NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL MEMORANDUM

No. 1144

ROCKETS USING LIQUID OXYGEN

By Adolf Busemann

Translation

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1. INTRODUCTION

It is my task to discuss rocket propulsion using liquid oxygen and my treatment must be highly condensed for the ideas and experiments pertaining to this classic type of rocket are so numerous that one could occupy a whole morning with a detailed presentation.

First, with regard to oxygen itself as compared with competing oxygen carriers, it is known that the liquid state of oxygen, in spite of the low boiling point, is more advantageous than the gaseous form of oxygen in pressure tanks, therefore only liquid oxygen need be compared with the oxygen carriers. The advantages of liquid oxygen are absolute purity and unlimited availability at relatively small cost in energy. The disadvantages are those arising from the impossibility of absolute isolation from heat; consequently, allowance must always be made for a certain degree of vaporization and only vented vessels can be used for storage and transportation. This necessity alone eliminates many fields of application, for example, at the front lines. In addition, liquid oxygen has a lower specific weight than other oxygen carriers, therefore many accessories become relatively larger and heavier in the case of an oxygen rocket, for example, the supply tanks and the pumps. The advantages thus become effective only in those cases where definitely scheduled operation and a large ground organization are possible and when the flight requires a great concentration of energy relative to weight.

With the aim of brevity, a diagram of an oxygen rocket will be presented and the problem of various component parts that receive particularly thorough investigation in this classic case but which are also often applicable to other rocket types will be referred to.

2. DIAGRAM OF DRIVE

The basic elements of a rocket drive using liquid oxygen are presented in figure 1. Fuel and oxygen are transferred by pumps

*"R-Gerät mit Sauerstoff." R-Antriebe, Schr. d. D. Akad. d. Luftfahrtforschung, Heft 1071, Nr. 82, 1943, pp. 127-143.

from the tanks to the combustion chamber where they burn at high pressure. The assumption is made that the pumps have a separate source of power, although conceivably this might not be the case. The combustion chamber may be cooled by one of the substances upon entering; this cooling is not indicated in the diagram. From the combustion chamber the actual rocket jet, which produces the forward thrust, emerges by way of a Laval nozzle. The reaction of the exhaust steam from the turbine driving the fuel and oxygen pumps may be put to use providing additional forward thrust.

3. TANKS

The fuel tanks need be designed in accordance with only the usual criterion of rocket technique, namely, large capacity with least possible weight and, when necessary, the ability to withstand great acceleration. For the oxygen tanks, however, additional measures are necessary because liquid oxygen with a low boiling point is being dealt with. Generally the double-walled, mirror-surfaced, evacuated Dewar flask is considered as a storage tank; nevertheless, simple sheet-metal tanks may also be used for liquid oxygen. On the somewhat warmer outside surface of such tanks the surrounding air does not become liquid but the water content of the air forms a frost coating of gradually increasing thickness. For tank sizes suitable for use in rockets, the evaporation loss is in any case so limited that the contents entirely evaporate after only 1/2 to 2 hours. The use of Dewar flasks allows this time to be extended to days or weeks but large storage tanks may be constructed with such good heat insulation that the contents require months to evaporate.

Consequently, if in rocket propulsion it is a question of storage periods of a few minutes, the evaporation losses in a simple sheet-metal tank are negligible. If it is desired to build a double-walled container, in doing so neither the inner nor the outer wall can be made very thin. After evacuation, the outer wall is subject to excess pressure from without and if made too thin will collapse in folds upon the inner wall. The inner wall must withstand the weight of the contents, which under certain circumstances is increased by acceleration, and usually the excess pressure of the boiling oxygen that is used to carry the liquid oxygen to the pump. The oxygen loss involved in cooling the tanks when filling is generally a burden only upon the ground organization and not upon the power plant in flight, therefore this question is less significant in the selection of propellants rich in energy and of containers for them.

The low temperature necessitates a certain caution in the choice of material from which to construct the tanks because most metals become very brittle at low temperatures. As oxygen, which does not burn but supports combustion better than air and can take part in combustion with metals, is being dealt with, certain precautions are necessary, which are also necessary with other oxygen carriers. Copper and certain bronzes merit chief consideration as constructional materials for tanks and piping. If iron or other metals combustible with oxygen are used for economy, the spread of any fire can at least be hindered by the use of copper or bronze separators but this measure should be taken in the laboratory where the unburned parts may be reused.

4. FUEL PUMPS

The general aspects of the advantages of pressure tanks against one of the various types of fuel pump have been thoroughly discussed (reference 1). In the cases in which liquid oxygen is used, the operating period is generally of such length that a pump is desirable. The character of liquid oxygen as a boiling liquid creates certain additional difficulties here only insofar as it necessitates that the pump pressure rise more rapidly than the vapor pressure created by the temperature rise occurring in the pump as a result of the unavoidable heat transfer. As soon as the oxygen leaves the supply tank and enters the pumping line, any vapor forming must be pumped along with it, whereby the work of the pump is increased and the familiar difficulties of pumping non-homogeneous fluids arise. In order to minimize these difficulties, at the beginning of the pumping period the oxygen container is often put under a certain excess pressure to serve as inlet pressure and pressure reserve for the first stages of the pump. Because of the short pumping period, this excess pressure may be produced by partial vaporization because the temperature difference between the superheated vapor and the liquid in the supply tank does not immediately equalize itself. The pumps require special construction to minimize the heating of the oxygen and because of the impossibility of lubrication with oil. Furthermore, suitable metals must be selected for high-speed centrifugal pumps with reference to mechanical strength and the danger of fire if the impeller rubs against the housing.

