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The Liquid-Propellant Rocket Motor — Past, Present, and Future

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At intervals in the history of transportation, certain major inventions have appeared which increased the previous speed of travel by an almost staggering amount within a few years after their introduction. The most obvious past examples of this are the locomotive and the airplane, each of which multiplied the previously existing speed record by a factor of about five during the first four decades of its development. In our present generation, a new vehicle has now appeared which promises a still more incredible increase in transportation velocity, as revolutionary in its possibilities and as challenging to the imagination as the locomotive and the airplane were in past generations. I refer, of course, to the modern high-power rocket.

The modern rocket, like the locomotive and airplane before it, is based on the use of a fundamentally new, unconventional light, and powerful engine. This engine is the liquid-propellant rocket motor, a device so light and powerful that it produces over half a million horsepower of useful propulsive power for an engine weight of less than a thousand pounds. It is the purpose of the present paper to give a brief survey of the history of this revolutionary new engine, the development of its operating principles, and the trends in its design in the present and future.

The original source of the liquid propellant rocket idea, like many other inventions, is still quite uncertain; but there is some evidence that the earliest practical working motor of this type was constructed about 1895 by Pedro E. Paulet, a young engineer of Peru, South America. (See Fig. 1) Paulet did not publish an account of his work until 1927, in an obscure news article in the Lima, Peru, "El Commercio", so that the validity of his claim may be rather doubtful; but it is interesting nevertheless, to quote Paulet's description of his motor, as abstracted in A. B. Scherschevsky's book "The Rocket for Transport and Flight".

The following is a free translation:

"I made my final tests with panklastite, which the discoverer of melinite, Turpin, had just introduced. As a structural material, I used vanadium steel, at that time a new metal. The rocket chamber, which was conical inside, was 10 cm high and had a 10 cm mouth diameter (thus the nozzle angle was 52°). As to the propellants: nitrogen, peroxide, and gasoline were fed into the combustion space in the upper part of the chamber through opposed injector ports with simple check valves. The ignition was electrical by means of a spark gap half way up the rocket chamber, similar to the spark plug of present day internal combustion engines. In the first tests the rocket chamber was provided with long flexible tubes in the form of external rings connected to the feed lines. The rocket motor was suspended so as to slide on two vertical wires, and the thrust was measured by a dynamometer. The tests were very satisfactory. The rocket motor weighed 2.5 Kg and gave a thrust of about 90 Kg at 300 explosions per minute. The tests showed that the rocket could operate for an hour without suffering any appreciable deformation. Because of the danger involved in tests with powerful explosives and owing to other personal reasons, the tests were terminated in 1897".

Paulet's device appears to have been the earliest example of a so-called bi-propellant rocket motor, in which the oxidizer and the hydrocarbon fuel are in separate tanks and are mixed only in the combustion chamber. His use of nitrogen peroxide as oxidizer also foreshadowed certain modern propellants such as nitric acid, and the set-up of his test stand was quite similar to types used in later years. The intermittent fuel injection which he employed has not been commonly used in more recent motors, which almost invariably employ a constant-pressure combustion cycle.

The next definite proposal for a liquid-propellant rocket motor was in 1903, by K. E. Ziolkowski of St. Petersburg, Russia. Ziolkowski, a mathematics professor, published an article in the St. Petersburg "Scientific Observer", dealing with the possibility of a rocket space ship for exploring outer space. While his paper dealt mainly with the mathematical conditions for space flight (and was in fact the first really scientific attempt to analyze this tantalizing problem) he also made several specific proposals for operating such a rocket, including the use of liquid oxygen and liquid hydrogen as propellants in a constant-pressure combustion chamber with a De Laval divergent expansion nozzle. This propellant combination has since received very extensive study, and still remains one of the most powerful fuels so far proposed, though it has not yet been used in practical rocket work owing to various practical difficulties.

In 1913, Robert Esnault-Pelterie, an aviation pioneer who invented the familiar "joystick" control for airplanes, read a paper before the French Academy of Sciences on the subject of rocket space flight in which he envisaged the use of liquid rocket fuels, though still in very general terms; and at about the same time Hermann Oberth, a Rumanian professor, began his early work on rocket theory, while Dr. Robert H. Goddard, of Worcester, Mass., likewise began work on the problem at Clark University.

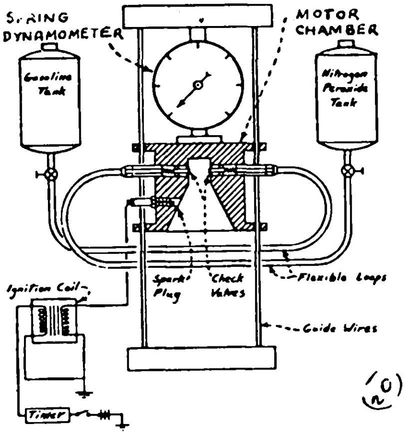
Both Oberth and Goddard were destined to play fundamental roles in the development of the new art of rocketry, which as a result of their efforts began to move out of the mathematics books and into the workshop and the test field. This new period began about 1915, when Dr. Goddard began his classical experiments on the smokeless powder rocket, which were published in 1919 in his famous Smithsonian Institution paper, "A Method of Reaching Extreme Altitudes." Unlike the early work of Ziolkowsky and others, this paper aroused much public comment and began a sort of ground-swell of interest in the subject among technical men which gradually spread further and further, in spite of all ridicule and indifference.

In his 1919 paper and in a 1914 patent, Goddard mentioned the possibility of using liquid propellants in rocket motors; and in 1920 he began active experimental work on the idea. By 1923 he had already developed a practical liquid oxygen and gasoline rocket motor, which was fed with fuel by a simple pump feed system, and on March 16, 1926 he succeeded in making a short flight of a small liquid fuel projectile. The general arrangement of this historic machine is shown in Fig. 2, which is reproduced from G. Edward Pendray's book "The Coming Age of Rocket Power". It will be noted that the motor of this rocket had a liner of alundum cement to protect the chamber from the intense heat. This method of construction is attractively simple, but has been largely abandoned in more recent designs, since it is necessary to resort to an inefficient rich fuel mixture to keep the chamber temperature down to a manageable figure.

Meanwhile, a new line of research began in Germany, following the publication in 1923 of Hermann Oberth's "By Rocket to Planetary Space". This remarkable book developed the general mathematical theory of rocket projectiles in much greater detail than any previous author, and also presented a number of strikingly advanced ideas on rocket motor construction. Oberth proposed to use dilute ethyl alcohol and liquid oxygen as propellants, a combination which has since proven highly successful. His proposed rocket motor, shown in Fig. 3, was probably the earliest scheme involving the important **regeneration** principle, in which one of the propellants is circulated around the combustion chamber in a cooling jacket, thus cooling the motor and simultaneously preheating the fuel prior to its injection into the motor. He also proposed to spray part of the fuel along the inner walls of the motor to provide a boundary layer of liquid and vapor for insulating the walls from the intense heat of the flame — an early example of another important cooling method sometimes called **film cooling.** His motor project also involved the use of multiple mixing parts in the motor head, each provided with concentric injection nozzles. Various variations of this layout have since been frequently used in large rocket motors.

Oberth's treatise led to many other speculative books on the subject of rocket propulsion and its possibilities, but it was not until 1930 that any really practical results began to appear. In that year a rather crude liquid-propellant motor was constructed in Germany by Max Valier for use in his rocket automobile experiments, but his work came to a rapid and tragic end when he was killed by an explosion during preliminary tests. Meanwhile, a group known as the "Society for Space Travel", which had been originally organized in 1927 by a group of amateur rocket enthusiasts, began experiments in Reineckendorf, a suburb of Berlin. During the years 1930 to 1933 they carried out a large number of tests on various simple types of liquid-propellant rocket motors and rockets, mainly under the technical direction of Rudolf Nebel and Klaus Riedel. Oberth and Johannes Winkler were also prominent in the earlier stages of this work.

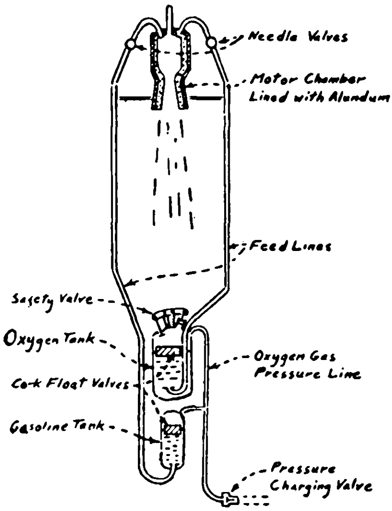
The early motors of this pioneer group were rather crude affairs, but gave good enough results for short test flights. The propellants used were liquid oxygen and alcohol or liquid oxygen and gasoline. The motor construction (See Fig. 4), was characterized by the use of a rather thin aluminum chamber and nozzle immersed in a small water tank for cooling, and also by "backshot" fuel injection in which the propellants were fed in close to the nozzle and sprayed toward the motor head, with a view toward improved turbulence and fuel mixing. Neither of these features has survived in modern motors, which usually require some form of forced-circulation cooling jacket to deal with the high temperatures attained with high-efficiency combustion, and which invariably have the propellant injection ports near the head of the motor, since this is both simpler and more efficient than the older "backshot" plan. However, these early motors of 1931 represented a first step toward the modern jacketed motor.

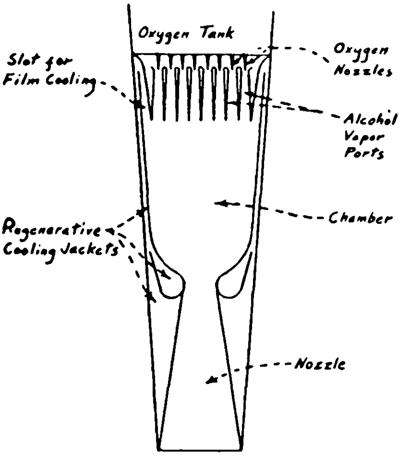


PAULET MOTOR—1895
(SCHEMATIC RECONSTRUCTION FROM DESCRIPTION)

FIG. 1

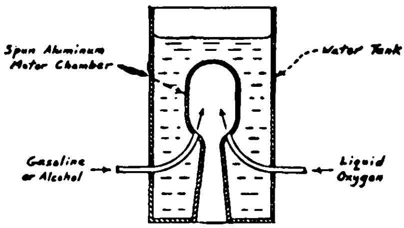
FIG. 2 GODDARD MOTOR—1926





OBERTH MOTOR DESIGN—1923

FIG. 3



GERMAN "REPULSOR" MOTOR—1931
(SCHEMATIC)

FIG. 4

About 1933, there was a "shake-up" in the affairs of the society, and its experimental work was taken over by the German Army, and became a "top secret" military development. The tests continued at the old test field in Reineckendorf, and were rapidly expanded. A young engineer named Werner Von Braun, who had assisted in the work prior to 1933, soon became prominent through his ability and energy and was put in charge of the experiments. The experimental technique was now greatly improved, and progress was much faster. Experiments were made during 1934 and 1935 with the so-called A-1 and A-2 rockets. These were the first rockets with regenerative motors, cooled by their own propellants. In the case of the A-1 and A-2 motors, this was accomplished by immersing the whole rocket chamber in one of the propellant tanks — a scheme which had already been unsuccessfully tried in one of the early 1931 motors of the "Society for Space Travel". The results attained have not yet been fully revealed, but it appears that they were not fully satisfactory either as to efficiency or reliability. It became evident that a much more elaborate research project would be needed to produce a satisfactory large rocket, and in 1936 a huge new research center was set up at Peenemunde on the Baltic Sea. During the next two years a greatly accelerated development program was energetically pushed, which engaged a full-time staff of several hundred men. The motor development section by 1938 had producted a satisfactory regenerative motor of over 2000 pounds thrust which was used to power the A-3 experimental rocket, (a reduced-scale prototype of the later A-4, commonly known as the V-2).

The preliminary calculations for the monster A-4 rocket date back as early as 1938, and the detailed design work on the project was begun late in 1940, the first successful flights being made in the fall of 1942. It is probable that the motor development was mostly carried out in 1941. Even at the present time, the boldness of the project is startling, and it ranks second only to the American atomic bomb as the most revolutionary military development of World War 2. It was necessary to increase the size of the previous A-3 motor, already by far the largest in the world, by a factor of about twenty to produce this gigantic new motor for the still more gigantic new rocket. The problems involved in such a project can perhaps only be fully appreciated by those who have carried out similar rocket developments under the insistent pressure of wartime conditions. It was necessary, for example, to work out fuel injectors that would properly vaporize and mix no less than 287 pounds of propellants a second, to burn them efficiently in a combustion space only three feet long and three feet in diameter, and to keep the chamber and nozzle cool in spite of a gas blast scouring the walls at a velocity of over a mile a second and a temperature of over 3000°F. — and all this had to be done in a contrivance weighing only a few hundred pounds.

The end result of this remarkable feat of engineering development is shown in Fig. 5. Perhaps the most striking feature of the A-4 motor is its very lack of striking features. It is remarkably simple and logical in its general lay-alcohol and liquid oxygen, an "old reliable" fuel combination which has out. The propellants used are 75% ethyl been used for years by other rocket experimenters. The alcohol is used to cool the motor chamber, which is of the regenerative type now in almost universal use. The alcohol coolant enters the jacket through a manifold ring around the nozzle, fed by several pipes from the alcohol pump. From there it circulates through a narrow clearance space between the jacket and the inner liner of the motor, eventually reaching a central port in the motor head which is kept closed by a large poppet valve until the motor is ready to fire. On opening the valve, the alcohol flows out into a distributing passage in the top of the head, which leads the fuel to the exterior of eighteen cup-shaped mixing chambers attached to the internal motor head; it is then injected into the interior of each cup through a series of special nozzles, producing a finely atomized spray. The liquid oxygen is fed directly from the oxygen pump to the injectors through several manifold pipes, and is injected through a dome-shaped perforated insert in the top of each mixer cup, somewhat like a large salt-shaker. The two propellants are mixed in the cups and start to burn immediately outside the cup mouths.

