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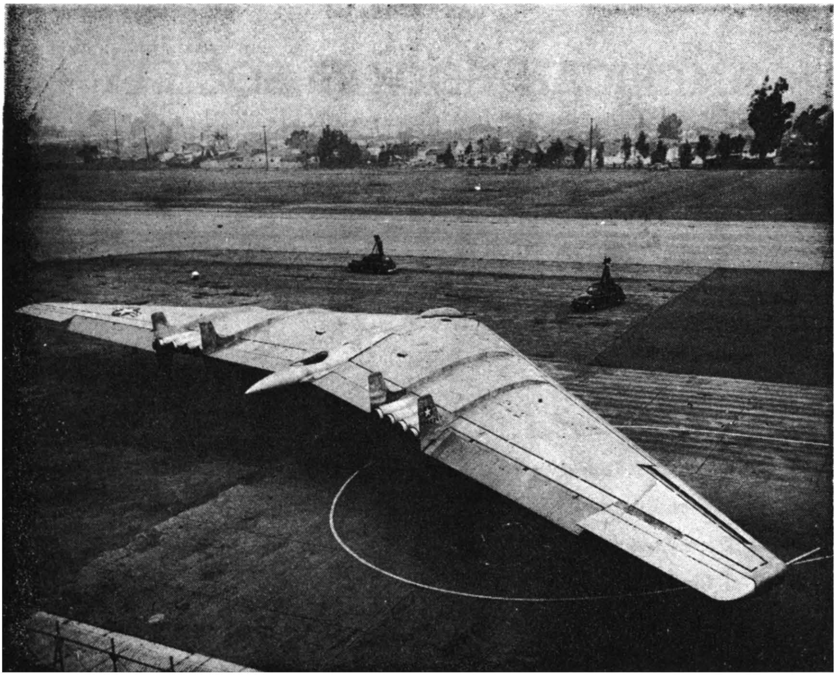
YB-49 Rollout



This is the U. S. Air Forces' mightiest bomber — the Northrop Flying Wing YB-49 — as it appeared rolled from its hangar at Northrop Aircraft, Inc., in Hawthorne, California. Eight jet engines, capable of developing 32,000 horsepower, will drive the boomerang-shaped craft through the air. Anticipated performance figures of the giant sky dreadnaught remain undisclosed. It is expected to operate at altitudes in excess of 30,000 feet. The latest of the Northrop Flying Wing planes, the YB-49 is the jet counterpart of the giant B-35 propeller driven bombers. Higher efficiency obtained in the Flying Wing design is made possible by elimination of drag-producing fuselage, tail assembly and protruding engine nacelles. All crew quarters, cargo space and power plants are housed wholly within the wing.

Limitations of Space Travel

By JAMES R. RANDOLPH

Formidable difficulties stand in the way of space travel even to the nearest planets. These difficulties increase greatly with the remoteness of the goal to be reached. Travel time to the outer planets would be measured in decades; to other stars in thousands of years. Temperatures beyond the orbit of Mars are below freezing. Temperatures beyond Pluto are below the freezing point of air.

All that we have been able to learn about atomic energy still indicates that this energy will have to be taken in the form of heat, and we will have to employ some kind of heat engine, with some conventional working fluid, in order to transform this heat into usable power.

With present methods of transportation by land, sea, and air the working fluid is either taken from the surroundings or is saved and used over and over. Atomic energy would take the place of fuel, and would mean, for instance, that a submarine could remain at sea for years instead of having to return to base every few weeks for refuelling.

But a rocket travelling in empty space must eject its working fluid and cannot replace it. Even with atomic energy, therefore, the space rocket is still restricted in its possibilities by the thermodynamic characteristics of its working fluid. Thus, if we limit our top temperature to 4,000° F, which is a few hundred degrees below the melting point of uranium, the highest jet velocity we could get with steam as a propellant is about 10,000 feet per second. Goddard has done better than this without atomic energy, by burning oxygen with an excess of hydrogen. (1) It seems likely that jet velocities of 20,000 or even 30,000 feet per second could be obtained with atomic energy using lithium as a propellant, but the thermodynamic properties of lithium are as yet but little known.

The mass ratio of a rocket is the ratio of its starting weight to the weight it still has left when all its propellant has been used up, and all its possible velocity changes made. This mass ratio is given by the equation:

$$M = e^{\frac{v}{c}}$$

in which e = 2.71828, c is the jet velocity discussed above, and v is the sum of the velocity changes to be imparted to the rocket.

In order to escape from the earth a rocket must be given a velocity of 6.95 miles per second. The first five miles per second of this velocity of escape must be imparted with the highest possible acceleration, for gravity is tending to pull the rocket down, and its effect is in proportion to the time the acceleration lasts.

Further velocity increments must be added in order to have the rocket go anywhere. These can be added more slowly, with a relatively small rocket Motor and a small rate of propellant consumption, but the equations for mass ratio still hold, and the increments are the same no matter how slowly they are added.

For preliminary computation of these increments the equation for the velocity of a planet or rocket in its orbit may be written in the form:

$$v \quad 18.5\sqrt{\frac{186}{r} - \frac{93}{a}}$$

in which r is the distance from the sun, and a is the semi-major axis of the orbit. Results are given in Figure 1.

The most economical way to go to Mars or Venus would be to pass very close to the moon, so that its attraction would throw the rocket into an orbit (orbit 1, Figure 1) having the same velocity as the earth (18.5 miles per second) at 93 million miles from the sun, but so much more elliptical as to be tangent to the orbit of Mars at its perihelion. Such a rocket could be gotten to Mars with only slightly more power than is required to get it free of the earth. But when it arrived it would have to make an air retarded landing from a velocity of approach four miles per second in excess of the velocity of escape. Very precise navigation would be required, and if any error occurred the rocket would wander helplessly out into space or into the sun, and there would be no slightest chance whatever of saving it.

A safer navigation procedure would be to give the rocket a velocity increment of 1.4 miles per second straight ahead of the earth (orbit 2, Figure 1). This would put it into an orbit which is tangent to the orbit of Mars at perihelion, and then a further increment of two miles per second would put it into the orbit of Mars.

A similar procedure could be used for reaching Venus, but the increments are 1.56 miles per second leaving earth, and 1.7 miles per second arriving at Venus.

A rocket could go to the moon, and around the moon, with no increments. But to land on the moon, which is airless, would take a reverse rocket action equal to the velocity of escape of the moon. This is 1.47 miles per second. An equal increment would be needed to get the rocket up again.

VELOCITIES OF PLANETS AND ROCKETS

Planet	**Distance from Sun**		**Velocity, mi. sec**			
	Perihelion	Aphelion	P	A	Increment	
Venus	66,738,000	67,653,000	21.8 avg.			
Earth	91,342,000	94,452,000	18.5 avg.			
Mars	128,330,000	154,760,000	16.46	13.7		
Halley's Comet	54,870,000		--33.7			
Mars	(at earth)					
Rocket						
Orbit 1	93,000,000	128,330,000	18.5	12.4		4.06
Orb. 2	93,000,000	128,330,000	19.9	14.42	1.4	2.02
Venus Rocket	67,000,000	93,000,000	16.94	23.5	1.56	1.7
Outer Space	93,000,000	∞	26.4		7.7	

Figure 1

Halley's Comet is another world which comes close enough to be interesting. It was near the earth in 1910, and is expected back about 1985. But this comet has a retrograde movement, circling the sun in the direction opposite to that of the planets. Hence it would be a difficult and complex problem to compute an orbit which could reach it without excessive expenditures of propellant.

Writing in The Cornell Engineer for April 1947, Alvin L. Feldman suggested the use of a ramjet instead of a rocket for that part of the space ship's velocity which can be attained in the air. This could be the 6.95 miles per second to escape from the earth or Venus, or the 3.13 miles per second needed to escape from Mars. But the mass ratio of the rocket, which alone could operate in the Vacuum of space, would still be considerable.

In making an air landing the space speed of the rocket has to be dissipated as heat. This heat amounts to 27,000 B.T.U. per pound for the speed attained in approaching earth, and it is 5400 B.T.U. per pound when making a landing on Mars.

To send a rocket to one of the remoter planets, or beyond the Solar System, an orbit increment of 7.65 miles per second would be required. This is obtained by putting:

$$a = \infty$$

in the velocity equation.

Rockets for space travel come down to manageable size when we figure on one way trips to Mars or Venus, with permanent colonies established there, and civilizations built up capable of constructing rockets before any attempt is made to return to earth. Building a rocket to return from Venus would be at least as difficult as building a rocket to go there in the first place. The two planets are of nearly equal size. Building a rocket to return from Mars would be enormously easier, for velocity of escape is less than half that of earth, and that makes the escape mass ratio the square root of the escape mass. ratio of earth.

A plan (2) which may soon prove practical is to send a colony to Mars which could establish industry there, build rockets, keep in touch with earth, and be in effect a pistol at the back of any aggressor who attempted to conquer the earth. America has played that role in two world wars, each time giving the peaceful nations of Europe an effective ally safely beyond the aggressor's reach. But in a third world war America herself would be vulnerable, and a base for counter-attack is needed which is beyond the reach of the weapons of this world. Mars appears to offer such a base. In the race to get there first with an adequate colony America has the same advantage she had in producing the Atom Bomb and in every other technical development of war. But atom secrets can be stolen. Mars, once effectively occupied, cannot.

One way journeys to Mars or Venus would take about six months. Round trips, because of the necessity of waiting for the proper configurations of planets in their orbits, about two and a half years. For the outer planets much longer travel times would be required. Halley's comet takes thirty years to go to the orbit of Neptune, and a rocket would probably take at least as long.

Light from Neptune takes four hours to reach the earth. Light from the nearest star takes four years. A rocket could hardly make the interstellar trip in less than a quarter of a million years. And most of the stars are infinitely more remote.

The question of temperature also tends to discourage space travel beyond Mars and Venus. The earth receives heat from the sun at the rate of 427.2 B.T.U. per square foot per hour. At any other distance from the sun the amount is greater or less, being inversely proportional to the square of the distance. Any surface also radiates heat off into space at a rate which is proportional to the fourth power of the absolute temperature. Putting these equal and solving for T we get:

$$T = 707 \frac{R_e^{1/2} (1-a)^{1/4} K^{1/4}}{R^{1/2} E^{1/4}}$$

in which T is the absolute temperature, 460° more than the Fahrenheit temperature, R is the distance from the sun, R_e is the earth's distance, or 93,000,000 miles, E is the emissivity of the surface, a is its albedo, and K is an exposure factor.

K is 1 for a surface normal to the sun's rays. It is the cosine of the latitude for a horizontal surface at noon during the equinox. It is divided by π to give the mean daily temperature of a horizontal surface, and is 1/4 for the average temperature of a perfectly conducting sphere.

Figure 2 is a chart based on this equation, showing the temperatures to be expected at different distances from the sun. Both curves are for black bodies (E = 1 and a = 0). The upper curve shows maximum temperature obtainable from unfocussed sunlight. The lower shows the average daily temperature at a planet's equator.

This chart shows a theoretical average of 70° F at the earth's equator. Weather Bureau records at Panama give an average for the year of 80°. This is a very close check, in view of the approximations involved. The same equation gives a theoretical temperature of —3° F at Edmonton, Alberta, at the equinox, and —6° at the poles of Mars in their summer. But the actual temperature at Edmonton at even the spring equinox is generally above freezing, and so, apparently, is the summer temperature at the poles of Mars.

The reason in both cases is that large areas covered with snow and ice prevent the dew point from rising above the freezing point. But the dew point approaches the freezing point very closely when the snow is melting in the sun. The result is a heat trapping effect, with clear skies by day and mist or clouds by night. Thus temperatures are maintained in which vegetation can grow, although theory gives a much lower temperature. And this heat trapping effect seems to be necessary to permit the growth of vegetation all over Mars, and not just at the poles. Mars appears to be a rather cold world, but one in which human beings could learn to live. (3)

Other studies we are making of Mars suggest that it probably has a high degree of civilization which could teach us a thing or two about getting along together in one world instead of quarrelling and fighting all the time.

We know very little about Venus, whose surface is permanently hidden by cloud. Reasonable guesses concerning it suggest that it has two polar regions which would be quite comfortable for human beings, and a tropical zone between them which is much too hot for any form of life we can conceive.

If we went still farther out into space we would find the heat absorbers on our space ship having a temperature up to 247° F as we left the earth. By the time we reached the orbit of Mars they would be at 113°. As we passed Jupiter their temperature would be —150°. At Neptune the air in the rocket would be turning liquid. At Pluto it would be solid.

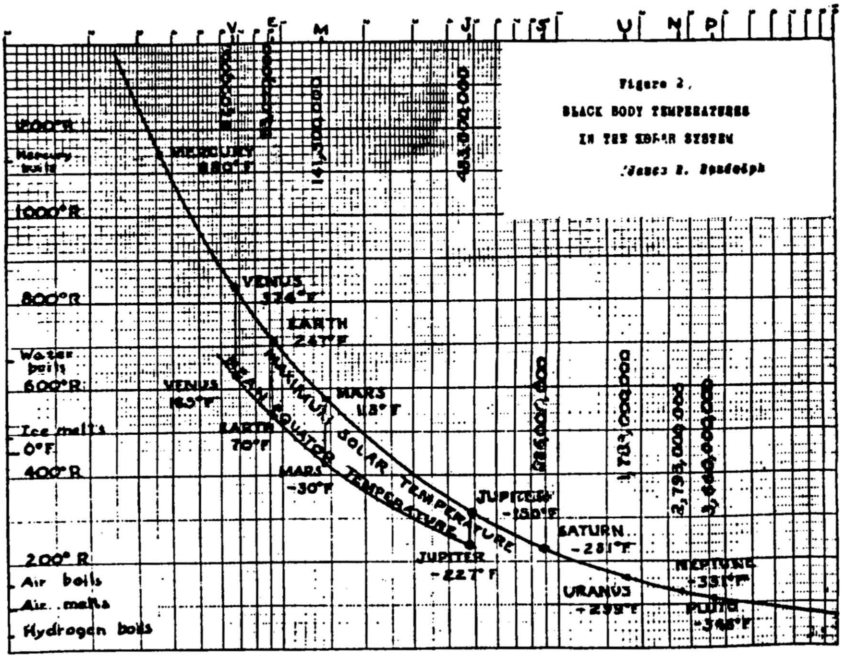


Figure 2,
BLACK BODY TEMPERATURES
IN THE SOLAR SYSTEM
James R. Randolph

For a longer journey the occupants of the space ship would have to freeze up solid and remain in a frozen sleep for millions of years until the approach of another sun warmed them into life, and they looked around at a strange family of planets, picked the one that looked habitable, and came down.

References:

(1) Rockets, by Robert H. Goddard (American Rocket Society).

(2) Occupation of Mars, James R. Randolph. Army Ordinance, Mar.-April 1947.

(3) The Climate of Mars, James R. Randolph. Journal of the American Rocket Society, June 1947.

Navy After Burner To Give Jets Added Boost

A workable thrust-augmentation technique for jet engines, known as an "After Burner", has been developed for the Navy by Ryan Aeronautical Company and is ready for actual flight test.