5. COMBUSTION CHAMBER

So far the necessary accessories have been dealt with; now the essential parts of the rocket power plant are being considered, the combustion chamber and the exhaust nozzle. The purpose of the combustion chamber is to combine the propellant components, which have previously been kept separate for safety reasons, under the most favorable conditions in order that the calorific value may be transformed as completely as possible into kinetic energy in the exhaust nozzle. Compared with all other oxygen carriers the rocket using liquid oxygen has the highest calorific value. Liquid oxygen is therefore particularly suitable for use where it is unnecessary to keep the temperature of the combustion chamber below a certain level by the addition of diluents and, on the contrary, the primary aim is the attainment of the very highest possible outlet velocities and with them the lowest consumptions. The temperatures to be expected lie between about 3000° and 4000° C, consequently there are no combustion-chamber walls that can withstand these temperatures without cooling. However, if one of the two liquid-fuel components is used for cooling, with appropriate size relations metal walls can be used with liquid cooling and the wall temperature on the combustion side need not exceed 300° to 600° C. Because in this manner the heat removed by the cooling process is absorbed into the fuel, no energy is lost from the balance. However, no substantial gain in the production of velocity is made because in connection with the heat transfer from 3500° to 600° K in the wall and to 300° K in the fuel a great increase of entropy occurs.

According to the thermodynamic rules of equilibrium, a reaction that develops heat remains less complete in proportion as the end temperature of the reaction is higher. If in a given reaction the atoms unite in such a manner that the molecular weight increases, this union can be promoted by increasing the pressure just as raising the temperature hinders it. At the same time as the increase of combustion-chamber pressure, a greater pressure drop in the nozzle for the production of velocity is obtained; therefore, this increase of pressure has a double result. Constructional difficulties increase with increasing combustion-chamber pressure but the outlet cross section becomes smaller with increasing pressure, as does the size of the combustion chamber for a given length of time that each atom of the fuel remains in the chamber, and with this decrease the surface area to be cooled. These are all considerations that make it advisable not to be discouraged by the greater constructional difficulties.

The completeness of the combustion varies with the temperature, as shown in figure 2. The case is that of an oil burning with some deficiency of oxygen; consequently, a complete combustion of the hydrocarbon to CO_2 and H_2O cannot take place even at the lowest temperatures and a certain proportion of H_2, corresponding to the deficiency of oxygen, must be in excess. As the temperature increases, at first the oxygen deficiency expresses itself in the formation of CO rather than H_2, in connection with which, of course, the partial pressures of H_2O and CO_2 must change. From 2500° K on, however, the partial pressures of both products of the oxygen deficiency, H_2 and CO, increase and at the same time still other gases of incomplete combustion, such as OH, O_2, H, and O, appear, and the completeness of combustion is considerably reduced. The heat of the combustion will produce a flame temperature of 3700° K at a pressure $P = 100$ atmospheres, whereas the lower temperatures plotted may be obtained by cooling and the higher ones only by artificial heating.

Figure 3 shows the enthalpy-entropy diagram for this same oil-oxygen mixture; the case shown in figure 2 is included in figure 3 as the isobar of $P = 100$ atmospheres. Corresponding to the initial enthalpy, the state of the burned gases in the combustion chamber lies at the intersection of the isobar of 100 atmospheres with the horizontal corresponding to $i_{initial} = 2800$ kilogram calories per kilogram and at that point the flame temperature of 3700° K previously mentioned is found. If the combustion pressure is now decreased, the flame temperature falls to about 3100° K at 1 standard atmosphere as a consequence of the less complete combustion. If an adiabatic expansion in the nozzle, the states of which correspond throughout to complete gas equilibrium, is secured, the theoretical outlet velocity according to Mollier will be that shown by the velocity scale given at the extreme left of the diagram. Such theoretical velocities of expansion to 1 standard atmosphere are plotted in figure 4 as a function of initial pressure, the individual curves corresponding to various mixture ratios s, that is, weight ratios of oil to oxygen. The optimum relations are attained when $s = 1:2.9$; the curve $s = 1:2.44$ in the diagram is fairly close to this value. With an excess of oxygen, lower velocities are obtained as shown by the curves $s = 1:3.8$ and $s = 1:6.15$. The reciprocals of the velocity represent the consumptions, which are evaluated in the scale given at the right of the diagram. Here, naturally, the dashed curve for the mixture ratio $s = 1:2.44$ falls in the lowest position. In actual fact the very rapid expansion in the nozzle will leave no time for the establishment of the gas equilibrium, consequently, closer agreement with reality will be found if the gas composition in the combustion chamber is regarded as "frozen" during the whole expansion process and the velocities computed accordingly.

The size of the combustion chamber is determined by the fact that the fuel should be burned as completely as possible before entering the thrust nozzle of the rocket; that is, the gas equilibrium should be approached as closely as possible. Because for safety reasons the fuel is made up of two components, which do not meet until they enter the combustion chamber, it is the task of the injection-spray devices to produce so uniform and finely divided a macroscopic blending that turbulence and diffusion will rapidly complete the microscopic mixing, which must still be accomplished. This task exists in all combustions and a comparison of the success attained in very diversified types of firing is highly informative. In the following table, therefore, the combustion-space loadings in kilogram calories per cubic meter are compared in conjunction with the pressures in the combustion space:

TABLE I - FIRE-SPACE LOADING OF VARIOUS COMBUSTION CHAMBERS

IN KILOGRAM CALORIES PER CUBIC METER HOUR

Type of Combustion			Fuel	Combustion-chamber pressure (standard atm.)	Combustion-chamber loading (kg cal/m^3h)
Air as oxygen carrier	Normal water-tube boiler[1]		Coal	1	0.7×10^6
	High-capacity ship's boiler[1]		Oil	1	3×10^6
	High-capacity boiler[1]		Oil	2	6×10^6
	Velox steam generator[1]		Oil	3	9×10^6
	Combustion chamber of rocket type		Oil	85	1500×10^6
Burned with pure oxygen	Rocket combustion chamber	Propellant not prepared	Oil	27	1830×10^6
			Oil	100	7030×10^6
		Propellant prepared	Oil or gasoline	21	19600×10^6

[1]According to Münzinger.