An interesting feature of the V-2 motor is the use of an auxiliary cooling system which injects part of the alcohol into the interior of the combustion chamber through several rings of small holes around the periphery of the motor. The fuel is injected at low velocity and spreads out on the inside wall of the chamber, forming a protective film which reduces the heat transfer from the hot gases. This expedient is an example of the film cooling scheme mentioned earlier; in the form shown here, the idea was first carried out by Dr. R. H. Goddard in some of his early experiments of about 1930, and is described in one of his patents dated 1935. Goddard succeeded in developing the film-cooling idea to considerable perfection, and was eventually able to operate motors as large as one ton thrust by this method without any cooling jackets. However, with this plan the fuel used for cooling cannot be completely burned (unlike the jacket coolant of the true regenerative motor) and this results in a reduction in efficiency; also, the size and location of the cooling holes require much laborious experimenting. Most present-day designers regard the method as a useful auxiliary to the use of cooling jackets, which can be used to clear up local overheating at certain critical points in the motor, but it is not widely used as the main cooling system.

It will be noted that the outer jacket of the V-2 motor has a number of v-shaped corrugations at intervals, which serve the purpose of expansion joints and take up the strains caused by the thermal expansion of the hot liner tube. This is apparently a minor detail, yet much difficulty has frequently occured with certain types of rocket motor whose design had not provided for such thermal strains, and it appears that even the designers of the A-4 had considerable trouble at first with leaks and distortions caused by them. The occurence of such strains in a motor of all-welded construction is not always at once apparent and often results in mysterious disturbances of fuel circulation and cooling which are difficult to trace. It might be said in passing that it is just such insignificant details as this that often determine the success or failure of a motor design, and make rocket engineering as much a fine art as a science. The simplicity of a rocket motor is often highly deceptive; it is a machine in which every part is worked up to its ultimate capacity, and in which even the smallest screw or fuel port must be carefully thought out and thoroughly tested.

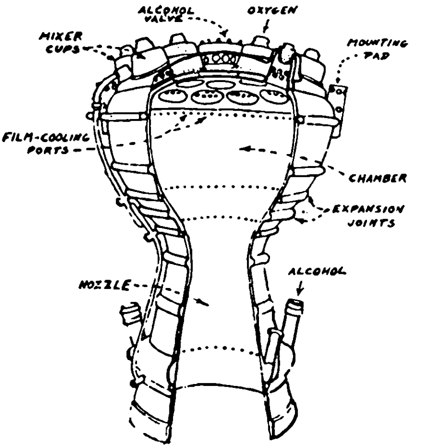
The mechanical construction of the A-4 motor represents the most extreme development so far made of welded sheet-metal construction in rocket motors. Except for a few of the injector parts, the entire motor is welded together from numerous pressings of low-alloy steel, most of them only about one-eighth inch thick. The roughly globular form of the combustion chamber, to gain additional strength and stiffness, is noteworthy. The thrust of the motor, amounting as much as 64,000 pounds, is transmitted to the rocket through only four bolts about 1½ inches in diameter, attached to the outside of the motor by ball and socket joints.

The A-4 motor undoubtedly represents the highest development so far attained in its field; but there have been many other interesting rocket developments involving new principles. Among these was the pioneer American work on rockets conducted by the American Rocket Society. This was begun in 1932, on the basis of some fragmentary information secured from the German "Society for Space Travel", and continued for nearly a decade. The earliest motors built by the Society were water-cooled, similar in principle to the German ones; an example is the multi-nozzle motor built in 1934 by John Shesta, now Research Director of Reaction Motors, Inc. (See Fig. 6). Another early design of 1934, due to G. Edward Pendray and Bernard Smith, is shown in Fig. 6; it represents an early attempt to apply the regeneration principle to rocket motors by immersing the motor in one of the propellant tanks. In 1935 a series of proving stand tests were made on another type of motor (See Fig. 7), involving the first application of what may be called "heat sponge" cooling. The motor was constructed of a series of thick blocks of aluminum alloy having a high thermal conductivity; these blocks were intended to "soak up" the heat transmitted from the combustion gases and thus prevent the motor from overheating seriously during a short test run of a few seconds. This idea was only partially successful at the time, but was later revised and improved in experiments at the California Institute of Technology, and has been applied to various types of short-burning solid and liquid-fuel rockets with fairly satisfactory results.

The regenerative cooling jacket made its first appearance in practice in a small motor constructed in 1933 by Harry W. Bull of Syracuse, N. Y., a member of the American Rocket Society (See Fig. 8). Bull's motor was fired on gasoline and oxygen gas, and gave very promising results. Bull also appears to have been the first to experiment with a monopropellant liquid fuel motor, on which full details have unfortunately never appeared. In this type of motor, which was later extensively developed in Germany for glide bomb propulsion and jet-assisted aircraft take-offs, the fule and oxidizer are combined in a single propellant, which is so arranged as to burn only in the motor combustion chamber. The obvious danger in this plan is the likelihood of a flashback from the motor to the main fuel tank. The dangerous nature of monopropellants was brought home to Bull when he was seriously injured in an explosion of his motor, and he soon afterwards dropped his experiments. The scheme has been widely worked on by others in later years, but has not yet attained as high a degree of safety, efficiency, or reliability as the bi-propellant type, in spite of the attractive simplicity of the idea.

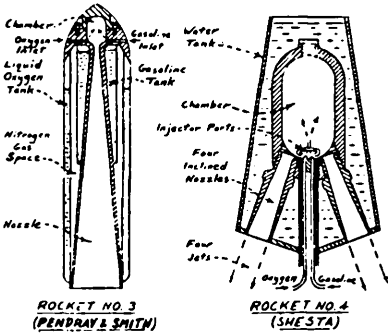
In the same year (1933) in which Bull constructed his first regenerative motor, Eugen Saenger, an Austrian areonautical engineer, began experiments in Vienna with a similar device. Saenger was one of the first to appreciate fully the advantages of a high chamber pressure in improving motor efficiency and jet velocity, and he claimed that his small experimental motor, operated on Diesel fuel oil and oxygen gas at about 1500 PSI, attained the remarkable jet velocity of 9800 feet per second.

Saenger developed his ideas considerably farther during the war years, and developed a very efficient motor of one ton thrust. This design was used in



A-4 MOTOR — 1942

FIG. 5



TWO A R S. ROCKET MOTORS - 1934

FIG. 6

experiments with a novel type of rocket propellant consisting of aluminum powder suspended in fuel oil, the oxidizer being liquid oxygen. This combination seems to have been first suggested by W. Zimmerman of the American Rocket Society, about 1937; its purpose, of course, was to increase the jet velocity by utilizing the intense heat of burning aluminum. Saenger, however, apparently found no striking improvement over conventional fuels. He made plans for much larger types of rocket motor and even began construction of an enormous motor of 100 tons thrust; however, in the latter part of the war his work was shifted to ram-jet research, probably because of some shift in Army policy on rocket development.

Another early regenerative motor was constructed by the present writer in 1938 (Fig. 9). This was the first American design to apply regenerative cooling to all parts of the motor. (Bull's earlier design had used regeneration for nozzle cooling only). The motor, first tested in 1938, proved very reliable and efficient, and after further tests in 1941 a small company, Reaction Motors, Inc. was formed by several members of the American Rocket Society to develop motors of this general type for military applications. The founders of the company, (who were Lovell Lawrence, Jr., John Shesta, H. F. Pierce and the writer), were not only the infant concern's sole officials, but also its sole employees and it may be added, its sole investors. Under a series of small research contracts from the U. S. Navy, numerous motors of rapidly increasing size were constructed. The first models (up to June 1942) were very similar to the original design of 1938, developing only 100 pounds thrust.

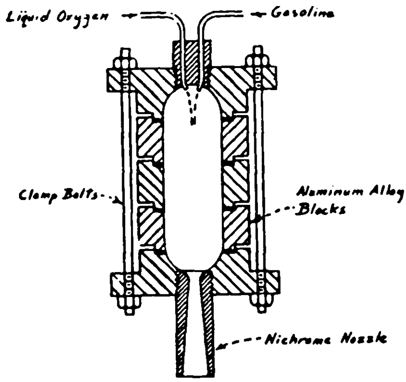
However, by November 1942, successful runs were being made at over 1000 pounds thrust, and this was increased to 3400 pounds by May, 1943, less than a year after the test work, at 100 pounds thrust. This work as accomplished by a total personnel of less than twenty men, and it may be observed without undue boasting that the corresponding German development work in 1934-1938, which was only disclosed after the end of the war, had required four years work by several dozen men and must have cost at least a million dollars, resulting in a motor having the same thrust as the American design, but a much poorer jet velocity. This seems a convincing proof that the lack of progress of American rocket research had not been due to any inherent lack of technical skill, but was caused solely by the prevailing attitude of ridicule and indifference to the whole subject which blocked almost all advance for at least ten years. It is true that there has been a very drastic change for the better in this respect in the last year or so, and the technical public now displays an almost feverish interest in rocket developments. The author, having observed the early struggles of rocket pioneers for over a decade, will merely remark that it is about time the public should run a fever.

Much of the most recent work in this country on liquid-propellant rockets is still confidential. However, some data has recently been released on a rocket engine built by Reaction Motors, Inc. for high-speed aircraft propulsion (See Fig. 10.)* This engine is the first example of a multiple-cylinder rocket engine, a new principle with important future possibilities. The three-ton thrust of the motor is divided between four individual motor units or cylinders mounted in a close array. Each motor is provided with its own valves and ignitor. Thus, by operating from one to four cylinders and simultaneously adjusting the fuel flow by throttle valves, any thrust from less than 1000 pounds to over 6000 pounds can be attained at high motor efficiency. This flexibility is of great importance in certain applications of rocket engines. The present type is intended as a power plant for a recently built transonic research airplane, which will make its first flights in the very near future. The motor will develop 12000 brake horsepower at the speed of sound (750 MPH) and rated 6000 pound thrust, on an engine weight of 210 pounds. The propellants used are liquid oxygen and alcohol.

Numerous other rocket motor projects are now being carried out by various American firms and government agencies. Special mention should be made of the work carried out by the California Institute of Technology and by the Aerojet Engineering Corporation of Pasadena, founded by some of the CIT members in 1942. The best-known example of their work is the GALCIT solid-fuel "jato" unit, now receiving extensive use in assisted aircraft take-off, but they have also done much research on both bi-propellant and mono-propellant liquid-fuel rocket motors, especially in the field of "jato" units. The GALCIT and Aerojet groups have made extensive use of propellants based on nitric acid as the oxidizer, which has advantages over oxygen for some applications because it can be stored in tanks for much longer periods and because it can be made self-igniting when combined with certain hydrocarbons. Similar fuels have also been used by Reaction Motors, Inc., and by many recent German experimenters. The GALCIT and Aerojet liquid-fuel motors at first used a simple "heat-sponge" method of cooling, but later models have been of the regenerative type in the interests of better efficiency and longer burning time.

Mention should also be made of the hydrogen peroxide motors developed in Germany by the Walter Works of Kiel, beginning about 1940. Their earlier designs were monopropellant motors, in which the energy was developed merely by the decomposition of the peroxide

*Described and illustrated in the Dec. 1946 A. R. S. Journal.



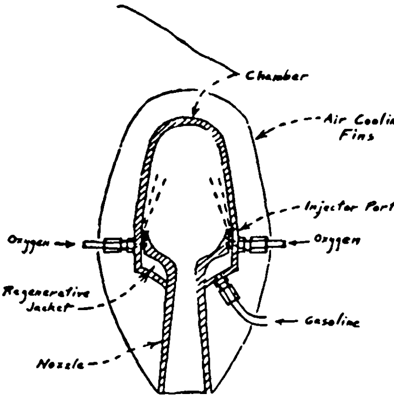
A.R.S. "HEAT-SPONGE" MOTOR—1935

FIG. 7

by a suitable catalyst; similar arrangements were later used to supply steam for operating the fuel pumps in the A-4 rocket and the Messerschmidt rocket interceptor. The main power plant of the Messerschmidt was a bi-propellant motor, however, in which the peroxide was burned with a mixture of alcohol and hydrazine hydrate in order to obtain greater energy and exhaust velocity. This fuel combination, however, proved quite tricky and dangerous in actual use, as well as expensive, and the idea has not been widely used by other designers. However, hydrogen peroxide as a monopropellant for driving auxiliaries such as pumps is still a very useful material.

The foregoing historical outline serves to give some general idea of how the rocket motbr has developed to its present state, though many interesting items have been omitted for reasons of space or security. It need hardly be added that the whole story to date is only an introductory chapter to the full history of the rocket motor, which it remains for the future to write. It would indeed be hazardous to try to predict from this early chapter how the whole book will turn out; but nevertheless it is interesting to attempt a brief survey of the present trends in rocket motor research, and to make a few guesses about its future.

One obvious difficulty in the way of future progress is the problem of rocket motor cooling. This has always been a persistent "headache" from rocketry's earliest days to the present, for as fast as cooling methods have improved the demands upon them have constantly increased. To get improved combustion and a more efficient expansion cycle, the chamber pressures are gradually being raised from the present-day figure of 200 or 300 psi, with a consequent increase in mass velocity through the nozzle as well as some increase in gas temperature. It is not unlikely that pres-



BULL REGENERATIVE MOTOR
(1933)

FIG. 8

sures of 500 to 1000 psi may soon be in use. The chamber gas temperature in present-day motors is in excess of 6000° R., with the thermometer steadily going up. The rate of heat transfer at the nozzle is already about 2,000,000 BTU per square foot per hour, and motor designers may soon be called upon to deal with two or three times this amount. It is evident that simple plans such as "heat Sponge" blocks or heat-resistant chamber linings cannot possibly deal with such conditions, except for special conditions where the firing time is only a few seconds. The problem will be further increased if new high-energy fuels such as liquid hydrogen are resorted to in order to get higher jet velocity — a development which is probably quite close. Some form of regenerative cooling must obviously be used. Experience indicates that it is not so much the overall heat transfer as the presence of local "hot spots" that cause most cooling troubles, and attention must be focussed on eliminating these. Such expedients such as extended-surface cooling ribs, improved coolant circulation, special materials combining high thermal conductivity and strength at high temperatures, and film cooling of danger zones, may improve matters considerably without drastic redesign. To reach still higher ratings, it may be necessary to use closed-cycle cooling jackets with special coolant fluids circulated by pump into heat exchangers where the heat absorbed can be more conveniently taken up by the propellants. Much experiment will be needed to clear up these difficult questions.

