced. The engine, designated F.2 and later developed into the Beryl [*95*], had distinctive features. They included a nine-stage compressor, a single annular combustion chamber with upstream injection, and a two-stage turbine, all on the same axis.

Until this period all other available effort was concentrated on the relatively advanced centrifugal type of engine, but a little more weight was added to the axial type when Armstrong Siddeley was given a contract in November, 1942. The result was a jet engine, the ASX, which ran five months later [*91*]. This was the first move toward a change in balance of effort, which gathered momentum only when the pressure of war demands eased and the advantages of the axial engine became more apparent. The RAE, however, had continued to foster this type and in addition to its association with Metropolitan Vickers, it cooperated with other firms on the axial compressor problems of the period. Relatively little was known of the requisites for high performance, but the series of compressors which were built and tested set future design on a much firmer basis [*93*, p. 452].

While ideas on axial engines were slowly crystallizing in the prewar days, the same struggle to win support was being waged for the centrifugal engine. Whittle has told the story in his Clayton lecture [*88*]. A company, known as Power Jets, Ltd., was formed, which in June, 1936, placed an order with the British Thomson-Houston Company for the manufacture

of an engine to Whittle's requirements. The engine was a simple jet propulsion gas turbine having a single-stage, double-sided centrifugal compressor driven by a single-stage turbine. In spite of the simplicity it was a remarkable venture, going well beyond all previous engineering experience in each of its main components. A compression ratio of 4:1 in a single-stage centrifugal blower was specified, compared with the best contemporary superchargers of $2\frac{1}{2}$:1, the single-stage turbine wheel of $16\frac{1}{2}$ inches outside diameter had to give more than 3000 shaft horsepower with high efficiency, and the combustion intensity was many times greater than existing practice. The first tests were begun on April 12, 1937, and the engine continued to operate, with several intervals for reconstruction, until 1941, when a turbine failure ended the career of this forerunner of a long line of turbojet engines.

It had proved its worth long before this and by the summer of 1939 the Air Ministry had recognized that Whittle had the basis of a practicable aircraft engine. It commissioned Power Jets to supply an engine for flight test and the Gloster Aircraft Company to build an experimental aircraft to house it. This first flight engine, known as the W.1, was cleared for flight in April, 1941. By 1940, however, the Air Ministry had anticipated the possibilities of using jet engines in the war and Gloster was authorized to design a jet-propelled, twin-engined interceptor fighter.[8] As originally planned, the engines were to be a more advanced Whittle design—the W.2. Unfortunately the design proved too optimistic and the required component efficiencies were not achieved. Although it was unsuccessful, knowledge was accumulating rapidly and a whole series of engine designs and modifications were being investigated. Out of all this appeared the W.2.B, which must rank as a classic. It had the reverse flow combustion system which was typical of the Whittle designs and it was eventually developed to give nearly three times the thrust of the W.1 without occupying any more space.

At this time other firms were drawn into the engine production. The first to be so engaged was the Rover Company in the spring of 1940. Their original contract was for production, development, and manufacture of the W.2 engine so that Power Jets might be free to concentrate on research and technical development. This latter role absorbed all Power Jets' efforts as the war progressed and in 1944 it was reinforced by amalgamation with the RAE team in a government-controlled company, Power Jets (Research and Development) Ltd.[9] [*96*]. Rover finally passed their work over to Rolls-Royce in 1943, but before this they pioneered a change in design from the reverse flow combustion chamber

[8] The F9/40—prototype of the Meteor.

[9] In 1946 this company's functions were reduced to those of consultants, holders of patents on behalf of the government, and a training school in gas turbine technology. The majority of the personnel was transferred to a new National Gas Turbine Establishment under the direct control of the Ministry of Supply.

to the straight-through type which has since been universally adopted. Their prototype engine, W.2B.26, ran in March 1942, and served as a basis for the Derwent I which was later developed by Rolls-Royce [*97*].

Rolls-Royce's interest had been stimulated by the flight of the E28/39 aircraft and in 1942 their first venture, the WR.1,[10] was started as a subcontract from Power Jets. It served mainly as an introduction to the new field and it was not long before the skill and experience of the firm were fully deployed. It assumed responsibility for Meteor power plant production and development and before the end of 1945 had put up jet thrusts from 1600 lb in the Welland to 5000 lb in the Nene, had increased thrust-to-weight ratios from 1.86 to 2.8, and had improved engine reliability so that 180 hour overhaul periods were approved [*91*].

Another company early in the field was DeHavilland [*98*]. In January, 1941, with the aid of Halford, designer of many successful piston engines, work commenced on the H.1, later known as the "Goblin." It represented a departure from Whittle's practice in having only a single-sided impeller and also a straight-through combustion system.

In 1941, official control of jet propulsion work was placed under a Director of Special Projects, R. Cox, within the Ministry of Aircraft Production. He inaugurated a Gas Turbine Collaboration Committee with representatives from all the main parties concerned in this important development, in order to coordinate new ideas and experience. Its formation was a unique example of direct competitors sharing information with complete frankness for the national benefit. It was in marked contrast to the German attitude and undoubtedly helped to place Great Britain much further ahead technically than any other nation.

Before completing the record of work in Great Britain, mention must be made of the engineers' awareness of potential developments of the basic design. Thrust augmentation by ducted fan was demonstrated in 1943 when Metropolitan Vickers modified the F.2 engine by the addition of a second turbine driving a two-stage, contrarotating fan. This unit, the F.3, increased the F.2 thrust by 66 per cent without any increase in fuel consumption and was the most economical jet engine in the world at that time, with a specific consumption of 0.65 lb/hr/lb of thrust [*99*]. Other thrust-boosting methods were tried on the bench, including water and alcohol injection into the intakes and afterburning.[11] Development of afterburning was in hand in 1943 at the RAE and engines were flight tested in a Meteor [*100*] the following year. An estimated increase in thrust of 25 per cent was obtained in flight and this was limited only by lack of a flightworthy variable area propelling nozzle.

United States. American interest was first recorded in 1922 when

[10] The initials were those of Whittle and Rolls-Royce and symbolized the close technical liaison.

[11] Originally referred to as "reheat" in Great Britain.

the Bureau of Standards was asked to investigate the practicability of jets. A study prepared by Buckingham and later published by the National Advisory Committee for Aeronautics [*101*] concluded that "propulsion by the reaction of a simple jet cannot compete in any respect with airscrew propulsion at such flying speeds as are now in prospect." The speeds envisaged were of the order of 250 mph and computations indicated that at this speed the jet would take about four times as much fuel per thrust horsepower as the airscrew. This report discouraged any further interest for many years and it was not until 1938 that a modest program was initiated at Wright Field to develop the gas turbine. In the interim S. Moss, of the General Electric Company, through his pioneering work on exhaust gas turbosuperchargers [*102*] had been a lone figure in maintaining some interest in turbine problems, particularly of high temperature materials. His persistence was fortunate, for the experience gained in that period proved of great value when the urgent requirement arose.

In 1939 the NACA was charged with basic research into turbojet problems and during the next two years a number of gas turbine research contracts were placed by different government agencies, but as late as June, 1940, a very discouraging report [*103*] was submitted to the Secretary of the Navy by a committee including such authorities as von Kármán and Millikan. This report suggested that the gas turbine could not be considered a serious aspirant in the field of aeronautics until much better materials were available. Thus it was not until the full extent of the British developments was revealed in the summer of 1941 that real pressure was put on the work. Once liaison had been established, full advantage of British experience was taken. By September, 1941, drawings of the Whittle engine had been studied, contracts placed with the General Electric Company for engines and the Bell Aircraft Corporation for airframes [*104*]. The following month a Whittle engine with a Power Jets test team was flown across the Atlantic, thus beginning a two-way collaboration which continued throughout the war. The G.E. version of the Whittle engine (type I) had its first successful run just over six months after work commenced and six months later the Bell aircraft became the second type of Allied jet-propelled aircraft to fly. Following this, General Electric produced a series of type I engines with greater thrusts, ending in 1944 with the I-40 having a thrust of 3750 lb [*105*].

One reason for the early pessimistic outlook in America was the failure to appreciate that new flight techniques would arise with jet propulsion. In consequence, emphasis was placed on obtaining fuel consumptions at least comparable to those of piston engines and the first projects involved turbopropellers, ducted fans, and compound piston-turbine arrangements. Northrop put forward a scheme for a propeller turbine engine, the Turbodyne, in 1940, but its development proved

slow and difficult and it was not until the end of 1944 that it actually ran on the test bed. Also in 1940 the NACA began construction of an experimental engine of the Campini ducted fan type (Art. 11). This operated successfully with afterburning in the fan exhaust during 1942, but it was abandoned the following year with another ducted fan designed by Allis Chalmers when the merits of pure turbojets had been appreciated. Two other groups were working at this time, one a private venture of Pratt & Whitney, whose PT-1 design consisting of a free piston reciprocating Diesel compressor with a gas turbine driving the propeller, was definitely regarded as a long term research project, the other a General Electric group at the firm's Schenectady plant, which was quite independent of the group working on the British turbojet designs. It was sponsored by the NACA gas turbine committee and considered only axial compressor engines. Its first unit was a turboprop, but a jet was started in 1943. This was designated TB-180 [*106*] and was comparable to the British F.2 engine, except that it had multiple straight-through combustion chambers. Its thrust was 4000 lb and it first ran in April, 1944.