If 10^6 kilogram calories per cubic meter hour is considered as a unit, which corresponds approximately to the normal water-tube boiler at 1 standard atmosphere, and air as the oxygen carrier, in good rocket combustion chambers a loading almost 20,000 times greater than usual is found. This increased loading is partly due to the higher pressure and partly to the use of pure oxygen as shown by a comparison of the combustion chambers with air and with pure oxygen. Nevertheless, these more favorable conditions do not account for all the increase obtained. An important part of the increase is gained by means of especially careful design of the injection-spray devices.

Excellence of combustion in the chamber may be recognized on a purely visual basis by the absence of afterburning in the emerging rocket jet stream. The two upper flame photographs in figure 5 show this difference for a gas-oil combustion and the two lower pictures for a gasoline combustion. In each pair the flames shown were photographed during the same experiment before and after making the sudden leap from the less favorable to the more favorable state, respectively, a change that may also be recognized by a change in the noise of the jet. Whereas the flame of afterburning was blinding to the observer, one could look directly at the exhaust when the gases were completely burned before leaving the chamber. This fact is, of course, less clearly shown in the photographs, which were taken at various shutter speeds.

6. ROCKET NOZZLE

The large pressure ratio involved in the expansion of the combustion gases from about 60 standard atmospheres in the combustion chamber to 1 standard atmosphere ambient pressure at sea level or to 0.1 standard atmosphere ambient pressure at high altitudes requires that the rocket nozzle be a Laval nozzle the cross-sectional area of which at first decreases and then increases again. At the narrowest cross section of this nozzle there exists the so-called critical pressure, corresponding approximately to one-half the initial pressure. Although the temperature of the jet falls as the velocity increases, the thermal load on the nozzle walls is decreased in respect to only the quantity of heat received by radiation, whereas the temperature drop, which is the determining factor in the heat transfer, remains almost unchanged because the boundary layer at the fixed walls also conveys part of the heat of friction to them. Therefore the calculation may be made approximately as if the temperature drop remained the same throughout the length of the Laval nozzle and only the heat-transfer coefficient changed increasing with the velocity. In regard to heat conduction, therefore an increasing quantity of heat per unit of surface area must be dealt with as the exhaust end of the nozzle is approached; whereas heat radiation is at a maximum in the combustion chamber and decreases toward the end of the nozzle because of the decreasing gas temperature and the screening of the radiant interior space, which is brought about by the form of the nozzle. These two components are superimposed to the greatest degree in the neighborhood of the narrowest cross section of the Laval nozzle where the radiation component is still approximately at full strength and the conduction component has already reached a value comparable to the final value. Nevertheless, the divergent part behind the narrowest cross section also absorbs heat to such a large degree that in order to lessen the necessary cooling it is desirable to use a short nozzle. The following considerations will show to what degree this use is possible.

For the sake of simplicity, an expansion in a vacuum, which is approximately the condition occurring at very high altitudes will be considered. Then with frictionless and complete expansion within a Laval nozzle a certain maximum velocity may be attained. The production of this maximum velocity during the pressure decrease is plotted in figure 6 using the coordinates, pressure and velocity. At the critical pressure a point of inflection is present in the s-shaped curve; this point divides the subsonic from the supersonic velocities. This state occurs in the narrowest cross section of the Laval nozzle. If the divergent part of the Laval nozzle is entirely omitted, the velocity ultimately attained is not the critical velocity corresponding to the point of inflection but instead that velocity found at the intersection of the tangent to the point of inflection and the ordinate value of the given back pressure. If the back pressure is postulated as that of a complete vacuum, then the intersection of the tangent to the point of inflection with the abscissa axis is the point in question. As may be seen from the figure, which is plotted on the assumption of ordinary air with a ratio of specific heats $\kappa = 1.4$, the final velocity even in the absence of any divergence of the nozzle reaches about 70 percent of the maximum velocity. (When the combustion gas has a value $\kappa = 1.25$, the final velocity is 60 percent.) If a divergent section is then added to the nozzle so as to produce a certain higher velocity at the end cross section, the pressure drop behind the nozzle will produce the velocities lying along another tangent, which touches the s-form loss-free curve at that point corresponding to the velocity and pressure attained at the end of the nozzle. The ratio of the area of the narrowest cross section to that of the end cross section is the reciprocal of the ratio of the slopes of the two tangents. The second tangent shown in figure 6 being about 2/3 as steep as the tangent at the point of inflection therefore corresponds to an extension of the Laval nozzle to reach a cross-sectional area 1.5 times as large as the narrowest cross section and shows that the extension will increase the outlet velocity [NACA comment: The expression "outlet velocity" refers to the velocity corresponding to the intersection of tangent and abscissa as shown in fig. 6.] to about 80 percent of the maximum velocity. Thus the divergent section has its use but the use is strictly limited because even without any divergence 70 percent of maximum velocity can be obtained, whereas the values above 90 percent can be attained only with very large divergence because of the manner in which the pressure-velocity curve is bent almost parallel to the abscissa. It will therefore be wise to find out the maximum possible velocity obtainable with a nozzle of a given length and given divergence in order to determine the optimum nozzle length.