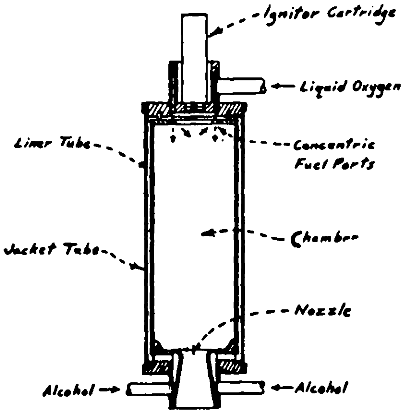
The structural problems of rocket motors also became more difficult as motor sizes and pressures increase and as weight reduction becomes more important. The modern tendency is to avoid the use of heavy cast parts, bolts, flanges, and flat surfaces and to resort to more refined design using relatively thin welded sheet-metal parts with dished surfaces and ribs to gain rigidity and strength. Extremely large motors such as the A-4 offer special difficulty because of the relatively thick wall sections, which produce a high temperature gradient in the wall. This results in increased cooling difficulties, as well as complicated thermal stress effects, which in some cases are aggrevated by various elastic instability conditions, as in the case of a nozzle externally cooled by a high-pressure coolant. Some materials research is called for here; at the present time there exist materials of high conductivity but poor high-temperature strength (such as copper, or aluminum alloys) and also materials of poor conductivity but high strength (such as stainless steel) but there seems to be no convenient material combining the two qualities.

Another problem with large motors is the fact that, generally speaking, the chamber weight for a given working pressure is directly proportional to the thrust (assuming a constant ratio of chamber volume to propellant feed rate) while the nozzle weight varies as the 3/2 power of the thrust. (This can easily be shown by elementary dimensional analysis). Consequently a very large motor will have an extremely large and heavy nozzle, representing most of the motor weight. This dilemma can theoretically be solved by using a multi-nozzle motor with many small nozzles instead of one large one, as proposed by Oberth more than twenty years ago; but in practice it appears that this leads to great difficulty with the coolant circulation. Another plan is to use a multiple motor, with several separate cylinders, as in the RMI 1500N4C motor (Fig. 10). This results in considerable weight saving in large sizes, and provides very flexible control of the thrust; it is also well adapted to rocket steering by swivelling some of the motor jets, or by differentially throttling them. There is also an increase in convenience in manufacturing and testing the motors; however, there is some increase in complication and motor bulk, as well as a control problem in synchronizing the thrust of the separate cylinders. This method seems one of the most logical approaches if very large rockets, exceeding the A-4, are to be built.

The problem of fuel injection also will require much future work. Modern practice is to use much more subdivision of the injector ports than was formerly customary; early motors had only two injection orifices, one for each propellant, but it is not uncommon for modern types to have over a hundred, and the A-4 motor has a total of over three thousand. Determination of proper shapes, sizes, and positions for these ports is a very difficult and tedious task, and there is still much disagreement as to the best solution. Many motors utilize fuel and oxidizer jets arranged in pairs, the two jets being aimed so as to strike one another at an angle to produce mixing; other motors, such as the A-4, produce a diffuse spray of each propellant, the spray particles being mixed at random; still others, such as the Walter motor, use concentric jets in with each oxidizer jet is surrounded by an annular jet of fuel. (See Fig. 11). All motors so far constructed employ "solid" fuel injection; but it seems possible that a form of "air" injection may eventually be used, in which part of the fuel is used to produce a preliminary gas stream for vaporizing the remaining propellant.

Another injection difficulty in rocket motors in the low pressure drop in the injection nozzles, which generally does not exceed 100 psi, owing to the excessive tank or pump weight needed if high pressure is used. If the motor is throttled down to low thrust, the injection conditions become particularly bad due to the very low injector drop, and it will no doubt be necessary to resort to various variable-area injectors in certain motors where a widely varying thrust is required. The Walter motor involved the use of a distributor valve which cut in varying numbers of injector orifices according to the required thrust; many other devices are also possible, and will no doubt be experimented with in the future. There is also room for much research on the interactions between injector design and the distribution of combustion zones and heated areas in the motor.

Finally, something should be said regarding modern propellant research and the possible output of future rocket motors. The propellants now in use with liquid oxygen, i.e.: ethyl alcohol and gasoline, are essentially the same as those of fifteen years ago, but the jet velocity has been raised from values of about 400 fps at that time to about 7000 fps at present, an improvement due primarily to better fuel injection and better cooling methods, as well as to some increase in chamber pressure. By using higher chamber pressures than the present ones, it appears that velocities of about 8000 or 9000 fps should be attainable. To get appreciably beyond this, special propellants will have to be used, possibly involving the high combustion heat produced by pure hydrogen. A number of combinations of this sort have been proposed which theoretically permit exhaust velocities as high as 12,000 fps, even after allowing for the various losses involved in combustion and expansion; and some small-scale experiments have indicated that such exhaust speeds can actually be attained in practice under favorable conditions. Because of considerations of expense and various difficulties in preparing and handling them, these "super-hot" propellants are only in the laboratory stage, but the future will un-



WYLD MOTOR-1938

FIG. 9

doubtedly see them in extended use wherever a very high degree of performance is required from a rocket motor. So far as thrust is concerned, there appears to be no practical limit in sight if multiple-cylinder units are used, although it does not seem feasible to construct single cylinders having much over 50 to 100 tons thrust—that is, 2 to 4 times as large as the A-4 motor—because of the rapidly increasing nozzle weight and cooling difficulties.

It is still questionable whether a jet velocity of 12,000 fps will permit the construction of practical transoceanic rockets or of some true space ships attaining an orbital velocity or even a full escape velocity. To be really sure of achieving such feats, some way must be found to utilize nuclear energy in a practical rocket motor. If this could be done with even reasonably good efficiency, all the books about jet velocity and mass ratio could be thrown out the window, for the energy available for propulsion would be so great as to render a trip to the moon no more difficult than a ferry boat ride from New York to Hoboken. However, it is as fruitless a task to try to imagine what an atomic energy rocket motor will actually look like as it would be to make up working drawings for the "Queen Elizabeth" from the plans of a toy steamboat. Obviously, the job cannot be done simply by blowing steam or hydrogen through a high-energy nuclear pile to develop hot gas, for we then have all the present difficulties with chamber temperatures in an exaggerated form, and no real improvement in jet velocity. On the other hand, we certainly cannot expect to let off a charge of plutonium or U-234 in a rocket chamber, unless we use a large hollow iron asteroid with a hole in one end; for the critical-mass effect makes it necessary to have either a tremendous explosion or no results at all. It remains for us to see whether some intermediate

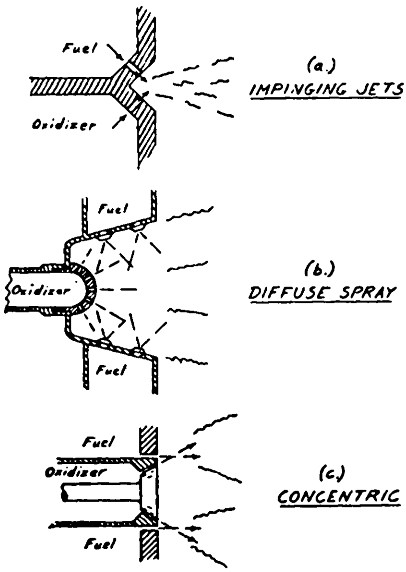


FIG. 11 TYPES OF INJECTOR

solution will eventually be discovered. The great recent progress in rocketry and in nuclear energy lead one to think that the technical difficulties involved will certainly be solved, perhaps within one generation; but this does not remove the major obstacle to space flight, which is not technical but political, so to speak. The development of an atomic energy motor powerful enough to drive a true space ship presupposes the existence of high power rockets capable of carrying atomic bombs to every corner of the earth, thus providing a means for our present technical civilization to commit suicide within a few minutes. A rocket engineer familiar with human history, if asked to predict whether rocketry will eventually lead to true space flight or whether it will lead to the complete extinguishment of all science, may well be forgiven some pessimism about the answer, and even some rather acute feelings about personal guilt. That the present-day rocket motor is only a tiny prelude to the monster engines of a future rocket age is absolutely certain; but (to borrow Thoreau's phrase of a century ago) whether we are to live like baboons or men is a little uncertain.

EMPLOYMENT OPPORTUNITIES

This space is available to both members and organizations at no cost for insertion of ads.

The Navy Department has announced vacanies in various professional classifications such as Physicists, Engineers (Civil, Aeronautical, Electronic, Mechanical, Ordnance, etc.), Architects, and Industrial Specialists. These positions are located in the Departmental service in Washington, D. C. and salaries range from $2645 to $10,000 depending on the formal education and specialized training of the applicants. Persons with research experience in the fields of guided missiles, jet propulsion, radioactivity, etc. are especially desired. Interested persons should contact the Special Projects Section, Navy Department, EXOS:AO:612, 18th and Constitution Avenue, N. W., Washington, D. C. for proper application forms and additional information.

The Climate of Mars

By JAMES R. RANDOLPH

Editor's Note: Major Randolph is the author of the prophetic article "What can we expect of Rockets" which was published in Army Ordnance in 1939. He is also the author of "The Occupation of Mars" which appeared in the March-April 1947 issue of Army Ordnance.

The surface of a planet receives heat from the Sun at a rate which varies inversely as the square of the distance. It reradiates this heat into space at a rate which depends on its surface temperature. An equilibrium temperature is reached when radiation emitted is just equal to the heat received from the Sun, and this temperature can be calculated when the various factors affecting it are known.

The calculation would be simplest if the surface were a black body, absorbing all the radiation which strikes it and emitting radiation according to a definite mathematical law. The actual calculation is complicated by the planet's albedo, or percentage of light reflected, and by its emissivity, which is usually less than one. Albedo and emissivity both vary in a complicated manner with temperature, surface conditions, and atmospheric conditions, so that they have to be determined experimentally for the particular conditions of the problem. They cannot be covered by simple mathematical laws.

The method of this article, therefore, is to compute significant black body temperatures of the earth and of Mars, and compare these calculated temperatures with observed temperatures in regions where the calculated temperatures are nearly the same, and in which albedo and emissivity appear to be the same. We can then assume that the difference between theoretical and actual temperatures is the same on both planets.

To derive an equation for this computation, let the heat received from the Sun per square foot of surface be

$$Q_s = 427.2 \frac{93^2}{R^2} (1-a) K$$

where R is the distance from the Sun in millions of miles (R = 93 for the earth or the moon), a is the albedo, and K is an exposure factor. It is unity for a surface normal to the Sun's rays It is sin 23½° for a horizontal surface at the Pole in midsummer. It is the cosine of the latitude for a horizontal surface anywhere else, at noon during the equinox. It is divided by ¶ to compute the mean daily temperature. It is ¼ for the average temperature of a whole planet, for the radiating area of a planet is the surface of a sphere; its absorbing area that of a disk of the same diameter.

Likewise we can write for the heat radiated the equation

$$Q_r = 0.172\, e \left(\frac{T}{100}\right)^4$$

where e is the emissivity, and T is the absolute temperature. This is 460° more than the Fahrenheit temperature.

Putting $Q_s = Q_r$ and solving for T we get:

$$T = 707 \left(\frac{93}{R}\right)^{\frac{1}{2}} K^{\frac{1}{4}} \frac{(1-a)^{\frac{1}{4}}}{e^{\frac{1}{4}}}$$

The equation is written in this form so that each of the variables may be considered separately. Thus, R is 67 for Venus, 93 for the earth, and 140 for Mars. $a = 0$ and $e = 1$ for a black body. K may have any of the values described above.

From this equation the following table has been computed.

Black Body Temperatures				Albedo 59%
	Earth	Mars	Venus	Venus
Equator, noon	707°	573°	834°	667°
Equator, average	530°	430°	625°	500°
Lat. 54° noon	609°	494°	719°	575°
Lat. 54° average	457°	371°	540°	432°
Pole, midsummer	559°	454°	660°	528°
Average for planet	500°	405°	589°	471°

The last column at the right gives temperatures in the top of the cloud layer which permanently surrounds Venus and hides all surface detail from our eyes. The temperatures at the planet's land or water surface would be increased by adiabatic compression, and so would depend on the altitude of the clouds. Until we know how high those clouds are we can have no idea of the planet's climate, except that it is probably very hot and very humid. Hence this article, from now on, will be confined to a comparison of the earth and Mars.

The table shows the theoretical average temperature at the earth's equator to be 530° absolute, or 70° F. The average temperature throughout the year at Panama is about 80°F, with a maximum recorded temperature of 90° and a minimum of 69°. This is a very close check on the computation. It applies to a narrow strip of land with big oceans on both sides, having a very large heat storage capacity. The fact that the average temperature is 10° higher and the maximum 20° higher than the theoretical can be explained by the heat trapping effect of the earth's atmosphere.

An important cause of this heat trapping effect is the condensation of moisture to form clouds when the temperature of the air goes below its dew-point, and the evaporation of these clouds when the Sun warms them. Outbound radiation is not merely checked by clouds we can see. It is also retarded to some extent by an almost invisible haze that forms at high altitudes on apparently clear nights. People sleeping in the open have often noticed a sudden chill in the air just before dawn, as the Sun's rays swing down into the stratosphere and clear away this lofty haze. Valleys are seen to fill with mist at dawn, where an hour earlier they had been clear.

As we go inland, into drier climates, the tendency to cloudy nights and clear days becomes more marked. The observed temperature rises higher above the theoretical average. It runs to greater and greater extremes. In the Sahara Desert, for example, rocks in the sunshine may become hot enough to fry eggs (they could theoretically have a temperature of 247°F.) Yet the following morning there may be ice in the water pitcher.

At Edmonton, Alberta (Lat. 54°) the table shows the theoretical average temperature at the equinox to be —3°F, and the maximum possible to be 157°F. But the average temperature at the March equinox is about 32°F, with a maximum of 70° and a minimum of —40°. This is the time of the Spring thaws in Edmonton, when snowfields are melting and river ice is going out. Even with winter just over the horizon it is a whole lot warmer than theory predicts, and the September equinox, with Summer behind it, is warmer still.

The summer temperature at the Poles of Mars is shown in the table as being 3° lower than the equinox temperature of Edmonton. Hence it would be a reasonable assumption that the actual temperatures in the polar regions in Summer are comparable to those of Edmonton in the Spring, and this appears to be the case. The polar regions of Mars, like those of earth, are covered with white caps in winter, having the appearance of snow. Then, when the sunlight warms them in the spring, they begin to shrink as snowfields do, and are surrounded while shrinking by dark blue bands that have the appearance of water. This water does not remain long. It drains away or evaporates, leaving desert behind.