During this period, the aircraft engine firms had only slight interest in the new forms of propulsion and it was not until they were convinced of their practicability by outside agencies that they became active in the field. This reluctance to depart from conventional power plants was also apparent in other countries. The first jet propulsion scheme came from the Lockheed Aircraft Corporation in 1940 but, like the Northrop turbopropeller, it suffered from ambitious attempts to give it a high efficiency and eventually it did not materialize. The only other turbojet scheme in this period came from the Turbo Engineering Corporation which received support from the Navy for a theoretical investigation in 1941. This resulted in a contract for a small booster turbojet to give 1100-lb thrust at sea level, but again it was very slow in development. A little later, Westinghouse was given a similar contract. It had been studying axial compressor engines for some time under the direction of the Navy Bureau of Aeronautics, but had no knowledge of any other projects. It produced the first American axial jet engine, the 19A, in March, 1943 [*107*]. This gave 1200-lb thrust in the remarkably small outside diameter of 19 inches, and completed a 100-hour endurance test by the following July.

From this time onward, firms engaged on the turbojet program increased rapidly. By the end of 1945 more than a dozen major engine firms were active in the field and at least the same number of aircraft firms. The latter had about twenty-two experimental jet fighter and bomber projects in various stages of design and five that had already reached the flight testing stage.

Germany. No comprehensive survey of German work has been prepared in Germany and consequently most of our information comes from the interrogation of the leading personnel carried out at the end

of the war by various technical investigation teams. The results of their work have been reviewed both in America [*90*, pp. 6–54] and in England [*108*], but some gaps are inevitable. As in England, the first impulses to practical aero-gas turbines came from individuals. Among these, von Ohain is worthy of mention. His first patents in 1935 were filed in the name of Max Hahn and assigned to the Heinkel Company [*87*]. He later designed the He-S3 turbojet which made the first jet flight in 1939. Heinkel, however, in spite of this early experience, was slow in exploiting it and was afterward overtaken by other firms, so that von Ohain never received due credit for his efforts.

In 1936 the German Air Ministry (RLM) became interested through one of its young engineers, H. Schelp. He was later placed in charge of gas turbine work but did not, at first, receive great support from the engine industry. By the end of 1938, however, he had succeeded in getting Junkers and the Bayerische Motoren Werke (BMW) to start project work which eventually led to the operational engines Ju-004 [*109*] and BMW-003 [*110*]. These engines were outwardly very similar in many respects to the English and American axial compressor types, but were crude in both design and performance. Normal engine life did not exceed 50 hours, thrust-to-weight ratio was only 1.3:1, and specific consumption was high at 1.4 lb/hr/lb. Restrictions on the use of high temperature materials were, in part, responsible for this poor performance, but it also appeared that some sacrifice was made in the interests of production. These points resulted in some interesting design features, particularly noteworthy being the elaborate cooling air system which absorbed some 7 per cent of the compressor output. In this field and especially in turbine blade cooling the Germans were far ahead of other countries.

A remarkable aspect of all designs was the complete preoccupation with axial compressor types, dictated by a desire for small frontal area. Almost all compressor design was based upon work on aerodynamic blade theory [*111*] at Göttingen and elsewhere by Betz and Encke. This differed from British ideas in having all the pressure rise occur in the moving blades. While this may have resulted in a more simple construction, it had few aerodynamic attractions and compressor efficiencies were low.

The complete exclusion of centrifugal compressor engines was the result of a rigid policy laid down by the RLM. It failed because it was not backed by coordinated research and development as in Britain. Consequently design stagnated and new engines did not show any real improvement, except in size, over their predecessors.

Other countries. Interest in the gas turbine was displayed before the war, particularly in Sweden and Switzerland, and in both these countries the idea of aircraft turbojet engines was eventually conceived.

Brown Boveri were the chief exponents in Switzerland and they produced designs with axial compressors and multistage turbines for aircraft propulsion. Their schemes were patented in Germany in 1939 [*81*, p. 51] and may have had some influence on German designs. In Sweden, Lysholm proposed a number of gas turbine aircraft applications, both by airscrew and jet propulsion, in a comprehensive American patent in 1933 [*112*]. His early ideas were based on multistage centrifugal compressors coupled to double-rotation Ljungström radial flow turbines. A Bofors experimental unit of this type was built and tested in 1934, but some trouble was experienced with compressor surging. Lysholm then had the idea of a rotary screw compressor [*113*] and by 1936 he had incorporated it in an engine using a multistage axial flow turbine. Descriptions of many of his schemes were published in a paper presented by him in London in 1947 [*114*].

A,13. Ramjet Development. The continuous air flow thermal duct jet has been variously described as propulsive duct, Lorin tube, ramjet, athodyd,[12] and even "flying stove pipe." The American term, ramjet, is now almost universally adopted.

Lorin, whose advocacy of thermal jet propulsion has already been noted (Art. 11) gave much thought to the compressorless motor and from him it derived one of its early names. Its possibilities were noted by both him and Marconnet as early as 1909, but it was not until 1913 that he propounded the more or less classical streamlined form of modern days [*115*]. The idea was premature, of course, since flying was in its infancy and attainable speeds were so low that little compression could be obtained. Consequently, little or no practical investigation was carried out. In spite of this, or perhaps on account of it, ideas continued to abound. Another French development, that of Hayot [*116*], introduced an interesting feature which attempted to meet, at least in part, the low ram criticism. The whole motor was to be housed in the leading edge of a wing, the intake being on the under surface where pressure was high and the exhaust on the upper surface where it was low. Additionally, a proportion of the hot gas was to be led back to the intake where it acted as an injector pump.

Also belonging to this early period are the suggestions of an American inventor, Lake, who obtained patents in 1909 [*117*]. In his system, mixing of air with the combustion gas to augment the thrust was proposed for the first time. An unrecognized advantage of this scheme was the increase in propulsive efficiency which resulted from the lower jet velocity. This system anticipated by some ten years that of Mélot, whose "propulseur à trompe" [*118*] became much better known. This augmentation method

[12] A convenient contraction of *A*ero-*th*ermo-*dy*namic *d*uct, first used by the Germans during the war.

was described by various other inventors of the same period, but in their cases it was applied to mechanical compressor motors. Mélot himself later applied it to rockets (Art. 5).

Up to this time, inventors were content with motors suitable for subsonic flight, but in the later part of the 1920–1930 decade the work of Mach, Prandtl, and ballisticians generally inspired consideration of supersonic flight motors, even though aircraft speeds had not risen high enough to make subsonic ramjets a reasonable proposition. In 1928 Fono described a thermal jet motor for high altitude flight for aircraft with high supersonic speeds. One of the characteristics of the motor was that "the cross section of the compression nozzle tapers at the front in the direction of flow." In 1933, René Leduc produced designs [*119*] which were very similar to present day schemes. They included a suggestion for surrounding a projectile body by a tube which "suppresses the impact waves at supersonic speed." He also considered boundary layer control for both inner and outer surfaces of his duct entry by suitable connection of high and low pressure regions. The combination of jet propulsion and boundary layer suction has been proposed frequently in more recent times [*120,121,122*]. In 1938, Leduc exhibited a model ramjet-powered aircraft at the Paris Salon de l'Aviation. This "machine of the future" was estimated to have an output of 14,000 hp and a speed of 625 mph.

Wartime developments were mainly in the hands of the Germans, who had been quick to appreciate the possibilities of the ramjet as a power plant for fighter and guided missiles. The Walter company at Kiel had explored the problems as long ago as 1936, but its work was mainly on a research basis and construction of a full scale model was only started shortly before the end of the war in Europe [*62*]. Sänger, of the Air Research Institute at Brunswick, was the next to take up the work and his earliest studies were reported in 1938 [*123*]. These were not very favorable as he concluded that the ramjet was a very inflexible power plant. On further consideration he modified his opinions of the value of the motor and quickly became a leader in the field. By 1943 he had built a full scale ramjet which was flight-tested on the back of a Do17Z [*124*]. This successful demonstration was followed by the design and test of very large units and by the end of the war he had various schemes for pure ramjet propulsion [*125*].

Sänger's work inspired much interest among the aircraft designers and both Focke-Wulf and Heinkel produced designs powered by ramjets. Pabst worked with the former company, studying particularly the combustion problem of the ramjet [*126*]. From his work he evolved a vaporizing burner which gave a short flame; this led to a short motor with a length-to-diameter ratio of only $2\frac{1}{2}$:1 compared with Sänger's 6:1.

Lippisch, who was a consultant to the Messerschmitt company, became interested in this propulsion in 1943. He considered the use of

solid fuel and carried out experiments with various shaped baskets packed with broken coal, charcoal, and briquettes [*127*]. His schemes were incorporated in several advanced aircraft designs including delta wing fighters [*128*]. Another German, Oswatitsch, studied the aerodynamic aspects and produced a theory on supersonic intake design using oblique shocks to give high efficiency. In tests he recovered 70 per cent of the isentropic pressure available in air initially flowing at a Mach number of 3 compared with a normal shock pressure recovery of 33 per cent [*129*].