In the case of an actual rocket nozzle, combustion gases having a ratio of specific heats κ equal to approximately 1.25 and an axially symmetrical exhaust nozzle are being dealt with. But in figure 7 only the two-dimensional case of an exhaust nozzle can be shown and air must be used as the exhaust gas with $\kappa = 1.4$. Nevertheless, substantial guidance is given by this figure. The figure summarizes the solutions found to the problem of parallel gas flowing at exactly sonic velocity through a given narrow cross section exhausting into a vacuum to produce the highest outlet velocities that can possibly be attained with the Laval nozzle ending at a given point. The attainable values in terms of the ratio to the maximum velocity are shown by the family of dashed curves. That is, each point on the 80-percent curve signifies that any nozzle that begins at the narrowest cross section and ends at the length and diameter corresponding to that point will yield at the most, with frictionless flow, an outlet velocity equal to 80 percent maximum velocity. Actually in order to obtain this optimum value, a certain definite nozzle form is, of course, necessary and this form is shown by the family of solid-line curves. Because of the twofold multiplicity of possible end points, in the sense that they differ both in distance from the narrowest cross section and in divergence, in general a twofold multiplicity of the best nozzles must also be expected. In the case of the two-dimensional exhaust nozzle, however, this multiplicity degenerates into one simple family of the best nozzle forms because there is only one best nozzle for a chosen length. This degeneration considerably simplifies the labor involved but this labor is unessential for the physical utilization of the result. It is much more essential to find the nozzle form that with the smallest arc length permits the attainment of the greatest outlet velocities and these forms are to be found where the dashed and the solid curves intersect at right angles. It is found that the angle of divergence increases with increasing nozzle length but the concave curvature is, on the whole, very small. If the diagram were extended to infinity, a 100-percent value would finally be attained with the plotted nozzle contour that has the greatest angle of divergence. By that time, however, the contour has so greatly altered direction that the length of the nozzle has become infinitely great in comparison with its outlet cross section.

7. ADMIXTURE OF SURROUNDING AIR

In a vacuum the rocket is the only power plant from which any thrust can be obtained. Because a portion of the path of the rocket must often traverse atmospheric air, however, it is desirable to improve rocket operation through the admixture of the surrounding air. Already reported elsewhere (Ber. 118 geh. d. Lilienthalges. f. Luftfahrtforschg.) is the extent of improvement of the thrust at various rocket speeds. But inasmuch as rockets are investigated first in the test rig and those are the circumstances that permit obtainment of gains with the use of surrounding air, the danger exists that we may deceive ourselves as to the effectiveness of surrounding air in the general case. Therefore, I should like to conclude with a brief discussion of the results obtained at that time. The thrust gain is plotted against the flight speed in figure 8. The velocity of sound, that is, 340 meters per second, is taken as the unit of measurement for the flight speed. The solid curves denote the thrust gain in the case of a rocket with very hot jet and high combustion pressure corresponding, for example, to the Walter power plant in hot condition. The dashed curves are computed for a cold jet and low pressure and correspond approximately to the Walter power plant in cold condition. All the curves are computed by allowing for only the mixing loss and no other friction losses in the mixing apparatus. Therefore, the curves represent optimum values, which never can be fully attained. In each of the two families of curves, the function is plotted for each of the mixture ratios 5, 20, and ∞, which mean that the rocket jet is mixed with 5, 20, and infinity times an equal weight of air, respectively. Although quite high gains, for example, more than 50 percent with a 5:1 mixture, may be obtained in the test rig, these gains decrease sharply as the flight speed increases and at a certain speed reach the zero value regardless of the mixture ratio. At this speed the use of a mixing nozzle produces no improvement in the most favorable case, and therefore in practice a decrease in efficiency in every case. Because the theory used in plotting these curves always postulates optimum mixing, after the speed is passed, positive improvements again occur. The criterions for the mixing in this range are, however, entirely different from those in the previous range and it is impossible to realize the thrust gains in both ranges with one nonadjustable nozzle. In the case of the colder rocket jet, the zero point of the thrust gain lies at a higher speed and the improvements in test-rig operation are also greater but conversely the gains after the zero point is passed are almost insignificantly small. The deduction to be drawn from this insignificant gain, namely, that the high-temperature jet produced by high-pressure combustion that is particularly designed for low consumption, permits smaller increases of thrust and reaches the zero-gain point at a lower speed, is of general validity and makes any use of a mixing nozzle appear particularly unsuited to rockets using liquid oxygen.

DISCUSSION

Wagner: May I ask whether in the last slide full account was taken of friction or whether the always positive thrust gains were only theoretical maximum values?

Busemann: The full amount of friction is by no means taken account of in the slide. On the contrary, all frictional effects that arise outside the mixing nozzle have been ignored and even within the mixing nozzle the friction with the walls has been left out of account. This omission was made because the one remaining loss, the mixing loss, is generally so great in itself that it makes the practical application of mixture very dubious. If, however, the mixture is assumed always to be brought about under the most favorable circumstances, the mixing loss can at any rate never lead to negative values of thrust gain. On this basis the slide shows only positive thrust gains or in the most unfavorable case zero gain. Of course, if any given case promises advantages in practice, the other losses must next be deducted and it can by no means be said that these losses are negligibly small in comparison with the mixing loss.

Translation by Edward S. Shafer,
National Advisory Committee
for Aeronautics.