The technique, which will increase the boost of a jet engine by slightly more than one-third its normal power, will be used on military aircraft in a similar manner to "water injection" on conventional engines, which provides a tremendous burst of power for a short period of time. However, unlike the water injection system, the increase in power is obtained with little additional strain on the critical moving parts of the jet engine.

Perfect combustion of gasoline is obtained with an air-fuel ratio of approximately 16-1. Jet engines normally mix air and fuel at a ratio of approximately 50-1. Therefore, after the normal combustion stage in the jet engine, oxygen is still available in the exhaust gases to support the burning of additional fuel, which in turn increases the final output of power.

To accomplish this secondary burning, fuel is injected into the tail pipe of the jet engine. It burns, adds mass and velocity to the already highly compressed stream of hot gasses, and increases final thrust through the tail pipe without additional strain on compressor or turbine.

Expansion Time Rate of Gases

By W. D. MUNROE

INTRODUCTION

The purpose of this paper is to develop the mathematics and some of the uses for a new property of gases, their time rate of expansion.

It has been assumed throughout that the reader is thoroughly familiar with the more common fluid flow equations and, hence, such equations have been presented without prior proof or explanation. For those who are not familiar with these equations, chapter VIII of Everett's "Thermodynamics" will furnish an excellent background.*

This paper is presented without benefit of prior experimental proof due to the unavailability of the proper technical aid and equipment.

THEORY

In figure I, imagine that you have a spherical unit mass of gas enclosed in a perfect heat envelope "A", which is in turn enclosed in perfect heat envelope "B". Now, suppose that "A" is instantly dissolved allowing the unit mass of gas to expand and occupy all of the space enclosed by "B". By inspection it is seen that molecule #8 has a much greater distance (8-8') to go in order to occupy its new position than does molecule #1 (1-1') **IN THE SAME TIME.**

This being true, the average velocity of molecule #8 must be somewhat greater than that of molecule #1. Therefore the velocity calculated by normal means:

(1) $V — 223.8\ (\text{delta } H)\ ^1/_2.$

where: V — velocity of expanding gas.

223.8 — (2 x acceleration of gravity x mechanical equivalent of heat) $^1/_2$.

delta H — change in enthalapy during expansion.

must represent the velocity of some average molecule and not the velocity of all the molecules in the expanding mass.

By inspection of figure II, it is seen that molecule #1 must move N distance in order to occupy its new position, #2 must move 2N, #3 must move 3N, and so on. (The value depending directly on the difference between the radius of "B" and "A", and inversely as the number of molecules along a radius of "A" prior to expansion.) Therefore, the distances the various molecules must move in order to occupy their new position after expansion vary as an arithmetical progression. Hence the molecule occupying the arithmetical mean position between the maximum and minimum distances moved, at the instant that molecule #8 touches "B" in expansion, must be the average molecule whose velocity is represented by:

$$V — 223.8\ (\text{delta } H)\ ^1/_2$$

Now, the maximum distance moved is represented by the radius of "B" less the radius of "A" and the minimum distance moved must be zero. This being true the average distance moved is:

$$(2)\quad D — \frac{(R_b — R_a) + O}{2}$$

$$\text{or}\quad (R_b — R_a)/2$$

* "THERMODYNAMICS" by H. A. Everett, 2nd Edition, D. Van Nostrand Company Inc., New York.

where: D — average distance moved by molecules in expansion
R_b — radius of envelope "B"
R_a — radius of envelope "A"

FORMULAS

Knowing these things we may now proceed to develop the formula for the time rate of expansion of gases for special cases, i.e., a particular mass of gas expanding from one definite set of conditions to another, using the following symbols as indicated:

1. delta H—change in enthalapy during expansion in BTUs.
2. W—work of expansion on delta H times the mechanical equivalent of heat.
3. D—average distance through which the molecules move in expansion as found by equation (2) above.
4. F—average force expended on the molecules throughout expansion.
5. R_a—radius of initial volume.
6. R_b—radius of final volume.
7. M—mass of expanding gas.
8. W—weight of expanding gas.
9. g—acceleration of gravity or 32.2 ft/sec/sec.
10. A—average acceleration of molecules in expansion.
11. T—time required for gas to expand.
12. E_{tr}—time rate of expansion of gas (Ft^3/sec).
13. V—average velocity of expanding gas as calculated in eq. (1).
14. V_a—volume of gas at initial conditions.
15. V_b—volume of gas after expansion.
16. S—rational distance from mouth to diameter in question of a deLaval nozzle.

3)
$$W — F\ (D)$$
$$F — W/D$$
$$F — \text{delta } H(777.8)/(R_b — R_a)/2$$
$$F — \frac{2 \text{ delta } H(777.3)}{R_b — Ra_a}$$

4)
$$F — MA$$
$$F — (W/g)\ (A)$$
$$A — (Fg)/W$$
$$A — \frac{2 \text{ delta } H\ (777.8)\ (32.2)}{W\ (R_b — R_a)}$$

5)
$$T — V/A$$
$$T — \frac{223.8\ (\text{delta } H)^{1/2}}{2(32.2)\ (777.3) \text{ delta } H/W\ (R_b — R_a)}$$
$$T — \frac{223.8(W)\ (R_b — R_a)}{2(32.2)\ (777.3)\ (\text{delta } H)^{1/2}}$$

6)
$$Etr — V_b — V_a/T$$
$$Etr — \frac{(V_b — V_a)\ (2)\ (32.2)\ (777.3)\ (\text{delta } H)^{1/2}}{223.8\ (W)\ (R_b — R_a)}$$

APPLICATIONS

Now that we have established how to acertain the time rate of expansion of gas for a special case, let us proceed to examine some of the possible applications of this new property.

The first and most obvious application is in the determination of rational lengths for deLaval nozzles, and, coupled with flame velocities, in the proper design of the combustion-nozzle type engine as used in the German V-2 (A-4).

To properly determine the dimensions of a deLaval nozzle, calculate the velocities and diameters for a series of points along the axis of the nozzle, as usual. Then to find the rational distance from mouth to diameter in question, substitute in formula (7) below:

(7) $S—\frac{1}{2}(A)(T)^2$

S—distance from mouth to diameter in question

A—From (4)

T—From (5)

substituting and cancelling:

$S—W(R_b — R_a)/2$

Except for general interest in comparing results with the more common methods, it is doubtful that this method of rational proportioning of deLaval nozzles or some modified form for proportioning combustion-nozzles, will come into general use. This is true because the present empirical methods give results well within experimental error of calculated exhaust velocities.

Another use of great interest would be in the comparison of various gases for feasible use in both closed and open cycle engines. The gas, all other things being equal, having the greater E_{tr} per BTU expended is the most desireable gas to use.

A true horsepower rating of jet engines has never been obtained because of lack of a time reference upon which to calculate an engine's output. This has resulted in the all too complicated method of rating engines as having so many horsepower at such and such a velocity. The E_{tr} theory eliminates this. Knowing the volume of the working area of the engine and being able to calculate the average time rate of expansion of the gas from initial to exhaust conditions, the time required for the mass of gas flowing to deliver its thrust may be calculated.

(8) $T_{HP}—V_e/E_{tr}$

T_{HP}—time reference in seconds for horsepower

V_e—volume of working aera of engine

E_{tr}—average expansion time rate of gas from initial to final conditions.

In addition to the three mentioned above, there is no doubt that many other applications will appear as the thermodynamists of the nation begin to probe the theory for its hidden worth.

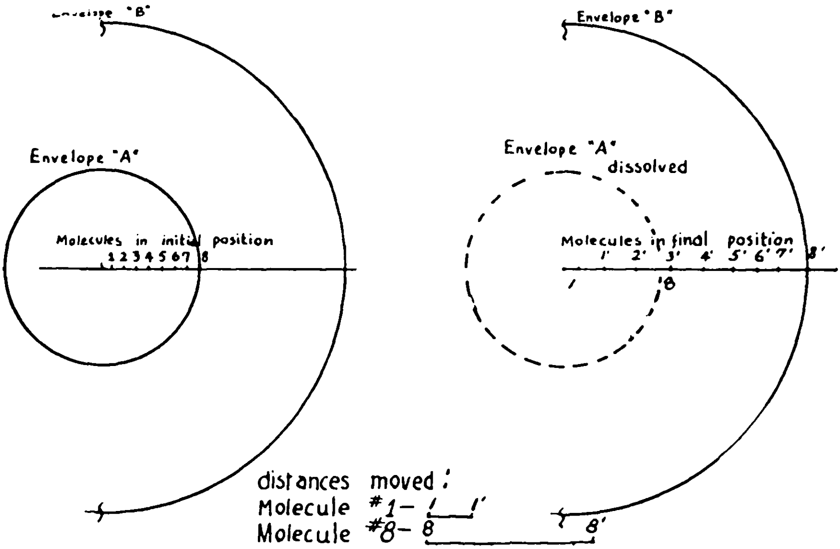


FIGURE 1

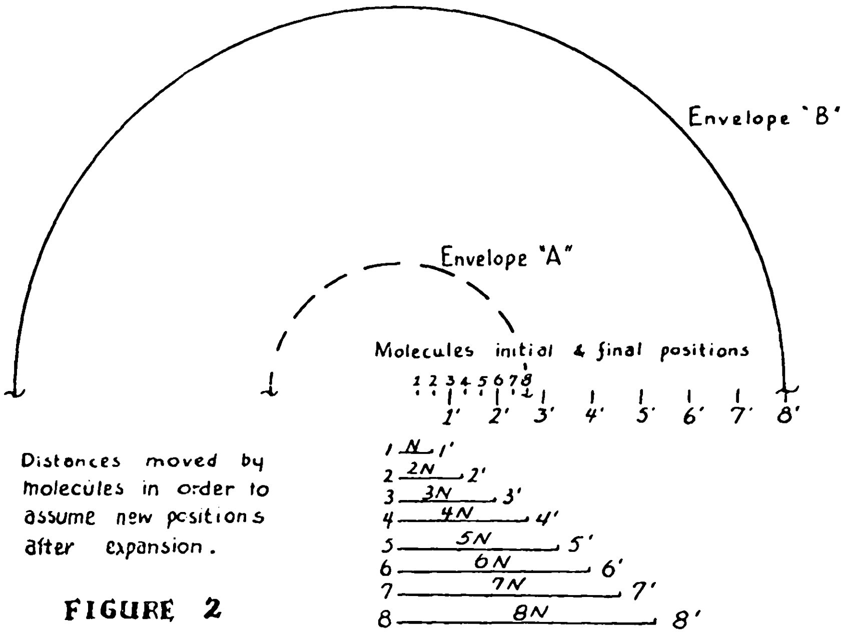


FIGURE 2

The Story Of The Nene

Pratt & Whitney Aircraft recently acquired the American rights to the Rolls-Royce Nene engine, most powerful turbojet engine in production in the world.

Rolls-Royce, in England, first took an interest in gas turbine development in 1938 and before the outbreak of war some design projects had been considered. Interest in gas turbine development during the early part of the war was maintained by the manufacture of various components for the producers of Air Commodore Whittle's jet-propulsion engine. The assistance thus rendered grew and, by 1941, test equipment was installed for the development of centrifugal compressors, work for which Rolls-Royce was particularly suited on account of its successful development of the high speed centrifugal supercharger.

At the end of 1941 Rolls-Royce undertook the manufacture of jet-propulsion engines based on the Whittle design. The first Rolls-Royce jet engine was known as the W.R.1, only two of which were made. These engines were produced for experimental test purposes and were designed with low turbine blade stresses, i.e. a comparatively big engine for a given thrust. The diameter of the W.R.1, was 54 inches, and the design thrust 2,000 pounds. It weighed only 1,100 pounds and the first engine ran for some 35 hours until trouble was experienced with the combustion chambers and turbine blades.

NEW MATERIALS DEVELOPED

The restricting factor on development at this time was the construction of the turbine blades due to limitations imposed by blade temperature and speed. The rapid progress which has since been made is a direct result of the materials which have been developed in the past few years.

By 1942, Rolls-Royce was going all out for the gas turbine engine and, favouring the double-sized impeller, their jet engine had achieved performance figures which were better than any other manufacturer's. On the score of reliability, Rolls-Royce was also making excellent progress and, by 1943, the R. R. Welland engine had passed its 100-hour type test. The Welland was 43 inches in diameter and gave a thrust of 1,600 pounds for a weight of 850 pounds. The Welland was the first of the Rolls-Royce "River Class" series of engines, each engine type in this series being named after a river to express the idea of flow, associated with jet-propulsion.

Production deliveries of the Welland began in May, 1944, and at this time the engine also passed its first 500-hour Air Ministry type test and went into service with 180 hours between overhauls.

Meanwhile, a new engine was being designed, known as the R.B. 38, and subsequently called the Derwent after the river which flows through the town of Derby, the home of Rolls-Royce.

Design work on the first Derwent started in 1942 and was completed in three and a half months; by the end of its first design year it had passed a 100-hour type test at a thrust rating of 2,000 pounds.

The Derwent I was followed by the series II engine, which gave a ten per cent improvement in thrust, delivering 2,200 pounds. The series III was an experimental engine to provide suction on the wing surfaces for boundary layer removal. Series IV gave another ten per cent increase in power, raising the thrust to 2,400 pounds, and the series V engines, which are fitted in the famous Gloster "Meteor", holder of the world speed record and used in large numbers by the R.A.F., have a thrust of 3,500 pounds and an overhaul period of 180 hours. The lower-powered Derwent I engine has achieved an overhaul period of 270 hours, which is much greater than any other jet engine in service.

NENE DESIGNED IN 1944

Between the design of the first Derwent and the Derwent V an even more powerful engine had been produced by Rolls-Royce; this was the famous "Nene"—the most powerful jet-propulsion engine which has yet flown.

The Nene was designed early in 1944 to conform to a design specification issued by the Ministry of Aircraft Production for an engine with a minimum thrust of 4,000 pounds and a weight not exceeding 2,200 pounds and having a maximum overall diameter of 55 inches. In general layout the Nene resembles the Derwent and the scaling up of a Derwent to meet the specification was considered. To achieve the 4,000 pound thrust specified would, however, have necessitated increasing the maximum diameter to nearly 60 inches, and a new design was therefore put in hand, which kept the over-all diameter down to 49½ inches. The new design was completed in the remarkably short period of 5½ months and on the proving run of the first engine a thrust of 5,000 pounds was successfully achieved.

The Nene is a pure jet-propulsion engine with a single-stage double-entry centrifugal compressor delivering air under pressure into nine straight-flow combustion chambers. There the burning process expands the gases, which, in exhausting rearwards, drive a single-stage, axial-flow turbine. The gases then eject rearwards past the exhaust-cone and through the jet-pipe. The reaction to the enormous increase in the momentum of these exhaust gases produces the thrust on the aeroplane. Once started, the combustion and functioning of the engine are continuous. No electrical ignition system is necessary except for starting. Engine speed and thrust are varied by regulating the supply of fuel to the combustion chambers.