In Britain there is little published information on wartime ramjet investigations and it can be assumed that very little effort was diverted from the main turbojet development for this purpose. G. G. Smith claims to have proposed a ramjet-propelled missile in 1943 for the bombardment of Germany from England [*81*, p. 288] and a little work on combustion systems has been reported [*93*, p. 470].

In America, interest in its possibilities was roused as soon as jet propulsion was demonstrated as a practical proposition. Theoretical studies were started on behalf of the Navy Bureau of Aeronautics and experimental work on subsonic ramjet combustion commenced at the Massachusetts Institute of Technology. In 1944 the work was extended to supersonic ramjets in the Applied Physics Laboratory, Johns Hopkins University, and since that date the effort has been actively pursued by groups of associated industrial and university contractors. The first successful flights of a ramjet at supersonic speeds were made in June, 1945 giving a demonstration of thrust exceeding drag in a six-inch diameter model [*130*]. The earliest information on aircraft ramjets relates to a subsonic unit constructed by Marquardt Aircraft Company. A pair of these were fitted on the wing tips of a Mustang aircraft and later to a Lockheed F-80 Shooting Star. This machine was the first piloted aircraft to fly with ramjet power only [*81*, p. 291]. Since that time there has been intense activity and considerable information has been released, but this applies to a period later than that covered by the present survey.

A,14. Development of Intermittent Jets. The intermittent jet engine has been variously described as a pulse jet, resojet, aero-resonator and buzz engine. Its principle is closely related to the well-known Kadenacy system for exhaust scavenging of two-stroke engines [*131*], but its origin goes back as far as that of the continuous flow ramjet. Two forms of compressorless intermittent explosion motors may be distinguished—nonresonant, where the frequency of firing is controlled mechanically, and resonant, where the operation is set by the natural frequency of the motor tube.

Only a few examples of the former class have been proposed, but these date from the time of Holzwarth's constant volume gas turbine combustion chamber. Typical of them is the proposal in 1931 of Goddard

[*132*] with inlet flap valves and plug exhaust valve coupled and linked with the spark ignition. A later idea, patented by The Fairey Aviation Company in England in 1944 [*133*], used only an automatic inlet valve and so avoided any difficulty with hot exhaust valves. It closely resembled the resonating type of motor, except for the heavy valve design which controlled the frequency, rather than the tube length.

Resonating types are first found in the patents of Marconnet in 1909 [*78*]. These give examples of both mechanical compression (Art. 11) and ram compression. The descriptions make it clear that he was fully aware of the possibilities of automatic operation. Independently, at the same period, Karavodine built and operated a resonating duct to provide energy for driving a small gas turbine. This combination had an efficiency of only 2.5 per cent [*134*] and may partially account for some of the subsequent neglect of the type, which continued for 20 years until the patent of Paul Schmidt [*135*]. Schmidt reached the typical tubular construction of the resonator jet duct by reasoning that pistonlike transfer of energy to an air column was more efficient than the ejector type of augmentation, proposed by Mélot [*118*]. From 1930 to 1940 he worked on the principle and during the latter part of the period received some German Government backing. This was not very enthusiastic until, early in the 1939–1945 war, he demonstrated a model giving nearly 1000-lb static thrust. Arrangements were then made for production by the Argus aero-engine firm and some intensive development and research was undertaken. The development culminated in the "flying bomb" motors used against England in 1944; the research was carried out at various institutes including the DFS, Ainring, and the LFA Brunswick, where Schultz-Grunow [*136*] examined the laws of wave propagation in resonator ducts.

No other country appears to have taken any serious interest in this engine until the impact of gas turbine jet propulsion turned thoughts to the possibilities of other thermal jet cycles. Late in 1943, Aerojet Engineering Corporation propounded an "aeropulse," and work was started under Zwicky with support from the Navy Bureau of Aeronautics. The motor was similar to Schmidt's tube except that it was to be completely filled with explosive mixture. By the spring of 1944, a motor having a small entry tube which acted as an acoustic valve had been built and was under test at the U.S. Naval Engineering Experimental Station, Annapolis. Later in the year copies of the German model were built and studies of the theoretical principle undertaken, but up to the end of 1945 no further major advances were made and the prospects for future applications were not very bright [*137*].

A,15. Other Forms of Air Flow Jet. The early use of piston engines as an intermediary in thermal jet propulsion has been noted (Art. 11), but it was not until 1932 that the use of exhaust gases as a means of

providing jet reaction to supplement the shaft power was suggested [*138*]. It was developed practically in the 1935–1940 era when aircraft speeds had reached a reasonable figure and was applied to many high speed aircraft during the war [*139*]. In the same period it was realized that some jet reaction, or at least reduction in drag, could be obtained by ducting the air flow through radiators of piston engines [*140*]. This, too, was extensively used during the war period.

The principle of utilizing the exhaust has since been applied to gas turbine propeller engines where the exhaust gas will supply quite an appreciable thrust [*141*]. The first such engine to be tested was the T-31-GE, a General Electric Company axial compressor turboprop, in the spring of 1943. By 1945, similar engines had been developed in England by Armstrong Siddeley [*142*], Bristol [*143*], and Rolls-Royce [*144*]. The last of these was a modified Derwent jet engine. It was installed in a Meteor and flew in September, 1945, the first turboprop to power an aircraft in flight. Since that time there has been a considerable broadening of interest in such mixed thrust engines because of their improved efficiency in the medium flight speed range.

A similar case has been made out for the combination of piston engine and turbocompressor unit in a manner which permits of considerable exhaust thrust being developed [*145*]. Such systems are generally complicated, but are justified by their increased thermal efficiency. Many possible combinations of compound engines were described by Bachle [*146*]. Two early examples started in 1944 were the Allison V-1710-127 and the British Napier Nomad [*81*, pp. 280, 284].

Proposals for thermal jets to drive rotors have been made since very early days. In 1904, Nernst suggested a scheme [*147*, p. 1219] for conducting air through hollow rotating arms where centrifugal force would compress it. Fuel was then injected and burned before the hot gases expanded through nozzles, so producing a torque on the central shaft. A similar scheme was developed in 1925 at the RAE by Carter [*148*], in which a hollow airscrew was driven by thermal jets at its tips.

An early helicopter scheme embodying jet-driven rotors was Avery's "flying machine," dating from 1903 and illustrated by Smith [*81*, p. 36]. The next helicopter proposal of interest was a single blade rotor conceived by two Frenchmen, Papin and Rouilly, in 1915 [*149*, pp. 169–172]. A similar scheme was proposed in America by M. Knight about 1936 and tests of a wind tunnel model were made in the following year. The same principle was adopted by the German Doblhoff, who used an engine in the fuselage to drive a compressor. This compressor supplied air to the blade tips where fuel was added and burned to produce the reaction jets. Several models were successfully flown during World War II [*150*]. In England, Pullin proposed the application of a turbojet engine to the same end [*151*], but in his case the hot gases leaving the turbine were

conducted to the blade tips without the addition of further heat. One other method of deriving rotor tip thrust envisaged in Germany was by the use of ramjets fitted to the blade extremities. This idea was put forward by Focke-Wulf in a remarkable rotating wing fighter design [*152*]. The wings were to have variable pitch so that the ram speed could be maintained at a high value for all aircraft speeds. Schemes similar to this and all the foregoing have been actively investigated since the close of World War II by several countries.

A,16. Postwar Turbojet Development. By the end of 1945 it was clear that jet propulsion would largely supersede the piston engine-propeller combination for high speed flight and consequently the development tempo, which might have been expected to slacken without the stimulus of war, actually increased. Unfortunately the secrecy of wartime conditions, after a temporary easement, was reimposed and it became very difficult to find what progress was being made. Careful consideration was given to all the theoretical combinations of engine components which might promise advantages [*153*], but no immediate action toward completely new types was apparent. In fact, the period became one of advance in detail more than in major features.

This steady progress is illustrated [*154*] by the increase in maximum engine static thrust from the 5000 lb of the Nene in 1945 (Art. 12) to the "over 15,000 lb" of the De Havilland Gyron in 1953. Similarly frontal area was reduced and the thrust area ratio increased from 500 lb/ft^2 in the Metro-Vickers Beryl to 1465 lb/ft^2 in the Armstrong Siddeley Sapphire 7. Static specific fuel consumptions were also greatly reduced and instead of figures around 1.0 lb/hr/lb the most economical engines gave consumptions around 0.76 lb/hr/lb, which was very close to the limit predicted only a few years earlier [*155*]. Reduction in engine weight was not so marked although several spectacular reductions pointed the way to lighter construction. The Bristol Orpheus [*156*] was a good example, having a thrust-to-weight ratio of 4.5 lb/lb compared with average values in 1945 of less than 3.0 lb/lb.

