NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL MEMORANDUM

No. 1106

A RAM-JET ENGINE FOR FIGHTERS

By E. Sänger and I. Bredt

TRANSLATION

"Über einen Lorinantrieb für Strahljäger"
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A. CHARACTERISTICS OF THE HIGH-TEMPERATURE JET TUBE

The power plant proposed in 1913 by René Lorin consists, as is known, of only one flow channel designed in such a way that the continuously inflowing air contrary to the direction of flight is first slowed down and dammed up, then heated and accelerated beyond the inflow velocity at the discharge orifice.

In the present paper the specific assumptions are made that the lag occurs in a frustum-shaped diffuser of 10° included angle; that at its end fuels are added to the air; that the combustion then takes place in a cylindrical combustion chamber, hence at increasing combustion gas velocity and decreasing combustion gas pressure; and lastly, that the combustion gas is discharged through a conical expansion nozzle, so that the whole motor assumes the shape of a pitot tube which henceforth will be called jet tube for short.

This design along with several pressure, temperature, and velocity conditions at a certain operational state are represented in figure 1.

1. Approximate Calculation of Thrust, Performance,

Fuel Consumption, and Efficiency

The subsequent approximate calculation of the jet tube permits a quick and fairly accurate account of the fundamental characteristics of the jet tube as a power unit and rests on the approximate assumptions that the incoming air is slowed down to zero speed, that the mass of the added fuels is negligible compared to the mass of

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*"Über einen Lorinantrieb für Strahljäger." Deutsche Forschungsanstalt für Segelflug E.V., Ernst Udet, Ainring, Oberbayern, 1943; Deutsche Luftfahrtforschung, Untersuchungen und Mitteilungen, Nr. 3509. Published by Z.W.B., Berlin-Adlershof.

air, that no flow losses occur in the jet tube, and that the specific heat before and after combustion is the same.

The thrust P as the vectorial sum of all excess pressures of the gas on the inside surface of the tube is then:

$$P = \rho_1 F_1 v_1 (v_4 - v_1) \approx \rho_1 F_1 v_1^2 \left(\sqrt{T_3/T_2} - 1\right) \tag{1}$$

where

F_1 cross-sectional area of inflow, meters^2

ρ_1 density, kilogram-seconds^2 per meter^4

v_1 speed of inflowing air, meters per second

v_4 speed of outflowing combustion gases

T_2, T_3 absolute stagnation temperatures of inflowing and outflowing medium

These stagnation temperatures follow from the energy equation at

$$T_2 = T_1 + \frac{Av_1^2}{2gc_p}$$

and

$$T_3 = T_4 + \frac{Av_4^2}{2gc_p}$$

where

A mechanical equivalent of heat, kcal/kgm

g gravitational acceleration, meters per second^2

c_p specific heat of gases at constant pressure, kcal/kg°

The heat input Q (in kcal/sec) required to produce this thrust is

$$Q = \rho_1 F_1 v_1 g c_p (T_3 - T_2)$$

Since with aerodynamically correct arrangement the flying speed v is about equal to v_1, the efficiency of the entire power unit can be expressed as:

$$\eta = \frac{APv}{Q} = \frac{Av^2}{gc_p(T_3 - T_2)}\left(\sqrt{T_3/T_2} - 1\right) \tag{2}$$

This efficiency is represented in figures 2 and 4 for $A = 1/427$ kcal/kgm, $c_p = 0.28$ kcal/kg°, $T_1 = 288°K$ and $216.5°K$, and $g = 9.81$ m/sec^2.

Both the thrust and the efficiency of the jet tube therefore increase as the square of the flying speed.

In consequence, the thrust can be represented by a propulsive efficiency c_v referred to the maximum cross-sectional area F_3:

$$c_v = \frac{2P}{\rho_1 F_3 {v_1}^2} = 2\left(\sqrt{\frac{T_3}{T_2}} - 1\right)\frac{F_1}{F_3} \tag{3}$$

These propulsive efficiencies of the jet tube also are shown numerically in figures 3 and 5 for $T_1 = 288°K$ and $216.5°K$ and for the value $F_1/F_3 = 0.16$ chosen after test experiences. Several c_v values obtained by accurate calculation (chapter A2) also were included. The added fuel mass involved effects, because of the necessary enlargement of F_4/F_1, an increase in c_v amounting to about 11 percent at the high temperatures; the considered variation of the c_p a further increase by about 3 percent without both influences causing any material change of c_p over v. The finite quantity of F_2 with $F_2 = 6.25F_1$ instead of $F_2 = \infty$

taken into account in the exact calculation of the thrust induces a decrease in c_v with increasing v owing to the compressibility, that is, the latter effect acts against the other two in such a way that the absolute magnitude of the c_v is fairly well confirmed by the approximation, but its variation, moderate by itself, decreases with v or even reverses at high temperatures.

Lastly, the fuel consumption in kilograms per second ton thrust can be simply expressed with

$$B = \frac{Q}{10^{-3}PH} = \frac{Av}{10^{-3}\eta H} \tag{4}$$

H denotes the heat value in kcal(kg).

This value is also included in figures 2 and 4 on the assumption of $T_1 = 288^\circ K$, $T_1 = 216.5^\circ K$, and $H = 10,000$ kcal/kg.

Figures 6 and 7 show the maximum thrust $\frac{P}{F_2} = \frac{P}{F_3}$ and the maximum thrust horsepower $Pv/75F_2$ per meters^2 of maximum cross-sectional area of jet tube with the respective fuel consumption B and efficiency η for stoichiometric hydrocarbon combustion on the basis of $H = 10,000$ kcal/kg; $F_1/F_2 = 0.16$; $c_p = 0.30$; $(T_3 - T_2) = 2060^\circ$ and standard atmosphere for all altitudes of flight and flying speeds in question.

The effect of flying speed and altitude on thrust, thrust horsepower, fuel consumption, and efficiency of the high-temperature jet tube is readily apparent from these diagrams.

2. Accurate Calculation of the Jet Tube

The exact check of the flow process in the jet tube has the double purpose of verifying the accuracy of the approximate calculation for thrust, power, fuel consumption, and efficiency and of securing data on the design of the jet tube and the quantities of state of the gases in the tube.

It is accomplished with the equation of energy, the momentum theorem, the continuity relation and equation of state with allowance for the added fuel masses, the variation of the specific heats and the adiabatic exponents in respect to temperature and mixture composition, as well as the actual maximum cross sections of the tube, hence with the following relations: Between F_1 and F_2 the equation of energy is

$$\bar{c}_p T_1 + \frac{A}{2g} v_1^2 = \bar{c}_p T_2 + \frac{A}{2g} v_2^2$$

equation of continuity

$$\gamma_1 F_1 v_1 = \gamma_2 F_2 v_2 = G_L$$

equation of phase change

$$\frac{T_1}{T_2} = \left(\frac{\gamma_1}{\gamma_2}\right)^{x-1}$$

between F_2 and F_3 the equation of energy is

$$G_L\left(c_p T_2 + \frac{A}{2g} v_2^2\right) + G_B H = \left(G_L + G_B\right)\left(\int_{T=0}^{T_3} c_p \, dT + \frac{A}{2g} v_3^2\right)$$

momentum theorem

$$F_2\left(p_2 - p_3\right) = \frac{1}{g}\left[\left(G_L + G_B\right)v_3 - G_L v_2\right]$$

equation of continuity

$$G_L + G_B = F_2 \gamma_3 v_3$$

equation of state

$$\frac{p_2 M_2}{\gamma_2 T_2} = \frac{p_3 M_3}{\gamma_3 T_3} = R_o$$

between F_3 and F_4 the equation of energy is

$$\int_{T=0}^{T_3} c_p \, dT + \frac{A}{2g} v_3^2 = \int_{T=0}^{T_4} c_p \, dT + \frac{A}{2g} v_4^2$$

continuity relation

$$G_L + G_B = F_4 \gamma_4 v_4$$

equation of phase change

$$\frac{x_u}{x_u - 1} \ln \frac{T_3}{T_4} + \sum_{i=1}^{9} \frac{p_i}{p_3} \left[\frac{\theta_i/T_3}{e^{\theta_i/T_3} - 1} - \frac{\theta_i/T_4}{e^{\theta_i/T_4} - 1} - \ln \frac{1 - e^{-\theta_i/T_3}}{1 - e^{-\theta_i/T_4}} \right] = \ln \frac{p_3}{p_1}$$

(whereby p_i denotes the partial pressures, θ_i the characteristic temperatures of H_2O, CO_2, N_2, and O_2).