DESIGN OF THE NENE

From the very first Rolls-Royce has favoured the double-sided impeller for the very important reason that the thrust which the jet engine can develop is determined by the amount of air which it consumes. It has therefore been the aim of Rolls-Royce designers to design an engine which, for a given frontal area, will consume the maximum quantity of air. To do this the central intake of the impeller must be as large as possible, and obviously a double-sided impeller with two central intakes can, for given dimensions, consume twice the quantity of air that the single impeller can consume. It is also important in the designing of jet engines to corelate the dimensions. In other words, having fixed the diameter of the eye of the impeller, then the diameter of the impeller is also fixed, and also the diameter of the engine.

For the same design criterion, therefore, the trouble-sided impeller will be 40 per cent smaller in engine diameter than the single-sided impeller for the same air consumed and the same thrust. It is, of course, necessary to leave a clearance round the blower casing when installed in the aeroplane to allow the air to reach the rear side of the engine, but this clearance need only be small, and in fact, in the case of the Nene engine it does not amount to more than about 2 inches on the overall diameter. It naturally follows that, if the dimensions of the engine are smaller for a given thrust, then its weight will be very much less, because the weight of any type of engine varies roughly as the cube of the linear dimensions.

There are other mechanical reasons which led Rolls-Royce to adopt and pursue the double-sided centrifugal impeller. One of the most powerful of these reasons is the fact that the double-sided impeller leads to a balanced turbine design. It is the object of any aero-engine designer to utilize all his materials to the maximum permissible stress consistent with mechanical reliability. With a double-sided impeller, twice as much air has to pass through the turbine blades and consequently the turbine blades have to be longer and the turbine discs smaller. Hence, the stresses in the turbine discs and the turbine blades tend to be equal and lead to a well balanced design.

NEW HIGHS IN EFFICIENCY

A further advantage derived from the adoption of the double-sided impeller is the fact that this type of compressor lends itself to proportions more favourable to high adiabatic efficiency; for example, the Nene has a working adiabatic efficiency of 81 per cent at a compression ratio of 4.3 to 1 which is the highest efficiency yet achieved by a centrifugal compressor.

The extreme importance of high component efficiencies will be readily appreciated when it is realized that it requires 10,000 horsepower to drive the compressor at ground level. As a result of this high compressor efficiency the Nene also shows remarkably good specific consumptions, development engines achieving 1.00 pound fuel/hour/pound thrust, whilst the average production figure is 1.05 pound fuel/hour/pound thrust. This again is the lowest specific consumption figure recorded for any production jet-propulsion engine.

It is fundamental that specific consumptions improve with altitude and it is also true to say that engines showing the best efficiency on the test bed for a given air intake efficiency also show the best figures in flight.

PROPERTIES OF HIGH TEMPERATURE MATERIAL

Material	Ultimate Strength Tons, Sq.Inch				Creep Strength for Fracture in 300 hrs. at 700°C.	Elongation at Fracture (Percentage)
	500°C. (932°F.)	600°C. (1112°F.)	700°C. (1292°F.)	800°C. (1472°F.)		
Rex 78	33.5	31.5	26.5	19.5	10 tons/sq.in.	7
Jesscps G. 18B	40	38	28	22.5	13 tons/sq.in.	10
Mond Nickel Nimonic 80	56.5	50	42	34	15.5 tons/sq.in.	2.5
Stayblade	29.5	22	14	--	--	--

Although production engines are rated at 5,000 pounds thrust, the Nene engine is still capable of great development and it is confidently expected that the eventual thrust rating will be much higher. The best thrust figure so far recorded on the test beds is 5,850 pounds, development engines averaging 5,150 pounds thrust. This figure has been used to measure the efficiency of the engine cycle and some extremely good results have been obtained, of which the following are examples: Compressor Efficiency, 81 per cent; Combustion Efficiency, 93 per cent; Expansion Efficiency, 93 per cent.

METALLURGICAL PROBLEMS

In the development of materials suitable for the turbine disc and blades, Rolls-Royce has, in addition to its own laboratory research, collaborated with other manufacturers in producing materials capable of withstanding the high temperatures and stresses of the modern gas turbine unit. The accompanying table shows some of the properties of the Jessop steel known as G.18.B, specially developed for turbine discs, and of the Mond Nickel alloy Nimonic 80, specially developed for turbine blades. The properties are compared with those of Stayblade and Rex 78 and the great improvement in ultimate tensile strength is clearly demonstrated.

The creep properties for fracture in 300 hours at 700°C. are also shown, and it will be noticed that Nimonic 80 only elongates before fracture a matter of 2½ per cent. This is no disadvantage since turbine blade tip clearance must be kept small to avoid losses, and consequently, too much elongation under creep conditions cannot be tolerated. In a well-designed turbine disc the creep can be confined to the root fixing of the blades only, and thus a considerable elongation before fracture can be permitted and is, in fact, desirable in order to avoid cracks in the roots while the plastic distortion is equalizing stresses.

MECHANICAL DEVELOPMENT

Development in this connection has been concentrated on improving the reliability and overhaul life of the engine at the 5,000-pound rating, most of the work being based on a series of rigorous 100-hour tests. Special tests have also been devised to simulate more nearly flight conditions and check the effect on combustion chamber life and performance.

One of the principal limiting factors in the early stages of development was the unreliability of the combustion chamber and a considerable amount of research has been done both as regards mechanical and thermo-dynamic characteristics for high altitude work. Combustion ware failed at less than 20 hours, but as a result of intensive development the reliability has been enormously improved and a 150-hour test can now be completed without any attention to the combustion chamber.

NENE-POWERED AIRCRAFT

The American interceptor fighter X-P80, or Shooting Star, was the first aircraft to be powered by the Nene and resulted in a considerable improvement in the performance of this machine. A later development was the fitting of Nene engines in the outboard positions of an Avro Lancastrian which rapidly became famous as the first passenger-carrying machine to be powered by jet engines. The installation of the Nene engines into the Lancastrian was an improvisation, made necessary to obtain flight experience of the engine as there were no military aircraft then designed capable of absorbing its enormous power.

The Nene/Lancastrian has since been demonstrated at air displays in England and France and the many hundreds of passengers, who have now flown in the machine, have been greatly impressed with the power and absence of vibration. Over 150 hours' flying has been done with the Nene/Lancastrian by the Rolls-Royce Flight Development Section, and valuable experience has been gained in operating the engines at varying conditions at high altitudes.

Military aircraft powered by the Nene include the Attacker intercepter fighter, manufactured by Vickers Armstrong, which is in the 600 m.p.h. class, and the deHavilland Vampire II which has achieved a 30 per cent increase in climb performance as a result of the additional thrust from the Nene engine A Vampire aircraft, powered by a Nene engine has also established an unofficial height record for a jet-propelled machine by climbing to 51,000 ft.

There are several high performance prototype aircraft under construction which will be capable of absorbing the enormous thrust from Britain's most powerful jet-propulsion engine, and until these are actually flying in the near future, it is not permitted to reveal any further information concerning them.

The Nene engine follows the usual Rolls-Royce tradition of advanced design at the outset, precision and high standards of workmanship in production, and an assured future of unsurpassed technical development. It is far in advance of any other proved unit in production anywhere in the world and is the outcome of over 25,000 hours' running of jet engines by Rolls-Royce, which is a record figure for research and development in the field of jet-propulsion.

Guided Missile Rocket Power Plant Design And Installation Problems

By ALFRED K. HUSE

Sr. Project Engineer, Reaction Motors, Inc.

Talk given at the 23 May 1947 Meeting of the American Rocket Society.

At the present time all topics on Guided Missile experimental development or production is on an Army or Navy classified status and the only projects that can be discussed are those that appear in current magazines. However, without violating any regulations, some of the guided missile power plant design and installation problems may be revealed.

A guided Missile may be most quickly pictured in everyone's mind by one letter of the alphabet and a couple of digits. Namely V-1 and V-2, the buzz bomb and the rocket bomb.

In a broader sense a Guided Missile is a pilotless device that flies thru the air or travels thru the water under supervision of an operator remote from the device.

The powerplant is not necessarily a pulse jet or rocket. It may be a reciprocating engine or gas turbine with a propeller.

In the 19 May 1947 issue of Aviation News, the following comment was at the top of a column titled, "Industry Observer". "Watch for propeller driven plane to appear soon in the joint AAF-Navy-NACA transonic research program. There is still hope for propellers to negotiate the transonic zone and the AAF is intensely interested in the development of an extremely long range propeller-driven guided missile."

A guided missile need not carry an explosive charge. It may carry scientific equipment. Its guidance need not be controlled by radar or radio. It may be controlled by wires and still be a guided missile.

Guided Missiles at present are classified by their military use into two groupings. The first is the offensive type. The second is the defensive type. The offensive type has the following flight and physical characteristics:

(a) Long range
(b) Heavy
(c) High speed is not essential on the offensive type
(d) Fast maneuvering is not important

The defensive type is characterized by just the opposite features:

(a) Short range
(b) Light weight
(c) High Speed
(d) Very maneuverable

Examples of the offensive types are the V-2 and we may consider as an offensive type the latest government release, the Neptune, manufactured by Glenn L. Martin Co. and powered by a Reaction Motors, Inc. rocket engine.

The V-2 is 75 feet long, 5'5" in diameter and weighs 14 tons at take-off. Its propellants are liquid oxygen and alcohol and guidance is done by radio control. Photographs were taken by cameras in a V-2 at a height of 102 miles on 7 March 1947, as you may recall. The highest V-2 flight that I know of is 116 miles.

Compare the V-2 with Neptune. The Neptune, it is expected, will be able to photograph the earth at a height of 235 miles. It is 45'3" long—has a diameter of 32", and weighs 11,410 pounds. Of this weight, 7000 pounds is propellant and the rest is tanks, equipment and frame. The powerplant weight is insignificant, amounting to only 250 pounds; yet the thrust produced will produce an average of 3,746 '/sec. It will take 335 seconds to reach the maximum height of 235 miles. This maximum height can be obtained if 100 pounds of equipment is carried along. If 200 pounds is carried, this height is reduced to 200 miles. With a 2000 pound load, the height is reduced to 84 miles.

The release, brief as it was, stated one interesting highlight which I would like to quote: "The Neptune introduces a number of new features in its propulsion and steering units." That statement simple as it appears to the average laymen, is the first indication by the Navy of an entirely new development in Rocket Powerplant design and may be the subject of an interesting talk in the future.

An example of the defensive type is the Tiamat. A Navy release on 23 December 1946, by NACA, told of a rocket powered, guided missile called the "Tiamat". It stated that the Tiamat was the first guided missile to be flown thru a predetermined program of maneuvers.

In Babylon religion, Tiamat was the Goddess of the ocean or marine waters and the wife of Apsu. Her emblem was a winged dragon. Now you need plenty of luck in the guided missile development field. I believe the AAF decided naming the missile "Tiamat" would make the problem of getting her to fly successfully thru a predetermined program of maneuvers would be less difficult. This female, I'm sorry to relate, is not built like Salome. She is 14'4" long, and weighs 600 pounds.

As a winged dragon, the Tiamat is

capable of approximately 600 miles per hour. As a guided missile, it was powered by a 200 pound thrust rocket engine. It is launched by a 7000 pound thrust rocket booster assembly mounted on its tail. After 3½ seconds the booster rocket is expended and the entire booster assembly falls away leaving the guided missile free for the remainder of its flight.

In order to eliminate the effect of the drag of the launching booster assembly on the course of the guided missile during the separation, special precautions must be taken.

Positive separation can be accomplished with the use of Rocket Power. The rocket power thrust being applied to the launching ramp is in a direction opposite to the thrust being produced by the missile.

POWER PLANT DESIGN AND INSTALLATION PROBLEMS

The first problem is to design the powerplant to fit the installation. If space was reserved for the powerplant in a missile layout, the design of the powerplant would be relatively simple. The rocket engine unfortunately is not yet respected and receives no more consideration than the lowly reciprocating engine in an aircraft installation. It is crowded by control devices and the usual gadgets until every available inch of space is utilized. Any change in design of the powerplant results in a major rework of the missile. This condition common to airplane installations as well as guided missiles makes powerplant design rather exacting. The quality required because of close clearances is far from that which you or I would consider satisfactory considering nothing can be found of the missile after its first, last and only flight.

Once the powerplant is desgined to fit the missile, the problem then is to attach the propellant lines from the tankage to the propellant inlets on the powerplant. Propellant lines in general should be capable of withstanding about 1000 pounds pressure, especially if acid is one of the propellants being used. Stainless steel tubing will withstand the pressure and is acid resistant; however, it is difficult to bend or flare and will not stretch or give to meet the connections in the guided missile. Stainless steel flexible hose lines could be used to alleviate the problem of meeting connections; however, it too had disadvantages.

Flexible hose lines are costly. High pressure acid resistant flexible lines are difficult to obtain. Changes in length or formed bends cannot readily be made as cheaply as can be done with tubing.

THRUST REQUIREMENTS

Powerplants are designed around the thrust requirements of the missile in question. The thrust requirements for level flight may be but a fraction of that required for a steep climb particularly if the guided missile is to travel at a constat high speed with little or no variation in that speed.

The methods of supplying the thrust requirements can be solved in several ways:

(a) by varying tank pressure—this method, however, is rather unsatisfactory for a defensive missile. If a rapid change in thrust is required there is insufficient time for the propellants to be expanded to allow for the expansion of the gases to a lower pressure in the tank. Therefore, varying the tank pressure is slow and the thrust from the powerplant would vary accordingly.

(b) by having the powerplant consist of a multiple of small engines. The advantages are twofold:

1. Each motor operates at its maximum specific impulse and efficiency.

2. Thrust can be obtained rapidly upon demand by starting or stopping the number of engines required to give the desired thrust.

(c) A variable thrust engine. The design will differ depending upon whether it will be used in a pressurized system or in a pump system. These systems require powerplants with different starter designs. Starting with a pressurized system, the compressed air which is stored in a tank provides the pressure to deliver the propellants to the powerplant. Designing a powerplant with satisfactory starting characteristics, is, for the above reason, not too difficult. The design of a powerplant for operation with a pump is a little more complicated as there is the added problem of getting the pump started.

Once the pump starts, propellants are fed to the powerplant and starting proceeds as before. The use of a pump adds a few more lines, namely, those from tankage to the pump and from the pump to the powerplant which take up precious space. To give you some idea of what an installation looks like, just imagine looking into a tin can full of rain worms.

I am restricted from mentioning the possible powerplant designs as far as varying thrust is concerned; however, Mr. Burdett indicated a possible solution that could be used, in his article titled: "The Variable Nozzle as a Means of Maintaining Engine Efficiency when Throttling".

Most guided missiles are designed to have sufficient thrust to maintain flight at some speed say 400 miles per hour, but auxiliary power or a booster is required to obtain this initial speed.

To obtain this initial speed a catapult or launching ramp may be used. As noticed previously on the Tiamat, a booster rocket is used which has considerable thrust for a few seconds duration.

The powerplant of this guided missile must therefore be designed so that each part can withstand a force of 20 times its own weight without affecting the operation.