There are no permanent bodies of water on Mars. Most of its surface is desert, of a reddish orange color. The rest is of a blue-green color, and could be vegetation. Because there is no water, but large areas of snow, it is unlikely that the dewpoint of the air ever rises much above 32°F. But it approaches that limit closely when the snowfields are melting, and so the heat valve for the whole planet opens and shuts at that temperature or below it.

Note now that the average temperature at the equator of Mars is theoretically —30°F, and the average for the planet is —55°F. These figures, coupled with the desert character of the planet, suggest that Arctic weather prevails over most of it most of the time, that temperatures below freezing are normal at night, even on the equator, and that **even on the equator it is probably necessary to wait for moisture from the melting snowfields to trap enough heat to make it possible for vegetation to grow.**

If we watched the earth from out in space we would see the green of spring creep up the latitudes with the sunshine, closely following the retreating snowfields. But on Mars we see this green begin at the edge of the snowfields and creep toward the equator and beyond. The canals appear. The blue-green areas darken. The deserts change tint from red toward yellow—a change that can be produced experimentally by thinly sprinkling their winter red with invisible bits of green.

This reversal of the progress of spring was one of the things which led Lowell to believe that the canals represented an actual irrigation network, and that vegetation had to wait for water before it could begin to grow. We can now see that the fact admits of a simpler and more natural explanation. It is not water alone for which the vegetation has to wait. It is the warmth that cannot be obtained without that moisture in the air; the clouds at night to shut in the heat received by day.

More than twenty years ago W. W. Coblentz of the Bureau of Standards made some radiometer measurements of surface temperatures on Mars, and these are in substantial agreement with the above estimates. He found the temperature in the polar regions to vary from 32°F in spring to 50° in summer, and to go to —104° in winter. On the equator temperatures at noon ranged from 14° to 50° in the deserts, and 50° to 68° in the blue-green areas. He believes these values are too low rather than too high. Sunset temperatures averaged around 32°, while sunrise temperatures were from —4° to —86° on the ground. Temperatures of clouds ranged from 14°F in the daytime to —40° at sunrise. He could not measure surface temperatures at night, as the night side of the planet was turned away from us. But he thinks they commonly go to —100°.

Vegetation on Mars would have to be very different in many respects from most of the vegetation on earth, but we find earth plants of limited range that are already adapted to such conditions There is only half the sunlight on Mars that we have on earth, but shade loving plants such as rhododendron can get along with even less. Hard freezing is common at night on Mars, but in our own arctic regions and in high mountains we find plants thriving under those conditions. Northern Alberta is a farming country, yet temperatures well below freezing are likely to occur any month in the year. Plants grow in deserts drier than those of Mars.

The maximum temperature sunlight could produce on Mars is only 113°F, whereas on earth it is 247°. Plants and animals exist on earth which can stand 113° without the necessity of cooling, so that the mechanism for evaporating water, on which earth plants and animals depend to keep their temperatures down, is probably quite unknown on Mars.

The straightness of the canals of Mars is the strongest evidence of their artificiality, but we do not need irrigation to explain them. We need only to suppose that water is always hard to get, so is used only on luxury crops, such as earth people have close to their homes. Imagine then a network of transportation lines (the arrangement and the distances suggest railroads), with people preferring to live close to these transportation lines, with orchards and gardens near their homes, and with back country used as grazing range. You have there a picture which fully explains the canals.

Some of the water could be obtained by irrigation canals. But it seems quite probable that most of it is obtained by less elaborate means. It could come from dew ponds. It could be obtained by scraping up snow in winter and spreading it on your garden to melt in the spring. But however it is obtained, it is not gotten easily, so the planet's main food supplies are grown without it, even as Alberta lives mainly on cattle and wheat, which are adaptable to her climate.

Some Possibilities For Rocket Propellants
PART II OF THREE PARTS*

By ARTHUR S. LEONARD

EDITOR'S NOTE: Because of the widespread interest in the subject matter of this series of articles, it has been decided to publish them in three parts instead of four, as originally planned. This will allow the members of the A.R.S. to peruse and evaluate the entire paper by Mr. Leonard within a reasonable period of time, since it is contemplated to publish the Journal ahead of schedule for the next few issues at least.

SUMMARY

In Part I equations were derived which gave the final velocity of the rocket in terms of the mass of the empty rocket, the volume of the propellant tank, and the density and the jet velocity of the propellant. These equations indicated that the highest final velocity will be obtained through the use of a variable propellant, starting with the combination which gives the highest value for the product of its density and jet velocity and ending with one which produces nearly the highest jet velocity.

In this installment is a discussion of some of the more important factors which affect the jet velocity; and equations are developed by which the jet velocity of various propellant combinations may be estimated. Tables of thermochemical properties of some of the compounds which may be used as propellant gases are presented.

From a study of these and other data (not presented) the conclusion is reached that fluorine and oxygen are the two best oxidants. With fuels composed of elements of low atomic weight, oxygen is just about as good as fluorine; but with fuels composed of elements of high atomic weight, fluorine is greatly superior. The highest jet velocities will be obtained through the use of elements of low atomic weight, while elements of high atomic weight will be useful where high propellant density is desired.

JET VELOCITY

The velocity of the products of reaction leaving the rocket motor may be expressed by the following equation:*

$$U_J = \sqrt{2J\left(h^{\circ}_{rc} - h^{\circ}_{rm}\right)} \quad (9)$$

This equation is based on the Law of Conservation of Energy, and states that the gain in kinetic energy by the reaction products as they flow through the nozzle is equal to the loss in their heat energy. In order to be able to see more readily what might be done to bring about the conversion of a maximum fraction of the available heat of the propellant into kinetic energy, we may express the jet velocity by another equation:

$$U_J = \sqrt{\frac{2JC_pT_{rc}}{M_r}\left[1-\left(\frac{1}{R_p}\right)^{\frac{\gamma\gamma}{\gamma}}\right]} \quad (10)$$

This equation is derived from Eq. (9) and is based on the following assumptions: (1) as the products of reaction pass through the nozzle, their specific heats remain constant; (2) they behave

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*For the meaning of the various symbols used, see Part I of this paper, Journal of the A.R.S. — Dec. 1946.

as a perfect gas; (3) no chemical reaction or combustion takes place; (4) no heat is transferred by radiation, convection, or conduction; (5) the angle of divergence of the nozzle is small; (6) the flow is frictionless; and (7) the jet discharges into a medium, the pressure of which is maintained equal to the static pressure of the gases of the jet at the mouth of the nozzle. Eq. (10) may be simplified further:

Let

$$H_e^o = C_p T \tag{11}$$

$$H_{erc}^o = C_p T_{rc} \tag{11a}$$

$$h_e^o = \frac{H_e^o}{M} \tag{12}$$

$$h_{erc} = \frac{H_{erc}^o}{M_r} \tag{12a}$$

and

$$\eta_n = 1 - \left(\frac{1}{R_p}\right)^{\frac{\gamma-1}{\gamma}} \tag{13}$$

Combining Eqs. (10), (11a), (12a) and (13), we get:

$$U_J = \sqrt{2 J h_{erc}^o \eta_n} \tag{14}$$

EFFECTIVE ENTHALPY

In order to obtain a reasonably accurate estimate of the effective enthalpy of the propellant gases in the combustion chamber, we will need to know, (a) the heat of formation and initial temperature of each component of the propellant; (b) the heat of formation and dissociation constants for each polyatomic molecule in the products of reaction, and (c) the specific-heat-temperature relationships for all elements and compounds involved. While data on the heat of formation of a great many elements and compounds are listed in the handbooks; specific heats and dissociation constants are given for only a few. We will, therefore, be able to make good estimates of the effective enthalpy of the products of reaction for only a very few propellant combinations. Rather than limit this investigation to these few propellants, as many combinations as possible will be compared without taking into account the effects, on the individual combinations, of high-temperature dissociation. While this is admittedly a poor basis on which to compare one propellant with another, it will allow us to eliminate many elements and compounds from further consideration, and will indicate in a general way which combinations show enough promise to warrant further investigation.

Theoretical considerations of the ability of the molecules of a gas to take up energy in the form of translational, rotational, and vibrational motion yield numerical values for molal specific heats, as given in the second column of Table 1. Experimental determinations of the specific heats of a number of gases indicate that for monatomic gases, such as helium or mercury vapor, the actual value is very close to the theoretical over a wide range of temperatures. The specific heats of polyatomic gases, on the other hand, start out at very low temperatures with values of around 5 calories per gram mole per degree centigrade, and increase with temperature, attaining their theoretical values in the neighborhood of 2000°C or higher. The experimental data indicate also that the specific heats of these compounds do not level off at this temperature, but continue to rise slowly above the theoretical value with increasing temperature.

The most generally accepted explanation for this behavior is that, at low temperatures, the polyatomic molecules do not take up energy in the form of rotational and vibrational motion as readily as their freedom of motion in this respect would indicate. As the temperature is increased, however, they begin to rotate and vibrate and thus take up additional energy. As a result of this, the specific heat increases. At very high temperatures the molecules become completely excited with respect to all degrees of freedom of motion. At this point the actual specific heat is equal to the theoretical value; and the total internal energy is equal to the product of the absolute temperature and the theoretical value of the specific heat. Since the total internal energy is always equal to the product of the absolute temperature and the average of the actual values of specific heat from absolute zero to the temperature, at very high temperatures, the average of the actual values from zero to the high temperature must be equal to the theoretical. For this to be true, and at the same time have the actual specific heat lower than the theoretical value over the lower temperature range, the actual must exceed the theoretical value over some of the higher temperature range. This explains why the observed values for specific heat of a gas can exceed the theoretical value at some temperatures.

Since the temperature of reaction, for most of the propellants about to be considered, will be quite high and since data on the specific heats, over the temperature range required, are lacking, it will be assumed that, for the gaseous products of reaction in the combustion chamber, the sum of the internal kinetic energy plus the external work done will be given by the product, $N\ C_pT$. This again is a poor assumption to make, but it probaly does not introduce anywhere near as much error in comparing one propellant with another as does the assumption that no dissociation occurs.

Any substance, on being heated from absolute zero until it is completely vaporized, must expand against its own internal forces of attraction as well as against whatever external pressure is applied. Because of the small change in volume, the energy required to expand a solid against its own internal forces is usually relatively small; and since the data required, in order to calculate this quantity, are lacking for most of the substances that will be considered, this energy will be neglected in the tables which follow. The energy required to expand a liquid against internal forces is likewise small for most substances; and it too will be neglected. Transformations from one crystal structure to another usually involve internal energy changes which are quite small and which also may be neglected. The processes of fusion and vaporization, on the other hand, usually involve much larger internal energy changes; and, since data are available for many elements and compounds, and may be estimated for many others, these energy changes will be included in the calculations.

From the above discussion, we may write an expression for the total heat content of one mole of the gaseous products of reaction at very high temperature:

$$H^{o} = \Delta E_F + \Delta E_v + C_p T \tag{15}$$

where ΔE_F and ΔE_v may be expressed as follows:

$$\Delta E_F = \Delta H_F - \frac{P\Delta V_F}{J} \tag{16}$$

$$\Delta F_v = \Delta H_v - \frac{P\Delta V_v}{J} \tag{17}$$

Now, the increase in volume due to fusion (ΔV_F) is so small that the last term in Eq. (16) may be neglected. The volume change due to vaporization (ΔV_v) is, on the other hand, large enough to justify its being taken into account in our calculations. The volume of the liquid at the boiling point is so small compared with that of the vapor at the same temperature and pressure that, for all practical purposes, the change in volume (ΔV_v) may be considered to be equal to the total volume in the vapor state (V_v). From this discussion and Eqs. (15), (16) and (17), we get:

$$H^{o} = \Delta H_F + \Delta H_v - \frac{PV_v}{J} + C_p T \tag{18}$$

Many saturated vapors at their boiling point occupy a volume so close to that indicated by the general gas equation that we may write:

$$PV_v = RT_v \tag{19}$$

From Eqs. (18) and (19), we get:

$$H^{o} = \Delta H_F + \Delta H_v - \frac{R}{J}T_v + C_p T \tag{20}$$

Since this equation should hold for any substance at any high value of T, it should hold for the reaction products in the combustion chamber of the rocket motor. Therefore, we may write:

$$H^{o}_{rc} = \Delta H_{Fr} + \Delta H_{vr} - \frac{R}{J}T_{vr} + C_p T_{rc} \tag{20a}$$

Combining Eqs. (20a), (11a) and 12a), we get:

$$h_{erc} = \frac{1}{M_r}\left[H^{\circ}_{rc} - \Delta H_{Fr} - \Delta H_{vr} + \frac{R}{J} T_{vr}\right] \quad (21)$$

When compounds having boiling or sublimation temperatures which are higher than the reaction temperature are formed in the reaction products, partial condensation of these compounds will result. Part of the latent heat of condensation will, therefore, be made available to heat whatever other gaseous elements or compounds are present in the products of reaction. Under such circumstances, the true value for the effective enthalpy may be considerably higher than that computed by Eq. (21).

NOZZLE EFFICIENCY

It should be noted that the term "nozzle efficiency" as defined by Eq. (13) is applied to the theoretical thermal efficiency of the nozzle rather than to the mechanical, or hydrodynamic, efficiency as is more commonly done. Consideration of the terms in Eq. (13) will show that the nozzle efficiency will be raised by increases in the pressure ratio, and will be higher the greater the value for the ratio of specific heats. In order to make the pressure ratio as large as possible, we should strive to increase the pressure in the combustion chamber, and reduce that at the nozzle mouth.

When we investigate the possibilities of increasing the chamber pressure, we find that a pressure will be reached, above which the added thrust, resulting from any further increase, will just accelerate the added weight at the same rate as that of the rocket as a whole. There is obviously nothing to be gained by raising the nozzle efficiency any more through further increases in the chamber pressure. The increase in weight of the rocket results from having to increase the weight of the propellant pumps and pump engine, or from having to increase the weight of the propellant tanks, if a pressurized system is used. In either case there will be an optimum value for the chamber pressure which will depend on the size and general design of the rocket.