Equation of state

$$\frac{p_4 M_4}{\gamma_4 T_4} = \frac{p_1 M_3}{\gamma_4 T_4} = R_o$$

The flow losses, heat losses, effects of dissociation, and incomplete combustion not taken care of by the calculation are evaluated separately; so, while the exact calculation offers no fundamental difficulties, the paper work involved is considerable compared to the approximate method, and the general representability of the results in closed form is lost.

Only for the exact calculation of the propulsion coefficients $c_v = P/qF_2$ can closed formulas be given in two specific limiting cases. One of these refers to the practically unimportant low flying speed where the flow can be regarded as incompressible; hence, with flow losses disregarded

$$c_v = \frac{1 - \frac{F_4}{F_2}}{1 + \frac{F_1}{F_2}} \frac{p_2 - p_1}{q} - \frac{1 - \frac{F_4}{F_3}}{1 + \frac{F_4}{F_3}} \frac{p_3 - p_1}{q} = \frac{\left(1 - \frac{F_1}{F_2}\right)^2 - \frac{F_1}{F_4}\left(1 - \frac{F_4}{F_2}\right)^2}{\frac{F_2}{F_4} + \frac{F_4}{F_2}} \tag{5}$$

If the jet tube is correctly designed, the propulsion coefficient in this case can be deduced from surface ratios or easily measurable excess pressures and surface ratios. With $F_2/F_1 \longrightarrow \infty$ and $F_4/F_2 \longrightarrow 1$, c_v approaches the limit value $c_v = 1$.

The second extreme case occurs when the input energy between sections F_2 and F_3 is exactly so great that $p_3 = p_4 = p_1$, and $F_4 = F_3 = F_2$.

In that event

$$c_v = \frac{p_2 - p_1}{q} + 2\frac{F_1}{F_2}\left(\frac{v_2}{v_1} - 1\right) \tag{6}$$

where

$$\frac{p_2 - p_1}{q} = \frac{\left[\frac{x - 1}{2}\left(\frac{v_1}{a}\right)^2 - \frac{x - 1}{2}\left(\frac{v_2}{a}\right)^2 + 1\right]^{\frac{x}{x-1}} - 1}{\frac{x}{2}\left(\frac{v_1}{a}\right)}$$

and

$$\frac{F_1}{F_2} = \frac{v_2}{v_1}\left[\frac{x-1}{2}\left(\frac{v_1}{a}\right)^2 - \frac{x-1}{2}\left(\frac{v_2}{a}\right)^2 + 1\right]^{\frac{1}{x-1}}$$

With $F_2/F_1 \longrightarrow \infty$ and $x = 1.4$ the extreme value of c_v approaches

$$c_v = \frac{\left[1 + 0.2\left(\frac{v_1}{a}\right)^2\right]^{3.5} - 1}{0.7\left(\frac{v_1}{a}\right)^2}$$

Restricted to flying speeds up to $v/a = 0.9$ so as to avoid local supersonic velocities, the propulsive efficiency of nozzleless tubes can at the most amount to $c_v = 1.22$ on the assumption of sufficient input energy.

This second theoretically interesting limiting case of a jet tube without special discharge nozzle, hence incorporating only the frustum-shaped diffuser and the cylindrical combustion chamber, is represented in figure 8, where the pressure rise in the diffuser (referred to $q = \rho v^2/2$) for loss-free diffuser flow, hence 100-percent diffuser efficiency, and c_v is plotted for different Mach numbers v/a independent of the height of energy input against all possible maximum cross-section quantities F_1/F_2.

Using high-grade hydrocarbons, such as octane, for fuel, the energy input with 658.3 kcal/kg, referred to discharge gas is limited and the then still possible propulsion coefficients themselves are limited to $c_v = 0.74$ for $F_1/F_2 = 0.266$ at sea level, and to $c_v = 0.79$ for $F_1/F_2 = 0.232$ at a 12-kilometer altitude of flight.

Utilizing other fuels, such as of light metal dispersions, metal alkyls or light metals, such as aluminum or magnesium which even burn additionally with atmospheric nitrogen, the upper limit of the obtainable c_v-values can be pushed still higher.

These considerations on nozzleless jet tubes are of theoretical interest only since the nozzleless jet-tube design has a number of practical drawbacks compared to the jet tube with discharge nozzle; for example, it does not yield the highest possible c_v-values, has greater fuel consumption, inferior strength properties, and high cold-resistance coefficients.

The arguments already indicate, however, how concrete the assumptions regarding fuel, flying height, ratio of surfaces F_1/F_2, and so forth, must be fixed for exact calculation if the connective data on phase quantities, design, and performance characteristics of a certain operating state are to be made.

In the subsequent exact calculation of more general conditions several concrete premises, interesting for the design of jet tubes, must be advanced again.

These special assumptions are chiefly as follows:

1. The chosen fuel is octane, with a low heat value of $H = 10,600$ kcal/kg and an air requirement of 15.11 kilograms per kilogram of fuel. The combustion was computed for stoichiometric octane-air ratio, and for excess of air, the factor m denotes the multiple of the stoichiometric fuel-air ratio. The values $m = 1.0$, 0.9, 0.8, 0.7, and 0.1 were taken into consideration.

In the cases of 1.0 and 0.9 the already traceable dissociation of CO_2 in CO and O_2 was disregarded. The error introduced into the unfavorable case of $m = 1.0$ and 12 kilometers' flying height amounted to $\Delta T_3 = -120^{\circ}$ and caused a deterioration in c_v by $\Delta c_v = -0.02$.

2. The calculations were made on the assumption of standard atmosphere for 0.4- and 12-kilometer altitudes. In the choice of these heights the height range important for the experimental towing flights played a part. Incidentally, it should be noted that, under otherwise identical conditions, c_v, v_4, and appropriate F_4 are more susceptible to air temperature variations T_1 than to changes in the air pressure p_1.

The acceleration $(v_4 - v_1)$ of the air mass $\rho_1 F_1 v_1$ depends, according to equation (1), not on the absolute height of the heat input volume Q per unit of weight but on the ratio of Q to initial energy content $\left(c_pT_1 + \frac{Av_1^2}{2g}\right)$, conformally to

$$P \approx \rho_1 F_1 v_1^2 \left[\sqrt{1 + Q\Big/\left(c_pT_1 + Av_1^2/2g\right)} - 1\right]$$

Thus the thrust can be raised by increased heat input as well as by reduced initial heat content, that is, lower T_1.

Therefore c_v and F_4 vary perceptibly with T_1, for equal energy input Q, equal v_1/a, and equal p_1; while a change in air pressure p_1 alone is ineffective, within the mathematical accuracy, as far as c_v and F_4 are concerned.

The increase of c_v with decreasing temperature T_1 is due to the fact that, with $T_1 \longrightarrow 0$ under otherwise constant conditions, the adiabatic pressure rise in the diffuser $(p_2 - p_1)$ becomes infinite. These facts must be borne in mind at departures of the actual atmosphere from the standard atmosphere.

3. In the selection of the ratio of diffuser inlet area to F_1/F_2 four factors affected by this ratio play a decisive part. They are:

The coefficient of propulsion c_v

The warm resistance coefficient c_{ww}, which acts against c_v

The fuel consumption B per second ton of thrust, and the total efficiency η

The fresh-air velocity v_2 and the combustion-chamber charge in kcal per hour and cubic meter discharge passage governing completeness of combustion, respectively

The most important of these quantities are represented in figures 9 and 10 for the flying speeds $v/a = 0.3$ and 0.9 according to the results of the exact calculation against F_1/F_2 for various flying heights without consideration to the flow losses.

The variation of all four quantities for $H = 12$ kilometers and $v/a = 0.9$ is indicated in the following table; the c_{ww} were computed from the wall friction on a tube of 2.50 meters maximum cross-sectional diameter:

F_1/F_2	c_v	c_{ww}	B (kg/sec)	v_2 (m/sec)
0.100	0.445	0.106	1.10	18
.160	.676	.093	1.16	29
.218	.802	.074	1.34	40
.232	.793	.067	1.43	43

The c_v-values therefore increase consistently with decreasing F_2 to a maximum near $F_2 = F_4$, and depend on H and v.

These maximums are due to the fact that, for given specific energy absorption and hence given F_1, the thrust starting at $F_2 = F_1$ increases continuously less, so that the c_v referred to F_2 must pass through an extreme.

Thus the highest c_v-values are obtained with jet tubes that have a small discharge nozzle with, say, 10- to 20-percent constriction of the maximum cross-sectional area. The coefficients of the free thrust $(c_v - c_{ww})$ give a flatter maximum because of the steady decrease of c_{ww} with increasing F_1/F_2.