REFERENCE

1. Zborowski, Helmut: Raketentriebwerke auf der Salpetersäurebasis und ihre spezifischen Antriebsgewichte. R-Antriebe, Schr. d. D. Akad. d. Luftfahrtforschung, Heft 1071, Nr. 82, 1943, pp. 91-119.

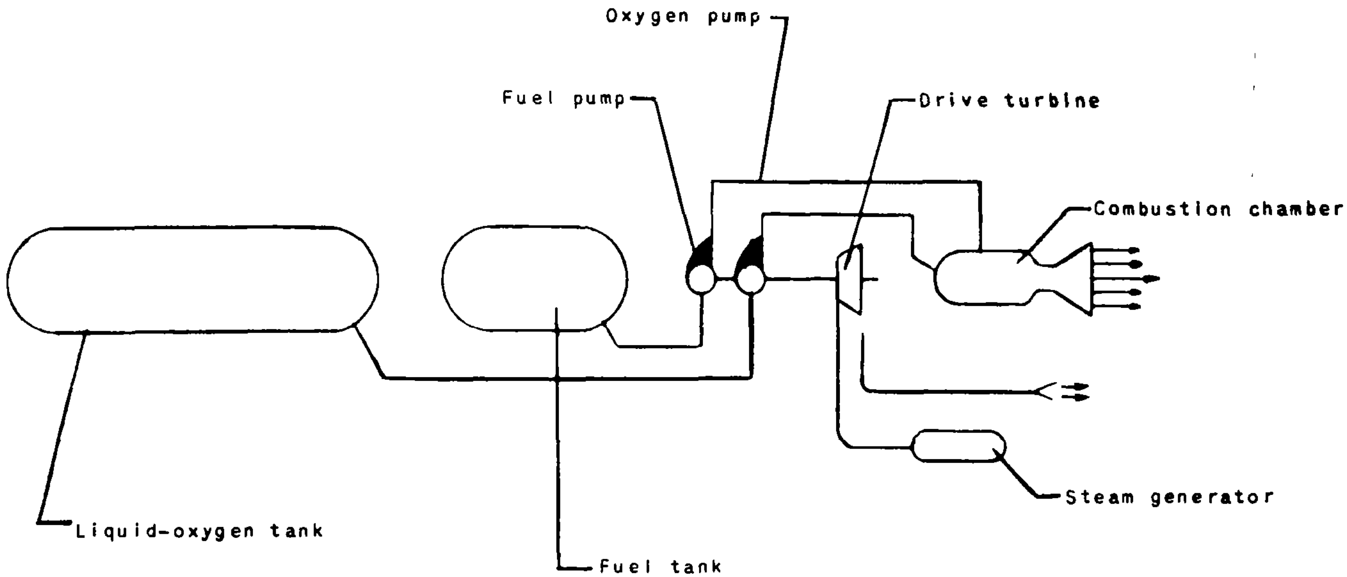


Figure 1. - Diagram of rocket drive using liquid oxygen.

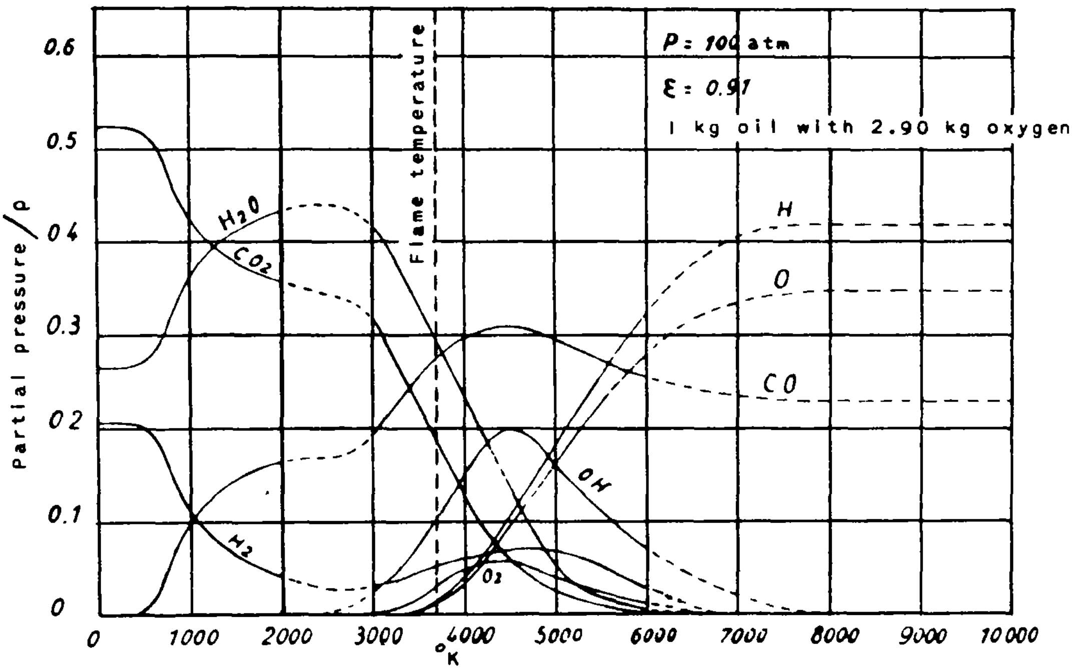


Figure 2. - Partial-pressure diagram for combustion of oil-oxygen mixture.