When we investigate the possibilities of increasing the nozzle efficiency by reducing the exhaust pressure, we find that here also, a pressure will be reached below which the added thrust will just accelerate the added weight at the same rate as that of the rocket as a whole. The reason for this is that the net thrust produced by any part of the nozzle is, for any given angle of taper of the nozzle, directly proportional to the static pressure of the gases at that point. As we add length to the mouth of the nozzle in an effort to carry the expansion of the gases further, the static pressure there becomes lower and lower. Finally a pressure will be reached which will just accelerate the mass of the last increment of length of the nozzle at the same rate as that of the rocket as a whole. This pressure also will depend primarily on the size and general design of the rocket and will be essentially independent of the physical properties of the propellant.

From this discussion, it can be seen that there will be an optimum value for the pressure ratio, which will depend primarily on the size and design of the rocket and motor, and which will be essentially independent of the physical properties of the propellant. For large rockets, such as the German V-2, in which the construction is by necessity rather heavy, the optimum pressure ratio may be as low as 25 to 1. In small rockets, or in other designs, in which the nozzle can be of lighter construction, the optimum ratio may be over 100 to 1.

In order to show the effects on nozzle efficiency of changes in the pressure ratio and in the ratio of specific heats, values for N_n have been calculated for two pressure ratios, 25 to 1 and 100 to 1, for a range of values of $\sqrt{}$. These are tabulated in the last two columns of Table 1. These calculations show the importance of high pressure ratio and a high ratio of specific heats. Since the value for the ratio of specitic heats depends almost entirely on the number of atoms per molecule in the reaction products, a propellant which produces reaction products having simple molecules will yield a higher jet velocity than one which has the same effective enthalpy of formation, but which produces reaction products having more complex molecules. As an example oi this effect, diatomic sodium fluoride with an effective enthalpy of formation of only 1870 calories per gram, but with a ratio of specific heats of 1.286, will produce a higher theoretical jet velocity than penta-tomic silicon tetrafluoride, with a value of Δh^0_{oe} of 3400 calories per gram, but with a value of $\sqrt{}$ of only 1.083.

Offhand, it might appear that, by taking the square root of the product of the effective enthalpy of formation and the nozzle efficiency, a value would be obtained by which we could rate the various compounds, as far as their potential jet velocity was concerned. However, such a comparison would be valid only for the cases in which a single compound was formed in the rocket motor. When two or more compounds, having different values for $\sqrt{}$, are formed simultaneously in the motor, the amount of heat energy converted into jet velocity will be greater than that indicated by the values in Table 1. For instance, if we were to add a fluorine-silicon mixture, which, by itself, gives a lower jet velocity, to fluorine and sodium, in any amount up to equal mole fractions, the jet velocity of the mixture would be higher than that of the sodium fluoride alone. Therefore, in order to calculate the theoretical jet velocity of a mixture of two or more compounds, we must first calculate the effective enthalpy and ratio of specific heats of the mixture as a whole; and then with these values, calculate the nozzle efficiency and jet velocity.

From this discussion, it can be seen that, while the ability of any pair of elements to produce a reaction product with high jet velocity will depend primarily on the effective enthalpy of formation and the number of atoms in the product of reaction, it will also depend on the physical properties and quantities of any other elements or compounds, with which it might be mixed in the combustion chamber, and on the pressure ratio of the nozzle. It should be pointed out, however, that while we cannot assign a definite value of nozzle efficiency to any given compound without knowing the circumstances under which it is to be used, we can be certain that in comparing any two compounds,, even under the most favorable circumstances, the one with the smaller number of atoms in the molecules will have a larger fraction of its heat energy converted into jet velocity.

HIGH-TEMPERATURE DISSOCIATION

While we need certain specific data in order to make a quantitative determination of the effects of high-temperature dissociation on any given reaction; in the absence of such data, we may at least give a qualitative description of some of these effects. First, the dissociation or ionization of any molecule or atom absorbs a relatively large amount of energy. The actual amount of heat released in the combustion chamber and, therefore, the true value of the effective enthalpy of the reaction will be smaller than that computed from the effective heat of formation. With many compounds dissociation sets in at such a relatively low temperature that the amount of heat which might actually be released in the combustion chamber may be well under 50 percent of the effective heat of formation.

Fortunately, the net loss in jet velocity may not be quite as large as that indicated by the dissociation in the combustion chamber. As the gaseous products of combustion pass through the nozzle, they expand and cool. As their temperature goes down, a certain amount of recombination, or association, may take place, thus recovering some of the heat lost in the combustion chamber. However, since this heat would be released after the gases have been partially expanded, it would not be as effective as if it had been released in the combustion chamber, before any expansion had taken place. Because of this fact, there would be some loss in jet velocity chargeable to dissociation even if recombination was complete by the time that the gases reached the mouth of the nozzle.

A second effect of dissociation is to raise the nozzle efficiency. In every case, dissociation results in an increase in the total number of separate molecules and atoms. This produces a reduction in the average number of atoms per molecule and raises the value of V for the mixture of reaction products. This effect is greatest for the most complex molecules and least for the simplest. Under the most favorable circumstances, the percentage increase in nozzle efficiency may be as much as one half of the percentage loss in heat release due to dissociation in the combustion chamber. With the effects of reassociation in the nozzle adding further to the recovery of energy we may, under some conditions regain as much as 75 percent of the energy lost through dissociation in the combustion chamber. With other compounds, however, the net loss in potentially available energy chargeable to the effects of dissociation, after allowing for recombination in the nozzle and increased nozzle efficiency, may be greater than 50 percent.

In conclusion, it should be pointed out that, while we need reasonably accurate thermochemical and dissociation data, in order to make a good estimate of the probable jet velocity which might be realized with any proposed propellant; we need not wait until such data is obtained before attempting to make use of any new propellant combination in a rocket. By running the new propellant in a small standardized test motor, we may very quickly and easily, and without any data on the reaction energy or dissociation constants, determine the actual jet velocity which it is capable of producing. Thus, with the aid of a few rather simple tests, we may determine the relative suitability of any proposed combination as a rocket propellant.

THE CHEMICAL ELEMENTS AS PROPELLANTS

One of the primary objects of this investigation was to determine the relative suitability of the elements as oxidizers and fuels. Tables 2A and 2B have been constructed for the purpose of making such a comparison. Table 2A lists heats of formation and melting and boiling-point temperatures of some of the oxides of representative elements, while Table 2B contains similar physical data for some of the fluorides.

If we start with pure elements in their standard states at some absolute temperature, T, allow them to react to form one mole of some compound, and then cool this material to the original temperature of the elements, the net heat given out to the surroundings is, by definition, the heat of formation, and is designated, ΔH_T.

If the elements start out at absolute zero, the heat given out is designated, ΔH_0. If the reaction were to be carried out in a thermally insulated space, so that no heat could be given out to the surroundings, the reaction products would be left with a total heat content equal to the heat of formation. Under these conditions:

$$H^\circ = \Delta H^\circ_0 \qquad (22)$$

Also, under these conditions:

$$H^\circ_e = \Delta H^\circ_{oe} \qquad (23)$$

Combining Eqs. (11), (20), (22) and (23), we get:

$$\Delta H^\circ_{oe} = \Delta H^\circ_o - \Delta H_F - \Delta H_V + \frac{R}{J} T_V \qquad (24)$$

From the relationship between enthalpy and molal heat content, we may write:

$$\Delta h^\circ_{oe} = \frac{\Delta H^\circ_{oe}}{M} \qquad (25)$$

It is this quantity which gives us a rough idea as to the jet velocity which might be obtained when using the combining elements as propellants. The jet velocity will be roughly proportional to the square root of the effective enthalpy of formation.

Values for the heat of formation are usually given for ordinary temperatures rather than for absolute zero. The heat of formation at absolute zero may be computed from these and other physical data by using the following equation:

$$\Delta H^\circ_{or} = \Delta H^T_{Tr} + H^\circ_{Tr} - \frac{N_O}{N_r} H^\circ_{TO} - \frac{N_F}{N_r} H^\circ_{TF} \qquad (26)$$

Values of the heat of formation at 18°C, taken from the handbooks, are tabulated in the 4th column of Tables 2A and 2B. These data, and estimates of the total heat content of the elements and compounds at 18°C, were used in Eq. (26) to calculate the heat of formation at absolute zero, tabulated in the 5th column. Where data on the heat of formation were not available, estimates were made. These estimated values are enclosed in brackets, and were made by comparing the heat of formation of the oxides, fluorides, chlorides, bromides, and iodides of the various elements when arranged, or plotted, as a function of their location in the periodic table. Values of the effective heat of formation, ΔH°_{oe} and the effective enthalpy of formation Δh°_{oe} are listed in the 6th and 7th columns and were computed with the aid of Eqs. (24) and (25).

The combining elements, listed in Tables 2A and 2B, are arranged in the order of increasing atomic weights. It will be noted that, in general, the value for the effective enthalpy of formation decreases with increasing atomic weight. This indicates that propellants composed of elements of low atomic weight should produce the highest jet velocities. Since, as has been shown, high density is under some circumstances as important as high jet velocity, and since the density of the elements and their compounds increase with atomic weight, we may find some of the elements of high atomic weight to be very useful as components of some propellant combinations. Therefore, until we have considered more evidence, we should eliminate very few of the chemical elements from our list of possible rocket fuels.

In comparing the values for the effective enthalpy of formation in Table 2A with those in 2B, it will be noted that for a few of the elements at the top of the list, the values for the oxides are roughly equal to those for the fluorides. For elements of high atomic weight, however, fluorine is greatly superior to oxygen as an oxidant. In fact, for many of the heavier oxides, not listed in 2A because of this, the term effective-enthalpy-of-formation is rather meaningless, because the compounds decompose, or dissociate at such very low temperatures, in many cases, below their boiling points.

If we were to make similar tables for the sulfide, selenides, tellurides, chlorides, bromides and iodides, we would find that all of the values for effective enthalpy were lower than those for the corresponding fluorides. With elements of low atomic weight, chlorine gives values for effective enthalpy of one third or less of that with fluorides. With the heavier elements, the ratio is about one half. The other oxidizing elements all give values lower than chlorine. While the densities of the heavier oxidizing elements and their compounds are somewhat greater than those of oxygen and fluorine, they are not enough so to offset the differences in the values for the effective enthalpy of formation. From this we conclude that, for fuels containing elements of low atomic weight, the oxidizer should be rich in oxygen or fluorine. For elements of high atomic weight, only oxidizers rich in fluorine should receive serious consideration.

REFRACTORY REACTION PRODUCTS

Some of the compounds listed in Table 2A are among the most refractory known. It might be thought that such materials, if formed in a rocket motor, would quickly foul it. Actually, this may not be the case. Over most of the interior surface of the combustion chamber and nozzle, a refractory substance would accumulate in a layer of rather uniform thickness which, because of the extremely high heat-transfer rates, would be limited in thickness to a few hundredths of an inch. Such a coating would be beneficial in that it would tend to make a reduction in the amount of coolant required by the motor, and it would protect the metal surface from erosion and oxidation by the hot propellant gases.

A very appreciable reduction in the refractoriness of the reaction products might be obtained by adding to the fuel a small amount of another element to form other compounds which would combine with the more refractory materials to reduce their fusion temperature. Even with this device, however, such materials may tend to accumulate to too great an extent around the fuel injectors and thus interfere with the proper atomization and combustion of the propellant. At the present time no clear-cut answer to this question can be given. Only through experimental research will we be able to tell how high a fusion temperature of the products of reaction can be tolerated in a rocket motor, and thus which elements must be excluded from use in propellants because of the formation of too refractory reaction products.

A third and concluding installment covering a wide range of compounds of different types, which might find use as rocket fuels or oxidizers, will appear in the next issue of the Journal.

TABLE 1 - THEORETICAL THERMAL PROPERTIES OF GASES AT HIGH TEMPERATURE

Number of Atoms in Molecule	Spec. Ht. at Const. Press., C_p $\frac{cal}{mole\ °C}$	Ratio of Spec. Hts., γ	Thermodynamic Eff. of Nozzle, η_n	
			For Press. Ratio, $R_p = 25$	For Press. Ratio, $R_p = 100$
1	4.97	1.667	.724	.842
2	8.94	1.286	.511	.641
3	13.90	1.167	.369	.482
4	19.86	1.111	.275	.369
5	25.82	1.083	.219	.298
6	31.78	1.067	.182	.250
7	37.74	1.055	.156	.215

Additional tables on following pages.