But with decreasing F_2 the fuel consumptions B themselves increase and especially so for F_1/F_2 values above about 0.16, so that the fuel consumption of the tube of maximum c_v is about 20 to 30 percent higher than that of the $F_1/F_2 = 0.16$ tube.

Therefore, if no other restrictions existed, the choice of F_1/F_2 could be made according to the purpose of use, that is, for short-period operation at highest possible thrust with greater F_1/F_2 values, for longer periods with greater demand on efficiency, with smaller F_1/F_2 values.

Actually, however, there is a limitation of the F_1/F_2 in the permissible fresh-air velocity v_2 and the discharge passage charge $2.5 \times 10^6 \gamma_2 v_2 / l$ in $(\mathrm{kcal/m^3h})$ (l denoting the length of the combustion chamber in meters).

For the first design of the experimental tubes a discharge duct charge of 27,000,000 $\mathrm{kcal/m^3h}$ referred to standard density was regarded as admissible, which, for stoichiometric combustion gives a still permissible $v_2 \leqq 34.9$ m/sec.

At other air densities the admissible discharge duct charges are proportional, so that v_2 remains the same.

From figure 10 it is seen that, in these conditions and when Mach number 0.9 at sea level is to be reached, F_1/F_2 should not exceed 0.165; while the corresponding value at 12-kilometer flight altitude amounts to $F_1/F_2 = 0.190$. These data follow from the relation

$$\frac{F_1}{F_2} = \frac{v_2}{v_1}\left[\frac{x-1}{2}\left(\frac{v_1}{a}\right)^2 - \frac{x-1}{2}\left(\frac{v_2}{a}\right)^2 + 1\right]^{\frac{1}{x-1}}$$

For the further theoretical and experimental investigations an $F_1/F_2 = 0.16$ was chosen.

With these three assumptions on fuel, atmosphere structure, and surface ratio F_1/F_2, the exact calculations of the jet tube can be carried out with flow losses disregarded.

Figures 11 to 16 contain the most important numerical data with which the constructor of the jet tube and the experimental engineer must work.

Figures 11 and 12 represent the gas flow velocities v_1, v_2, v_3, and v_4, the gas temperatures T_1, T_2, T_3, and T_4, at 0-kilometer flight altitude for all flying speeds v/a and for the fuel ratios $m = 1.0, 0.9, 0.8, 0.7, 0.1$ the pressure increases referred to dynamic pressure $q = \rho v^2/2$

$$\frac{\Delta p_2}{q} = \frac{(p_2 - p_1)}{q}; \quad \frac{\Delta p_3}{q} = \frac{(p_3 - p_1)}{q}$$

the c_v-values derived from it and the important quantity of the nozzle mouth area in the form of F_1/F_4.

Figures 13 and 14 contain the same quantities for 4-kilometer flight altitude, figures 15 and 16 the same quantities for 12-kilometer altitude.

The more accurate jet-tube calculation represented here makes no allowance for a number of minor factors of potential influence on the results secured, such as, incomplete combustion, discharge gas dissociation, heat loss due to convection radiation and convection of discharge gases, flow losses in diffuser and in the other flow passages, and so forth.

The various types of loss and their characteristic nature are briefly discussed.

Completeness of combustion depends upon the completeness and the rapidity of the mixture formation between fuel and fresh air, which is reached by a very large number of well-atomizing fuel sprays with high relative speed against the fresh air; upon the intensity of the ignition effected by injection of small quantities of zinc diethyl in the ignition zone or by electric spark plugs; and upon the duration of the combusting mixture in the combustion chamber, which is assured by a sufficiently low discharge passage charge as already indicated for the choice of the permissible F_1/F_2.

At high discharge temperatures the completeness of the combustion is, of course, very high, as manifested later by the discharge temperature measurements; while at low T_3 values special constructive measures, such as partial combustion at high temperature and subsequent admixture of air, should be resorted to. In the experimental flights at 1- to 3-kilometer altitude it ranged above 70 percent. Altitudes above 18 kilometers are probably questionable owing to the decreasing completion of combustion in the thin air.

A special form of incomplete combustion is presented by the thermal dissociation of the discharge gases at very high combustion temperatures. It can, as already stated, become traceable in the cases $m = 1.0$ and 1.9, especially by the decomposition of CO_2 in CO and O_2 and in unfavorable conditions, as, for example, at $m = 1.0$ and 12-kilometer altitude results in a temperature drop by about $\Delta T_3 = 120^\circ$ and a decrease in c_v by $\Delta c_v = 0.02$.

Of course, it is to be expected that this dissociation actually occurs for correspondingly high combustion energies and in sufficiently high altitude, since the duration of the combustion gases for 10^{-2} to 10^{-1} second in the combustion chamber is sufficient for it and a recovery of dissociation losses in the nozzle is unlikely owing to existing high discharge temperatures and low mean molecular shock factors; but for the range of application in question the path range of very high discharge temperatures and at the same time very low air pressures is traversed in such short periods that the losses play only a secondary part and for the time being do not warrant the substantially greater task of a calculation with allowance for the dissociation.

The heat losses due to discharge gas radiation can be approximated by assuming the radiation at about 20 percent of that of a black body of equal temperature T_3 and selecting a spherical surface of the same diameter as the maximum cross section as radiating surface.

The relative heat losses increase further with F_2/F_1, since the absolute radiation at constant rate of heat flow increases with F_2/F_1.

By this method the pure radiation losses figure at about 10.5 percent for $T_3 = 2400^\circ K$ at 12-kilometer altitude for 0.3 times sonic velocity and $F_2/F_1 = 6.25$.

For the jet tube with $F_2/F_1 = 6.25$ employed in the fighter discussed here the radiation losses are expected to average 2.5 percent of the total heat input at 12-kilometer altitude for 0.9 times sonic velocity. They appear for a very short period only, since the tube after completed ascent continues its flight with greatly throttled combustion and temperatures of around $T_3 = 800^\circ K$, where only about 0.06-percent radiation losses are to be expected.

These data do not include the losses by luminous radiation of the chemical reaction, the fundamental data of which are still lacking.

The heat losses by convection for octane combustion in the stoichiometric ratio will amount to about one-third of the temperature radiation losses. This fraction increases with decreasing T_3.

By diffuser efficiency is meant the ratio of excess pressure actually reached at the diffuser end to the adiabatic excess pressure computed lossfree; the losses due to accumulation and separation of the gas particles flowing slowly past the wall appear in direction of increasing pressure. It is defined as $\frac{p_2{}^x - p_1}{p_2 - p_1}$, $p_2{}^x$ denotes the pressure actually recorded in F_2 relative to the pressure p_2 computed by means of the adiabatic laws of flow. For the circular-conical enlargement of the jet-tube diffuser and its aerodynamically smooth inside walls the diffuser losses are chiefly dependent upon the included angle of the diffuser and the flying speed v on the assumption that the section of F_4 is aerodynamically designed, so that the flow fills the entire tube exactly correct.

The relationship between diffuser losses and included angle is based on investigations which were confirmed by personal towing tests and according to which the diffuser efficiency first decreases slowly then increasingly more with increasing included angle.

To avoid the drawback of an excessive diffuser length, a 10° included angle was chosen for the theoretical and the experimentally tested jet tube, at value that also represents the limit at which the steep drop in diffuser efficiency starts.

With this included angle and flying speeds of around $v/a = 0.3$ personal flight tests in Göttingen yielded, at even lower speeds, the known model tests by Peters (Ing. Archiv, 2, p. 92, 1931), diffuser efficiencies at 90 percent.

For the diffuser efficiencies at high speed near the Mach number 1 the wind-tunnel tests of the Italian, Aymerito (Ricerche Ing., No. 1, p. 16, 1935) are available, who worked with a diffuser of 6° included angle. Aymerito defined the efficiency at $\eta_{diff} = p_2{}^x/p_2$ and quotes a 99.5-percent decrease in efficiency between 228 and 335 meters per second which then continues to drop steeply on reaching the boundary of sonic velocity.

At 0.9 times sonic velocity about $\eta_{diff} = 0.975$ would accordingly be obtainable or $\eta_{diff} = 0.94$ according to the definition employed here, when the included angle of the diffuser is 6°. For the explored diffuser of 10° the efficiency will be slightly worse.

The effect of this pressure loss in the diffuser on the thrust coefficient and on the dimensions of F_4 in the incompressible as in the compressible range of flow is the object of a special investigation.