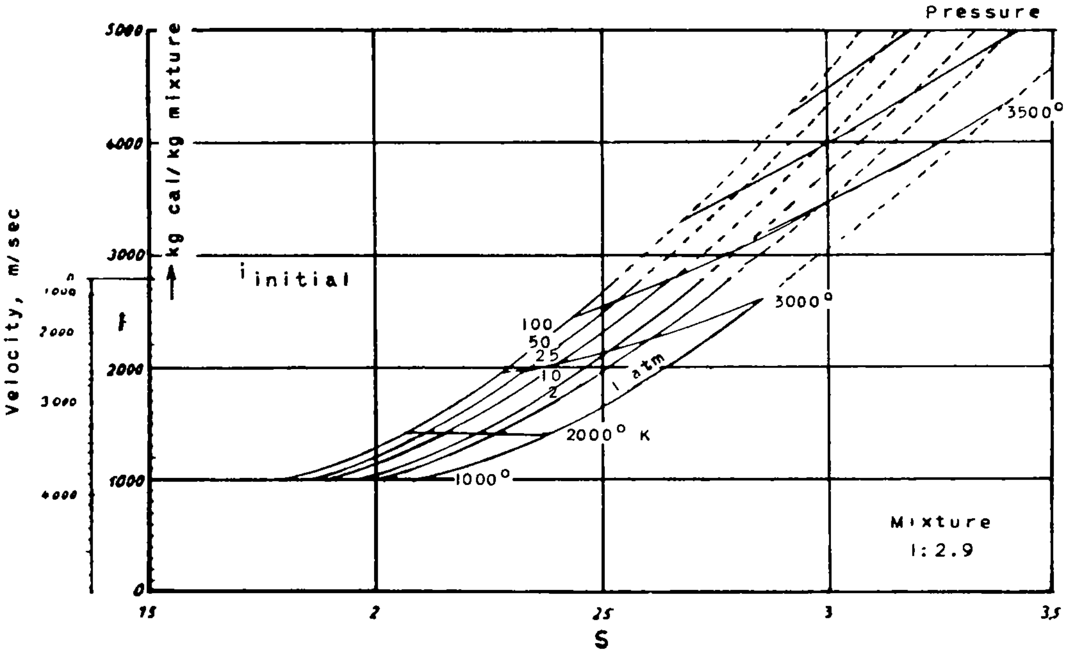


Figure 3. - I-S diagram for combustion of O_2 and $C_{14}H_{25}$.

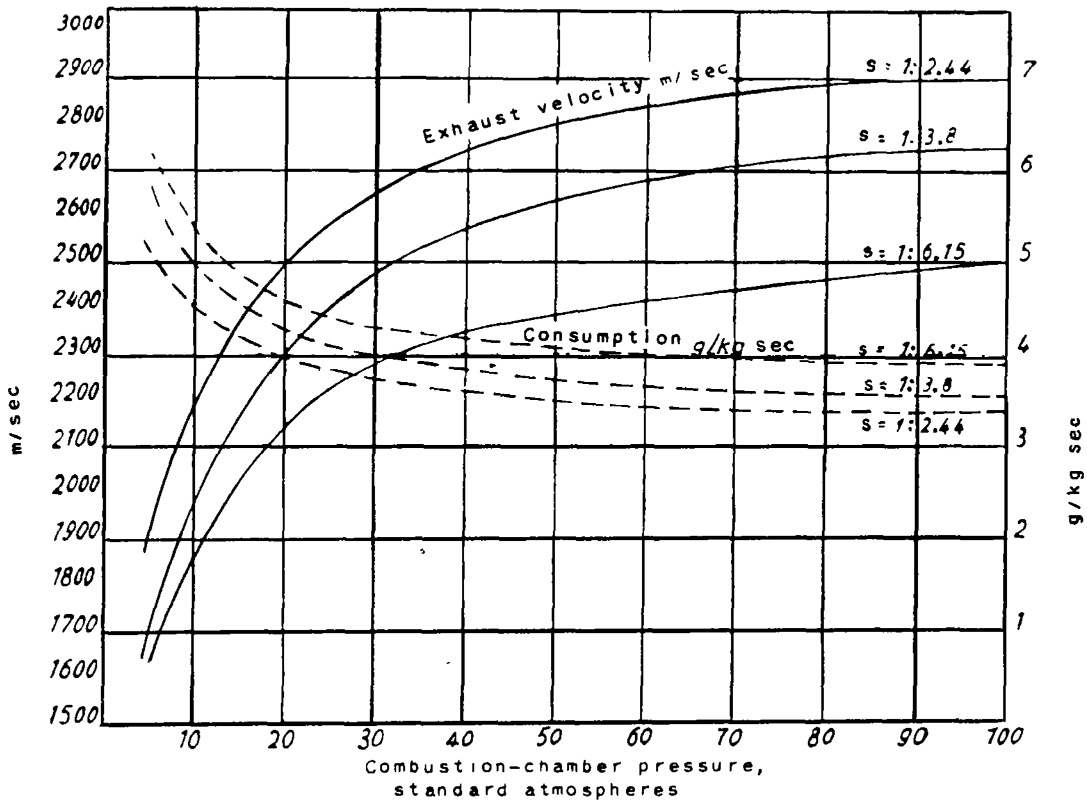
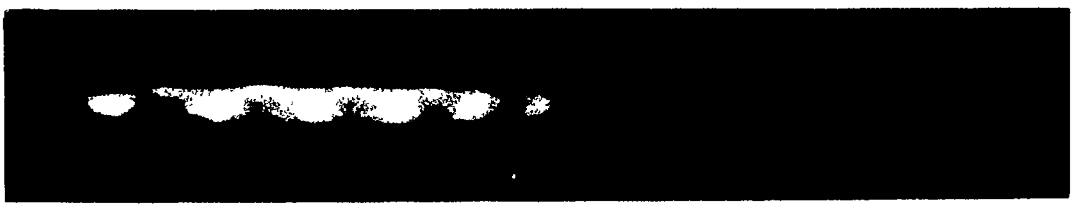


Figure 4. - Possible exhaust velocities and consumptions as functions of combustion-chamber pressure.