The maneuvers of a guided missile also affect the powerplant design. The guided missile may be directed to fly upside down. In this case you can readily see that the powerplant valving and controls must be capable of operating in the inverted position. I may add another problem applicable to guided missiles and common to airplanes, is that of keeping the engine supplied with fuel and oil during maneuvers. This problem, also, had to be solved for the guided missile powerplant. The problem, of course, is to keep the pump, or a pump version, from cavitating. On a pressurized unit the problem is to keep the pressurizing air from entering the propellant feed lines to the powerplant.

Powerplant design is affected seriously by the fact that maximum range is desired of the guided missile.

A powerplant is sometimes designed and used in many different installations. Peak operation and efficiency of a powerplant is attained usually in a definite range of propellant inlet pressure. Since line losses from tanks to the engine propellant inlets vary with different installations, a rocket motor is usually designed to operate with equal pressures on both the fuel and oxidizer propellants as they enter the powerplant.

The powerplant is therefore designed and guaranteed to operate within specified limits of thrust, specific impulse and mixture ratio, only when propellants are supplied to the inlets of the powerplant with the propellent pressure equal. Since maximum range is desired guided missile tankage is designed on the basis of the powerplant mixture ratio. This may result in one propellant tank being five times as great at the other where the mixture ratio of oxidizer to fuel is 5:1. The propellant flow from the acid tank during operation will be in the same ratio of 5:1. This large flow from one tank, you can readily understand, will require much larger lines to supply the propellant from the tank to the powerplant.

Now we will enter the forbidding realms and mention a few of the disasterous effects of this simple fact on the powerplant.

EFFECTS OF UNEQUAL PRESSURE

Let us assume for the sake of discussion that the propellant lines have not been designed to supply equal pressures to the powerplant at the fuel and oxidizer inlets.

The maximum range of the guided missile is immediately reduced. Why? Because, the tanks were designed to run dry together when the engine operated at the basic design mixture ratio of 5:1. This mixture ratio was no longer 5:1 when unequal pressures were applied to the powerplant.

What are the possible effects on the powerplant?

(a) If due to unequal inlet pressures the tank containing the engine coolant runs dry first, the powerplant will burn out due to lack of coolant.

(b) The powerplant will experience a hard start due to incorrect mixture ratio. When using spontaneous combustibles for propellants, the mixture ratio for a satisfactory start is very critical. Unequal inlet pressures here cause havoc.

A hard start, you may well remember, is the result of ever so gently pushing the starting button on a rocket installation and having all those special devices that took so long to build disappear before your eyes.

Wan Hu is a synonymous name for hard start. The reason is that a Chinese Mandarian by the name of Wan Hu anxious to go from one province to another in a hurry, devised a fancy sedan chair with 47 rockets and had himself and his chair lifted by two large kites as boosters for the launching. 47 coolies lit the 47 rockets and everything disappeared. Dr. Pendray may not agree, but I feel Wan Hu and "hard start" should become synonymous in Rocket diction.

In a much broader sense powerplant design also depends upon whether the guided missile is to be subsonic or supersonic.

ADVANTAGES OF ROCKET ENGINES FOR GUIDED MISSILE INSTALLATION

Reaction Motors, Inc. and other companies in the powerplant field are well aware of all the different types of powerplant requirements and their installation problems in regards to guided missiles.

For subsonic guided missiles, rocket engines have certain advantages over other types of powerplants.

1. The propellants are dense resulting in a reduction in propellant volume.

2. If acid-aniline is used the propellants are spontaneously combustible and problems of storage and handling are simplified.

3. Missile volume is small.

4. Rocket engines are proven.

5. They are readily available.

6. They are inexpensive to build in production.

7. Their use results in light wing loading at the end of their flight which permits better guidance, maneuverability with resulting better target accuracy.

8. Since the engine is usually installed in the rear of the missile, and no ducting is required, the nose is left free for guidance equipment.

Compare these advantages with the disadvantages of other powerplants such as the boost required for a ram jet, the vibration of a pulse jet or the expense and complication of a turbo jet.

For the supersonic powerplant, rocket engines again are definitely superior.

1. Acid and Aniline are propellants that can be stored.

2. Problems of other types of powerplants requiring ducting are eliminated.

3. A rocket does not use air and therefore extreme altitudes are possible.

4. The greater efficiency at higher altitudes increase the range at higher altitudes.

5. The maximum Is available has not been reached for liquid propellant rockets. Improvements in Is (specific impulse) are expected.

6. The missile size as a result of using a rocket is small.

7. If acid-aniline is used, complicated ignition systems are eliminated as acid-aniline is spontaneous.

8. Production cost of rocket engines is low compared with turbine engines.

9. Due to density of propellants and size of engine, the missile size can be reduced.

10. A rocket powerplant is small and light.

11. The engines can be readily designed for a specific installation.

12. And last but not least, rocket engines are available and have been demonstrated to be practical.

Rocket engines answer most problems for guided missiles and it will be but a short time before the design will be simplified by their continued use.

MARTIN XB-48

Ground view of the six-jet Martin XB-48, newest Army Air Forces high speed jet bomber which made its initial flight Sunday from The Glenn L. Martin Company airport in Baltimore to the Patuxent River (Maryland) Naval Air Station, remaining aloft 37 minutes. Powered by six General Electric J-35 gas turbine engines housed three in each wing, the Martin XB-48 has a speed of over 480 miles per hour and carries a bomb load of more than 10 tons. It employs a new type "bicycle" landing gear, because of the difficulty of retracting heavy gears into the extremely thin wings required for high speeds. Two pairs of main wheels are located tandem-style under the fuselage, and two smaller "outrigger" wheels farther out under the wing, to give stability during ground operations. The large main gear folds into the fuselage and the smaller wheels retract into the wings.

BOOK REVIEW

The Interplanetary Travels by the Solar Radiations, by Luigi Gussalli. Translated from the Original Text by Jeanne Wiest. Giulio Vannini, Brescia, Italy, 1946. 92 pages, illus.

The Gussalli system is based on the throwing of spray-dust from rockets and the utilization of the pressure of solar light for the propulsion of a space vehicle. Rockets, such as the V2, when out of the terrestial atmosphere at an altitude of 120 km. (75 miles) would eject spray-dust by a mechanical means located in the fore section. The period of casting the spray-dust at the top of the trajectory would last only a few minutes.

It is stated that based on steam formulas a sq. meter cordon containing 1 mg. of spray-dust in each cubic meter when animated to the speed of 4 km. per sec. by solar radiation pressures would theoretically produce 42.6 h.p. A cross cordon system employing two or more spray-dust casting rockets is considered most efficient as additional cordons mean great pressures.

The push of the nebular cordon is to act on large surfaces mounted on the astroship. Adjustment of these bearing surfaces would control the pressure received and the space-ship's forward speed.

A grain of spray-dust with a diameter of 0.0002 mm. is deduced large enough for solar pressures to push against gravity. The pressure of solar rays on 1 sq. meter of flat absorbent surface is placed at 0.4 mg., and on 1 sq. meter of reflecting surface 0.8 mg., with the pressure figure of 1 mg. commonly used in calculations.

Dawn of the Space Age, by Harry Harper. Sampson Low, Marston & Co., Ltd., London, 1946; 142 pages, illus. 8s. 6d.

Following a foreword by Kenneth W. Gatland, co-founder of the Combined British Astronautical Societies, this non-technical presentation is divided into three parts, each being treated in turn by the author.

The Story of the Rocket, and the Coming of Rocket Power, gives the history and development of the rocket from earliest times up to the birth of the V2 in World War II.

The second part, Objectives of Space-Flight—The Moon, Mars and Venus, deals with conditions to be encountered on the planets and the problems possible of being solved through space flight.

Part three, Design and Construction of Vessels for the Navigation of Space, features the British Interplanetary Society's pre-war lunar space-ship design and discusses the investigations on proper instruments, equipment and supplies required for journeys into space. Future needs and possible developments are considered for the post-war era. A number of photographs and drawings illustrate the text.

Moon Rocket, by Arthur Wilcox. Thomas Nelson and Sons, Ltd., London, 1946; 161 pages, illus. 7s. 6d.

This semi-technical book discusses interplanetary travel and kindred subjects pertaining to a moon rocket. After a short history of rocketry, chapters consider earth escape velocity, speed and acceleration, fuels and motors, and gravity.

The British Interplanetary Society's space rocket, reported first in 1939, is described and is detailed in line drawings. The problems and conditions likely to be met in a trip to the moon and return are presented to the reader.

A number of the photographs and drawings pertain to A.R.S. tests and the Greenwood Lake mail rocket plane flight. The book has an introduction by Prof. A. M. Low, British scientist, editor and rocket enthusiast.

The Gravitational Reaction Motor, by Arnold G. Guthrie. Tacoma, Washington, Apr. 1947; 13 pages. $.25.

An interesting pamphlet on the theory of gravity and its many applications. The repulsion gravitation theory of Le Sage, and the various ideas advanced on gravity, electrical radiations, forces, and fields are described.

Chemical fuels are regarded as practically obsolete and emphasis is placed on discharging atomic particles at high velocities in an electric rocket motor design. A motor consisting of a linear accelerator and an ionizer is found the most possible type of motor for future interplanetary travel.

FICTION

The Best of Science Fiction, by Groff Conklin. Crown Publishers, New York, 1946; 785 pages. $3.00.

A collection of 40 stories printed in recent years in science fiction magazines. The contents of the book is divided into six parts: The Atom, The Wonders of Earth, The Superscience of Man, Dangerous Inventions, Adventures in Dimensions, and From Outer Space.

A preface by John W. Campbell, Jr., editor of Astounding Stories, acquaints the reader with the features and trends of science fiction.

Adventures in Time and Space, by Raymond J. Healy and J. Francis McComas. Random House, New York, 1946; 997 pages. $2.95.

An excellent collection of 34 modern science fiction stories, all except three taken from Astounding Stories, 1937-1945.

A non-fiction article on the V-2 by Willy Ley is included in the book.

—C. G.

Society News

The first meeting of the 1947-1948 season of the American Rocket Society was held on September 19 at the Engineering Societies Building. Major James R. Randolph of Pratt Institute gave a lecture on "Possibilities of Space Travel" to a group of about ninety members and friends of the Society.

Before Major Randolph presented his material, an election of members of the Board of Directors of the newly formed New York Section of the American Rocket Society was held. The following men were elected to the Board of Directors of that Section for a term ending December, 1948:

Henry K. Georgious, M. W. Kellogg Company.

Morton Gerla, Industro-Matic Corp.
Lawrence Heath, Reaction Motors, Inc.
George Messerly, M. W. Kellogg Co.

S. G. Nordlinger, Ranger Aircraft Engines.

Edward F. Schwartz, Wright Aeronautical Corp.

Charles A. Villiers, Grumann Aircraft Corp.

James Wheeler, Sperry Gyroscope Co.

James H. Wyld, Reaction Motors, Inc.

Meetings of the American Rocket Society for the remainder 1947-1948 season have been scheduled as follows:

November 20—Joint ASME-ARS meeting — January 16 — February 20 — March 19 — April 16 — May 21.

The December meeting will be the annual National Convention held during the first week of December in Atlantic City, N. J. in cooperation with the ASME.

Notices of individual programs will be sent to members as early as possible each month.

LEONARD AXELROD, Secretary

Tentative program for the American Rocket Society participation in the American Society of Mechanical Engineers Annual Convention at Atlantic City during the first week of December, 1947:

Wednesday 3 December

A.R.S. will co-sponsor, with Gas Turbine Power Div. of A.S.M.E. two technical sessions;

Morning — Technical Session — Joint with Fuels Division, A.R.S. and Oil & Power Division.
Chairman — A. E. Hershey
Recorder — H. E. Melton

1. "Ignition of Flame Stabilization Processes in Gasses"—Dr. Bernard Lewis—U. S. Bureau of Mines.
2. "High Output Combustion of Ethyl Alcohol and Air"
   A. H. Shapiro—Asst. Prof. of Mech. Engrg., M.I.T.
   D. Rush—Division of Industrial Cooperation, M.I.T.
   W. A. Reed—Research Associate Dept. of Chem. Engrg., M.I.T.
   D.G. Jordan—Division of Industrial Cooperation, M.I.T.
   G. Farnell—Ingersoll-Rand Co.
3. "The Influence of Reaction Interface Extension in the Combustion of Gaseous Fuel Constituents" — Prof. W. J. Wohlenberg, Yale University.

Afternoon—Technical Session — Joint with Fuels Division, A.R.S. and Oil & Power Division.
Chairman - R. V. Kleinschmidt
Recorder — J. M. Campbell

1. "The Concept of Gas Turbine Combustion Chamber Efficiency"—Prof. A. L. London—Stanford University.
2. "Temperature Measurements and Combustion Efficiency in Combustors for Gas Turbine Engines"—Dr. W. T. Olson—NACA Engine Lab.—Everett Bernardo.
3. "Determination of Gas Turbine Combustion Chamber Efficiency by Chemical Methods"—Peter Lloyd—National Gas Turbine Establishment, England.

Thursday 4 December

12 noon—The A.R.S. is negotiating with Aerojet Engineering Corp. for a demonstration of JATO on the "Navion" airplane, to be held between 12 noon and 1 P.M. at the Atlantic City Naval Air Station.

Annual ARS Dinner—Co-sponsored by

Gas Turbine Division ASME.

Negotiations are underway with the object of obtaining Dr. Clark B. Millikan, Jr., Director of GALCIT, as guest speaker. The suggested subject is "Ten Years of Rocket Research at GALCIT", and it is hoped that motion pictures and slides will also be shown.

Friday 5 December

9:30 A.M.—Gas Turbine & Aviation Divisions ASME Cooperating.

Chairman—H. Burdett, Reaction Motors, Inc.

Recorder—L. Axelrod, M. W. Kellogg Co.

1. "Rocket Development at the Naval Engineering Experimental Station" —R. C. Truax, Lt. Comdr., Bureau of Aeronautics, Wash., D. C.

2. "Pumps and Turbines for Rocket Engines"—Wm. P. Munger, Reaction Motors, Inc., Dover, N. J.

12:15 P.M.—ARS Luncheon—Co-sponsored by Gas Turbine Div. ASME.

Negotiations are underway with the U. S. Ordnance Department to permit Dr. Werner von Braun, Director of Germany's Peenemunde Research Center, to appear as speaker. The technical problems encountered in the development of the V-2 missile will be discussed if Dr. von Braun is available. If it should not be possible to obtain his services, or an alternate German research leader, the Ordnance Department will be asked to designate one of the outstanding American officers now participating in the White Sands tests.

2:15 P.M.—Gas Turbine—Power Div. ASME Cooperating.

1. "Supersonic Research Aircraft"—P. B. Klein, Colonel, AAF Chief Fighter Branch, Aircraft Projects Section, Eng. Div. AMC.

2. "Testing of Rocket Engines" — Robertson Youngquist, Glen L. Martin Company, Baltimore, Md.

3. "'Thermo-Chemistry of Rocket Propellants"—George P. Sutton, Aerophysics Laboratory, North American Aviation, Inc., Los Angeles, California.

Chairman — Roy Healy, A.R.S.

Recorder — Morton Gerla, Industro-Matic Corp., N. Y.