The subject is concluded with a brief discussion of the loss due to friction of flowing gas at the walls of the cylindrical combustion chamber which becomes evident in a pressure loss $p_3 - p_3^x = \lambda \frac{l}{d} \frac{\overline{\rho v}^2}{2}$, where $\overline{\rho}$ and $\overline{v}$ signify the mean density and mean flow velocity between F_2 and F_3. For the jet tube with 4-meter chamber length, 4.91 meters^2 maximum cross section and 0.9 times sonic velocity in flight at sea level, $p_3 - p_3^x$ amounts to about 25 kilograms per meter^2 or 0.0025 at atm on the assumption of a resistance factor $\lambda = 0.02$. This friction loss is of interest for the reason that, according to the definition of the thrust as the vectorial sum of all internal gas excess pressures, the result is a gain of propulsion, which, it is true, is more than consumed by the simultaneously occurring high frictional forces.

3. Technical Characteristics of the High-Temperature Jet Tube

The outstanding technical characteristics of the jet tube described are:

1. The static thrust is zero and practically increases like the air forces with the square of the flight speed and with the root of the discharge temperature and at 300 meters per second at sea level reaches more than 3000 kilograms per square meter of maximum cross-sectional area, corresponding to a $c_v = 0.57$, possible with stoichiometric hydrocarbon combustion.

2. The static efficiency is also zero and increases with the square of the flight speed to reach 6 percent at 300 meters per second in vicinity of the ground with stoichiometric hydrocarbon combustion and more than 10 percent at lower discharge temperatures.

3. The thrust-horsepower increases accordingly with the third power of the flight speed and at 300 meters per second in ground vicinity reaches about 12,000 horsepower per square meter of maximum cross-sectional area.

4. The fuel consumption per second per ton thrust drops approximately proportionally to the reciprocal value of the increasing

flight speed to values of around 1.2 kilograms per second at 300 meters per second in ground proximity for stoichiometric hydrocarbon combustion and to 0.7 kilogram per second at lower discharge temperatures.

5. For constant flying speed at higher flight levels the absolute thrust and the thrust horsepower decrease slightly less than the air density; propulsive coefficient and efficiency, on the other hand, increase owing to the decreasing fresh-air temperature, so that at stratosphere temperature even at 265 meters per second flight speed, propulsion values up to 0.68 are to be expected; while the efficiency for stoichiometric hydrocarbon combustion is around 5 percent and at lower discharge temperatures is more than 9 percent. Accordingly, the fuel consumption ranges around 1.2 to 0.6 kilograms per second.

6. Thrust and thrust horsepower of the jet tube are, for given flight altitude and flight speed from maximum to zero, controllable by adjustment of the discharge nozzle end section and the fuel injection, with the efficiency increasing at first with increased throttling.

7. For the design of high-power units it is important that the enlargement of the absolute dimensions of the device are neither limited by rotating parts, such as on continuous compressor units, nor by nonstationary flow and combustion processes, such as on periodic compressorless jet-power units, so that units with thrust horsepowers of the order of magnitude of five-place horsepower figures seem feasible.

8. Used as airplane power unit the weight per horsepower of the jet tube referred to maximum output is expected at around 0.02 kilogram per horsepower according to past design experience.

9. In correspondence with the low structural weight and the absence of all moving parts the outlay for manufacture and maintenance is low and the life expectancy great.

10. Since the operating process of the jet tube consists merely of a steady combustion process, the power plane is very insusceptible to the type of fuel employed, so that instead of gasoline even heavier hydrocarbons as well as special fuels, such as alcohols for obtaining invisible flames for night fighter use or metal dispersions or metal alkyls for further increasing the flame temperature and hence of the propulsion efficiency, come into question.

11. For practical flight operation the low susceptibility against climatic conditions, such as frost, sand, and so forth, is important, and so also the ready availability without warming up and the invulnerability of the entire unit.

12. For military purposes the little noise of the jet tube at full operation is significant.

B. EXPERIMENTS

The theoretical results of the high-temperature jet tube were so favorable that a quick experimental proof seemed desirable.

In view of the increasing danger to the country by enemy bomber attacks the tests were planned for immediate application to aircraft. Thus the time-consuming construction of stationary test stands was ruled out and the towing-test method chosen.

The lesser accuracy of the towing tests is overbalanced by the advantages to the extent that in the towing test the actual state of the fresh air, especially regarding turbulence, density, humidity, and so forth, was utilized and that the temperature conditions of the combustion chamber walls occurring in actual flight and the relative sizes of the total unit are actually realized in the test.

1. Street Towing Tests on Automobile

up to v = 25 m/sec

First it was necessary to check whether in the comparatively large combustion chambers of the high-temperature jet tubes the necessarily high discharge temperatures and at the same time the necessary discharge duct loads of more than 20,000,000 kcal/m^3h were actually obtainable.

Since, for structural reasons the length of the combustion chamber was not to exceed 4 meters and since with hydrocarbon combustion about 850 kcal are convertible per $meters^3$ air of normal conditions, fresh-air velocities of the order of speed of fast trucks appear suitable in first approximation.

Thus it was possible to imitate the processes in the cylindrical combustion chamber of the flying jet tube to a certain degree in a cylindrical pipe towed by means of an automobile.

Several such towing tests are illustrated in the photographs (figs. 18 and 19). Gasoline was injected and atomized by conventional spray nozzles near the forward pipe edge and the gasoline-air mixture ignited by continuous injection of small quantities of zinc diethyl.

After a greater number of test runs the combustion in the pipe was successfully brought to a silent state and to fuse a 0.05-millimeter gage, freely stretched Pt-Ir wire of 30-percent Ir content and 2180 °K fusion point 2.20 meters in a pipe of 880-millimeter diameter aft of the fuel injection. Considering the still substantial heat radiation of the fusion wire, it may be assumed that the theoretical combustion temperature of the gasoline-air mixture of around 2400 °K was fairly reached.

Other tests in which the gasoline was preheated by the combustion flame gases and vaporized or replaced by other fuels, such as gas oil, aluminum-gas oil dispersions or aluminum dust resulted in ignition and installation difficulties and were for the time being abandoned.

But a still larger number of towing runs with complete jet tubes were made, as indicated in photograph (fig. 20). In these runs the pressure in the discharge passage was carefully measured and the included angle of the diffuser systematically varied, in order to find out whether angles greater than 10° could be used. Since at these low running speeds and the prescribed orifice ratio F_1/F_2 the fresh-air velocity v_2 can be approximately neglected, the diffuser efficiency can be represented by the ratio of recorded pressure rise in the diffuser to the dynamic pressure. The obtained decrease in diffuser efficiency $\Delta p_2^{x}/q$ with increasing including angle is represented in figure 17 and does not let shorter diffusers appear suitable.

2. Flight Towing Tests with 2400-Horsepower Tube on the Do 17 up to v = 100 m/sec

The jet tube had an inlet section diameter of 200 meters, a 10° diffuser included angle, and a combustion chamber length of

2960 millimeters with a diameter of 500 millimeters, hence a rated output of about 2400 horsepower at $v = 300$ meters per second.

The fuel was injected through several spray nozzles at the diffuser end and at the tube periphery, the ignition was effected by zinc-diethyl injection.

In the 70-airplane towing tests of this phase of the tests the discharge nozzle end sections F_4, the number and direction of the sprays of the injection nozzles, and the injection pressures were varied.

The method of suspension of the jet tube on the towplane is seen from the photographs (figs. 21 to 24).

It permits the recording of drag and thrust by recording spring dynameter within the airplane fuselage.

The excess pressures relative to the reference pressure of the outside air at the start and the end of the cylindrical combustion chamber were also recorded.

The individual test generally lasted more than 300 seconds; the fuel was supplied by pressure tanks.

The tests were evaluated as follows:

If c_Δ denotes the coefficients referred to dynamic pressure and maximum cross section of the difference in force recorded by dynamometer with and without combustion in the tube c_{wk}, the resistance of the cold tube, c_{ww}, the resistance of the hot tube, and c_v the propulsion, the desired c_v is equal to

$$c_v = c_\Delta + c_{ww} - c_{wk} \tag{7}$$

with c_Δ and c_{wk} directly measurable in flight test, while c_{ww} can only be estimated. Thus the dynamometer measurements of the flight test give for the present only $c_v - c_{ww}$, the free (remainder is missing from the original).

The estimation of c_{ww} may proceed from the assumption that the residual drag of the tube in operation is predominately friction, and then affords, with allowance for the increased roughness of the combustion-chamber wall and the friction portion of the discharge gases within the tube, a value of $c_{WW} \approx 0.2$.