TABLE 2B THERMOCHEMICAL PROPERTIES OF SOME INORGANIC FLOURIDES

Chemical Formula	Molecular Weight, M $\frac{gm.}{gm.mole}$	Physical State at 18°C	Heat of Formation at 18°C, ΔH_{291}^{291} − $\frac{cal.}{gm.mole}$	Heat of Formation at 0°K, ΔH_0^0 − $\frac{cal.}{gm.mole}$	Effective Heat of Formation, ΔH_{oe}^0 − $\frac{cal.}{gm.mole}$	Effective Enthalpy of Formation, Δh_{oe}^0 − $\frac{cal.}{gm.}$	Melting Point Temp., t_F − °C	Boiling Point Temp., t_v − °C
HF	20.01	liq	71.4×10^3	71.7×10^3	63.8×10^3	3190	−92	19
LiF	25.94	sol.	145.6	144.2	94.7	3650	870	1676
BeF_2	47.02	sol.	—	[230]	188	4000	800	[1300]
BF_3	67.82	gas	256.9	259.2	254.4	3750	−127	−101
CF_4	88.01	gas	163	164	161	1830	−184	−128
NaF	42.00	sol.	136.3	134.9	78.6	1870	990	1700
MgF_2	62.32	sol.	261.4	258.5	192	3090	1396	2239
AlF_3	83.97	sol.	329.0	325.1	282	3350	1040	[1200]
SiF_4	104.06	gas	361.3	363.4	360.8	3470	subl.	−95
PF_5	125.98	gas	—	[350]	345	2740	−94	−85
SF_6	146.06	gas	262	263	258	1770	subl.	−64
KF	58.10	sol.	134.5	133.2	85.8	1480	880	1500
CaF_2	78.08	sol.	290.2	287.4	224	2870	1360	[2450]
TiF_4	123.90	sol.	—	[400]	382	3080	[280]	284
CrF_3	109.01	sol.	266.1	261.9	217	1990	subl.	[1100]
FeF_2	93.84	sol.	164.5	161.9	125	1330	[1100]	[1300]
NiF_2	96.69	sol.	157.5	154.8	118	1220	[1100]	[1300]
CuF	82.57	sol.	—	[52]	28	340	908	1100
ZnF_2	103.38	sol.	172.7	169.6	131	1260	872	[1300]
GaF_3	126.72	sol.	—	[270]	230	1810	subl.	[950]
GeF_4	148.60	sol.	—	[340]	321	2160	[300]	[300]
AsF_5	169.91	gas.	—	[310]	303	1780	−80	−53
SeF_6	192.96	gas	246	248	242	1250	subl.	−46
SrF_2	125.63	sol.	289.0	286.3	226	1800	[1300]	[2490]
ZrF_4	167.22	sol.	—	[480]	445	2660	[600]	[600]
MoF_6	209.95	liq.	—	[450]	442	2110	17	35
AgF	126.88	sol.	48.7	47.4	13	100	435	[1300]
CdF_2	150.41	sol.	162.2	159.1	105	700	1100	1758
InF_3	171.76	sol.	—	[250]	207	1210	1170	[1250]
SnF_4	194.70	sol.	—	[320]	295	1520	[600]	705
SbF_3	178.76	sol.	216.6	212.4	197	1100	subl.	292
TeF_6	241.61	gas	315	317	311	1290	subl.	−38
BaF_2	175.36	sol.	287.9	285.4	229	1310	1280	2137
LaF_3	195.92	sol.	—	[420]	374	1910	[1280]	[1600]
HfF_4	254.60	sol.	—	[520]	475	1870	[900]	[900]
HgF_2	238.61	sol.	—	[100]	80	335	645	650
TlF	223.39	sol.	—	[68]	60	270	subl.	300
PbF_2	245.21	sol.	159.5	156.8	120	490	855	1290
BiF_3	266.00	sol.	211	207	180	680	730	[800]
ThF_4	308.12	sol.	—	[550]	500	1620	[1200]	[1200]

TABLE 2A. THERMOCHEMICAL PROPERTIES OF SOME INORGANIC OXIDES

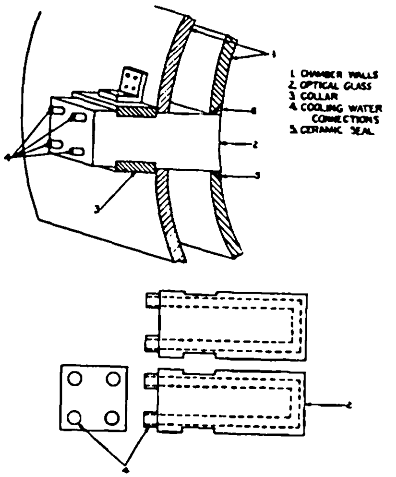
Chemical Formula	Molecular Weight, M − $\frac{gm.}{gm.mole}$	Physical State at 18°C	Heat of Formation at 18°C, ΔH_{291}^{291} − $\frac{cal.}{gm.mole}$	Heat of Formation at 0°K, ΔH_0° − $\frac{cal.}{gm.mole}$	Effective Heat of Formation, ΔH_{oe}° − $\frac{cal.}{gm.mole}$	Effective Enthalpy of Formation, Δh_{oe}° − $\frac{cal.}{gm.}$	Melting Point Temp., t_F − °C	Boiling Point Temp., t_V − °C
H_2O	18.02	liq.	68.4×10^3	67.8×10^3	57.4×10^3	3180	0	100
Li_2O	29.88	sol.	142.3	140.6	107	3580	subl.	1230
BeO	25.02	sol.	135.0	133.3	25	1000	2570	3900
B_2O_3	69.64	sol.	335.8	330.5	285	4090	577	[1600]
CO_2	44.01	gas	94.4	98.6	93.0	2110	subl.	−78
Na_2O	61.99	sol.	99.2	97.7	50	810	Subl.	1275
MgO	40.32	sol.	145.8	144.0	33	820	2800	3600
Al_2O_3	101.94	sol.	402.9	397.4	321	3150	2050	2250
SiO_2	60.06	sol.	203.3	199.8	126	2100	1470	2230
P_2O_5	141.96	sol.	371.7	363.0	346	2440	569	591
SO_2	64.06	gas	70.9	76.3	69.0	1075	−73	−10
K_2O	94.19	sol.	86.2	84.7	48	510	[800]	[1200]
CaO	56.08	sol.	151.7	150.0	66	1180	2572	2850
TiO_2	79.90	sol.	219.0	215.7	116	1450	1825	[3000]
Cr_2O_3	152.02	sol.	268.9	264.4	192	1260	1990	[2200]
ZnO	81.38	sol.	83.5	81.8	28	340	Subl.	1800
Ga_2O_3	187.44	sol.	256	251	182	970	1900	[2100]
GeO_2	104.60	sol.	—	[170]	125	1200	1100	[1200]
As_2O_3	197.82	sol.	154.1	149.5	140	710	313	457
SeO_2	110.96	sol.	56.4	53.4	34	300	subl.	317
SrO	103.63	sol.	140.8	139.2	70	680	2430	[2500]
ZrO_2	123.22	sol.	258.1	255.0	115	930	2715	4300
MoO_3	143.05	sol.	176.5	172.2	141	990	745	1151
Sb_2O_3	291.52	sol.	165.4	161.0	142	490	656	1550
TeO_2	159.61	sol.	77.6	74.7	52	330	subl.	450
BaO	153.36	sol.	133.1	131.6	80	520	1923	[2000]
La_2O_3	325.84	sol.	456.9	452.4	350	1070	2315	4200
HfO_2	210.60	sol.	271.5	268.6	130	620	2812	[4400]
PbO	223.21	sol.	52.5	51.1	0	0	890	1472
Bi_2O_3	466.00	sol.	137.1	132.9	100	210	820	1890
ThO_2	264.12	sol.	293	290	155	590	[2900]	4400

Letters to the Editor

Editor, Journal of the

American Rocket Society:

Many interesting papers were presented at the First Annual Convention of the Society. However none of them mentioned any information which would indicate that some means of getting into the combustion chamber and having a look at what actually goes on has been used.



I have enclosed some drawings of a device which might prove practicable. A rectangular piece of fine optical glass is inserted through the walls of the combustion chamber and is held in place by a collar mounted on the outside wall. The glass is cored for cooling water. The cores should be as close to the surface as possible. This will cut down on the volume of glass subject to extreme temperatures and will increase the size of the viewing area. Since we will be looking at the inside of the combustion chamber through a sheet of water, the cores must be designed to reduce turbulence to a minimum.

The combustion chamber end of the glass is provided with a graphite or ceramic filling to seal off the opening. By experiment it could be determined whether or not the glass must be ground to the same curvature as the chamber walls. If it must conform exactly then we will probably have to provide a correcting lens at the external viewing end.

Now that we have entry into the combustion chamber, we must provide means for obtaining our information, taking into account at all times the safety requirements for operating personnel.

We can mount a periscope so that the operators can look directly into the combustion chamber through a very dark filter. There are several advantages to be gained. Impending materiel failure could be detected in time to shut off the fuel and save the rocket motor. But more important, we could obtain information on fuel nozzles, ignition troubles and flame pattern. This information could very easily lead to more efficient motor design.

It should be easy to mount a high speed camera for the purpose of preserving the pictures for latter comparison with other motors. The camera will have to be provided with a proper filter, and the pictures should be taken at high speed to minimize vibration distortions.

Several means for obtaining temperature readings present themselves. Optical pyrometry through filters has been successfully used to determine extreme temperatures. Perhaps we could use a thermopile which has been carefully calibrated using the identical testing arrangement on measurable temperatures and the calibration curve extended. Accurate knowledge of temperatures attained by different fuels would be gratifying indeed.

A spectroscopic analysis of different sections of the chamber and nozzle is indicated. It is conceivable that we could obtain information with which to attack the problem of dis-association.

I may have missed the boat entirely in this approach. However, we greatly need information on what goes on in the combustion chamber and I will be satisfied if this only leads to greater discussion of the problem.

CHARLES WEBER, Jr.

A Contribution To The Levitation Problem

By CEDRIC GILES

Since the publication of a previous article* on means of lifting objects against gravity several additional findings have come to the attention of the writer of which two of the most interesting are presented.

An early concept of reaction propulsion in which a gas mass was to be successively impelled was contemplated by the Polish engineer Franz Abdon Ulinski. As shown in Fig. 1 the system consists of a large chamber, an injector nozzle, a compressor driven by any outside source of power, and a working supply of inert gas, such as helium or nitrogen, moving continu-

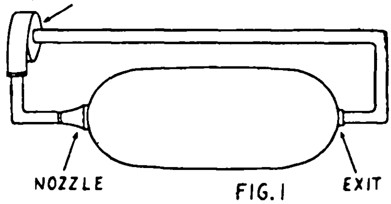


FIG. 1

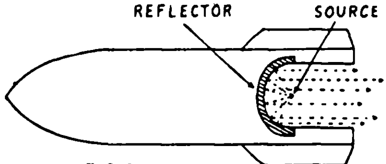


FIG. 2

ously through the system. A cycle would constitute a mass of gas after leaving the compressor being discharged at high velocity through the nozzle into the chamber of gas maintained at ordinary atmosphere pressure. Gas drawn at low velocity from the chamber would be again compressed by the compressor.

The reaction force was to be produced by the difference between the high jet velocity at the nozzle end and the low velocity at the chamber exit. A small reaction loss was readily conceived due to the discharge into an atmosphere rather than a vacuum.

ATOMIC REFLECTOR THEORY

A few years ago a suggestion[1] was made that a special type of reflector might be used to control the direction of atomic particles for providing a reaction to the rocket. The general idea may be considered similar to the principle of reflecting light rays in straight lines from a parabolic mirror which has a source of light at its focus point.

As in Fig. 2, atomic particles would emanate from a fixed source of radioactive energy and on meeting a form of electromagnetic parabolic reflector would be reflected in parallel lines opposite to the direction of travel by the rocket. As discharged particles are now controlled in the Cyclotron by magnetic forces the possibility of eventually developing such a source of reflected energy was not considered too remote.

REFERENCE

[1] U.S. Naval Institute Proceedings, June 1942.

*Journal of the American Rocket Society, No. 68, December 1946.

Some Remarks on the Temperature Problems of the Interplanetary Rocket

By ROBERT L. STERNBERG
Mathematics Department, Northwestern University

Continuing the discussion started by Commander Ellerton in the December 1946 issue of the JOURNAL I should like to report the following calculations based upon standard astronomical procedures. Considering a spherical rocket gondola made of heat conducting material which has been coated with lamp black to approximate an ideal black body, and presuming it to be operating in space in the vicinity of the earth, we may determine its equilibrium temperature (i.e. the temperature at which its rate of absorption of solar energy exactly equals its rate of emission of black body radiation) from the following equation in which the energy absorbed per unit time by the projected area of the sphere exposed to the sun's rays is equated to the energy radiated per unit time by the entire area of the blackened sphere at the unknown absolute temperature T. Thus we have,

$$\pi R^2 S = 4\pi R^2 K (T^4 - T_0^4) \qquad (1)$$

where

R = Radius of the gondola in cm.

S = Solar Constant at the earth's orbit $= 1.34 \times 10^6$ ergs cm. $^{-2}$ sec. $^{-1}$

K = Stefan-Boltzmann Constant $= 5.67 \times 10^{-5}$ ergs cm. $^{-2}$ deg. $^{-4}$ sec. $^{-1}$

and Taking $T_0 = 0$ as the rocket is operating in space practically devoid of matter we obtain, upon solving for T the following,

$$T = \sqrt[4]{\frac{S}{4K}} = 277^\circ \text{ ABSOLUTE} \qquad (2)$$

or $T = +4^\circ$ Centigrade. (3)

To calculate the equilibrium temperature for the gondola at any distance from the sun other than the earth's distance we need only correct the Solar Constant, S, by the inverse square law as follows, denoting by S', the Solar Constant at a distance D astronomical units from the sun, recalling that one astronomical unit is 92,870,000 miles, or the mean radius of the earth's orbit.

$$S' = S\left|\frac{\text{Distance: Sun to Earth}}{\text{Distance: Sun to Rocket}}\right|^2 = \frac{S}{D^2} \qquad (4)$$

So that substituting S' for S in equation (2) we have,

$$T' = \sqrt[4]{\frac{S'}{4K}} = \sqrt[4]{\frac{S}{4KD^2}} = \frac{277^\circ}{\sqrt{D}} \text{ ABSOLUTE} \qquad (5)$$

for the temperature T' of a black gondola at a distance D astronomical units from the sun.

Using equation (5) we find at Mars the equilibrium temperature, T', of the gondola is 224° Absolute or —49° Centigrade and at Venus 326° Absolute or +53° Centigrade. Thus a considerable change in temperature due to solar radiation will be experienced on any interplanetary journey unless some artifice is provided to regulate the rates of absorption and emission of heat radiation by the gondola.

However, solar radiation is by no means the only source of power which will tend to heat the rocket. Like any other piece of machinery a rocket will not be 100% efficient but a certain percentage of the energy released by the rocket motors, etc., will ultimately appear as heat within the gondola and will have to be dissipated into space by radiation. During the period in which the rocket is operating relatively near to a planet and hence in an intense gravitational field the power output required of the rocket motors will be quite large and even a highly e....cient rocket may produce enough internal heat to raise the equilibrium temperature sufficiently to require that the rocket be slowed down to prevent overheating of the entire system.

Whether or not an actual rocket would best be black is a question which is open to discussion. Black has merely been taken as the color of the hypothetical gondola treated above to facilitate calculation. There are, I believe, paints and lacquers available which would considerably reduce the absorption of solar heat by the gondola and only slightly reduce the radiation of heat from the gondola as compared with an ideal black body of the same size and shape. Since in all probability it will be most practical to keep the rocket as cool as possible to prevent over heating when in an intense gravitational field and then deliberately heat it when necessary, such an exterior finish as this which would provide maximum emissivity and minimum absorptivity may be most desirable.

References:

Astronomy — Russell, Dugan, and Stewart, Vols. I and II; 1927.

A Text Book of Heat—Part II—Allen & Maxwell; 1945.

Handbook of Chemistry & Physics—Hodgman; 1944.