Week-long Display of Atomic Energy, Jet Propulsion and Rocket Equipment.

New Navy Fireball Powered By Larger Jet Engine

Continuing its experimental work in the field of combination-engine aircraft, the U. S. Navy announces its third prop-plus-jet fighter airplane, the XFR-4, built by the Ryan Aeronautical Company, which uses a combination of a reciprocating engine in the nose and a thermal jet engine in the tail for its well over 450 MPH top speed.

Because of the installation of a much more powerful jet engine in the tail, the new Ryan fighter exceeds considerably the performance of its predecessor, the FR-1 "Fireball". It has an extremely high and well sustained rate of climb, and its high altitude performance is considerably improved over the FIREBALL.

The earlier FIREBALL, developed during the latter part of the war and the only semi-jet fighter presently assigned to a carrier based fighter squadron of the Pacific Fleet, is equipped with a Wright conventional engine of 1300 horsepower in the nose, and a General Electric I-16 thermal jet engine in the tail. The I-16 delivers 1600 pounds of static thrust. Top performance on both engines is over 400 MPH.

Second in the Navy fighter series built by Ryan was the XF2R-1, an experimental model. In this plane, the original Wright engine was replaced by a General Electric prop-jet engine, the TG-100. Exceptional performance in climb, speed and altitude have been recorded.

In the XFR-4, also classed as a Navy experimental fighter, the Wright conventional engine is retained as the forward power plant, but the new Westinghouse 24C axial flow turbo-jet-replaces the smaller I-16 engine in the tail.

Also undergoing engineering evaluation on the XFR-4 is a flush entry duct system. When the rear power plant is not in use, its air intake ducts are closed by sliding doors, electrically operated, thereby decreasing drag for single-engine operation.

The XFR-4 still maintains the excellent take-off characteristics of the FIREBALL by virtue of greater propeller efficiency at low speeds. The increased jet boost in the tail, provided by the new 24 inch Westinghouse engine, accounts for an increase of almost 100 MPH in the top speed of the new plane.

NEW ALLISON JET ENGINE

The Lockheed P-80R which set a new world's speed mark of 623.8 MPH is powered with a new Allison turbo jet engine with a top rating of more than 4600 pounds thrust.

Designated the Allison Model 400, this jet engine is of the same general type as the production AAF J33-21 engine which powers production models of the Lockheed P-80 Shooting Star.

The take-off rating of the Model 400 makes it the highest powered jet engine ever manufactured in this country. Late this year Allison will put this model in production for a new production series of the Lockheed P-80.

Performance data on the new engine, all of which have been exceeded in test stand operation are:

Take-off power (dry) — 4600 pounds static thrust.

Military power — 4600 pounds static thrust.

Maximum continuous power — 3600 pounds static thrust.

Military power S.F.C. — 1.13 pounds/pound thrust/hour.

Maximum continuous power S.F.C. — 1.12 pounds/pound thrust/hour.

Cruise S.F.C. — 1.14 pounds/pound thrust/hour.

Engine weight — 1735 pounds.

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The Rocket Research of Dr. Robert H. Goddard

Paper by Alfred Africano* presented December 6, 1946, at the First Joint Annual Convention of the American Society of Mechanical Engineers and the American Rocket Society, Hotel Pennsylvania, N. Y.

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I. HISTORICAL

Except for the content of the confidential reports on his war research which are not yet declassified, the work of Dr. Robert H. Goddard, American physicist and world pioneer in rocket research can be followed with a fair amount of accuracy by reviewing his classic 1919 paper, "A Method of Reaching Extreme Altitude", his 1936 paper, "Liquid Fuel Rocket Development", and by studying his 69 Patents issued from 1914 to Oct. 8 of this year, which cover a wide variety of the difficult problems in this field.

The 1919 paper was the result of his early experiments with powder rockets which began shortly after his graduation from Worcester Polytechnic Institute in 1908, when he began to search for a method of sending recording equipment to altitudes beyond the 20 mile range of sounding balloons. By 1910 he had rejected the old black gunpowder used for firework rockets for the past thousand years, finding that the average velocity of ejection of the powder gases from the nozzle was only about 1000 ft. per sec. or 2% of the heat value of the powder mixture.

He next tried smokeless powder, using strong steel chambers, with throat diameters of about 1/8 and 1/4 in. and 8 degree divergent nozzles with various expansion ratios, and succeeded in attaining the surprisingly high effective gas velocity of about 8000 ft. per sec., a value not exceeded even by modern rocket projectiles, probably because these are fired at much lower pressures. Dr. Goddard's velocities were calculated from ballistic pendulum measurements and showed that about 50% of the heat energy of the smokeless powder was now being converted into the kinetic energy of the jet gases.

Satisfied with this result, Dr. Goddard then tackled the problem of proving that the rocket would work in the vacuum beyond the atmosphere as well as in air. Quoting his own modest words: "It remained to determine to what extent these velocities represented reaction against the air in the nozzle, or immediately beyond." He knew that the notion prevalent at that time—that the rocket gases had to have air to push against in order to create the reaction —was erroneous, but evidently he expected to find a slight effect so he very carefully set up the equipment for the "vacuo" experiments.

Without going into too much detail, the essential point was to have a large enough evacuated receiver tank so that the explosion and subsequent expansion of about one gram of smokeless powder would still leave a negligible final pressure in the tank. He measured the rocket reaction, or recoil, by direct displacement and also by the simple harmonic motion induced in a spring supporting the whole assembly inside the tank. He eliminated the gaseous rebound from the bottom of the tank by filling the lower half with coils of half inch square mesh wire fencing, and showed by means of a tissue paper detector that the slight impulses bouncing back to the rocket was negligible.

The results were even better than he had hoped for. They showed—not a decrease in the thrust and gas velocities but an actual increase of from 12 to 22% over the corresponding values when the rockets were fired in air. Thus, the reported increase of 16% in the thrust of the V-2 at the end of its burning period, 20 miles up, was predictable on the basis of these vacuo experiments. It would be of interest to repeat these experiments with low pressure rockets to get quantitative data on the effects of over and underexpansion, a subject which has received much theoretical study but not enough satisfactory experimental verification.

Combining the theoretical equations developed during the period of his fellowship at Princeton University in 1912 to 1913, and the results of the air and vacuo tests at Clark University made in 1915 to 1916, Dr. Goddard sent a report to the Smithsonian Institution. The Smithsonian became interested, alloted him small grants to assist in the research and finally published the 1919 paper. Interesting problems considered in this paper included besides the minimum mass required to send a lb. of magnesium to make a visible flash on the surface of the moon, the probability of a collision with meteors, which is small; parachute recovery of instruments; and the theory underlying his measuring equipment. The fundametnal equation of rocket motion given in this paper will be reviewed critically as the principal purpose of the present discussion, in the light of the V-2 data now known.

Before tackling that, let us finish sketching the highlights of Dr. Goddard's career. In 1918, he was asked to develop a military rocket and almost had a repeating bazooka ready when the armistice was signed and official interest in the research ended. Because of the severe criticism from his fellow scientists following publication of "A Method of Reaching Extreme Altitudes" (which we now know was entirely unwarranted) Dr. Goddard issued no further report on his progress until 1936 when the Smithsonian Institution published the results of his liquid fuel and liquid oxygen rocket tests. Most of these were made at the Mescalero Ranch in Rosewell, New Mexico, and supported from 1930 to 1932 and aqair from 1934 to 1935 by grants from the Carnegie Institution of Washington, Daniel Guggenheim, and the Daniel and Florence Guggenheim Foundation. This report, "Liquid Propellant Rocket Development" disclosed that the world's first liquid fuel rocket had been shot on March 16, 1926 at Auburn, Massachusetts. Since the first German Rocket Society experiments we heard of followed this historical date, it is reasonable to assume that they had heard about it and received their inspiration from Dr. Goddard's work.

Earlier experiments with liquid oxygen and various liquid hydrocarbons, such as ether and liquid propane, as well as gasoline had been made at Clark University in his spare time from 1920 to 1922. Since the most practical combination appeared to be liquid oxygen and gasoline, he concentrated his work on developing a motor using this mixture, attaining an average effective gas velocity of over 5000 ft. per second at the conclusion of the 1932 static tests.

On the resumption of the work in 1934 Goddard began the development of the gyroscopic control of deflecting vanes placed in the nozzle blast. This feature of rocket design together with the film cooling of the combustion chamber were the most important of the many details in the V-2 apparently copied wholesale from his work.

The motion picture of the flight tests will give you a better idea of these than I could so I will simply state that the maximum rocket velocity attained exceeded the speed of sound for the first time, establishing another record, with a maximum altitude of about a mile and a half, and a maximum range of about 2 miles and a half.

The 1936 progress report stated that further work would include reduction of the weight to a minimum. In accordance with this plan tests with light weight fuel pumps and gas driven turbines were made during the next few years as well as tests for continued development of the motor.

Now let's examine the 1919 paper.

II. DR. GODDARD'S MINIMUM MASS EQUATION

Dr. Goddard started out with what he terms an "ideal rocket amenable to theoretical treatment" shown in Figure 1. The simple cone shape is at all times geometrically proportional to its original shape. Using his symbols, the mass, H, weighing one lb. is the payload to be raised to the highest possible altitude with the least expenditure of propellant, P, and the container or structure dead load, K. The propellant is assumed to be discharged downward with constant velocity, c. He further assumes that the casing drops away continuously as the propellant burns, approximating the case where cartridge shells are ejected as fast as they are fired.

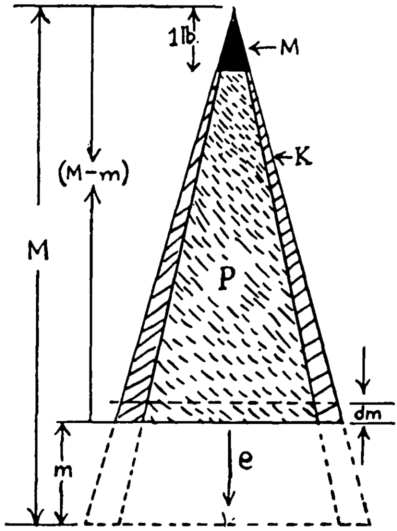

Figure 1

The symbols used in the derivation are defined as:

M—the initial mass of the rocket

m—the mass that has been ejected up to time, t

v—the velocity of the rocket at the time, t

c—the effective gas velocity of the mass expelled

R—the force, in absolute units, (poundals) due to air resistance

g—the acceleration of gravity

dm—the mass expelled during the increment of time, dt

k—the constant fraction of the mass dm that consists of casing K, expelled with zero velocity relative to the remainder of the rocket, and

dv—the increment of velocity given the remaining mass of the rocket

Following the law of conservation of momentum, we can equate the momentum existing at time t to the momentum existing at time (t + dt) taking due account of the impulses exerted on the rocket by air resistance and gravity which are equal to momentum since force times time or impulse, equals mass times velocity, or momentum:

$$(M-m)\,v = \overset{(1)}{dm\,(1-K)(v-c)} + \overset{(2)}{vK\,dm} + \overset{(3)}{(M-m-dm)\,(v+dv)} + \overset{(4)}{R\,dt} + \overset{(5)}{(M-m)\,g\,dt}$$

The term of the left side of the equals sign is the total momentum existing at any time, t, and considered positive in the upward direction.

The five terms on the right side are (1), the new momentum of the ejected propellant, which is the mass fraction (1—k)dm times the absolute velocity remaining or (v—c). The gas velocity in all rockets built to date is larger than the maximum rocket velocity attained. (In the V-2 the maximum velocity of 5000 ft./sec. has approached clqser to the gas velocity of 6400 ft./sec. than any other rocket, but the equation in no way excludes the case of the rocket going up faster than the oppositely directed velocity of the gases.); (2) the momentum of the casing kdm times the upward velocity v, since it is assumed to be cast off without change of velocity; (3) the new momentum of the remaining rocket mass (M-m-dm) times its increased velocity (v+dv); (4) the impulse due to Air Resistance, R, times the time, dt, in which it acts considered positive upward since it represents a part of the lost rocket momentum which otherwise would have been upward; and (5) the impulse due to gravity, or the weight of the whole mass (M-m) times the time, dt, during which lift against gravity must be supplied.

This looks formidable but it reduces nicely, after multiplying out the terms in parentheses, canceling like terms of opposite sign, and neglecting the product of the two infinitesimals, to:

$$c(1-k)dm = (M-m)dv + [R+g\ (M-m)]\ dt \qquad \text{(Eq. 2)}$$

In order to find the minimum mass m of propellant needed to give a payload any given upward velocity, or altitude, the derivative dm has to be equated to zero, all variables on the right side expressed in terms of one variable, say, the altitude, and the equation solved for m.

Dr Goddard states in the paper that the procedure necessary to do this is a new and unsolved problem in the Calculus of Variations so he proposed the method of intervals as an approximate solution, assuming the air resistance, gravity, and the net acceleration upward as constant during each interval.

Dividing Equation (2) by dt and substituting for dv/dt the letter a for net acceleration, he gets the linear differential equation of the first order:

$$\frac{dm}{dt} = \frac{(M-m)\ (a+g) + R}{c(1-k)} \qquad \text{(Eq. 3)}$$

which can be put into the form for a standard solution

$$\frac{dm}{dt} + Pm = Q \qquad \text{(Eq. 4)}$$

where P is equal to $\dfrac{a+g}{c(1-k)}$

and Q is equal to $\dfrac{M(a+g) + R}{c(1-k)}$

After performing the necessary steps for the standard solution which can be found in any text on differential equations, the equation becomes:

$$m = \left(M + \frac{R}{a+g}\right)\left(1 - e^{-\frac{a+g}{c(1-K)}t}\right) \qquad \text{(Eq. 5)}$$

This result is then converted to one giving the initial mass, M, required when the final mass in the interval is one pound by use of the defining equation:

$$M - m = 1 \qquad \text{(Eq. 6)}$$

which substituted in (Eq. 5) to eliminate m gives his final equation for the mass at the beginning of each interval:

$$M = \frac{R}{a+g}\left(e^{\frac{a+g}{c(1-K)}t} - 1\right) + e^{\frac{a+g}{c(1-K)}t}$$

(Eq. 7)

If R and g are set equal to zero in (Eq. 7) the simplified equation is left which gives the mass ratio used for testing for a minimum effect of these variables, or

$$M = e^{\frac{at}{c(1-k)}} \qquad \text{(Eq. 8)}$$

When the values of air resistance and gravity are such that ratio of the mass required with them (Eq. 7) to the mass required without them (Eq. 8) is a minimum, then it is concluded that M for (Eq. 7) is a minimum for the interval considered.

Finally, the total initial mass for several intervals is the product of the several individual initial masses from the first to the highest interval.

In applying (Equations 7 and 8) to get the minimum mass ratios, Goddard assumed the atmosphere divided up into nine intervals as the least which could be used without having the density vary too widely for the mean in each interval.

Since he obtained 8000 ft./sec. gas velocity for the smokeless powder, he considered 7500 ft./sec. a fair value for c, and 1/15 as at least a possible limiting value for the ratio of structure dead load to the total load. The velocity c(1-k) thus becomes 7000 ft./sec. and no further loss is considered due to structure load in the calculation of the minimum masses shown in Table 1 below.