For approximately incompressible flow c_v can be determined once more from the pressure measurements p_2 and p_3 and from the assumption that with correct jet operation p_1 and p_4 are equal to the reference pressure p of the outside air:

$$c_v = \frac{\dfrac{1 - F_1}{F_2}}{\dfrac{1 + F_1}{F_2}} \frac{p_2 - p_1}{q} - \frac{\dfrac{1 - F_4}{F_2}}{\dfrac{1 + F_4}{F_2}} \frac{p_3 - p}{q} \tag{8}$$

This relation is used approximately also when on improperly dimensioned tubes the pressure of flow in F_1 differs from the undisturbed outside pressure, since the error due to the steep pressure rise is negligible in a certain range.

Various results of these flight tests are represented in table I.

Although the results of the measurements are not very accurate and hence scattered by reason of the improvised character of the towing tests, the fundamental confirmation of the theoretical results is nevertheless apparent and particularly the significance of suitable mixture preparation for the action of the jet tube.

3. Flight Towing Tests with 20,000-Horsepower Tube on the Do 217 E-2 up to v = 200 m/sec

The employed jet tube had a 600-millimeter-entrance section diameter, a 10° diffuser opening angle, and a 4000-millimeter combustion chamber length by 1500-millimeter-diameter, hence a nominal output of around 20,000 horsepower at $v = 300$ meters per second.

The fuel was injected through 60 to 120 spray nozzles at the diffuser end at 4 to 11 atms. The injection nozzles were seated on the ring and cross visible in figure 26 and faced in flight direction, hence injected against the incoming fresh air.

The suspension system of the tube is seen from the photographs (figs. 25 and 26). A streamlined strut crosswise through the diffuser at about two-thirds diffuser depth served for attachment. Figures 27 to 29 show the experimental carrier with jet tube in flight.

The tests had the purpose of recording thrust and fuel consumption at speeds ranging from 100 to 200 meters per second and of observing the fuel consumption in these conditions.

The fuel was forced from a pressure tank of 700-liter capacity into the combustion chamber by means of nitrogen.

During the tests the excess pressures over the reference pressure of the outside air were indicated in the cross sections F_1, F_2, and F_3 by manometer in the cabin of the test carrier and recorded by a robot camera.

Since the test carrier with mounted tube and its own motive power reaches only a horizontal speed of around 90 meters per second and after starting the combustion up to about 120 meters per second, the higher flying speeds were obtained by slightly pushing the machine out of higher flying level.

Thus in accelerated gliding flight the speed ranges between 100 and 200 meters per second.

By reason of the constant gasoline feed the excess of fuel decreases with increasing flying speed, so that each flight gives a series of measurements at different v and m.

Table II contains the data of some of the most informative test flights; the most important quantities are represented by circles and quantities derived therefrom by lines. They confirm the results of the theoretical calculations and of the test flights with the 2400-horsepower tube.

Up to the highest recorded speeds of 197 meters per second the combustion was substantially quieter than with the small tubes and remained safely in the chamber for all fuel additions.

The diffuser efficiencies for combustion with stoichiometric $(m = 1)$ or excessive $(m > 1)$ fuel additions are in part very high in spite of the ordinarily too low injection pressures, and manifest a growing tendency with the injection pressures.

New, compared to table I, are the measurements with insufficiency of fuel $(m < 1)$ and unchanged cross-section dimensions which at the flying speeds employed give, on the whole, favorable c_v-values. The air inflow volume per second was throughout assumed at $\rho_1 v_1 F_1$.

At the highest flying speeds m had to be kept $m < 1$, since the nose-heavy moment owing to the tube thrust could in the end not be balanced any longer by the horizontal tail surfaces and the experimental carrier became unstable, according to the pilot's report. Otherwise, no special phenomena were observed.

The outside temperatures of the combustion-chamber jacket estimated by means of heat-susceptible paint ranged below 600°C at the low speeds, lower still at the higher speeds, and in the latter case well withstood by normal carbon steels.

The slightly incomplete combustion evidenced by the luminous exhaust flame and computed from the ratio of the theoretical to the recorded $(p_2 - p_3)$ will have to be improved by further refined mixture preparation, where the preheating and vaporization of the fuel by its own combustion-chamber heat, after splitting the required total length of vaporizer tube in a larger number of short vaporizer tubes arranged in paralled, acts very favorable.

4. Flight Tests with Special Airplane Structure

up to v = 300 m/sec

The need of this fourth experimental phase, not yet carried through, results from the fact that the present airplane structures in subsequent combination with large jet tubes can not be accelerated to the required terminal velocity of 300 meters per second for strength reasons, whereas the testing of jet tubes in the speed range of from 200 to 300 meters per second, that is, within range of its subsequent application, is inevitable.

This testing can be accomplished with the jet unit mounted in a special structure such as illustrated in figure 30.

This experimental machine can then be towed by towplane to a few thousand meters height where it receives the required flying speed by diving in order to put the jet unit in operation with sufficient thrust.

In this experimental phase thrust and propulsive efficiency at the desired speeds can be obtained by pressure measurements. At the same time the most immediately interesting performances, such as maximum speed, rate of climb, and, if a pressure-tight cabin is used, the ceiling, can be determined.

This project is under way.

C. RANGE OF APPLICATION

1. Fundamental Possibilities of Application

The range of application of high-temperature jet tubes is largely defined by the two facts that its optimum operation is developed a little below the velocity of sound, that is, at speeds aimed at, at present, particularly by airplanes, gliding bombs, air torpedos, armor piercing bombs, and so forth; and that the thrust-cross section loading of the jet tubes does not exceed 3000 kilograms per meter^2.

Another restrictive factor is that the experiments reported here refer to combustion chambers of considerable cross-sectional dimensions, as is necessary for the application as airplane propulsion unit and that the conclusions are not directly applicable to small cross sections, such as of the caliber of artillery shells.

The acceleration of a bomb dropping vertically can, as long as the air resistance is small, be raised by the ratio (lying near unity) of the cross-section loadings of thrust and weight, that is, by about one gravitational acceleration. So the speeds of impact from equal heights of drop increase by 25 percent at the most. An acceleration of vertically ascending missiles is possible only when the sum of the cross-section loads of weight and air resistance can be kept below about 3000 kilograms per meter^2. The propulsion of horizontally flying missiles, such as gliding bombs, resembles the airplane propulsion unit treated subsequently, and is well possible with jet tubes.

In the aero-mechanical investigation following in this connection diffuser efficiency and combustion-chamber efficiency were assumed at unity, since the values actually departing from unity are of secondary effect on the time of ascent, whereas duration of flight and length of flight decrease in proportion to the product of the two efficiencies, for example, drop to about 60 percent of the cited values at $\eta_{\text{diffuser}} = 0.86$, $\eta_{\text{combustion chamber}} = 0.70$.

2. Fighters and Combat Airplanes at Low Altitude

The most promising range of application for the high-temperature jet tubes is their use as short-period propulsion units of airplanes. A logical example is the airplane represented in figure 30. The jet tube of 2500-millimeter maximum diameter, the fundamental form of which is already shown in figure 1; forms the fuselage of the jet fighter, with a cabin for the crew and the fuel tank, with tail boom for horizontal and vertical control surfaces set on top, while the landing gear and eventually also the wing structure are attached at the lower side of the fuselage. If landing skids are preferred, the wing structure can be mounted in midwing arrangement and continued through the free diffuser space. The double-wall light metal shell design of the diffuser forms the constructive backbone of the fighter, while combustion chamber and discharge nozzle of the jet tube designed as thin sheet-steel shells because of their high temperatures keep distance from the other structural parts. The necessary adjustability of the discharge nozzle end cross section can, for example, be secured in such a way that the circular section F_3 toward F_4 gradually merges in a rectangular or square section, where the upright, flat end surfaces are designed as flaps which can be adjusted by the tailplane by means of spindles.

With an estimated structural weight of 3400 kilograms and a gross weight of 6000 kilograms, with 2400-kilogram fuel supply, the whole airplane has no moving parts, but represents a pure shell design, when discounting the few auxiliaries for landing-gear operation, cabin air system, fuel injection, nozzle and flap setting, and so forth. The fuel is to be supplied by a centrifugal pump which, like the other auxiliaries, is actuated by an impeller mounted in half diffuser depth.

For take-off two rockets of conventional type at both sides of the fuselage with $1\frac{1}{2}$-ton thrust each for 30 seconds and of 700-kilogram total weight are assumed, making the take-off weight altogether 6700 kilograms. The take-off rockets could also be placed within the tube to minimize the eccentricity of its thrust and increase its magnitude at low flying speeds.