Photograph 1. Incomplete combustion of O_2-gas-oil mixture 1:3.22; roaring flame.

Photograph 2. Complete combustion of O_2-gas-oil mixture 1:3.38; whistling flame.



Photograph 3. Incomplete combustion of O_2-gasoline mixture 1:3.79; roaring flame.

Photograph 4. Complete combustion of O_2-gasoline mixture 1:3.4; whistling flame.

Figure 5. - Flame photographs.

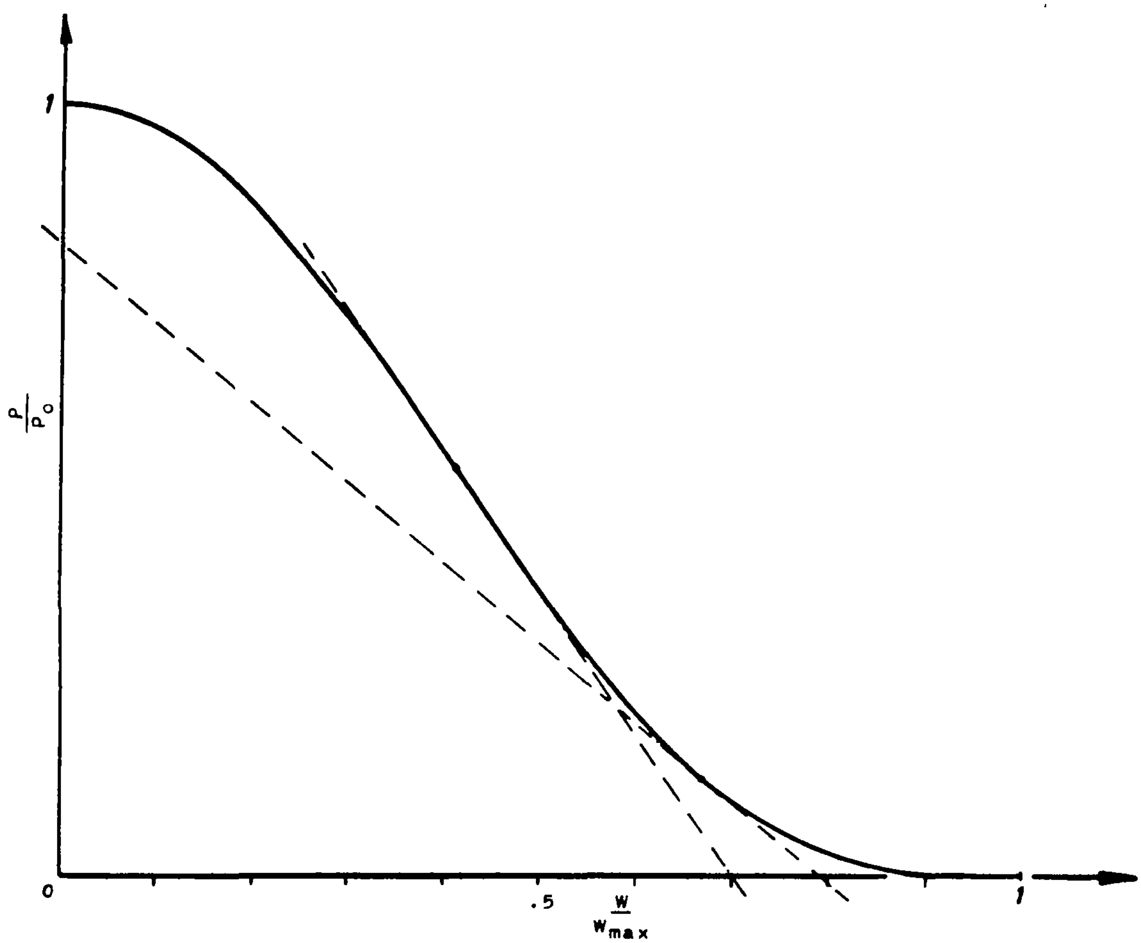


Figure 6. - Pressure-velocity curve.