The method of calculation is to assume a series of velocities acquired during the interval, adding to the previous velocity if it is not the first interval, then dividing the height of the interval by the average velocity to get the time to cross the interval. This time divided into the increase in velocity gives the average acceleration for the interval. The mean air resistance from the initial to the final velocity is used to determine R, and one sq. in. is assumed as the cross-sectional area for the one lb. mass to be projected to high altitudes.

The location of the minimum masses for the first 6 intervals is well defined but would occur for the seventh and eighth intervals only for very high velocities requiring a higher acceleration than the rocket and its instruments could withstand. Therefore an arbitrary limit of 150 ft./sec.^{2} was set for the acceleration for these and the last or ninth interval.

The variation of the acceleration of gravity with altitude was considered negligible for the first eight intervals, but was reduced to 68% of the familiar 32 ft./sec.^{2} at sea level in the 9th interval in accordance with the inverse square equation:

$$g = g_o\left(\frac{R}{R+h}\right)^2 \qquad \text{(Eq. 9)}$$

where g is the value at altitude h, above sea level, g_o is the surface value, and R is the radius of the earth (3960) miles.

"COASTING FLIGHT":

The altitude gained is a projectile after acceleration stops is calculated from the law of conservation of energy by equating the kinetic energy to the work done against gravity and the work done against air resistance, with the mean density of each interval above being used, or:

$$\frac{mv^2}{2} = mgh + psh \qquad \text{(Eq. 10)}$$

TABLE 1
GODDARD'S MINIMUM MASS RESULTS

Interval	Height of Upper End of Interval above Sea Level ft.	Mean Density to Sea Level	Velocity of Interval ft./sec.	Acceleration ft./sec²	M lbs./lb.	M Products	Greatest Altitude ft.
S_1	5,000	0.928	1,000	100.	1.25	1.25	14,000
S_2	15,000	0.730	1,200	22.	1.24	1.55	21,000
S_3	25,000	0.520	1,400	25.8	1.22	1.90	34,300
S_4	45,000	0.278	1,600	15.0	1.30	2.46	65,700
S_5	85,000	0.080	1,800	8.5	1.28	3.16	126,000
S_6	125,000	0.015	2,000	9.5	1.16	3.67	184,500
S_7	200,000	0.0026	5,160	150.	1.74	6.40	610,000
S_8	500,000	0.000025	10,790	150.	2.65	12.3	2,310,000
S_9	9,310,000	0	33,790	150.	48.8	602.	Infinity

Where m is the constant mass W/g, moving at initial velocity v; h is the height lifted when the velocity becomes zero; and p is the mean pressure in lbs. force per sq. in. of cross section, s. Thus the additional free flight distance is:

$$h = \frac{v^2}{2g}\left(\frac{w}{w + ps}\right) \qquad \text{(Eq. 11)}$$

Note in Table 1 that the product of mass ratios to reach an altitude of about 100 miles is a little under 6.4/1, actually about 6/1. This is about double that required for the V-2 to reach the same altitude, and the V-2 didn't have the advantage of losing any of its structure dead load. In other words, k equals zero for the case of an ordinary liquid fuel rocket during acceleration. The average gas velocity of the V-2 averaging its initial 6400 ft./sec. and its 16% reported final increase was 6900 ft./sec. or also less than Goddard's 7000 ft./sec. assumed effective velocity.

The discrepancy is readily explained by the fact that Goddard's assumption of a cross-section of 1 square inch per lb. of rocket was entirely too large as he knew and very definitely states in his paper. The cross sectional area of about 3300 sq. in. for a 5 ft. 5 in. dia., divided by its initial weight is only about 1/8 of a sq. in. per lb. Thus Goddard's minimum masses occur for values of accelerations which are about half the accelerations of the V-2 with consequently greater gravity losses. They are no doubt the lowest values for a small rocket, but are not the minimum values for a rocket the size of the V-2.

But before beginning the critical examination and modification of the fundamental mass ratio equal to make it usable for a rocket of the size of the V-2 and larger, it would help to become familiar with the performance of the V-2 and the next section is devoted to a study of the acceleration, velocity, and air resistance for this rocket.

III. V-2 PERFORMANCE

For this calculation the following assumptions were made on the basis of published data:

(a) Initial thrust, 55,000 lbs. increasing to 16% greater or 64,000 lbs. at 20 miles altitude (for this approximation, linear variation with time was used instead of the actual variation with altitude). The corresponding effective gas velocities are 6400 to 7400 ft. per sec.

(b) Initial weight of V-2, 28,200 lbs., decreased by 500 lbs. to allow for "idling" losses, making the initial weight for the mass ratio calculations, 27,700 lbs.

(c) Final weight, on the basis of 275 lbs. per sec. fuel consumption, and 64 seconds burning time, 10,100 lbs., making the initial to final weight ratio, 2.75/1, and the total fuel consumption, 17,600 lbs.

(d) Cross-sectional Area for 5 ft. 5 in. maximum diameter, 3300 sq. in., and a ballistic shape coefficient of one.

(e) Acceleration of gravity change to 100 mile altitude proves negligible, so a constant value of 32 ft. per sec. was used throughout.

The basic equation used for the numerical integration is the simple application of Newton's 2nd Law that acceleration of the rocket mass is produced by the unbalanced force, or the rocket reaction, less the air resistance and the weight, or:

$$a = \frac{F - R - W}{W} \cdot g \qquad \text{(Eq. 12)}$$

where a is the acceleration in ft./sec.^{2}, F is the rocket reaction, lbs., R is the air resistance, lbs., W is the average weight during the time interval assumed (5 sec.) and g is the acceleration of gravity, 32 ft./sec.^{2}

The air resistance is found from the Gavre Retardation tables* by use of the defining equation:

$$A_R = \frac{G_v}{C} \quad \ldots\ldots \text{(Eq. 13)}$$

where A_R is the retardation in ft./sec.^{2} for a projectile (or rocket having a ballistic coefficient, C, and a value of G_v found from the table for the average velocity of the interval. In the table C is assumed as one.

Since the ballistic coefficient is defined as:

$$C = \frac{W}{i\,\sigma\, d^2} \quad \ldots\ldots \text{(Eq. 14)}$$

where W is the weight of the projectile in lbs.,

i is the shape coefficient

σ is the air density ratio to sea level, and

d is the diameter, in inches,

The air resistance, R, in lbs. is given by:

$$R = \frac{i\,\sigma\, d^2 G_v}{g} \quad \ldots\ldots \text{(Eq. 15)}$$

The resulting values of acceleration, velocity, air resistance, and altitude reached are shown in Table 2 following and in Fig. 2.

The effect of air resistance in decreasing the acceleration is shown by the shaded area between the acceleration curves with and without air resistance.

The fact that air resistance at the end of the burning time becomes zero makes it a simple matter to extend the V-2 results to find out what a series of step-rockets with the same performance can do in the way of higher altitudes for a comparison with Goddard's results.

* Condensed table in Appendix.

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Figure 2

CALCULATED V-2 PERFORMANCE

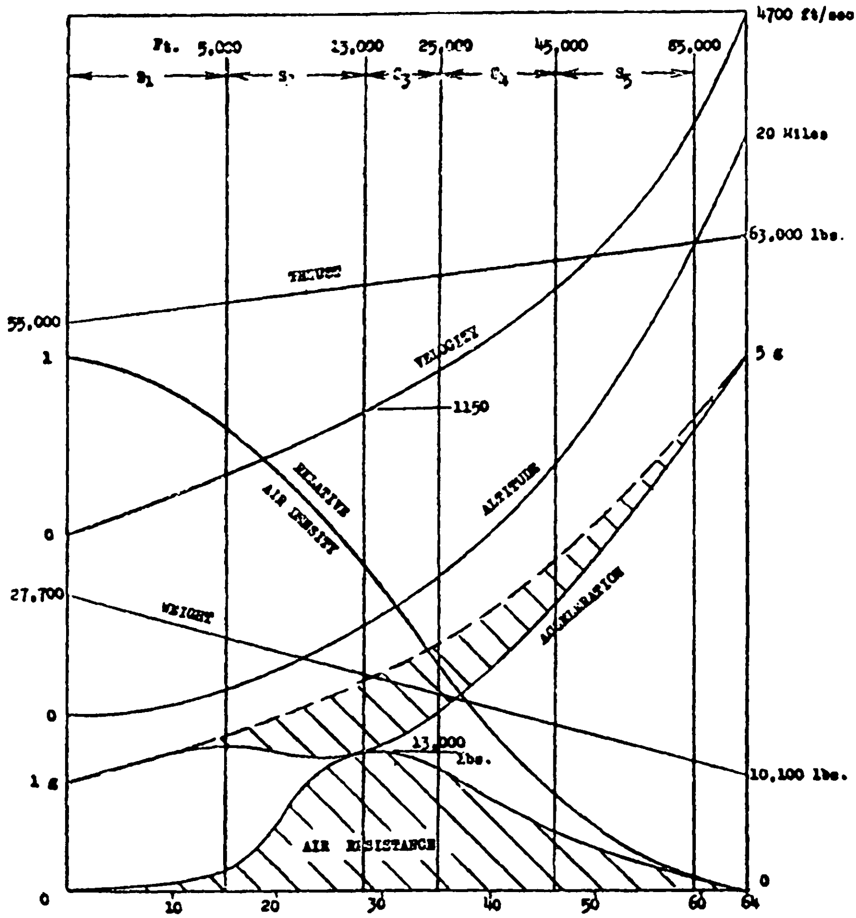

TABLE 2
RESULTS OF APPROXIMATE NUMERICAL INTERGRATION FOR V-2

Time t Secs.	Weight W Lbs.	Thrust F Lbs.	Acceleration a ft./sec.^{2}	in Velocity Δ v ft./sec.	Final Velocity v ft./sec.	Air Resistance R, Lbs.	Altitude Reached h ft.
0	27,700	55,000	32	0	0	0	0
5	26,300	55,700	40	180	180	0	500
10	24,900	56,400	45	212	392	500	1,900
15	23,500	57,100	45	225	617	1,500	4,400
20	22,100	57,800	42	217	834	6,500	8,000
25	20,800	58,500	40	205	1,039	11,500	11,700
30	19,400	59,200	45	212	1,251	13,200	18,400
35	18,000	59,900	56	252	1,503	10,900	25,300
40	16,700	60,600	68	310	1,813	8,600	33,600
45	15,300	61,300	84	380	2,193	6,300	43,600
50	13,800	62,000	106	475	2,668	3,000	55,700
55	12,500	62,700	126	580	3,248	1,700	70,500
60	11,200	63,400	150	690	3,938	400	88,500
64	10,100	64,000	161	775	4,713	0	110,000

IV. STEP-ROCKET CALCULATIONS

The first question to be considered is whether a rocket larger than the V-2 can be expected to give the same performance, i.e., reaching an altitude of about 20 miles at the end of burning and about 100 miles at the top of its free flight. If we assume that the larger steps will be geometrically similar to the V-2, we can make use of Goddard's equation relating the changing area and mass of such rockets. When we do this we are tacitly making a second assumption, namely, that the average density of the enclosed volume— about ⅔ that of water for the V-2—will remain constant for the larger rockets. This may or may not be the case.

However, using the relation between the area and weight of the V-2 to obtain the relation for any other geometrically similar rocket, we have:

$$\frac{A_1}{A_2} = \left(\frac{W_1}{W_2}\right)^{\frac{2}{3}} \quad \text{or } A_1 = \left(A_2 / W_2^{\frac{2}{3}}\right) W^{\frac{2}{3}} = \frac{3300}{28{,}200^{2/3}} W_1, \text{ and FINALLY } A = 3{,}57 W^{2/3}$$

(Eq. 16)

It is evident from this equation that the relative areas for the larger rockets do not increase as rapidly as the weights. Therefore, the larger first step having the same mass ratio as the V-2, will get through the atmosphere with less air resistance.

The fraction of fuel spent in overcoming air resistance for the V-2 can be estimated by integrating the area under the air-resistance-time curve, or the lost impulse or momentum, $\int$ Rdt, which appeared in (Eqs. 1 and 2). This average resistance over the 64 seconds assumed burning time is about 5000 lbs., which is only 8% of the average thrust of 59,000 lbs. acting over the same time.

Thus, a two-step rocket weighing 182 tons, or the 13.5 tons of the V-2 squared, would have a cross-sectional area from (Eq. 16) of 18,200 sq. in. and a ballistic density of 20 lbs. per sq. inch, as against 8.4 lbs. per sq. inch for the V-2 alone.

It follows that the retardation would be less than half as great, or less than 4% for the 182 ton rocket.

The happy conclusion is then that as we go to larger rockets we can practically forget about the air resistance, provided we keep the accelerations and velocities similar to those existing for the V-2 climb through the atmosphere.

This conclusion is so stimulating, that it seems only logical to continue on with the simplified step-rocket calculations taking account only of gravity losses. Even these decrease as the effect of the decreasing value of the acceleration of gravity begins to become apparent.

Dr. Goddard assigned a top value of 5g in his high altitude calculations as a protection against damage to the delicate instruments intended as payload. We shall do the same, thus lining up maximum requirements for even human cargo with the 5g maximum acceleration developed by the V-2.

An exact equation for the final velocity of the step rockets without air resistance can be derived by direct integration of the basic acceleration,

$$a = \left(\frac{F - W}{w}\right) g$$

(Eq. 17)

where the terms are the same as defined before (Eq. 12).

Substituting the product: mass rate of flow of the gases through the nozzle, $\frac{w}{g}$ times the constant gas velocity c in place of the reaction, F, and the difference of the original weight of the rocket, W_1 and the propellant w consumed in time t, can express (Eq. 17) in terms of the variable t, or

$$a = \frac{\frac{w}{g}c - (W_1 - wt)}{W_1 - wt} g$$

which multiplied by dt to give dv, is then integrated to give the velocity as

$$v = c \log_e \left(\frac{W_1}{W_2}\right) - gt$$

(Eq. 18)

where v is the velocity acquired during acceleration time t for a mass ratio of W_1/W_2 and a constant jet velocity, c. This acceleration of gravity was considered constant which is not far from the case for the change during the time each step is firing.

The logarithm to the base e for the constant practical mass-ratio of 2.75/1 assumed for all the steps is 1.01 so that we can conclude immediately from (Eq. 18) that the maximum velocity attained after each step fires and before subtracting the loss due to gravity—is equal to 1.01 times the jet velocity.

The jet velocity varies only for the first step thru the atmosphere, then remains constant at the higher value quoted for the V-2 at the 20 mile level. Since we are neglecting air resistance entirely it is only fair to balance it off by neglecting this increase for the first step, so in Table 3 the velocity acquired before the gravity term is subtracted is only 6450 ft./sec. The loss in velocity due to 32 ft. per sec. per sec. acting downward for 64 seconds is 2060 ft. per sec. leaving a net gain of 4450 ft./sec. for step No. 1.