The starting (take-off) processes are represented in figure 31.

The rolling friction at incipient take-off follows from the 6700 kilograms take-off weight and an average ground friction factor $\mu = 0.07$ at 470 kilograms. The air resistance is deduced from the estimated polar (fig. 31) of the jet fighter in operation.

The take-off process is followed beyond the lift-off from the ground up to a flying speed of 200 meters per second at sea level. For the then still existing gross weight of about 5800 kilograms the lift coefficient is $c_a = 5800/(2500 \times 4.91) = 0.473$, which according to the polar is associated with a drag coefficient of 0.12 and air resistance of $W = 0.12 \times 2500 \times 4.91 = 1475$ kilograms.

Between these two drag values an approximately linear resistance curve is assumed.

The thrust of the take-off rockets is chosen constant at 3 tons for 30 seconds: the thrust of the jet tube with wide open nozzle setting is derived from $c_v = 0.55$.

From the thus known propulsion and resistance forces and the variable gross weight the progressive acceleration, time and path with respect to speed can be computed and plotted.

These three curves, the solid lines in figure 31, indicate that the take-off acceleration reaches maximum values up to 10 meters per second^2 and that the full flying speed of 720 kilometers per henry is reached after about 38 seconds and a distance of around 3400 meters.

With a lift-off speed estimated at 70 meters per second the jet fighter lifts off after about $17\frac{1}{2}$ seconds and 650-meter rolling distance.

Figure 31 further shows the specific fuel consumption and the curve of the variable gross weight computed from this consumption and the take-off rocket consumption estimated at 15 kilograms per second, which begins at 6700-kilogram take-off weight and ends at about 5800 kilograms, so that the jet fighter, after reaching its full flying speed and with numerical values of figure 30, still has a supply of 2200 kilograms of gasoline on hand.

With this hypothetical gasoline supply two extreme cases are briefly treated: namely, a flight in vicinity of the ground up to landing, and an ascent into the stratosphere.

If the flight near sea level is to be made with low angles of attack at 720 kilometers per henry speed, fuel injection and discharge nozzle can be throttled until the thrust of the jet tube is equal to the air resistance estimated to average 1340 kilograms.

The propulsive efficiency required $c_v = p/qF_2 = 1340/(2500 \times 4.91) \approx 0.11$ is, according to figure 3, already obtainable at low discharge temperatures of less than 600°K, hence, according to figure, obtainable with a fuel consumption of about 0.97 kilogram per second. Thus with the available fuel supply of 2200 kilograms the fighter can, near sea level, fly as a combat plane for 2200/1.3 = 1690 seconds at 720 kilometers per henry, or a distance of 338 kilometers. The length of the distance decreases with higher flying speed, because in the employed lower polar range the air resistance on approaching sonic velocity increases by a little more than the square of the speed of flight and the required discharge temperatures rise. With the jet tube throttled to 4 ton thrust the flying speed near sea level reaches around 1100 kilometers per henry, that is, for 730 seconds, and for a distance of 222 kilometers. In other words, the fighter can operate over a range of more than 100 kilometers at speeds between 720 and 1100 kilometers per henry with a load of 1000 kilograms in bombs or other armament. The potential application is of certain tactical significance against land and sea targets as combat plane, for example, owing to the penetrating force of dropped missiles connected with the high flying speed, and on account of the element of surprise resulting from the absence of loud airplane noises.

3. Fighter or Bomber at High Altitude

The second extreme case of application involves a take-off with immediately following climb to the ceiling of the fighter. For the climb it is assumed that the fighter first climbs with constant dynamic pressure up to a flying speed of 0.9 times sonic velocity with small angles of attack, and during the further ascent at the respective height of 0.9 times sonic velocity remains above 11,000-meter altitude, that is, at $v = 265$ meters per second.

The ascending conditions of the jet fighter illustrated in figure 32 were obtained by progressive path calculation, the respective path angle follows from the sum of the forces on the airplane in longitudinal direction of the path.

Mathematically, the theoretical ceiling is around 21,000 meters, but at the low absolute air pressures existing at that level the combustion in the jet tube should have some difficulties, so that, for the first, a ceiling not above 20,000 meters should not be expected, even with low wing loading. Thus, when the jet fighter inclusive of take-off time of 40 seconds reaches flight levels of 6000 meters in 70 seconds, 12,000 meters in 100 seconds, and

18,000 meters in about 190 seconds, it may be said to have supremacy of this height range, utilized by combat planes and only partially protected by convential fighter planes and flak artillery. The theoretical fuel consumption for the actual climb to 12,000 meters, that is, without take-off, is 505 kilograms. Calculations were also made with smaller v/a and d_2, and the actual time of ascent to 12,000 meters, along with the respective fuel consumptions compiled in figure 32(a). The admissibility of lower Mach number to $v/a = 0.7$ for the ascent is readily seen.

Another interesting point is the length of time the jet fighter can remain at a required flight altitude and the horizontal distance it can cover with the remaining fuel supply. At 12,000 meters, for example, the remaining gross weight is, according to figure 32, 5295 kilograms, hence the fuel supply 1695 kilograms, the average lift coefficient required for level flight is

$$c_a = A\ qF_2 = (5295 + 2600)/2 \times 1117 \times 4.91 = 0.72$$

and the air resistance by the $v/a = 0.9$ polars of figure 31 about $c_w = c_v = 0.17$, hence the thrust required is $P = 932$ kilograms. This thrust is obtainable by the approximate equation

$$P = \rho_1 F_1 v_1^2 \left(\sqrt{F_3/F_2} - 1\right)$$

with the stagnation temperature $T_3 = 590^\circ K$ and - conformably to the relation $B = Q \times 10^3/PH$ - with a specific fuel consumption $B = 0.554$ kilogram per second. The air resistance therefore requires a gasoline consumption of 0.516 kilogram per second; that is, the remaining supply lasts for $1695/0.516 = 3280$ seconds, or for just 1 hour and a distance of $3280 \times 0.2655 = 870$ kilometers. The duration and distance in horizontal flight at different Mach numbers and for different flight levels as computed by this method are represented in figure 33. The path lengths of the maximum value computed for sea level to 370 kilometers increase monotonic with the altitude, and reach a maximum of over 1100 kilometers at 1800 meters. A noteworthy fact is that the greatest path lengths at lower levels are reached with lower flying speeds, but in the high flying levels with the greatest speeds, for in high flying levels the favorable effect of the absolute air resistances, low at low flying speeds, is vitiated by the high propulsion coefficients required and consequently high discharge temperatures and fuel consumptions. The absolute increase in the path length with the altitude of flight is due chiefly to the favorable lift-drag ratios which are utilized at higher flight levels. Employed for antiaircraft defense a maximum endurance in horizontal flight is, under certain circumstances, more important than a maximum path length. According to figure 33 the endurance also increases, at first, with the altitude of flight, reaches a maximum of over 4800 seconds at 14,000-meter height, and then decreases a little. For, if the airplane at a certain altitude flies just so fast that its polar at a certain point below that of optimum lift-drag ratio is utilized, the maximum range is reached. This point is determined by the fact that at it the fuel consumption per kilometer of gross weight and m flight path, that is, the expression $c_w/c_a(c_wF_2/4F_1 + 1)$ becomes smallest.

Naturally, the airplane flies longer with still a little lower speed. The flying speeds most favorable for duration of flight are therefore lower, as a rule. Figure indicates the important practical fact that the maximum range and endurance of fighters in the 10,000 to 12,000-meter altitude range are obtained with $v/a = 0.7$, hence with aerodynamically safely controllable flying speeds. This short take-off and climb to any altitude up to 18,000 meters together with horizontal flight paths over 1100 kilometers and durations up to more than 4800 seconds at speeds up to 1000 kilometers per henry makes the jet fighter an excellent weapon against bombardment planes in the 8000 to 18,000-meter height range.

Another potential use is as combat plane in fast, low, level flight over distances of up to 100 kilometers, and especially for combat-dive bombing up to distances of nearly 500 kilometers.

Used as fighter-bomber the rapid climb to ceiling level is not necessary. Therefore figure 34 shows a flight path with only 12° angle of ascent 16-kilometer ceiling, which at 1230-kilometer total length has about the same range as a corresponding path with steep ascent. An initial gross weight of 6000 kilograms and a useful load of 1000 kilograms for 2400-kilogram fuel load serve as basis. Estimating the practical depth of penetration of the fighter in enemy territory at only 35 percent of the flight length, it gives a safe depth of penetration of 430 kilometers, which in conjunction with a 950-kilometer per henry flying speed at 16 kilometers altitude appears tactically very valuable. Aside from the actual flight path, figure 34 gives the most important data, such as absolute and specific fuel consumption, thrust, thrust horsepower, discharge temperature, flying speed, duration of flight, and gross weight, also a diagram of the airplane with weight distribution and a polar of the air forces referred to the aerodynamic supporting surface of 30 meters^2.