Evaluation Of Several Mass-Ratio Phases

Fractional Versions of Fuel-Velocity Relationships

By CEDRIC GILES

Examination of fuel components in the power flight of a standard fuel-weight ratio rocket suggests a hitherfore unexplored line of thought. By attacking the fuel problem through an initial fuel breakdown a better opportunity may be provided for systematically reducing the high fuel load.

A second subject contemplates the relationships of fuel and velocity values and flight altitudes present at halfway points while the rocket is under power thrust. Consideration of such midpoint functions reveals emphatically the high percentage of fuel used in the early flight stages.

A third proposition deals briefly with the various launching and flight assists for extending the range of a rocket, and suggests some practical solutions to the problems outlined throughout the article.

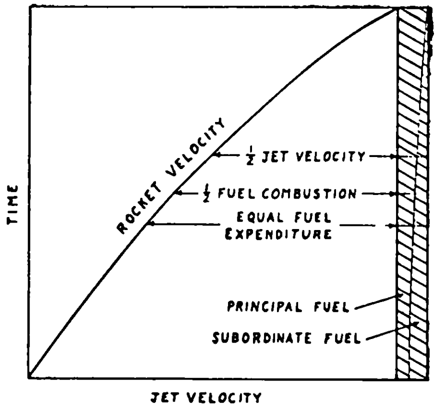
FUEL-WEIGHT RATIO

As defined in an article on the law of exponential motion in a recent issue of the JOURNAL* the equation of rocket motion is based on that function of rocket mathematics where initials rocket weight to initial fuel weight is 1:1.72, and indicated by the approximate figure 2.72, the e number. The mass-ratio rule, stating that a 2.72 rocket will at-

*JOURNAL of the American Rocket Society, Nos. 66 and 67, June and September 1946.

tain the speed of its gases the moment all fuel is expended, is considered a simple yardstick for basing complicated calculations of rocket performance.

As shown in the accompanying figure fuel consumption is indicative of duration of time, since with constant fuel expenditure equal portions of consumed fuel are used each unit of time. As fuel is burned, with the rocket be-



Velocity curve of a 2.72 rocket in relation to fuel components.

coming constantly lighter, the acceleration and velocity of the rocket will increase proportionally. Throughout the powered flight acceleration and velocity increase for any individual time period remain equal to each other. The accumulated rocket velocity will equal the jet velocity.

The fuel mass can be divided into two distinct parts; the principal fuel necessary to accelerate the rocket proper to jet velocity, valued at 1, and the subordinate fuel required to transport fuel, estimated at .72 or almost 2/5 of the entire fuel load.

Values given in this paper were obtained from points plotted on an enlarged graph; a process more tedious than complicated which does not warrant detail reproduction at this time.

THRUST DEVELOPMENT

In the case of a rocket having its overall weight equal or greater than pounds of thrust, the rocket will remain momentarily motionless. As fuel is consumed with increasing rocket lightness the thrust will take effect and the rocket will accelerate until all fuel is exhausted and become momentumborne. Assumption here is that the rocket at start of power flight will be of the 2.72 class, with the omission of air resistance and other outside affecting factors, and under a steady thrust will have exhausted all fuel when rocket velocity and jet velocity become equal.

FUEL EVALUATION

In evaluating the principal and subordinate fuel amounts, it becomes quickly apparent that although entire fuel mass is utilized at a constant rate, a certain percentage of each constituent must be used each instant, not first the principal fuel mass and then the remainder.

The graduated amount of each fuel part, illustrated in the fuel column, shows that in the initial moments of firing a greater mass of the adjunctive fuel is consumed, while the principal fuel consumption only is greater in the final increments of time. With the net rocket weight remaining fixed, the constant rate of fuel consumption to meet rocket velocity requirements is such that at the instant of take-off the principal fuel only needs to lift 1.00, the net rocket weight, while the subordinate fuel must sustain the entire 1.72 fuel load.

As fuel is decreased with the duration of firing the fuel component percentages graduate and have an equal expenditure when 5/12 of the entire fuel load (or time element) has elapsed. From this point on, the principal fuel supersedes the subordinate fuel as the paramount source of energy.

FUEL MIDPOINT

From calculations it has been found that at the halfway mark of the fuel expenditure (which is synonymous with the halfway point of the duration of time) velocity of the rocket is only slightly more than 1/3 of the speed of the exhaust gases with about 1/5 of the power flight completed. This means that in the expenditure of the second half of the fuel the rocket velocity must attain nearly ⅔ of the final velocity to cover the balance of the accelerated flight altitude. It also has been noted that at the halfway mark some 40% of the principal and 65% of the subordinate fuel has been spent.

VELOCITY MIDPOINT

A second consideration wherein rocket velocity equals one-half jet velocity with 1/3 of the altitude attained occurs when approximately 60% of the entire fuel is used. This demonstrates the high increase of rocket velocity per period of time as it approaches the maximum velocity. A breakdown of the fuel shows that nearly ½ of the principal fuel and ¾ of the subordinate fuel has been expended.

LAUNCHING AIDS

Providing an assisted start against gravitational pull before the rocket becomes thrustborne has the advantage of extending the otherwise normal altitude of the rocket. Reference to some of the ideas on the subject considered during the past years is of interest.

Early counter-gravity schemes suggested launching the rocket from a cannon, or catapulting it on a trajectory from the circumference of a giant rotating wheel, or circular track. Lately good results have been attained in the high acceleration launching of winged missiles by jet driven booster apparatus.

Less violent is the proposal of taking off from a high elevation or aircraft. This plan finds much favor for reaching very high altitudes and calculations show the great saving in fuel weight when compared with rockets starting at sea level.

Through the denser atmosphere at the slower speeds the use of thrust augmentors, a series of coaxial cones surrounding the jet for inducing air to increase jet mass, would aid the rocket in many cases. After use the augmentors could be discarded by parachute for future flights.

WEIGHT REDUCTION

Reduction of rocket and fuel weight during flight are most important as any weight elimination method could very likely be adapted to rockets in general.

The present approved method for lessening the rocket weight employs the multi-rocket principle, where fuel containers of each step are dropped when empty. Any diminishing of the net rocket weight means a smaller amount of principal fuel, resulting in less subordinate fuel.

Fuel weight may be reduced through some form of refueling in flight though a difficulty lies in synchronizing the rapid motion of rocket and fuel base. Most studies on this subject theorize on space flight with the rocket obtaining fuel from an interplanetary supply base.

A weight reduction seems possible in constructing fuel containers of a substance suitable as a fuel. Some work with dry fuel rockets was done years ago but little information is available at present.

CONCLUSION

The mass-ratio approach presents a convenient standard means for theoretically solving within certain limits specific flight cases. Some form of mass-ratio scale or table could be worked out which was applicable for evaluating flight performances.

The breakdown of fuel components demonstrates effectively the high percentage of subordinate fuel required to sustain the principal fuel, as well as providing a method of determining relative weight of components at any instant of thrust ascent.

Numerical results show the poor rocket efficiencies in the early thrust stages and the considerable amount of increase in latter time periods. This seems to warrant the necessity of supplying some type of booster aid in the starting moments.

Society News

Report on the annual meeting of the American Rocket Society held on Thursday, April 24th, 1947 in Room 503 of the Engineering Societies Building, 29 West 39th Street, New York, N. Y.

Mr. Pendray welcomed the members and guests of the American Rocket Society and reported to them, on behalf of the Board of Directors as follows: "To the Members of the Society:

The American Rocket Society has made very great progress in the last three or four years.

A large part of this progress has been made possible by the unprecedented growth of interest in rockets and jet propulsion. It has been aided by developments within the Society which have been initiated by the Board of Directors and well supported by the members. Among the developments which have been most significant are the following:

1. Affiliation with the American Society of Mechanical Engineers, and the opening of an office in the Engineering Societies Building.

2. Publication of "Rockets" by Dr. Robert H. Goddard, as the first step in what is expected to become a broad publishing program of the Society.

3. Development of a program for making available the growing literature on rockets, jet propulsion and related fields to members of the Society. The sales of books and other literature have been of great value in supporting the Society's other activities, and have also provided a central source from which information of this kind could readily be obtained by members.

4. The establishment of a paid secretarial staff to manage the business of the Society in a prompt and efficient way.

5. The enlargement and improvement of the Journal of the Society.

6. The first national convention of the Society last December, inaugurating annual national get-togethers of our members.

7. The "Dutch Treat Dinners" which have brought additional strength and guidance from the active membership of the Society.

In addition, the Society is now taking a step which may be more momentous than any: the adoption of a new constitution, which provides for chapters throughout the country, and makes direct participation in the Society and its meetings available to members in all parts of the country. By this step, the Society becomes a truly national organization, and takes its place alongside the other effective engineering and technical societies of the country.

The growth of membership of the Society is illustrated by these comparative figures for the last three years:

Membership: **Comparative Figures**

	1945	% Total	1946	Total	1947	% Total	% Increase 1947 over 1946
Active	51	16%	107	21%	204	26%	91%
Associate	221	69%	328	64%	462	61%	41%
Junior	46	14%	72	14%	93	11%	29%
	318		507		759		66%
Journal Subscribers (Institutions)	60		78		152		51%

As the figures indicate, the proportion of active members has increased 10 per cent in two years and the active membership is steadily increasing. The percentage of junior members is decreasing.

The Society must have other sources of income to meet expenses and be self-supporting. The two most logical sources are advertising in the Journal and the sale of publications.

A program for advertising in the Journal has now been undertaken, and should begin to bear fruit in the coming year.

Our present income, aside from membership, is derived principally from the sale of books, including the Goddard books, back numbers of "Astronautics" and other literature. Comparative figures for the income from these sources is given in the following table:

Publications **Comparative Figures**

	1946	1947
Books (not Goddard	$ 923.19	$ 516.43
Publications	950.43	1,265.22
Goddard book	241.99	2,583.07
	$2,115.61	$4,364.72

The total of $4,364.72 represents 56% of the total income of the Society for the fiscal year ending in 1947.

The Society is worth, in cash, bonds, equipment and saleable material, approximately $19,383.57, including 1080 Goddard books. The inventory of saleable material should remain at present valuation for several years, with the expenditure of only a small amount from time to time to replace materials which may run out of print.

"I have been closely associated with the development of the Society for more than 15 years. As Secretary of the Society, it has been my privilege to help carry out a number of the projects which have contributed to the Society's growth since 1942. I believe the organization is now well launched on its post-war career, and that it will become one of the most significant factors in the future technological development of the rocket and jet propulsion industry and of our nation."

At this meeting the following seven men were elected directors of the Society for the interim term ending in December, 1947.

ALFRED AFRICANO
LOUIS BRUCHISS
LEONARD AXELROD
CEDRIC GILES
HARRY BURDETT
ROY HEALY
E. L. CHANDLER

Under the constitution of the Society, the Board of Directors elects the officers. At the last meeting of the Board, the following officers were elected:

ROY HEALY, President

ALFRED AFRICANO,
Vice President and Chairman of the Technical Sections Division

LEONARD AXELROD, Secretary

CEDRIC GILES, Treasurer

LOUIS BRUCHISS, Editor of the Journal of the ARS

G. EDWARD PENDRAY, Chairman, Public Relations Committee

H. BURDETT, Chairman, Program and Convention Committee

E. CHANDLER,
Chairman, Membership Comm.

L. LAWRENCE,
Chairman, Nominating Comm.

As announced at the Annual Meeting, the Society will hold its Second National Convention in conjunction with the ASME in Atlantic City early in December. Members are earnestly requested to submit papers on any phase of jet propulsion devices or their application for possible presentation at the Convention or publication. Please forward these papers or a detailed outline by August 15th, 1947. Papers should be addressed to the Chairman of the Program Committee, office of the Society.

The new Board of Directors, on behalf of the entire membership of the American Rocket Society, takes this occasion to express its deep appreciation for the untiring services rendered by Dr. Pendray during his many years as secretary. G. Edward Pendray will always remain synonymous with the A.R.S.

NEW ACTIVE MEMBERS

Keith K. McDaniel
Marvin Wall
D. Marshall Klein
Jose Paso
Lawrence Maisak
David H. Ross
Vernon J. Basore
George W. Flynn
George T. Allen
Richard W. Darrow
Jack Isreeli
Alberto M. deLecca
William C. Cooley
John A. Kerr
Gordon M. Murtaugh
Nathaniel H. Rickles
Alton B. Crampton
Wilford P. Lakin
Bertrand des Clers de Beauheats

General News

Harry F. Guggenheim, President of The Daniel and Florence Guggenheim Foundation, announced recently that the Foundation has engaged the services of G. Edward Pendray as counsel on rockets and jet propulsion.

The Daniel and Florence Guggenheim Foundation has long been identified with the development of rockets and jet propulsion. Beginning in 1930, it supported the pioneer researches of Dr. Robert H. Goddard, founder of the modern field of rocketry and jet propulsion, and shares ownership with the Goddard estate in the important basic rocket and jet propulsion patents resulting from that research.

Mr. Pendray, well known as a counselor on industrial public relations and management, has also been closely identified with rockets and jet propulsion for more than 16 years. A pioneer rocket experimenter, he was one of the founders of the American Rocket Society, national professional association of rocket and jet propulsion engineers.

Use Of Flying Laboratories Highly Successful, Says G-E Engineer

Development testing of aircraft gas turbines through utilization of Army bombers, converted into flying laboratories, has proved to be a safe and expedient way of conducting tests under altitude conditions and more economical than establishing altitude wind tunnels, according to General Electric Company engineers.

Explaining that the jet power plants are installed as an auxiliary unit of the flying laboratories instead of a substitute engine, W. O. Meckley, G-E engineer, pointed out that a modified B-29 Superfortress is among the flying test beds now operated at the company's Flight Test Center near Schenectady. Tests under varying flight conditions, he said, are conducted with the company's powerful TG-108 gas turbine —power plant of the newest Army and Navy jet-propelled aircraft.

The flying laboratories also have proved "highly satisfactory" in the testing and development of new gas-turbine units before actual installation aboard aircraft, according to Mr. Meckley, who is assigned to flight test projects of the G-E Aircraft Gas Turbine factory at Lynn, Mass.

Where important components of a newly-designed jet engine are available before the complete unit, he pointed out, flight tests of the new components are conducted by utilizing some parts of the older engines.