The altitude gained at the end of firing is about 20 miles as before the V-2, and since the acceleration of gravity is still about 32 ft. per sec. per sec. no correction of the velocity is needed yet.

Step No. 2 and all subsequent ones make use of the increased efficiency of the rocket nozzle in a vacuum so that 7380 ft./sec. is used corresponding to a thrust of 64,000 lbs. in the V-2, and kept constant thereafter. A slight correction in the value of g is found after the new altitude is calculated, and the final corrected velocity used for subsequent calculations.

The initial weight for the series of steps is the continuing product of the initial weight per unit of payload, since the payload is considered to be the next step, and so on. It is important to understand that the structure dead load which appears in the ordinary expression of initial to final load must be rejected before the succeeding step rocket can become effective.

In the case of the V-2, the payload is about 1 ton, the remaining structure about 4 tons, and the initial weight 13.5 tons. The total weight of n steps is therefore:

$$w = (13.5)^n$$

(Eq. 19)

Table 3 shows the altitudes which can be reached by various numbers of steps, ending in the requirement of a 5,500,000 ton 6-step rocket to reach the escape velocity from the earth, 6.1 miles per sec. at the 1135 mile altitude then attained.

This is lower than the familiar figure of 1 miles per second at sea level, in accordance with the equation:

$$V_h = V_0 \sqrt{\frac{R}{R+h}} \quad . \; . \; . \; . \; . \quad \text{(Eq 20)}$$

where v_h is the escape velocity corresponding to altitude
h miles above the surface of the earth,
v_o is the escape velocity at sea level, miles per sec.,
and R is the radius of the earth, miles.
The escape velocity at the surface of any planet or satellite is:

$$V_o = \sqrt{\frac{2gR}{5280}} \quad \ldots\ldots \text{(Eq. 21)}$$

where v_o is in miles per sec., g in ft./sec.^{2}, and R is the radius in miles.

The satellite velocity is also of current interest, and is found by equating the weight of the body to the centrifugal force

$$W = \frac{WV^2}{gR} \text{ or solving for } V$$

$$V_{sat} = \sqrt{\frac{gR}{5280}} \quad \ldots\ldots \text{(Eq. 22)}$$

where v_{sat} is the satellite velocity in miles/sec. For the earth at the surface it is 4.9 miles per sec. so that the five step rocket would be required. At 780 miles altitude Eq. (20) shows it to be 4.5 miles per sec.

ESTIMATE OF SURFACE RANGES FOR STEP ROCKETS

Esnault-Pelterie in his book, L'Astronautique, gives the results of calculations of elliptical trajectories for various velocities as calculated from Kepler's laws. The ranges coresponding to the velocities for each of the six step rockets were obtained by interpolation from his table and are shown in Table 4.

The glide due to wings with an assumed lift-drag ratio of 6.4/1 is estimated by equating the kinetic energy of the rocket to the drag times distance. Thus

$$\text{K.E.} = \frac{WV^2}{2g} = \frac{W}{6.4} \times d$$

or

$$d = \frac{V^2}{10} \text{ (approx.)} \quad \text{(Eq. 23)}$$

In addition to this distance, there is the gain due to centrifugal force, which decreases the amount of the weight requiring lift approximately according to the relation:

$$\frac{W}{W_o} = \frac{\left(\frac{2}{3}V\right)^2}{g(R+h)} \quad \ldots\ldots \text{(Eq. 24)}$$

Where W is the reduced weight, lbs.
W_o is the unreduced weight, lbs.
2/3 V is the mean velocity, ft./sec.

TABLE 3
STEP ROCKET CALCULATIONS

(1) Step No.	(2) Wt. Ratio Tons*	(3) Gas Velocity ft./sec.	(4) Rocket Velocity without gravity loss ft./sec.	(5) Velocity gravity loss gt ft./sec.	(6) Velocity Increase 1st Trial ft./sec.	(7) Final Velocity 1st Trial ft./sec.	(8) Aver. Velocity for step ft./sec.
1	13.5	6450	6500	2060	4450	4450	1780
2	182	7380	7450	2060	5390	9850	7150
3	2460	7380	7450	1920	5530	15510	12680
4	30,300	7380	7450	1730	5720	21400	18500
5	408,000	7380	7450	1630	5820	27300	24400
6	5,500,000	7380	7450	1400	6060	33500	30400

(1) Step No.	(9) Powered Flight, ft. per Step	(10) Total Powered Altitude ft.	(11) Reduced g ft./sec.^{2}	(12) Revised gt. loss ft./sec.	(13) Corrected Final Velocity ft./sec.	(14) Free Flight Miles	(15) Maximum Altitude Miles
1	114,000	114,000	32.0	2060	4450	62	84**
2	457,000	571,000	30.0	1920	9980	350	450
3	811,000	1,382,000	27.0	1730	15700	920	1200
4	1,180,000	2,562,000	25.5	1630	21500	2000	2500
5	1,560,000	4,120,000	22.0	1400	27500	3800	4600
6	1,950,000	6,000,000	19.0	1200	33700	Inf.	Inf.

* per ton of payload.

g is the acceleration of gravity, ft./sec.[2]

R is the radius of the earth, ft.,

and h is the altitude, ft.

Naturally, the results by this method are very rough approximations, but they give some indication of the great gain in range which result at the higher velocities and the value of making more detailed studies of the possibilities of large winged rockets.

TABLE 4
SURFACE RANGES

(1) No. of Steps	(2) Initial Weight Tons	(3) Max. Vel. Miles/sec.	(4) Approx. Elliptical Traj. range miles	(5) Approx. Range with Wings*	(6) Approx. Surface Initial Dia. ft.	(7) Approx. Height on V-2 basis ft.	(8) Ballistic Density lb./sq. in. "Punch Thru ability"
1	13.5	1	160	400	5.5	46	8.4
2	182	2	700	2000	13	108	20
3	2450	3	1900	6000	30	255	50
4	30,300	4	4000	16,000	72	610	110
5	408,000	5**	Inf.	40,000	170	1460	260
6	5,500,000	6***	Inf.	Inf.	400	3400	620

* Including glide with 6.4/1 wing lift-weight ratio, and favorable effect of centrifugal force.

** This is the circular orbital velocity for a satellite traveling around the earth outside the atmosphere. The drag of the atmosphere would cause it to slow down and land as shown in Col. (5).

*** This is less than the familiar 7 miles/sec. escape velocity for the earht because at the altitude of the last step the acceleration of gravity has decreased to about two-thirds of the value at sea level, making 6 miles per sec. the appropriate escape velocity.

V. CHECK OF V-2 FOR MINIMUM MASS

In order to check the V-2 performance by Dr. Goddard's minimum mass method we must first generalize his last equation, (Eq. 5) to make it applicable for any size rocket larger than one lb.

Starting with his equation:

$$m=\left(M+\frac{R}{a+g}\right)\left(1-e^{-\frac{a+g}{c(1-K)}t}\right) \quad \text{(Eq 5)}$$

let us first convert the masses to weights by substituting $\frac{W_1 - wt}{g}$ for m, and

$$\frac{W_1}{g}$$

for M; then multiplying through by g and considering k=O for no change in structure dead load, we have:

$$W_1-W_2=\left(W_1+\frac{Rg}{a+g}\right)\left(1-e^{\frac{a+g}{c}t}\right) \quad \text{(Eq.25)}$$

Dividing through by W_2 and solving for the weight ratio W_1/W_2 we obtain the modified equation

$$\frac{W_1}{W_2}=e^{\frac{a+g}{c}t}+\frac{Rg}{W_2(a+g)}\left(e^{\frac{a+g}{c}t}-1\right) \quad \text{(Eq. 26)}$$

Eq. (26) is the same as Goddard's result except for the term W_2 which acts to decrease the air resistance loss and the required mass-ratio for larger rockets.

If Eq. (25) is divided by W_1 instead of W_2, and again solved for the same weight ratio, W_1/W_2 we obtain another form of the result which was found more convenient in indicating the additional weight due to air resistance, or

$$\frac{W_1}{W_2}=\frac{e^{\frac{a+g}{c}}}{1-\frac{Rg}{W_1 a+g}\left(e^{\frac{a+g}{c}t}-1\right)} \quad \text{(Eq. 27)}$$

Where R is the air resistance in lbs. for the constant cross-sectional area corresponding to the initial weight, W_1, and can be calculated from the ballistic coefficient relations already given.

The procedure to determine the minimum weight ratio for each interval is the same as used by Goddard.

The test criterion is that the ratio of

$$\frac{W_1}{W_2}$$

in Eq. (27) with gravity and air resistance included to

$$\frac{W_1}{W_2}$$

in a vacuum without gravity shall be a minimum. For convenience in the calculations the expression Rg/W_1 in Eq. (27) may be expressed as:

$$\frac{Rg}{W_1} = \frac{4 i \sigma A_1 G_v}{\pi W_1} \quad \ldots \text{(Eq. 28}$$

where the terms are as previously defined.

The calculations for the first 5000 ft. interval (as used by Goddard) are shown in Table 5 following:

TABLE 5
FIRST 5000 FT. INTERVAL

Velocity V, ft./sec.	Mean Veloc.	Time t, sec.	$\Delta v/t$ Accel. a, ft./sec.^{2}	$\frac{at}{c^*}$	$e^{\frac{at}{c}}$	$\left(\frac{a+g}{c}\right)t$
400	200	25.0	16	.0625	1.065	.1875
500	250	20.0	25	.0782	1.081	.1781
600	300	16.7	36	.0938	1.098	.1771
800	400	12.5	64	.1250	1.133	.1865

Velocity V, ft./sec.	$e^{\left(\frac{a+g}{c}\right)t}$	Gv	Mean** Rg/W1	Denom. of Eq. (27)	W1/W2 Eq. (27)	$\frac{W1/W2}{e^{at/c}}$
400	1.206	7.0	0.33	.999	1.207	1.132
500	1.195	10.6	0.50	.998	1.197	1.106
600	1.194 ***	14.9	0.71	.998	1.196	1.089
800	1.205	28.8	1.36	.997	1.208	1.065

* 6400 ft./sec.

** Since the air resistance varies as the square of the velocity for the low velocities, the mean value was taken as one-third the end value.

*** Minimum shown here.

No minimum weight-ratio is shown by the test method of dividing the weight-ratio with the opposing factors of the resistance and gravity Eq. (27) by the weight-ratio required for acceleration alone Eq. (8) However, the weight-ratios in (Col. 8) for acceleration and gravity but no air resistance show a minimum value at a velocity of 600 ft./sec. and an acceleration of 36 ft./sec.^{2} As the air resistance during this interval was shown to be negligible in the V-2 performance calculations and the kinetic energy at the top of the interval is a small fraction of that existing at the final velocity, the conclusion is forced on us that the minimum weight ratio should be solved for without regard to Eq. (8) or the magnitude of the velocity at the end of the interval but only on the basis of the least ratio required for simple lift through the interval.

Thus, the weight ratio in the equation:

$$\frac{W_1}{W_2} = e^{\frac{(a+g)t}{c}} = e^{\frac{v}{c}} \quad \ldots \text{(Eq. 29)}$$

will be a minimum if the exponent is a minimum.

Further, we need only test the numerator of this exponent for a minimum since the gas velocity, C, in the denominator is a assumed constant.

Thus;

$$v = (a + g)\,t \qquad \text{(Eq. 30)}$$

Eliminating the variable, t, by use of the relation

$$S = \tfrac{1}{2}\, at^2 \qquad \text{(Eq. 31)}$$

Then setting the derivative of the

velocity with respect to the acceleration equal to zero yields the interesting result that the initial acceleration for minimum weight for any rocket is:

$$a_0 - g \qquad \text{(Eq. 32)}$$

We conclude, therefore, that the proper velocity for the first step is one g just as was used in the V-2. The corresponding velocity for the end of the interval and for minimum weight-ratio would have been 567 ft./sec. but the ratio for the 600 ft./sec. velocity, 1.20/1, is close enough to it to continue on to test the next step for a minimum.

It is of interest to note that the calcalculations for the acceleration of 37 ft./sec.^{2} indicate the distribution of the propellant consumed as 50% for acceleration, 48½% for gravity, or lift, and only 1½% for air resistance, checking very closely the law expressed by Eq. (32) that the favorable fraction of acceleration thrust, or propellant must balance the unfavorable traction.

The next interval—from 5000 to 15,000 ft.—starts with the new initial weight of 27, 700/1.20, or 23,100 lbs. The cross-sectional area is the same; gas velocity is assumed the same; the mean air density drops from 0.93 to 0.735; and the initial velocity is 600 ft. per second.

The calculation proceeds as before, except that the mean velocity of course is half the increment (at) added to the 600 ft./sec., and the height traveled is 10,000 ft. This time we find a minimum weight ratio in the right column by the Goddard method.

TABLE 6

SECOND INTERVAL, 5,000-15,000 FT.

Velocity V ft./sec.	At or Δ	Mean v	Time t sec.	Accel. a ft./sec.^{2}	$\frac{at}{c}$	$\frac{at}{e^{c}}$	$\left(\frac{a+g}{c}\right) t$
1000	400	800	12.5	32.0	.0625	1.065	.1250
1050	450	825	12.1	37.1	.0703	1.073	.1307
1100	500	850	11.8	42.5	.0782	1.081	.1374

Velocity V	$e^{\left(\frac{a+g}{c}\right) t}$	Gv	Mean Gv	Mean $\frac{Rg}{W}$	Denom. of Eq. (27)	$\frac{W1}{W2}$	$\frac{W1/W2}{e^{at/C}}$
1000	1.133	68	41.5	5.54	.988	1.148	1.077
1050	1.140	88	51.5	6.87	.986	1.156	1.076
1100	1.147	113	64.0	8.53	.983	1.166	1.077

The minimum weight-ratio for this interval is found to be 1.16/1 for an acceleration that is still close to one g, or only 37 ft./sec. The distribution of propellant requirement for this interval is shown to be: 47% for acceleration, 43% for gravtiy, and 10% for air resistance or still within the 3% ratio of the 50-50 distribution.

The new weight for the next interval is then 23,100/1.16 or 20,000 lbs.; the new density ratio for the height from 15,000 ft. to 25,000 ft. is 0.0520; the initial velocity is now 1050 ft./sec.; and other values are kept the same, leading to the results shown in Table 7.

TABLE 7
THIRD INTERVAL 15,000-25,000 FT.

Velocity V ft./sec.	At or V	Mean Vel.	Time t, secs.	Accel. Δ v/t	$\frac{at}{c}$	$e^{\frac{at}{c}}$	$\frac{a+g}{c}t$
1400	350	1225	8.15	42.9	.0547	1.056	.0953
1500	450	1275	7.85	57.4	.0704	1.073	.1096
1600	550	1325	7.55	72.8	.0860	1.090	.1236

Velocity V	$e^{\frac{a+g}{c}t}$	Gv	Mean Gv	Mean Rg/W	Denom. of Eq. (27)	$\frac{W1}{W2}$	$\frac{W1/W2}{e^{at/C}}$
1400	1.100	262	152	19.1	.974	1.130	1.070
1500	1.116	305	197	21.5	.972	1.148	1.069
1600	1.131	347	218	23.8	.970	1.167	1.070

The minimum weight ratio for this interval is found to be 1.15/1 for an acceleration of 57 ft./sec., or approaching 2g, and an end velocity of 1500 ft./sec. The distribution of propellant is 49% for acceleration, 29% for gravity and 22% for air resistance. The weight for the next interval is 20,000/1.15, or 17,600 lbs.; the new air density ratio, 0.305; and the height travelled is 20,000 ft.