The fact that the application as a horizontal bomber necessitates considerably less thrust than when used as a fighter by reason of the absence of the steep climb might suggest a special jet bomber with weaker jet tube. Figures 35 and 36 therefore show for the same initial gross weight of 6000 kilograms, the same useful load of 1000 kilograms and the same aerodynamic supporting surface of 30 meters^2 of two jet-engine airplanes with jet tubes of 2 meters and 1.5-meter diameter, along with the corresponding theoretical flight paths, the polar of figure 34 being used in both cases on the assumption that the decrease in air resistance due to the reduced surfaces would approximately cancel the necessarily more unfavorable design in the area of the cabin. The flight lengths decrease substantially with the thickness of the jet tubes, since the shorter tubes must, in order to reach the required traction forces, fly with higher temperatures, hence, lower efficiency; the medium tube gives a length of only 1120 kilometers, the small tube of only 780 kilometers, corresponding to depths of penetration of only 390 kilometers and 270 kilometers, respectively.

The airplane arrangement recommended as a fighter with a jet tube of 2500-millimeter diameter is therefore the best for the bomber also.

It is characteristic of this tube in both spheres of application that the minimum flying speed from which the airplane is able to accelerate to higher speeds again with the jet tube alone amounts to about 430 kilometers per henry for full load at 12 kilometers altitude and decreases with decreasing load and height until it is nearly equal to the landing speed with the empty airplane in ground vicinity, as indicated in figure 31.

The study of a diagram of the thrust and drag with respect to the flying speed indicates that the jet fighter exhibits a propulsive instability similar to that of the conventional propeller airplane in slow flight which must be compensated by careful gas lever operation or an automatic thrust regulator.

CONCLUSION

The continuous compressorless Lorin jet-propulsion unit with cylindrical combustion chamber and high discharge temperatures makes it possible, at flying speeds slightly below the velocity of sound, to obtain thrust horsepowers of the order of magnitude of five-place horsepower figures at efficiencies of one-third to one-half of the total efficiencies of conventional propeller propulsion units.

The theoretical considerations are confirmed by towing tests on existing airplanes with units of more than half natural size at flying speeds up to 200 meters per second, while corresponding flight tests at speeds up to 300 meters per second with special airplane structures are in preparation.

With this jet-propulsion unit very simply constructed fighters can be designed, which according to the data available can climb to nearly 18,000 meters altitude within 3 minutes and remain up to an hour at those levels while covering a horizontal distance up to 1000 kilometers.

The apparently obtainable flying speeds lie at 1100 kilometers per henry at sea level and at 950 kilometers per henry in the stratosphere.

Such airplanes are ready to use on low-attack missions for distances up to 100 kilometers, for level and dive-bombing missions from high altitudes for distances up to 500 kilometers and as fighters in the stratosphere for up to one hour duration.

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TABLE I

FLIGHT TOWING TESTS WITH HIGH-TEMPERATURE LORIN JET TUBES

$(F_1 = 0.0314\text{m}^2;\ F_2 = F_3 = 0.20\text{m}^2;\ F_4 = 0.129\text{m}^2;\ N_{\text{nom.}} = 2400\text{ hp})$

No. test	Date	Altitude (m)	Speed (m/sec)	Fuel	Injection nozzles (Ø in mm)	Injection pressure	Direction of injection (flight duration)	Measured fuel volume (kg/sec)	Fuel factor m (-)	$\Delta p_2/q$ (-)	$\Delta p_3/q$ (-)	Δ (kg)	$c_\Delta = \Delta/q_2$ (-)	c_{wk} (-)	$c_v = c_{ww} = c_\Delta - c_{wk}$ (-)	$c_v = (c_v - c_{ww}) + 0.2$ (-)	$c_v = 0.730\Delta p_2/q - 0.219\Delta p_3/q$	Remarks
41	27.5.42	500	80.2	Gasoline with octane rating 40; density 0.715 kg/dm^3	6Ø1.65; 1Ø2.7	13		0.225	1.16	0.66	0.21	52.7	0.69	Average - 0.45	0.24	0.44	0.44	Continuous injection necessary
43	3.6.42	1950	88.2		6Ø1.65; 1Ø2.7	11		0.220	1.19	0.66	0.23	51.2	0.65		0.20	0.40	0.43	Diffuser intake rounded after this continuous injection test
44	3.6.42	1200	86.3		6Ø1.65; 1Ø2.7	11		0.223	1.20	0.66	0.23	51.0	0.63		0.18	0.38	0.43	Continuous injection
45	4.6.42	1900	82.6		7Ø1.65	10		0.181	1.04	0.72	0.25	52.3	0.77		0.32	0.52	0.47	5 injection nozzles forward, 2 backward, continuous injection
48	4.6.42	2000	86.9		7Ø1.65	32		0.215	1.19	0.80	0.27	57.6	0.76		0.31	0.51	0.52	Only initial ignition
49	6.6.42	550	81.0		7Ø1.65	32		0.235	1.20	0.71	0.29	60.3	0.78		0.33	0.53	0.46	
50	6.6.42	850	82.9		7Ø1.65	32		0.232	1.20	0.73	0.29	60.8	0.75		0.30	0.50	0.47	
51	6.6.42	850	82.9		6Ø1.65; 1Ø2.7	11		0.186	0.96	0.64	0.26	51.2	0.66		0.21	0.41	0.41	
52	6.6.42	1200	86.3		6Ø1.65.1Ø2.7	14		0.213	1.09	0.68	0.27	54.4	0.71		0.26	0.46	0.44	
53	8.6.42	1000	84.3		6Ø1.65; 1Ø2.7	15		0.222	1.15	0.70	0.29	58.6	0.72		0.27	0.47	0.45	
55	8.6.42	1000	82.6		7Ø1.65	36		0.257	1.35	0.77	0.28	62.8	0.81		0.36	0.56	0.57	2 nozzles forward, 5 upright; injection pressure and excess of fuel high
56	8.6.42	1000	86.4		6Ø1.65; 1Ø2.7	16		0.249	1.25	0.88	0.32	62.0	0.73		0.28	0.48	0.53	2 small nozzles forward, the rest upright
57	8.6.42	1000	80.7		7Ø1.65	36		0.263	1.41	0.81	0.27	60.2	0.81		0.36	0.56	0.57	
58	9.6.42	1500	83.8		12Ø1.65	7-1/4		0.215	1.17	0.73	0.29	56.9	0.74		0.29	0.49	0.47	
59	9.6.42	1300	84.0		12Ø1.65	7-1/4		0.216	1.15	0.73	0.29	53.0	0.67		0.22	0.42	0.47	
61	11.6.42	400	81.6		14Ø1.65	22		0.232	1.17	0.75	0.22	57.4	0.71		0.26	0.46	0.50	
62	11.6.42	2000	85.8		14Ø1.65	22		0.214	1.20	0.71	0.22	55.1	0.73		0.28	0.48	0.47	High injection pressure, high excess of fuel effect, high diffuser efficiencies, and coefficient of propulsion
63	11.6.42	2000	86.5		7Ø1.65	34		0.270	1.50	0.86	0.31	63.8	0.83		0.38	0.58	0.56	
64	11.6.42	2000	87.5		7Ø1.65	34		0.254	1.39	0.86	0.30	65.4	0.83		0.38	0.58	0.56	
65	11.6.42	2000	85.3		7Ø1.65	34		0.252	1.42	0.83	0.29	60.5	0.81		0.36	0.56	0.55	With streamlined strut across and at 213 depth of the diffuser
66	11.6.42	2000	87.5		7Ø1.65	34		0.249	1.36	0.83	0.29	63.6	0.81		0.36	0.56	0.54	
67	12.6.42	1000	84.1	Avia. gas octane rat. 87	14Ø1.65	8		0.206	1.06	0.75	0.27	60.8	0.75		0.30	0.50	0.49	Substantial solution of gasoline spray gives good results even at low injection pressure
68	12.6.42	1200	85.5		14Ø1.65	12		0.245	1.27	0.81	0.27	62.2	0.76		0.31	0.51	0.53	
69	12.6.42	1200	83.5		7Ø2.7	4		0.225	1.19	0.67	0.20	54.9	0.70		0.25	0.45	0.44	Injection pressure too low
70	12.6.42	100	78.5	Methanol	14Ø1.65	36		0.50	1.08	0.85	0.28	59.7	0.78		0.33	0.53	0.56	Colorless flame