These examples of British progress were repeated in American designs and were the fruits of the intensive research efforts undertaken [*157*] in this period. The main work was on four lines, aimed at increasing compression ratio, cycle temperature, combustion efficiency and intensity, and the operating temperature of heat-resisting materials.

The first of these encouraged a most important postwar development—the diversion of effort from the centrifugal to the axial compressor turbojet. The necessity for this was quickly recognized in the United States, but it provoked much argument in Great Britain [*158*] and produced many detailed problems [*159*]. That it was a sound move has since been clearly demonstrated by the almost universal adoption of the axial

layout in new designs. It also ended the era of simplicity in which engines were designed and manufactured within six months.

Limitations on the maximum pressure ratio of a single rotor axial compressor caused by poor off-design characteristics led to proposals for "two spool" engines in which independent turbines drove separate high and low pressure compressors [*160*]. This form has been largely responsible for the improved specific fuel consumption noted above. Such arrangements were first considered in 1938 by the Royal Air Establishment [*161*] but were abandoned in favor of a more simple engine layout. In 1944 a combination of axial and centrifugal compressors on one shaft was used on the Bristol Theseus turboprop [*143*] and in 1947 Rolls-Royce used separately driven axial and centrifugal compressors in the Clyde [*162*]. Efforts to reduce the size and weight of compressors turned attention to supersonic compressors [*163*] in which one or two stages would do the work of six or eight subsonic stages. The Germans were the first to consider the advantages of such compressors and Weise of the DVL, Berlin, took out patents in 1936 [*164*], but his experimental compressor was a failure. Westinghouse carried out some work in 1941–1942, but the modern work was initiated by Kantrowitz at the NACA Langley Laboratory [*165*]. So far, however, it does not appear to have been applied to an actual engine.

Higher maximum temperature cycles offered advantages in specific thrust output and could be achieved by better materials or cooling. At the end of the war the use of the high nickel-chromium, heat-resisting alloys [*166*] was firmly established in Great Britain for temperatures up to 1073°K. These were further developed so that operating temperatures up to 1200°K became possible [*167*]. The main advances were, however, in cooling techniques. The Germans had again been pioneers in this field, with both air cooling [*108*] and liquid cooling [*168*]. Extensive experimental work was carried out by NACA after 1945 and a series of reports issued [*169*]. Similar work in England at the National Gas Turbine Establishment resulted in experimental operation at turbine gas temperatures up to 1400°K [*170*]. Most of the work involved internal cooling flow, but one interesting proposal was for external turbine blade cooling in conjunction with afterburning [*171*]. In this system, liquid fuel was sprayed onto the blades and was burned further downstream.

The third line of development was in combustion systems. The position in 1945 [*172*] was far from satisfactory, and a great deal of both practical and theoretical study effort was applied to increasing combustion intensity, stability at altitude, ignition, and efficiency [*173*]. This was accompanied by a trend toward annular chambers, first using a number of individual flame tubes, but in later designs containing a single annular primary zone.

Apart from the progress in these main components, the exhaust

system underwent considerable development. The advent of supersonic flight increased its importance both with reference to thrust augmentation by afterburning and to efficient propelling nozzle jet expansion. The first afterburning systems have been mentioned [*100*], but their full value could not be realized without the use of variable area propelling nozzles. Originally it was considered that two-position nozzles would meet the needs [*174*] and various geometrical forms with such descriptive titles as letter box, eyelid, caliper, and clamshell were tried. By 1950 it was found desirable to have fully variable nozzles and an iris construction was developed [*175*]. This had a high efficiency, the opening was always circular and concentric, and it could be conveniently cooled. Mechanically operated nozzles have, however, the disadvantages of considerable additional weight and complication and this was claimed to be overcome by the French [*176*], who produced an aerodynamically controlled nozzle in which a radial flow of compressed air injected at the jet exit section caused the effective area to vary.

All these nozzles have been of the simple convergent type, but at supersonic flight speeds convergent-divergent types are necessary for good efficiency. An annular supersonic nozzle with good off-design operation was proposed by Griffith in 1952 [*177*]. This made use of a Prandtl-Meyer type of expansion around the external lip of the annular throat. Its chief disadvantages were the difficulty of cooling the central bullet and the fact that the maximum diameter of the nozzle was necessarily greater than the jet diameter.

Further development efforts on turbojet exhaust systems were directed toward thrust deflection, thrust spoiling, and thrust reversal. The first has been tried in connection with both shortened take-off and landing [*178*], and the others with landing only. An early thrust spoiler was made in 1945 and described by Cox [*91*]. A more sophisticated system using "eyelid" shutters over the normal jet exhaust orifice and deflecting the gas through airfoil vanes on the side of the pipe was demonstrated by Rolls-Royce in 1956 [*179*]. In the meantime the French had applied their aerodynamic nozzle control system to thrust spoiling; by 1951 they had completed wind tunnel tests with a De Havilland Goblin [*180*].

The propelling nozzle was also the subject of considerable study in connection with the problem of jet noise. The first success in dealing with this was by the use of deflector "teeth" on the nozzle, proposed by Westley and others working at Cranfield, England [*181*]. This was developed by Rolls-Royce into the "petal" nozzle [*182*] which gave a marked reduction in noise level without loss in thrust.

A,17. Other Recent Forms and Applications. Postwar turboprop engines developed in parallel with the turbojets, taking advantage of all the improvements in compressor design, combustion, and materials to

provide power/frontal area ratios increasing from 295 ehp/ft² in 1946 to 600 ehp/ft² in 1955, specific consumptions falling from 0.60 lb/ehp-hr to 0.40 lb/ehp-hr, and specific weights dropping from 0.80 lb/ehp to 0.53 lb/ehp [*183*]. Maximum single-shaft engine powers remained in the neighborhood of 4000 ehp, largely because of the limitations on power absorption by a single airscrew. An effective doubling of the power was provided by the use of twinned engines. Two separate engines drove either a single contrarotating propeller, as in the Bristol "coupled Proteus" [*184*], or two separate concentric propellers, as in the Armstrong Siddeley "Double Mamba" [*185*]. A later development for ensuring large power at altitude, described as the "supercharged turboprop," was put forward in 1954 by Bristols [*186*]. This concept required that the potential sea level power output of the engine should be very large, but that the mechanical design and stressing should limit the actual power to some lower figure. This figure, however, could be maintained to quite high altitudes because the theoretical performance could become more and more real with increasing altitude without exceeding the limiting stresses. The great advantage of this was the much lighter engine at height.

The ducted fan engine and its near relation, the bypass turbojet, were the subject of much design thought because of their good specific consumption and other virtues, but the first practical example was the French Turboméca Aspin ducted fan type which produced a thrust of 440 lb with a specific consumption of 0.628 lb/hr/lb [*187*]. This engine was flown in a Fouga aircraft in 1951. About the same time a much larger bypass engine, the Rolls-Royce Conway, was being developed and run at over 10,000-lb thrust [*188*].

While the main turbojet applications were to high speed aircraft, the foregoing were all intended for normal subsonic operation. At the same time new thoughts were forming on possible uses at very low speeds. The jet-driven rotor for helicopters was under active development (Art. 15) and an interesting variant of this was the proposal to mount actual turbojet engines at the rotor tips to provide the thrust [*189*]. A less ambitious but more promising scheme was the use of a single engine to provide both vertical lift and horizontal thrust as proposed by Fairey [*190*] and incorporated in their Rotodyne. This machine had gas turbines which provided shaft power to air compressors for driving rotor tip jets during hovering flight. These were declutched and the power applied to driving propellers in forward flight. This was one form of the so-called "convertiplane" which aroused serious thought about 1949 [*191*]. Another, and more direct, form achieved vertical take-off by starting with the aircraft pointing vertically and relying on the thrust generated by the propeller being greater than the all-up weight. When adequate height and speed had been obtained the aircraft could be turned on to a horizontal course. Preliminary designs of this type were made in the Research

Division of the Bureau of Aerodynamics and experiments on such an arrangement were carried out by Convair in 1954 [*192*] and were made possible by the low specific weight of gas turbines. Similar experiments with direct lift jets were made by Rolls-Royce with their "flying bedstead" about the same time [*193*]. The ultimate objective was to provide an aircraft with a large number of lifting jets which could be diverted to give horizontal thrust in flight, or alternatively could be switched off while other engines gave forward thrust.

A radically different approach to low speed landing and take-off was proposed by Davidson [*194*] with the "jet flap." In this the engine jet was blown out along the trailing edge of the wing through a narrow slot which deflected it downwards. This stabilized the air flow over the wing at very low forward speeds, and very high lift was achieved in combination with useful propulsive thrust. The use of such a jet for boundary layer control was not new and had been demonstrated as early as 1931 by Bamber [*195*], but neither he nor the others prior to Davidson had appreciated the vital principle of the combined lift and thrust effect. It was not until Attinello advanced the idea of supersonic blowing that the concept became practical. His ideas were like a breath of spring after a long winter. The Attinello system was soon tested and installed in both United States and British airplanes.