TABLE 8
FOURTH INTERVAL, 25,000-45,000 FT.

Velocity V ft./sce.	At or Δ V	Mean Vel.	Time t, sec.	Accel. v/t	$\frac{at}{c}$	$e^{\frac{at}{c}}$	$\left(\frac{a+g}{c}\right)t$
2100	600	1800	11.12	53.8	.081	1.084	.130
2200	700	1850	10.81	64.7	.095	1.100	.141
2400	900	1950	10.26	87.8	.122	1.130	.166

Velocity V	$e^{\left(\frac{a+g}{c}\right)t}$	Gv	Mean Gv	Mean Rg/W	Denom. of Eq. 27	$\frac{W1}{W2}$	$\left(\frac{W1/W2}{e^{at/C}}\right)$
2100	1.139	552	375	28.3	.954	1.193	1.101
2200	1.151	593	395	29.7	.954	1.210	1.100
2400	1.180	679	438	33.0	.950	1.243	1.101

The minimum weight-ratio for this interval turns out to be 1.21/1 for an acceleration of 65 ft./sec. or over 2 g. The distribution of propellant consumption is 48% for acceleration, 24% for gravity and 28 %for air resistance.

Enough minimum values have now been calculated for the comparison with the V-2 weight-ratios for the same critical intervals through the transonic ranges. Table 9 shows the accelerations, individual ratios and product of the interval ratios.

TABLE 9
V-2 MINIMUM WEIGHT CHECK

Interval	Acceleration For Actual V-2	Acceleration For Min. Weight	Velocities For Actual V-2	Velocities For Min. Weight	Weight Ratios For Actual V-2	Weight Ratios For Min. Weight	Weight Ratio Products For Actual V-2	Weight Ratio Products For Min. Weight
S_1	38.4	36.0	630	600	1.180	1.200	1.18	1.20
S_2	41.6	37.1	1160	1050	1.175	1.156	1.39	1.38
S_3	48.0	57.4	1500	1500	1.110	1.148	1.54	1.58
S_4	70.4	64.7	2250	2200	1.196	1.210	1.84	1.90

The agreement of the actual with the calculated minimum weight ratio within 2% for the first interval, 1% for the second, and 3% for the third and fourth intervals is impressive.

Such close agreement simply cannot be fortuitous. The available accelerations for various thrusts for the V-2 must certainly have been fitted to some minimum weight calculation. And Goddard knew the method was sound in 1916, thirty years ago. It is a great tribute to his courage that he continued to work in this field in the face of so much adverse criticism. Now that the V-2 is history, he is completely vindicated, and I am proud to have had this opportunity to show how right he was.

We can all be proud that an American scientist was the first to conceive the modern rocket. And we can claim it as one more of our major American inventions.

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TABLE 10

AIR PRESSURE AND DENSITY RELATIVE TO SEA LEVEL*

Altitude Feet	Pressure	Density	Altitude Feet	Pressure	Density	Altitude Feet	Pressure	Density
0	1.000	1.000	35,000	.236	.302	70,000	.0440	.0580
1,000	.962	.973	36,000	.225	.289	71,000	.0415	.0550
2,000	9.30	.945	37,000	.215	.278	72,000	.0395	.0529
3,000	.898	.914	38,000	.204	.266	73,000	.0375	.0503
4,000	.865	.890	39,000	.196	.253	74,000	.0358	.0481
5,000	.833	.861	40,000	.185	.242	75,000	.0340	.0459
6,000	.802	.835	41,000	.175	.231	76,000	.0324	.0438
7,000	.772	.810	42,000	.167	.220	77,000	.0310	.0418
7,000	.743	.785	43,000	.158	.210	78,000	.0295	.0397
9,000	.715	.760	44,000	.151	.200	79,000	.0285	.0379
10,000	.690	.737	45,000	.145	.191	80,000	.0275	.0361
11,000	.663	.712	46,000	.138	.182	81,000	.0264	.0342
12,000	.636	.690	47,000	.131	.173	82,000	.0256	.0328
13,000	.614	.668	48,000	.125	.168	83,000	.0248	.0312
14,000	.590	.645	49,000	.120	.159	84,000	.0240	.0298
15,000	.568	.624	50,000	.115	.152	85,000	.0231	.0285
16,000	.545	.603	51,000	.109	.145	86,000	.0223	.0271
17,000	.525	.585	52,000	.103	.138	87,000	.0215	.0260
18,000	.502	.568	53,000	.099	.131	88,000	.0207	.0248
19,000	.483	.550	54,000	.094	.127	89,000	.0199	.0238
20,000	.464	.532	55,000	.090	.119	90,000	.0190	.0228
21,000	.443	.513	56,000	.085	.112	91,000	.0182	.0219
23,000	.406	.481	58,000	.078	.103	93,000	.0166	.0201
24,000	.390	.467	59,000	.074	.098	94,000	.0158	.0192
25,000	.372	.450	60,000	.070	.094	95,000	.0150	.0182
26,000	.355	.435	61,000	.068	.090	96,000	.0140	.0174
27,000	.340	.420	62,000	.064	.086	97,000	.0133	.0167
28,000	.325	.402	63,000	.061	.081	98,000	.0125	.0158
29,000	.310	.388	64,000	.058	.078	99,000	.0117	.0149
30,000	.297	.372	65,000	.058	.073	100,000	.0110	.0140
31,000	.283	.359	66,000	.053	.070			
32,000	.270	.344	67,000	.050	.068			
33,000	.260	.330	68,000	.048	.063			
34,000	.247	.317	69,000	.047	.060			

* Values taken from a plot of I.C.T. data.

TABLE 11
APPENDIX
Condensed Table of the Gavre Retardation Function G*

NOTE: The value of G_v in the following table is the retardation in ft. per sec. effected by atmosphere of standard density (sea level) for the velocity in ft. per sec. of a projectile (or rocket) having a ballistic coefficient, C, of one.

100	0.50	2100	552	4100	1731
V	Gv	V	Gv	V	Gv
200	1.95	2200	593	4200	1814
300	4.16	2300	635	4300	1899
400	7.03	2400	679	4400	1987
500	10.6	2500	723	4500	2080
600	14.9	2600	769	4600	2170
700	20.6	2700	817	4700	2260
800	28.8	2800	867	4800	2360
900	42.6	2900	919	4900	2460
1000	68.0	3000	973	5000	2560
1100	113	3100	1030	5100	2660
1200	166	3200	1088	5200	2770
1300	216	3300	1149	5300	2870
1400	262	3400	1213	5400	2980
1500	305	3500	1280	5500	3090
1600	347	3600	1349	5600	3210
1700	388	3700	1421	5700	3320
1800	429	3800	1496	5800	3440
1900	470	3900	1572	5900	3560
2000	511	4000	1650	6000	3680

* Condensed for Slide Rule approximations from Table I, p. 76 Herrmann's Exterior Ballistics, 1935, U.S. Naval Institute, Annapolis, Maryland.

TABLE 12
APPENDIX
List of Robert H. Goddard U. S. Patents

1	1,102,653	July 7, 1914	Rocket Apparatus (2-step Powder Rocket)
2	1,103,503	July 14, 1914	Rocket Apparatus (Cartridge Reloading)
3	1,137,964	May 4, 1915	Means for Producing Electrically Charged Particles
4	1,154,009	Sept. 21, 1915	Apparatus for Producing Gases
5	1,159,209	Nov. 2, 1915	Apparatus for Producing Electrical Impulses or Oscillations

(Cont. on Page 46)

EDITOR'S NOTES

1. The third and final installment of Arthur Leonard's article, "Some Possibilities for Rocket Propellants", scheduled for the September issue of the Journal, has been postponed to the December issue, due to last-minute changes by the author.
2. Primary reason for the late appearance of the September issue was the work involved in painstaking corrections of the paper, "Rocket Research of Dr. Robert L. Goddard", done by Alfred Africano. This significant article, first presented at the initial annual convention of the American Rocket Society in 1946, is hereby made available to the jet propulsion engineering profession.
3. The December, 1947, issue of the Journal will be a special annual convention number, containing several of the papers to be presented at the convention in lieu of preprints. As such, it will prove a very valuable number. It is planned to have it ready during the first week in December.
4. The Journal is open to advertising, as the membership will have noted. Members are requested to bring this to the attention of the advertising manager or president of engineering organizations with whom they may be affiliated. A request will bring pertinent information, rate card and sample copy of the Journal. Revenue obtained from advertising will go directly into the Journal in the form of increased paging with more articles.

6	1,191,299	July 18, 1916	Rocket Apparatus (Improved Cartridge)
7	1,194,496	Aug. 15, 1916	Rocket Apparatus (Less Weight)
8	1,206,837	Dec. 5, 1916	Rocket Apparatus (Improved Performance)
9	1,311,885	Aug. 5, 1919	Magazine Rocket (Movable Breach Block)
10	1,341,053	May 25, 1920	Magazine Rocket (Balanced Loading)
11	1,363,037	Dec. 21, 1920	Means for Producing Electrified Jets of Gas
12	1,609,540	Dec. 7, 1926	Sound Reproducing Device
13	1,661,473	Mar. 6, 1928	Accumulator for Radiant Energy
14	1,700,675	Jan. 29, 1929	Vaporizer for use with Solar Energy
15	1,809,115	June 9, 1931	Apparatus for Producing Ions
16	1,809,271	June 9, 1931	Propulsion of Aircraft (Rocket Driven Propellers)
17	1,834,149	Dec. 1, 1931	Means for Decelerating Aircraft (Flap Open)
18	1,860,891	May 31, 1932	Apparatus for Pumping Low Temperature Fluids
19	1,879,186	Sept. 27, 1932	Apparatus for Igniting Liquid Fuel (Idling Flame)
20	1,879,187	Sept. 27, 1932	Mechanism for Directing Flight (Gyroscope Controlled Air and Jet Vanes)
21	1,929,778	Oct. 10, 1933	Propulsion of Aircraft (Ring Turbine Around Propeller)
22	1,951,403	Mar. 20, 1934	Heat Absorbing Apparatus for Use with Solar Energy
23	1,951,404	Mar. 20, 1934	Focusing Mirror and Directing Mechanism
24	1,969,839	Aug. 14, 1934	Apparatus for Absorbing Solar Energy
25	1,969,840	Aug. 14, 1934	Method of Welding Thin Metal Structures
26	1,980,266	Nov. 13, 1934	Propulsion Apparatus (Pulse Jet)
27	2,016,921	Oct. 8, 1935	Means for Cooling Combustion Chambers (Film Cooling)
28	2,026,885	Jan. 7, 1936	Aircraft (Combination Propeller and Rocket)
29	2,039,217	Apr. 28, 1836	Nozzle for Welding Thin Metals
30	2,085,800	July 6, 1937	Combustion Apparatus (Film Cooling)
31	2,090,038	Aug. 17, 1937	Aircraft Construction (Inflated for Rigidity)
32	2,090,039	Aug. 17, 1937	Igniter (Spark Plug with Separate Fuel Supply)
33	2,109,529	Mar. 1, 1938	Reinforced Construction for Light Hollow Members (Wire Wound)
34	2,122,521	July 5, 1938	Cooling Jacket Construction
35	2,127,865	Aug. 23, 1938	Seal for Centrifugal Pumps (Liquid Seal)
36	2,158,180	May 16, 1939	Gyroscope Steering Apparatus (Tilting and Rotation)
37	2,158,181	May 16, 1939	Delayed Action Reversing Switch
38	2,158,182	May 16, 1939	Heat Insulating Coupling (Liquid Oxygen Pump)
39	2,183,311	Dec. 12, 1939	Means for Steering Aircraft (Shell-wall Vanes)
40	2,183,312	Dec. 12 1939	Fuel Storage and Discharge Apparatus (Hollow Gyro)

41	2,183,313	Dec. 12, 1939	Combustion Chamber for Aircraft (Cooled Porous Wall)
42	2,183,314	Dec. 12, 1939	Gyroscope Apparatus for Directing Flight
43	2,217,649	Oct. 8, 1940	Combustion Chamber for Rocket Apparatus (Film Cooling)
44	2,271,224	Jan. 27, 1942	Parachute Attachment for Aircraft (Gyroscope Controlled)
45	2,281,971	May 5, 1942	Fluid Cooled Bearings (Oxygen Pump)
46	2,286,908	June 16, 1942	Auxiliary Turbine for Rocket Craft (Jet Operated)
47	2,286,909	June 16, 1942	Combustion Chamber (Porous Wall and Film Cooling)
48	2,307,125	Jan. 5, 1943	Launching Apparatus for Rocket Craft
49	2,394,852	Feb. 12, 1946	Liquid Feeding Apparatus (Concentric Tanks)
50	2,394,853	Feb. 12, 1946	Liquid Storage Tank (Flow for any Position)
51	2,395,113	Feb. 19, 1946	Mechanism for Feeding Combustion Liquids to Rocket Apparatus
52	2,395,114	Feb. 19,1946	Rotating Combustion Chamber for Rocket
53	2,395,403	Feb. 26, 1946	Rotatable Combustion Apparatus for Aircraft Apparatus
54	2,395,404	Feb. 26, 1946	Aerial Propulsion Apparatus (Rocket Driven Turbine)
55	2,395,405	Feb. 26, 1946	Landing Apparatus for Rocket Craft (Pneumatic Tube)
56	2,396,406	Feb. 26, 1946	Combustion Apparatus (Igniter and Film Cooling)
57	2,395,435	Feb. 26, 1946	Emergency Control Mechanism for Aircraft (Powder Rockets)
58	2,395,809	Mar. 5, 1946	Rocket Directing Apparatus (Air and Jet Vanes)
59	2,396,321	Mar. 12, 1946	Apparatus for Controlling Acceleration and Deceleration in Aircraft
60	2,396,566	Mar. 12, 1946	Rocket Apparatus (Successive Charges)
61	2,396,567	Mar. 12, 1946	Combustion Apparatus (Cooled Walls)
62	2,396,568	Mar. 12, 1946	Apparatus for Steering Aircraft (Wall Flaps)
63	2,397,657	Apr. 2, 1946	Control Mechanism for Rocket Apparatus (Sequence Timing)
64	2,397,658	Apr. 2, 1946	Combustion Apparatus (Wall Expansion)
65	2,397,659	Apr. 2, 1946	Control Mechanism for Rocket Apparatus (Timing)
66	2,397,998	Apr. 9, 1946	Propelling Apparatus for Aircraft (Rocket Turbine Propeller)
67	2,397,999	Apr. 9, 1946	Propelling Apparatus for Aircraft (Division of Above)
68	2,405,785	Aug. 13, 1946	Combustion Chamber
69	2,409,036	Oct. 8, 1946	Feeding Device for Combustion Chambers