Plans are underway, he said, to flight test jet power plants of advance design now being developed at the Aircraft Gas Turbine factory.

Pointing out that the I-40 gas turbine, developed by General Electric as the power plant for the Lockheed P-80 Shooting Star, also was further developed aboard a B-24 flying laboratory, Mr. Meckley said that the flying test cells provide a great number of the facilities and advantages of a wind tunnel, with "considerably less cost and greater availability." Use of the flying laboratories, he explained, also allows flight space for design engineers to observe operations under flight conditions and provides a means of acquainting them with the problems attendant to flying.

General Electric inaugurated its flight-test program early in 1942 when a B-23 was obtained for turbosupercharger flight investigations. Since then the company's Flight Test Division has operated nearly a dozen different flying laboratories, including three B-29s, six B-24s and other aircraft.

Currently maintained at the new Flight Test Center at Schenectady are two B-24 Liberators, a B-29 and other aircraft.

The project is sponsored jointly by General Electric and the Army Air Forces. A Pan American Airways crew mans the B-29 and other bombers on the flight tests. G-E personnel fly the other aircraft.

Pioneer Rocket Motor Test Station Built For General Electric - U.S. Army Ordnance Research

A test station which will static-test rocket propulsion motors up to 50,000-pounds thrust, equivalent to 500,000-hp, has been constructed near Schenectady as a part of the General Electric-U.S. Army Ordnance Department joint long-range rocket research program, it was announced by Ray Stearns, Manager of the Company's Aeronautic and Marine Engineering Division.

Mr. Stearns revealed that the test station, the first of its kind, has been in operation for over a year, contributing invaluable data for basic rocket motor design and fuel development.

Situated some distance from Schenectady, and about a mile from any habitation, the test station has three heavily-constructed steel-re-inforced concrete rocket-motor static-test pits. Each pit is set into the forward slope of a small hill, and earth is mounded around its sides.

Three covered stairways lead down into the interior of the test pit, which is divided into four rooms; control room, motor room, and two reactant rooms. Three-foot steel-re-inforced concrete walls separate the motor room from the rest of the structure, while all other walls and the roof are two feet in thickness. Heavy steel doors seal off each room when tests are in progress.

The control room, lying parallel to the motor room, contains all the valves and controls used in regulating "runs" of rocket motors under test in the pit. Roughly, two-thirds of the wall space is filled with observation ports and panels containing valves, switches, and gages. The rest of the area contains banks of photoelectric recorders which calibrate and record data continuously during the tests.

Three observation windows or ports pierce the three-foot wall between the control room and the motor room. Each of these windows is composed of three 3-inch bullet-proof glass panes, abutted on each side with 2-inch steel plates. Individually, each 3-inch bullet-proof glass pane will withstand the penetration of a 0.50-calibre projectile. Additional protection is provided in the observation ports by an air space of approximately one foot between panes.

Open at the rear for the passage of the rocket motor exhaust flame, the motor room is situated in the center of the test pit. Access to the motor room from the interior of the pit is through a narrow right-angle passageway off the reactant room passageway, sealed by a heavy steel door. Construction of this narrow right-angle passageway acts as an additional safety factor in case of misfire.

The operating stand to which the rocket motor is attached during test firings is on the forward wall of the motor room. Reactant and cooling water lines are connected to the motor at this point. Bolted to the floor opposite the observation ports of the control room are mirrors which reflect the motor and its exhaust to the observers during the "run". Attached to the wall above the observation ports, a totally-enclosed steel and asbestos box contains a camera, operated from the control room, which photographs the rocket exhaust during firing.

Behind the pit and in direct line with the motor room is a specially-constructed flame deflector of concrete and expanded metal screen. The expanded metal screen or flare baffle breaks up the rocket motor exhaust flame in cases where it extends beyond the confines of the motor room.

The reactant room directly forward of the motor room contains a specially-fabricated and calibrated tank which holds the oxygen propellant for the rocket motor. Liquid oxygen for this tank is supplied by a specially designed oxygen "buggy" which is connected to a fueling line outside the pit. Extending across this room from the wall of the motor room is a thrust bar which records the thrust of the rocket motor during test.

The second reactant room, parallel to the motor room, houses a smaller fabricated and calibrated tank for the alcohol propellant. Both propellant tanks are pressurized from gas bottles housed on the exterior of the pit.

Fire is an ever-present danger in rocket motor development and extensive precautions have been taken throughout the entire test station.

No smoking is permitted in or near the test pits. Power driven ventilation fans clear any fumes from the reactant rooms. Automatic alarm systems are located in the interior and outside of each test pit.

Within the pits a high-pressure water system, at 140-psi, can operate either automatically or manually by means of controls located in the control room. This system draws upon a 30,000-gallon reserve with a pumping station that can be run either by electric or gasoline pumping motors.

In the motor room, three different flooding systems can be used. An overhead sprinkler system has been installed near the roof while high-pressure nozzles are trained from above directly on the motor-operating stand. Piping running around the three sides of the room can completely flood the floor. In each of the reactant rooms, high-pressure nozzles are directed down upon the reactant tanks. Carbon dioxide extinguishers are located throughout the sites.

A fire department comprising two pumper trucks and a large capacity tank truck are on standby day and night with firefighting crews immediately available. Throughout the immediate test area a 6-inch high-pressure main has been laid with hydrant houses and hose strategically located. Fire guards patrol the entire area.

Sirens, starting five minutes before and costinuing through each "run", as well as flashing red lights above the pits, warn personnel in the test area. Guards are also placed a safe distance from the pit under test to prevent persons entering the area. Sirens continue to sound some time after the "run" to allow a safety period after shutdown.

Navy Unveils Powerful Jet Fighter

The Navy Department recently announced successful flight tests of its newest and most powerful fighter, the 600 mile per hour BANSHEE. It was designed and built by the McDonnell Aircraft Corporation, St. Louis, Missouri, under contract with the Bureau of Aeronautics.

Officially designated as the XF2D-1, the BANSHEE, bears a marked resemblance to the McDonnell PHANTOM, the Navy's first carrier based all-jet fighter. Production deliveries of the PHANTOM are now being made in St. Louis.

The BANSHEE is powered with two Westinghouse 24-inch diameter axial flow turbo-jet engines installed in the wing roots. These engines, the culmination of a long range turbo-jet development program, make the BANSHEE the most powerful single seat fighter flown in the United States today.

The McDonnell BANSHEE was built to conform to the exacting requirements of carrier operations, and to meet the ever increasing demand for higher speeds and increased rates of climb. Even though carrier based, it is in the 600 mile per hour class and climbs at a rate of over 9,000 feet per minute.

In addition to its fast speed and rate of climb the BANSHEE has unusually long range. Five self-sealing internal tanks carry its large fuel load. It is capable of normal take-off from aircraft carriers or landing fields, and is also fitted with catapult hooks, should this type of take-off be desired. The arresting gear is a conventional Navy type hook, housed in a well in the lower rear part of the fuselage. All these components, such as wheels, catapult hooks, arresting hook and wing flaps completely retract in flight, and are covered by flush type doors, giving the jet plane a sleek, compact appearance.

The BANSHEE, like its predecessor the PHANTOM, is designed to cruise on one or two engines. The Westinghouse 24 inch jets are placed close to the center line of the airplane so that little yaw results when one engine is shut off. By cruising on one engine the range is extended at lower altitudes and added safety for over water operations is provided.

The pilot sits well ahead of the wings and has unusual vision in all directions. He is protected from gun-fire by armor plate and a bullet-resisting windshield. The pilot's seat and supporting structure are crash resistant. They are designed to resist a crash impact 40 times the weight of the pilot. The wings, wheels, and flaps of the BANSHEE retract electrically avoiding the use of hydraulic lines and providing additional protection against gun-fire damage.

The BANSHEE has a wing span of 41 feet (18 feet when wings are folded for storage aboard an aircraft carrier). Its length is 39 feet over-all, height 14 feet and take-off weight is over 14,000 pounds.

Robert Edholm, engineering test pilot of McDonnell Aircraft in St. Louis, has put the BANSHEE through a series of satisfactory flight tests.

LIST OF ADVERTISERS:

Description Of Westinghouse Engine Powering Navy's New Fighter Plane

The pair of slim, axial-flow Westinghouse turbo-jet engines, which make the Navy's spectacular McDonnell "BANSHEE" (XF2D-1) the America's most powerful fighter plane, put at the command of the pilot more power than is found in any other fighter plane whether powered with turbo-jet, turbo-prop or reciprocating engines.

Such an outpouring of power in a fighter is unprecedented. Two thousand horsepower was considered about maximum for a fighter during the war. The engine of the most powerful preceding American jet plane produces the equivalent of a reciprocating engine of about 5,500 horsepower. The power of the "BANSHEE" engines is ever greater.

When Charles A. Lindbergh flew to Paris 20 years ago, he got 200 horsepower from his then remarkable engine.

Capacity to get into or ready for action fast is nearly as important a qualification for a Navy shipboard fighter as mere air speed. The enormous power of the BANSHEE'S Westinghouse jets gives the plane a rate-of-climb performance heretofore unapproached.

The engine is designated the YANKEE 24C to indicate its American origin and its diameter in inches. Like the Westinghouse 19XB Yankee which powers the PHANTOM, Navy's first all-jet plane, the prototype 19B engine which was the first wholly-American designed jet engine to be tested in flight, the 24C turbo-jet is an axial flow, straight-line, design. From the time the air enters the front until it streams forth with a gigantic push from the rear at almost twice the speed of sound, its direction of flow is never changed.

Such axial-flow design, pioneered by Westinghouse, packs the greatest power and least air-resisting frontal area into the lowest possible weight.

Each turbo-jet unit consists of an aluminum alloy and stainless steel cylinder 24 inches in diameter. There is but one major moving part, the two-piece aluminum alloy and steel shaft running lengthwise through the engine. On this shaft, are mounted the air compressor at the front and the gas turbine at the rear, just ahead of the point at which the hot gases that result from the burning of fuel and air are ejected through the especially-shaped jet orifice to produce the engine's propulsive thrust.

The compressor, which supplies the enormous quantities of air needed to achieve the jet volume and velocity consists of a forged aluminum alloy rotor containing numerous discs to which are attached spoke-like blades. These blades, each shaped like a tiny square-ended propeller blade, diminish in length with each successive "stage", or disc, squeezing the incoming air into an ever-smaller space as it is rammed rearward toward the combustion chamber in the engine's center.

The torrent of air enters the combustion chamber compressed to a fraction of normal volume. Inside the combustion chamber are stainless steel "combustion baskets", their sidewalls latticed with holes to admit part of the inrushing air. The air entering the baskets is mixed with gasoline sprayed from many atomizing nozzles. When the engine is started, this fuel-air mixture is ignited by a pair of spark plugs, but once in operation, combustion is self-sustaining, and the electric ignition shuts off.

As the burning gases travel through the combustion chamber, they are cooled down by the air which bypassed the combustor basket. Then, hurtling 400 miles an hour, the gas speeds through the turbine, which extracts the horsepower from its heat energy to drive the compressor. The remaining energy in the gas is utilized in the form of a high-velocity jet to yield the engine's propulsive thrust as it rips from the tailpipe nozzle at 1,200 miles an hour.

The engine's electric starting motor, built in the Westinghouse Small Motor Division at Lima, Ohio, is scarcely larger or heavier than the starter on an automobile.

Because of its "straight-line" design, the maximum diameter of this Westinghouse engine, including such accessories as fuel pumps and governor, is so small that the whole engine can be "buried" in the "BANSHEE" wing root and leave scarcely a ripple to interfere with the plane's swift passage through the air.

UNITED STATES PATENTS

The following patents were compiled from issues of the Official Gazette of the U.S. Patent Office. Copies of patents may be obtained from the Commissioner of Patents, Washington 25, D.C., at 25 cents each.

No. 2,280,835, "Aircraft"; Alf Lysholm, Stockholm, Sweden, assignor, by mesne assignments, to Jarvis C. Marble, Leslie M. Merrill, and Percy H. Batten.

No. 2,300,024, "Rocket Gun"; Edward F. Chandler, Brooklyn, N. Y.

No. 2,391,864, "Repeating Rocket Gun"; Edward F. Chandler, B'klyn, N. Y.

No. 2,391,865, "Self-Propelled Projectile"; Edward F. Chandler, B'klyn, N. Y.

No. 2,409,904, "Rocket"; Conrad David Schermuly, Alfred James Schermuly, and Charles Schermuly, Parkgate, Newdigate, England, assignors to The Schermuly Pistol Rocket Apparatus Limited, Parkgate, Newdigate, Surrey, England.

No. 2,410,538, "Prime Mover"; George William Walton, Farnham Common, England.

No. 2,411,895, "Nozzle Control"; David M. Poole, Summit, N. J., assignor to United Aircraft Corporation, East Hartford, Conn.

No. 2,412,134, "Projectile"; Carolus L. Eksergian, Detroit, Mich., assignor, by mesne assignments, to the United States of America.

No. 2,412,173, "Projectile"; Winslow B. Pope, Detroit, Mich., assignor by mesne assignments, to the United States of America.

No. 2,412,266, "Reaction Propelled Device"; Reginald W. Hoagland, Detroit, Mich., assignor, by mesne assignments, to the United States of America.

No. 2,412,825, "Jet Propulsion Apparatus"; Henry J. De N. McCollum, Chicago, Ill., assignor to Stewart-Warner Corp., Chicago, Ill.

No. 2,414,579, "Rocket Launcher for Aircraft"; Carl D. Anderson, Robert B. Leighton, and Charles H. Wilts, Pasadena, and Aldon L. Melzian, Altadena, Calif., assignors to United States of America.

No. 2,415,584, "Motive Power and Driving Means for Rotating Propeller of Helicopters"; Victor P. Fleiss, New York, N. Y.

No. 2,416,389, "Torque Balancing of Jet Propulsion Turbine Plant"; Fritz Albert Max Heppner, Leamington Spa, and John Denis Voce and David Rhys Evans, Coventry, England, assignors to Armstrong Siddeley Motors Limited, Coventry, England.

No. 2,418,488, "Power-Plant Apparatus"; Albert S. Thompson, Swarthmore, Pa., assignor to Westinghouse Electric Corporation, East Pittsburg, Pa.

No. 2,419,866, "Aerial Torpedo"; Walter Gordon Wilson, Martyr Worthy, Winchester, England.

—C. G.