At the high speed end of the flight spectrum, ramjets developed rapidly, especially for propulsion of guided missiles [*196*], but no completely new ideas were put forward. Leduc (Art. 13) continued his work in France which was interrupted by the war and by 1949 had flown his 010 machine successfully. During the following years his integrated ramjet-aircraft flew on a large number of occasions at high subsonic speeds, with the ramjet as the sole power plant [*197*]. Proposals for hybrid power plants in which ramjets were combined with rockets or turbojets were mentioned from time to time [*197*], but the most interesting was the air-turborocket which has been actually constructed by Aerojet [*198*] and Experiment Incorporated. It consisted essentially of a turbojet in which the compressor was driven by a turbine running on rocket propellants and the compressed air was burned in a tailpipe with auxiliary fuel. The engine was much less dependent on altitude and forward speed than the conventional turbojet and had better consumption characteristics than the rocket.

The advent of nuclear fission as a practical source of energy led to many attractive but ill-considered proposals for its application to aircraft. Serious studies indicated that, at least in early designs, atomic energy would only be used indirectly as a substitute for the combustion system to provide heat energy which could be transferred to a more conventional gas turbine system. Nevertheless, in spite of the many

peculiar problems it presented, several American firms undertook practical design studies and it was claimed that an atomic engine would be airborne in 1955 [*199*].

In conclusion, it may be observed that in the ten years following the 1939–1945 war, air flow jet engines supplanted piston engines in almost all high performance military aircraft, they made large inroads on the civil aviation field including the first regular turbojet passenger service in 1951 [*200*], they made supersonic flight a commonplace, and finally they practically doubled the world air speed record [*201*].

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SECTION B

BASIC PRINCIPLES OF JET PROPULSION[1]

MAURICE ROY

CHAPTER 1. DEFINITIONS AND SIMPLIFICATIONS

B,1. Classification of Jet Propulsion Engines. In the absence of all contact with an external solid, a prime mover placed in a fluid of finite or even zero density can propel itself by ejecting a fluid or solid mass toward the rear. By convention we say that this mover is propelled "by jet propulsion," although the propulsive thrust really results from the effects of pressure and friction exerted on the wall of the hollow interior of the mover by the solids or fluids moving in the interior toward the exhaust nozzle.

The mechanical power plant that involves a propeller acts in a similar manner but externally, in the sense that it imparts to the outer medium an additional momentum directed toward the rear. Because the ejection of the exhaust gases of a turboprop (that is, a gas turbine driving a propeller) can contribute more or less, to the total thrust of the system, we include this engine, which constitutes a mixed form of jet propulsion, in the present exposition of basic principles.

In all jet propulsion engines, just as for all propulsion of thermal origin, the source of available energy is an exothermic chemical transformation of solids, liquids, or gases, *carried on board* and called propellants. For brevity, we include all these propellants under the single term "fuel."

For propulsion in the atmosphere, where ambient air is available, this air may be inducted by the jet engine in order to participate essentially in the chemical transformation of the fuel: the engine is then properly called an air flow jet engine in contrast to the rocket, which does not use any air and is the only jet engine which can be used for propulsion in vacuum.

A flow of ambient air may also be inducted exclusively for augmenting the thrust by increasing the mass flow of the discharged gases: thus we can conceive of an air flow jet engine with two flows, an example of

[1] This section was written in 1951.

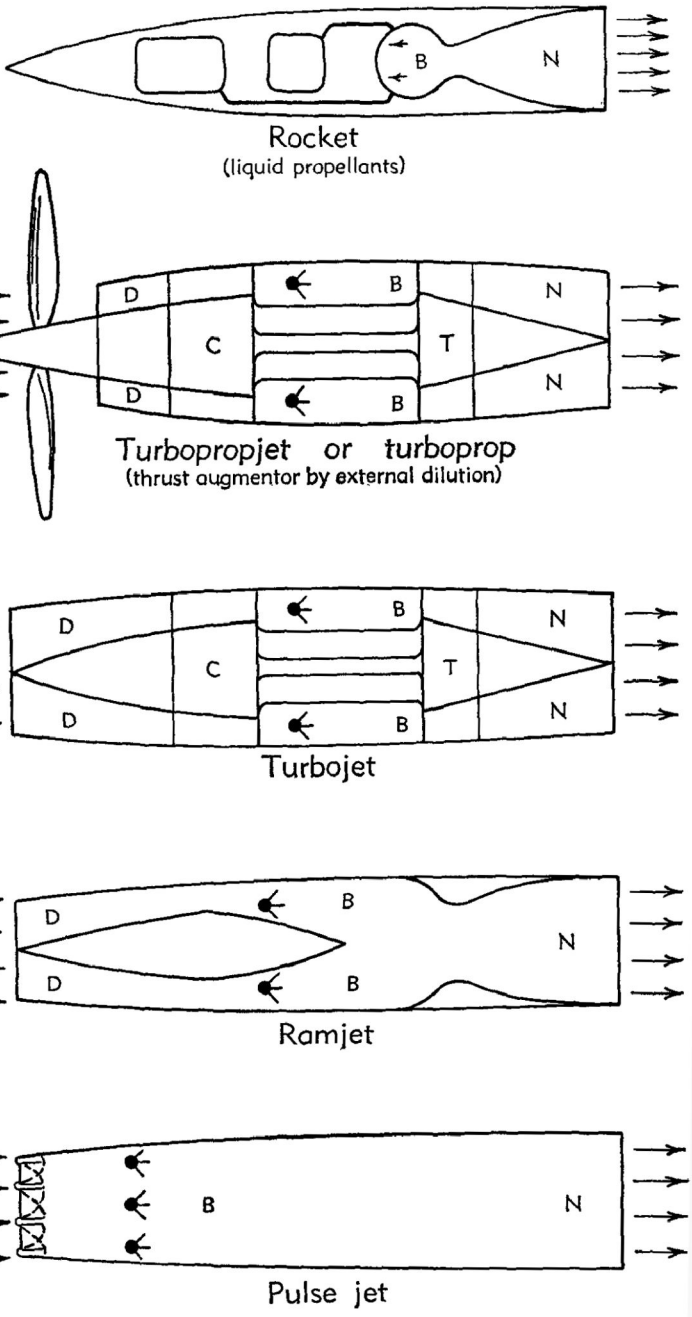
B
N
Rocket
(liquid propellants)
D
C
D
B
B
T
N
N
Turbopropjet or turboprop
(thrust augmentor by external dilution)
D
C
D
B
B
T
N
N
Turbojet
D
D
B
B
N
Ramjet
B
N
Pulse jet

Fig. B,1.

which is given by a turbojet with a ducted fan. In fact, we can even conceive of a rocket whose thrust is augmented by inducting ambient air—this air may be induced by an ejector effect or by the ram pressure due to flight velocity. The first system is called an ejector rocket and the latter a ramrocket. At low speeds, the latter operates as an ejector rocket.

Whereas the internal flow of a rocket is usually continuous, and even quasi-steady, that of an air flow jet engine can be continuous or discontinuous, i.e. intermittent. The machines which work on the internal flow of a jet engine and have a continuous flow require purely rotating machines, i.e. turbomachines; the engine is called a turbojet.

If any of these machines are reciprocating, the internal flow is discontinuous and the engine is called a motojet. Up to now this kind of jet engine has not been used, although it was proposed as early as 1908 by R. Lorin [*1*].

If compression and expansion of the internal flow is not produced by such rotating or reciprocating machinery but by oscillatory motion of the flow, i.e. wave motion, the jet engine is called a pulse jet.

If, in a continuous flow jet, only the compression resulting from ram effect in the inlet diffuser is utilized, the compression and expansion machinery is eliminated, and the turbojet becomes a "ramjet," the principle of which was proposed by Lorin in 1913 [*2*]. This very simple jet engine, which is also called an "athodyd" by abbreviation of "aerothermodynamic duct," is necessarily inefficient at low speeds.

The principal types of engines as classified are represented schematically in Fig. B,1.

We can further subdivide this classification by distinguishing among the many types of air inlets, internal flows, exhaust nozzles, and ways of introducing fuel or active substances such as the "coolants" in the various flows, etc. Combinations then appear possible between the preceding fundamental types: for example, the well-known turbojet with "reheat" can be considered as a ramjet supercharged by a turbojet.

B,2. Thermodynamic Evolution and States of the Internal Flow. All air flow jet propulsion engines are composed of several constituent parts, in which the changes in all or part of the active substances (fuel, coolant, air) take place. These constituent parts include *stationary* members such as the inlet diffusers, burners, and tailpipes, and *moving* members such as the distributors, compressors, pumps, and turbines. Practically, it is necessary to study the change of state, i.e. thermodynamic evolution, of the internal flow across each such constituent member by fixing the state of the flow in a conventional entrance or exit section of the specified member, the state at the exit of each member being considered as the state at the inlet of the member following.

In any reference section chosen arbitrarily the area is designated by

the symbol A_x; the corresponding state of the flow is symbolized by the subscript x. This state is characterized by physical parameters (pressure p, density ρ, absolute temperature T, etc.), by chemical parameters (hereafter, a single parameter denoted by κ, which is zero before the reaction and equal to 1 when the transformation is complete), and finally by a parameter of motion, composed of the specific relative kinetic energy $E_k = \frac{1}{2}V^2$ (V, velocity referred to the casing or stator of the engine), each one of these quantities being marked by the subscript x.