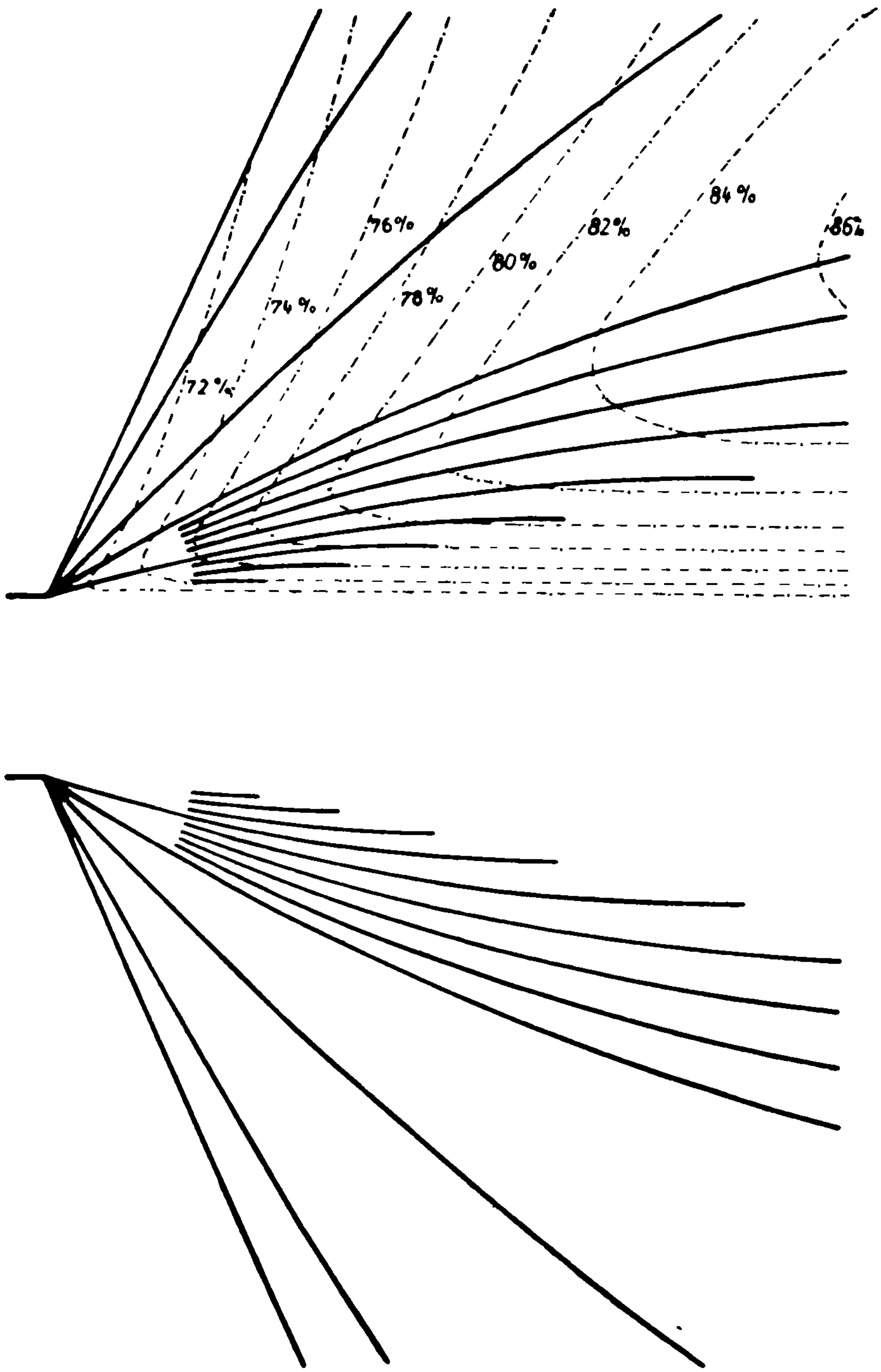


Figure 7. - Two-dimensional exhaust nozzles.

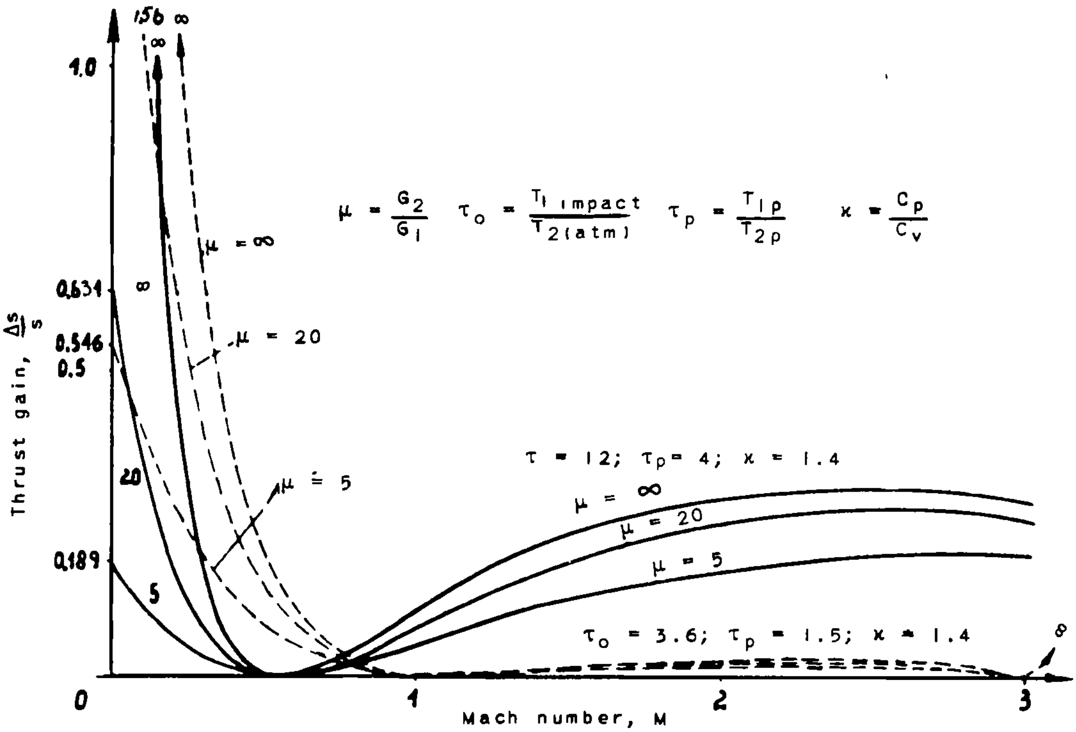


Figure 8. - Thrust gain for flight at sea level.