The state of flow is thus defined arbitrarily as p_x, T_x, κ_x, E_{k_x}. Actually, this state is variable from one point to another across the section A_x, as well as during the duration of the period τ of the "fixed set of running conditions," granting the existence of such a fixed set of conditions.

The quantities p_x, T_x, κ_x, and E_{k_x} then represent *mean values in the section A_x during the period τ*, these averages being weighted in relation to the elementary masses of fluid $\delta m \delta\tau$ which are flowing across the element of cross section δA_x during the element of period $\delta\tau$. With the exception of motojets and pulse jets, the conditions of the internal flows are practically steady and the preceding averages are then weighted only in relation to the steady mass flow δm across the element of cross section δA_x.

The calculation of the *thrust* of a propulsive engine is mainly based on the axial component of the relative flow velocity in the inlet and exit sections. The exit sections A_e are practically orthogonal to the x axis, and are in the end opposite to the direction of the velocity V_∞ of the engine in translation; therefore the relative exhaust velocity V_e is practically parallel to the x axis. Hence it is particularly the average (over the period) of V_e, mass-averaged with respect to δm, which comes into play in the calculation of the thrust. This *simple average*, denoted by the symbol $\overline{V}_e$, differs from the *quadratic* average $\overline{\overline{V}}_e$, which is defined by $E_{k_e} = \frac{1}{2}\overline{\overline{V}}_e^2$. It is important to remember that we always have

$$\overline{V}_e < \overline{\overline{V}}_e = \sqrt{2E_{k_e}} \tag{2-1}$$

With regard to the air inlet section A_i they are defined in such a way that the air flow velocity V_i is in each section practically parallel to the x axis, so that in the calculation of the thrust we have again $\overline{V}_i < \overline{\overline{V}}_i \equiv \sqrt{2E_{k_i}}$.

In order that the normal force of the flow, over the sections A_i and A_e which are practically perpendicular to the x axis, be identified with the reversible pressure p, we assume in these sections that the effect of viscosity is small and, in particular, that the thickness of the boundary layer in the exit flow is sufficiently thin in the sections A_e.

The preceding conditions and reservations are necessary in order to justify the general expressions to be discussed below.

If it is assumed that the operation is accomplished under a fixed set of conditions, then here it can only be a question of *horizontal* flight. In this case, gravity does not affect the internal operation and the state (with subscript ∞) of the undisturbed atmosphere, that is, the atmosphere far upstream of the engine and at the same altitude is defined by p_∞, T_∞ and the specific relative kinetic energy $E_{k\infty} = \frac{1}{2}V_\infty^2$.

B,3. Thrust and Drag. The jet propulsion engine is housed in a streamlined container which is a hollow body connected to the outer surface of the engine by the exhaust nozzle and possibly by the air inlets. In order to define thrust or drag, it is necessary to separate by convention the jet propulsion engine from the other bodies to which it can be connected. Thus such a convention is quite natural for a totally isolated engine, like a rocket without wings or tail surfaces, or even for a streamlined engine plainly separated from the airplane to which it is attached, by pods for example. The convention is more arbitrary for an engine more or less completely enclosed in a wing.

Whatever the conventional definition may be, here we make the following designations (Fig. B,3a where the forces are positive):

1. External drag F_{ext} is the axial resultant of the normal and tangential forces of the ambient atmosphere on the *external* surface Σ_{ext} of the engine
2. Internal thrust F_{int} is the axial resultant of the internal flow forces on the *interior* surface of the engine
3. Effective thrust F_{eff}, is the difference between the internal thrust and external drag ($F_{int} - F_{ext}$) possibly augmented by the thrust F_p of the propeller, i.e. the axial resultant of the forces of the ambient atmosphere on the propeller blades which protrude from the engine cowling.

We then have, by definition:

$$F_{eff} = F_{int} - F_{ext} + F_p \tag{3-1}$$

The solid external surface Σ_{ext} is bounded on one hand by its conventional frontiers with one or more bodies (airplane, wings, and tail surfaces) connected to the engine, and on the other hand by the fictitious surfaces of inlet A_i and of exit A_e.

In the case of an air inlet of the "scoop" type, the fictitious surface of inlet (and of area A_i) is drawn across the entering flow and practically follows a plane perpendicular to the x axis, while the portion of the streamline surface located downstream of the cross section A_i, which establishes the boundary of the inducted flow, must be considered a part of the external surface Σ_{ext} of the propulsive engine. Of course, the exact shape of this portion of the streamline surface varies with the "fixed set of running conditions" and accordingly with the flight speed V_∞. At zero speed, in particular, the drawing of a surface A_i practically perpendicular to the x axis may be impossible, because the flow of

inducted air enters the inlet following various directions more or less oblique to the x axis. In such a case we would have to integrate the axial components of pressure and velocity V_i of the flow for each element of a better appropriate surface A_i, but for simplification we do not consider this particular case hereafter.

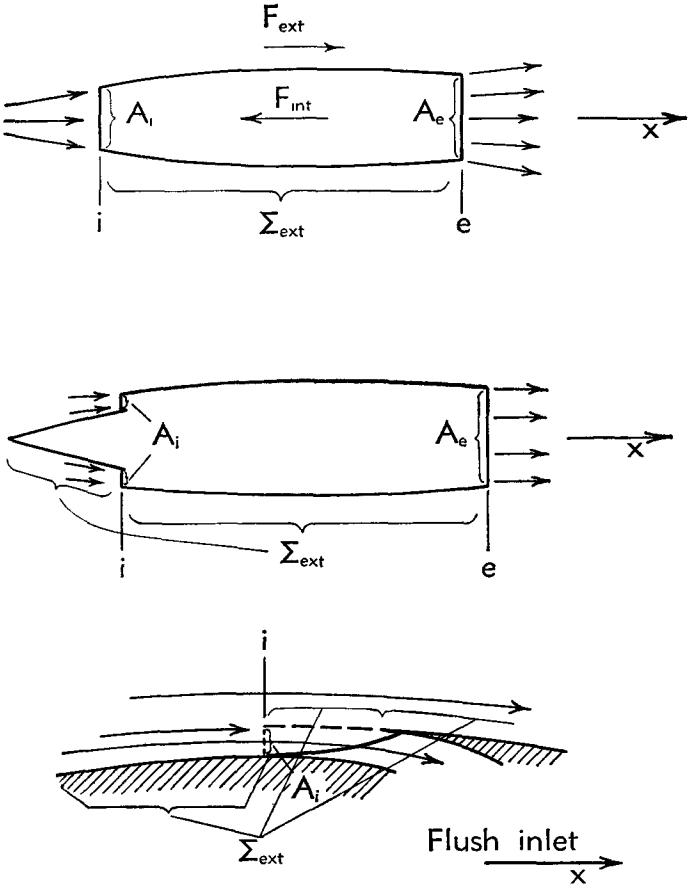

Fig. B,3a.

It is important to note, particularly as a result of the presence of shock waves in the neighborhood of the sections A_i and A_e, that the pressure of the internal flow in these sections is not always uniform or equal to the pressure p_∞ of the uniform atmosphere infinitely far upstream.

Therefore let us draw (Fig. B,3b), across the internal flow extended upstream of A_i and downstream of A_e, two sections A_∞ and A_f perpendicular to the x axis and where the pressure p_∞ exists, these sections being located as near as possible to A_i and A_e.

At A_∞, the air at p_∞ and T_∞ is practically at the axial and uniform relative speed V_∞. Let us designate by Σ'_{ext} and Σ''_{ext} the portions of the streamline surface, extended to A_∞ and A_f, which bound the internal flow, and which are to be added to Σ_{ext}; and by ΔF_{ext} and ΔF_{int} the corresponding additions to F_{ext} and F_{int} which correspond respectively to the axial resultant of the forces of the ambient and external flow, or the internal flow, on the additional lateral surfaces Σ'_{ext} and Σ''_{ext}.

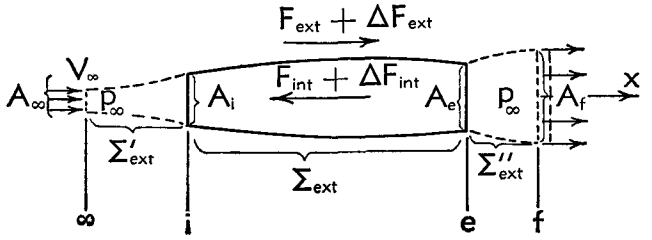

Fig. B,3b.

Since by definition $\Delta F_{ext} = \Delta F_{int}$, Eq. 3-1 is written as

$$F_{eff} = (F_{int} + \Delta F_{int}) - (F_{ext} + \Delta F_{ext}) + F_p \qquad (3\text{-}2)$$

Let us suppose $F_{ext} + \Delta F_{ext}$ is known. The internal thrust F_{int}, or its corrected value $F_{int} + \Delta F_{int}$ can then be obtained from the momentum theorem, applied during the period τ to the portion of the flow included respectively between A_i and A_e, or between A_∞ and A_f.

Let us designate by m the average mass flow across a fixed section during the period τ, affixing the subscripts a, f, and w respectively, to the air, to the propellant or fuel, and finally to the coolants such as water or water plus methanol, which one can inject into the flow in order to reduce its temperature.

Noting that the fuel and the coolant can be considered in their reservoirs[2] to be at the pressure p_∞, and withdrawn at negligible speed, we have

$$F_{int} = (m_a + m_f + m_w)V_e - m_a V_i + (p_e A_e - p_i A_i) \qquad (3\text{-}3)$$

$$F_{int} + \Delta F_{int} = (m_a + m_f + m_w)V_f - m_a V_\infty - p_\infty(A_\infty - A_f) \qquad (3\text{-}4)$$

Either one of these equations can be used to calculate F_{int}.

If we call d_{ext} and Δd_{ext} the drags F_{ext} and ΔF_{ext} that are calculated by subtracting p_∞ from the normal pressures on the surfaces Σ_{ext} and $\Sigma'_{ext} + \Sigma''_{ext}$, and if we observe that the resultant of the uniform pressure p_∞ on the surfaces completed by A_∞ and A_f is zero, we can write

$$F_{ext} + \Delta F_{ext} = d_{ext} + \Delta d_{ext} - p_\infty(A_\infty - A_f)$$

[2] If these substances are stored in the liquid or solid state, the pressure in the reservoir is of negligible influence.

Then from Eq. 3-2 and 3-4, we have

$$F_{\text{eff}} = (m_a + m_f + m_w)V_f - m_a V_\infty - (d_{\text{ext}} + \Delta d_{\text{ext}}) + F_p$$

The "kinetic thrust of the jet" is designated by the term[3] F_k, where:

$$F_k = (m_a + m_f + m_w)V_f - m_a V_\infty \tag{3-5}$$

This differs from the effective thrust of the jet only by the sum $d_{\text{ext}} + \Delta d_{\text{ext}}$, which is generally small and which can be calculated as specified above.

In that which follows, F_k is taken as a measure of the thrust of the jet, and $F_k + F_p$ as a measure of the *total thrust*, propeller included. If the section A_f practically coincides with the actual exit section A_e, then V_f becomes the simple average V_e of the axial exhaust velocity.

It should be noted that in the case of supersonic propulsion, one might prefer to take $F_k - F_n$ as a measure of the principal part instead of F_k, with

$$F_n = n_e A_e - n_\infty A_\infty$$

where n designates the "dynalpy" $P + \rho V^2$, and the downstream section of flow is A_e instead of A_f with $P_e \neq P_\infty$.

B,4. Powers and Efficiencies. The engine being considered consumes, on an average and during unit time, the mass of fuel or propellant m_f (in one or more distinct bodies). All the powers are hereafter referred to this mass, which is evaluated as "specific power." Provided that we assume air to take part in the chemical transformation of the fuel, the relative air-fuel supply ratio m_a/m_f is *at least sufficient* to render possible a complete transformation ($\kappa = 1$).

In addition to the states ∞, i, e, and f considered above for air (subscript $_a$) and the burned gases (subscript $_g$) we distinguish the following states: 0, the state of the fuel, coolant, and air at rest ($E_{k_0} = 0$) and at p_∞, T_∞, the initial state for the definition of the heating value Q_0 of the fuel at constant pressure and per unit mass of the fuel, in the presence of the respective proportions m_w/m_f and m_a/m_f of coolant and air; fi, the final state of the gases of complete combustion at p_∞ ($\kappa_{fi} = 1$), these gases having been brought back to rest ($E_{k_{fi}} = 0$) and to T_∞; fr, the final state of the actual combustion gases ($\kappa_{fr} = \kappa_g$), taken at $p_g = p_{fr} = p_\infty$, and brought back to T_∞ and to rest ($E_{k_{fr}} = 0$), under the constant presspre p_∞.

Using these notations and designating the specific enthalpy[4] and

[3] This term is called "universal thrust" by Zwicky [3], and "gross thrust" by the author [4]. The designation "kinetic" is used here in preference, in order to emphasize the momentum change.

[4] In the above equations, the changes in h_f and h_g include what is usually called "chemical energy" as well as the enthalpy of the working medium.

entropy by h and s, the (specific) *heating value* Q_0 is defined by

$$m_f Q_0 = m_a(h_a)_0 + m_f(h_f)_0 + m_w(h_w)_0 - (m_a + m_f + m_w)(h_g)_{fi} \quad (4\text{-}1)$$

Then the (specific) *energetic value* Φ_0 of the fuel, that is, the diminution of its (specific) free energy when undergoing the "reversible" transformation from 0 to fi at the stationary condition in one atmosphere at T_∞, is defined by

$$m_f\Phi_0 = m_f Q_0 + T_0[(m_a + m_f + m_w)(s_g)_{fi} - m_a(s_a)_0 - m_f(s_f)_0 - m_w(s_w)_0] \equiv m_f(Q_0 + q_0) \quad (4\text{-}2)$$

Concerning the engine in translation at a speed V_∞, the masses m_f and m_w taken aboard and discharged in unit time possess the specific and absolute kinetic energy $E_{k_\infty} = \frac{1}{2}V_\infty^2$, taken away from the source of energy of fuel previously consumed, with the result that the total *available power* $m_f P_{av}$, related to the absolute coordinate system, is:

$$m_f P_{av} = m_f\Phi_0 + (m_f + m_w)E_{k_\infty} \quad (4\text{-}3)$$

On the other hand, the "combustion efficiency" η_b of the actual and incomplete combustion is defined by

$$\eta_b m_f Q_0 = m_a(h_a)_0 + m_f(h_f)_0 + m_w(h_w) - (m_a + m_f + m_w)(h_g)_{fr} \quad (4\text{-}4)$$

This being granted, then we denote: $m_f P_p$ as the *mechanical power* furnished to the eventual propeller on its shaft; and the quantity $m_f P_{pr}$ as the *effective power of propulsion*, or propulsive power, for which we identify the effective thrust to be the sum $F_k + F_p$. Then it follows that

$$m_f P_{pr} = (F_k + F_p)V = (F_k + F_p)\sqrt{2E_{k_\infty}} \quad (4\text{-}5)$$

The quantity $m_f P_{th}$ is the *thermal power* of the transformation of the active substances, starting with the states i and 0, to the state f, for the engine in translation at the speed V_∞ (to be defined later).

By convention, we define the *over-all efficiency* η_0 by the ratio of the effective power to the available power, or

$$\eta_0 = \frac{P_{pr}}{P_{av}} = \frac{(F_k + F_p)\sqrt{2E_{k_\infty}}}{m_f\Phi_0 + (m_f + m_w)E_{k_\infty}} \quad (4\text{-}6)$$

the *thermal efficiency* η_{th} by the ratio of the thermal power to the available power, or

$$\eta_{th} = \frac{P_{th}}{P_{av}} \quad (4\text{-}7)$$

and finally, the *propulsive efficiency* η_{pr} by the ratio of the effective power of propulsion to the thermal power, or

$$\eta_{pr} = \frac{P_{pr}}{P_{th}} \quad (4\text{-}8)$$

so that the first efficiency is the product of the other two:

$$\eta_0 = \eta_{th}\eta_{pr} \tag{4-9}$$

From Eq. 4-6 and by definition of the propeller efficiency η_p, the thrust F_p of the propeller is related to the mechanical power $m_f P_p$ that it receives on its shaft by the relation

$$F_p V_\infty = \eta_p m_f P_p \tag{4-10}$$

As for F_p, η_p is here relative to the propeller in operation on the engine under the fixed set of conditions considered, that is, the aerodynamic interaction of the engine on the propeller is taken into account by η_p, just as the external drag of the engine F_{ext} includes the interaction of the propeller on the engine.

Concerning the mechanical power $m_f P_p$ let us note that the expression for it may be derived from the principle of conservation of energy by reason of the assumed existence of a fixed set of running conditions

$$m_f P_p = m_a(h_a)_i + m_f(h_f)_0 + m_w(h_w)_0 - (m_a + m_f + m_w)(h_g)_f + m_a E_{k_i} - (m_a + m_f + m_w)E_{k_f} - m_f q_{if}^e$$

where the term $m_f q_{if}^e$ represents the total heat transferred in unit time by radiation and conduction to the external atmosphere by the engine and the internal flow from the section A_i to the fictitious section A (Fig. B,3b).

Then, from state ∞ to state i upstream of the engine, the flow of inducted air undergoes a steady state and adiabatic transformation, that is, at constant stagnation enthalpy, even if the flow crosses one or more stationary shock waves upstream of the inlet section A_i, so that:

$$(h_a)_i + E_{k_i} = (h_a)_\infty + E_{k_\infty}; \quad (h_a)_\infty = (h_a)_0$$

Taking into account these relations and Eq. 4-4, the above expression for $m_f P_p$ becomes:

$$m_f P_p = m_f(\eta_b Q_0 - q_{if}^e - q_f) - m_a E_{k_\infty} - (m_a + m_f + m_w)E_{k_f} \tag{4-11}$$

where, for simplification, we place

$$m_f q_f = (m_a + m_f + m_w)[(h_g)_f - (h_g)_{fr}] = (m_a + m_f + m_w)\int_{T_\infty}^{T_f}(c_p)_g dT \tag{4-12}$$

The *thermal power* $m_f P_{th}$ cited above is defined as the sum of two terms: the first is the total mechanical power that would be received through the same transformation $i \to f$ of the active substances for the engine in flight at the speed $V_\infty = \sqrt{2E_{k_\infty}}$, but by bringing the gases back to absolute rest by means of a perfect impulse turbine. This would

add to $m_f P_p$ the additional mechanical power

$$(m_a + m_f + m_w)(E_{k_f} - E_{k_\infty})$$

The second is the supplement of the effective power of propulsion, which corresponds to the thrust due to the discharge of the supply $m_f + m_w$ at the speed V_∞, and would then be $(m_f + m_w)2E_{k_\infty}$.

From this definition, we therefore have

$$m_f P_{th} = m_f(\eta_b Q_0 - q^e_{if} - q_f) + (m_f + m_w)E_{k_\infty} \tag{4-13}$$

One will note that this expression, which is necessary, depends only on the transformation $i \rightarrow f$ of the internal flow, the consumed supply of the active substances taken aboard, and the flight speed $V_\infty = \sqrt{2E_{k_\infty}}$.

If the engine were to operate at the stationary condition, that is at zero speed ($V_\infty = 0$), with the same transformation $i \rightarrow f$ of the internal flow and with the same active substances, but taken and restored to absolute rest, the *thermal power at the stationary condition*, denoted by $m_f P^0_{th}$, would consist only of the mechanical power obtained by (1) adding to $m_f P_p$ the additional power $(m_a + m_f + m_w)E_{k_f}$ furnished by a perfect impulse turbine which would bring the discharged gases back to absolute rest, and (2) subtracting the power $m_a E_{k_\infty}$ absorbed by a perfect impulse fan which would pass the air, at p_∞, T_∞, from absolute rest to the state ∞.

By analogy with the definition of η_{th} (Eq. 4-7), we will call *thermal efficiency at the stationary condition*, denoted by η^0_{th}, the ratio of $m_f P^0_{th}$ to $m_f \Phi_0$, without the additional kinetic term. Then

$$\eta^0_{th} = \frac{\eta_b Q_0 - q^e_{if} - q_f}{\Phi_0} \tag{4-14}$$

As we will see further on, it will be especially convenient to consider this term, which depends only on the transformation $i \rightarrow f$.

Let us note that the *maximum mechanical power* which can be furnished by this transformation completed by the return to absolute rest and at p_∞ of the discharged gases in the state f is:

$$m_f P_{th} - (m_f + m_w)2E_{k_\infty}$$

It is therefore convenient to place

$$P_p = \zeta \left[P_{th} - \left(1 + \frac{m_w}{m_f}\right) 2E_{k_\infty} \right] \tag{4-15}$$

where ζ represents the fraction of the preceding maximum power transferred to the propeller. The case $\zeta = 0$ corresponds to the case of a pure jet or rocket, whereas $\zeta = 1$ corresponds to a pure turboprop or a propeller driven by a rocket motor, the burned gases of either engine being exhausted with the relative velocity V_∞.

As ζ ranges between 0 and 1 (we evidently exclude ζ negative, which would correspond to a propeller operating like a windmill), an actual propeller necessarily corresponds to

$$P_{th} - \left(1 + \frac{m_w}{m_t}\right) 2E_{k_\infty} > 0 \tag{4-16}$$

Finally let us recall that, in Eq. 3-5, V_f represents a *simple average* of the axial exhaust velocity in one or several sections A_f, which very frequently coincides with the actual exit section or sections A_e, where this simple average is not necessarily equal to $\sqrt{2E_{k_f}}$.

Also, if the change of state of the internal flow between the sections A_e and A_f is accomplished under quasi-steady conditions, almost adiabatically, with constant chemical composition, and with a very small change $(p_e - p_f)$ of pressure, we have

$$(h_g)_e - (h_g)_f \cong E_{k_f} - E_{k_e} \cong \left(\frac{\gamma_g - 1}{\gamma_g}\right) \frac{p_e - p_f}{(p_g)_e} \tag{4-17}$$

where γ_g represents the ratio of specific heats of the exhaust gases. We see that if $p_e - p_f$ is sufficiently small, it is admissible, to a first approximation, to let the states f and e and the corresponding section A_f and A_e coincide, even more so in Eq. 4-11 and 4-12 than in Eq. 3-5. For greater rigor, we forego this simplification here.

CHAPTER 2. GENERAL FORMULAS

B,5. Jet Engines with a Single Discharged Flow. We first suppose (Art. 5 to 11) that there is only a single discharged flow, i.e. that the exhaust nozzle, even if it is composed of distinct parts, provides for the discharge of a flow in which the various portions are characterized by the same state f.

Furthermore, for simplification, we will agree hereafter to identify the exhaust axial velocity (simple average) V_f with the quadratic average $\sqrt{2E_{k_f}}$. Then:

$$V_f \cong \sqrt{2E_{k_f}}$$

Let us place, with respect to the various supplies of active substances,

$$\delta = \frac{m_a}{m_t}, \quad \nu = \frac{m_w}{m_t}$$

δ representing the inverse of the usual fuel-air ratio.

Let us also place

$$\lambda_\infty^2 = \frac{E_{k_\infty}}{\Phi_0} = \frac{V_\infty^2}{2\Phi_0}, \quad \lambda_f^2 = \frac{E_{k_f}}{\Phi_0} = \frac{V_f^2}{2\Phi_0}$$

with $\Phi_0 = Q_0 + q_0$ while noting that q_0 is generally neglected for mixtures of air and gasoline or air and kerosene so that the specific heating value Q_0 is then substituted for the (specific) energetic value Φ_0. Consider also that our "thermal efficiency" η_{th} is then identical with the "combustion efficiency" in the usual theory of internal combustion engines, with the addition of the term $(1 + \nu)E_{k_\infty}$ which corresponds to the absolute kinetic energy of the active substances transported aboard at the velocity $V_\infty = \sqrt{2E_{k_\infty}}$.

Let us refer back, on the other hand, to the definition (Eq. 4-14) of the thermal efficiency η^0_{th} at the stationary condition and to the definition (Eq. 4-15) of the coefficient ζ of transmission of mechanical power to the propeller. The thermal efficiency η_{th} *in flight* is expressible then, from Eq. 4-7, 4-3, and 4-14, by:

$$\eta_{th} = \frac{\eta^0_{th} + (1 + \nu)\lambda^2_\infty}{1 + (1 + \nu)\lambda^2_\infty} \tag{5-1}$$

Then, from Eq. 4-11 and 4-10, *the square of the specific exhaust velocity* λ^2_f is expressible as

$$\lambda^2_f = \lambda^2_\infty + (1 + \zeta)\frac{\eta^0_{th} - (1 + \nu)\lambda^2_\infty}{1 + \nu + \delta} \tag{5-2}$$

Hence from Eq. 4-6, 3-5, 4-10, and 5-2, the *over-all efficiency* η_0 is

$$\eta_0 = \zeta\eta_p \frac{\eta^0_{th} - (1 + \nu)\lambda^2_\infty}{1 + (1 + \nu)\lambda^2_\infty} + \frac{2\lambda^2_\infty}{1 + (1 + \nu)\lambda^2_\infty}$$
$$\left[(1 + \nu + \delta)\sqrt{1 + (1 - \zeta)\frac{\eta^0_{th} - (1 + \nu)\lambda^2_\infty}{(1 + \nu + \delta)\lambda^2_\infty}} - \delta\right] \tag{5-3}$$

Finally, from the definition of Eq. 4-8, the *propulsive efficiency* is:

$$\eta_{pr} = \zeta\eta_p \frac{\eta^0_{th} - (1 + \nu)\lambda^2_\infty}{\eta^0_{th} + (1 + \nu)\lambda^2_\infty} + \frac{2\lambda^2_\infty}{\eta^0_{th} + (1 + \nu)\lambda^2_\infty}$$
$$\left[(1 + \nu + \delta)\sqrt{1 + (1 - \zeta)\frac{\eta^0_{th} - (1 + \nu)\lambda^2_\infty}{(1 + \nu + \delta)\lambda^2_\infty}} - \delta\right] \tag{5-4}$$

The obvious advantage that is offered in the use of η^0_{th}, in preference to η_{th} in the above equations, results from the fact that η^0_{th} varies much less than η_{th} with λ^2_∞ for a given set of conditions of *internal operation* $i \rightarrow f$ of the engine. If we regard η^0_{th} as fixed, for example, then Eq. 5-1 immediately expresses the variation of η_{th} with λ^2_∞; in particular, whatever η^0_{th} may be, η_{th} always tends toward 1 when λ_∞, and therefore the flight speed V_∞, increases indefinitely.

B,6. Hypervelocities. The necessary condition (Eq. 4-16) for the propeller to be able to effectively receive mechanical power, requires that,