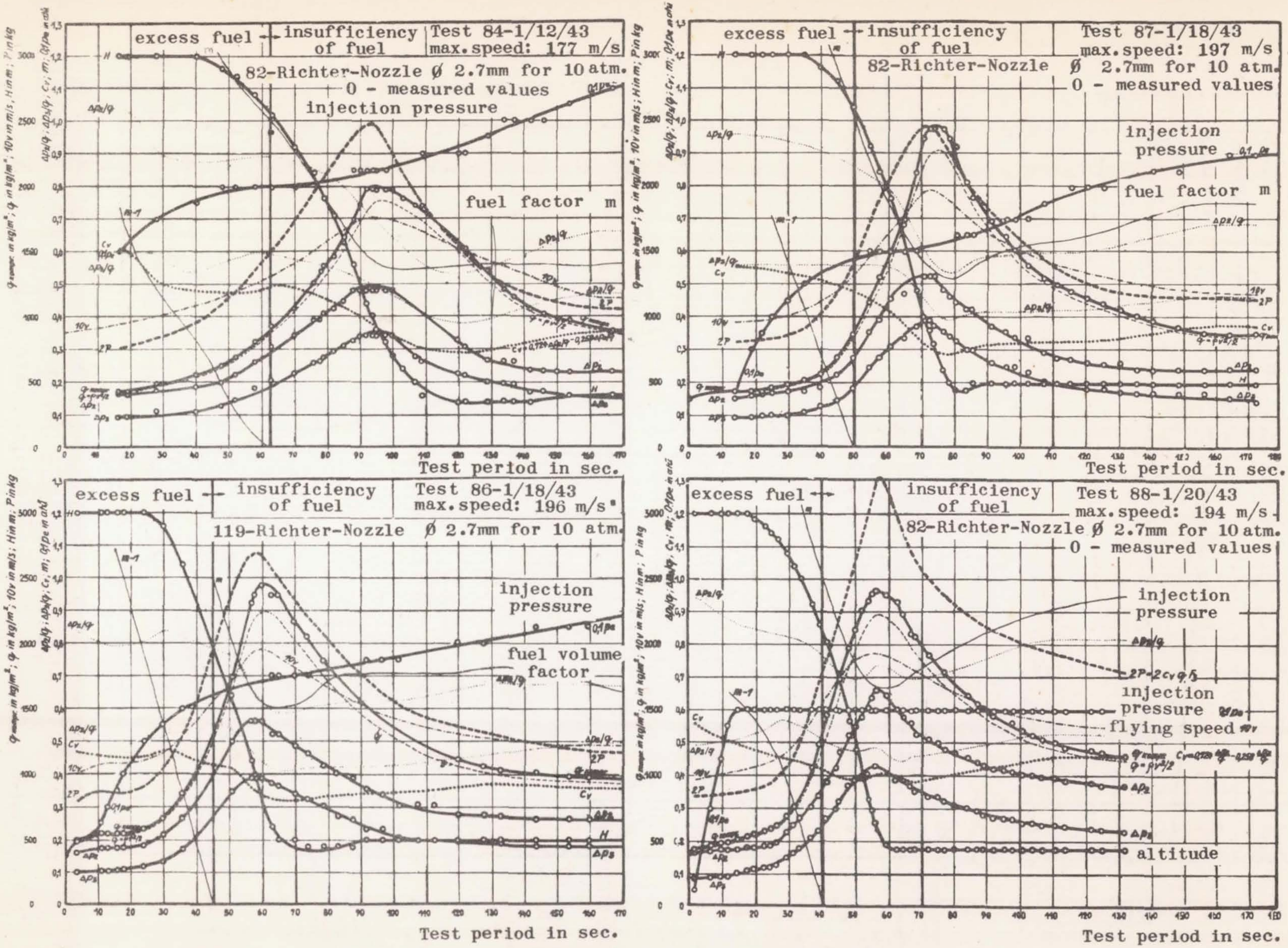

TABLE II.- RESULTS OF FLIGHT TOWING TESTS WITH HIGH-TEMPERATURE JET TUBE OF $F_1 = 0.283m^2$; $F_2 = F_3 = 1.76m^2$; $F_4 = 1.043m^2$ AND 20,000-H.P. RATED OUTPUT

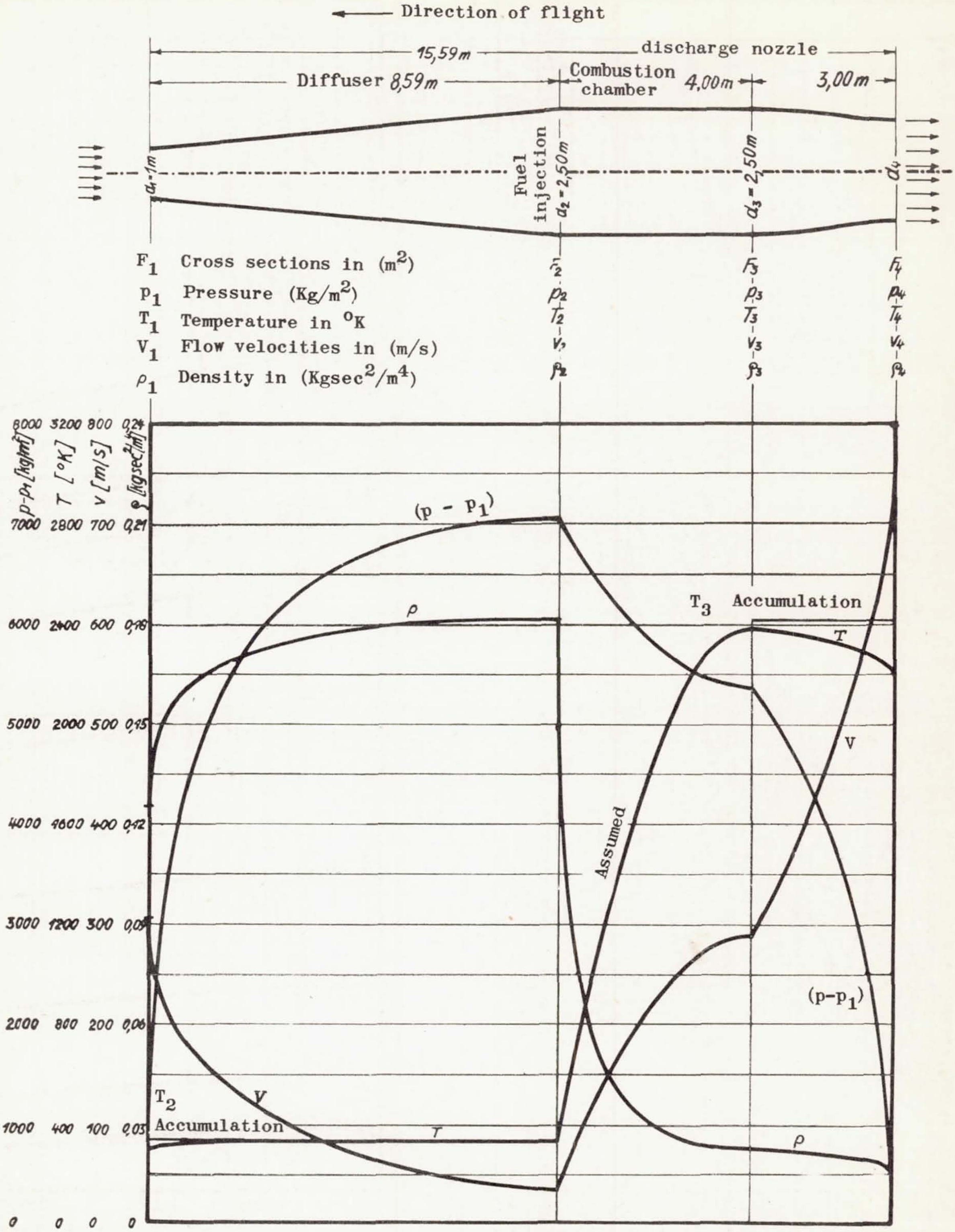

Figure 1.- Jet tube of 2.5m diameter at flight near sea level with stoichiometric gasoline consumption and 1100 Km/h (v/a = 0.9) flying speed. (Thrust: 16.4 tons, thrust coefficient: $c_V = 0.57$; gasoline consumption: 1.19 Kg/sec t; efficiency: 5.7%; thrust horsepower: 67,000 hp.)

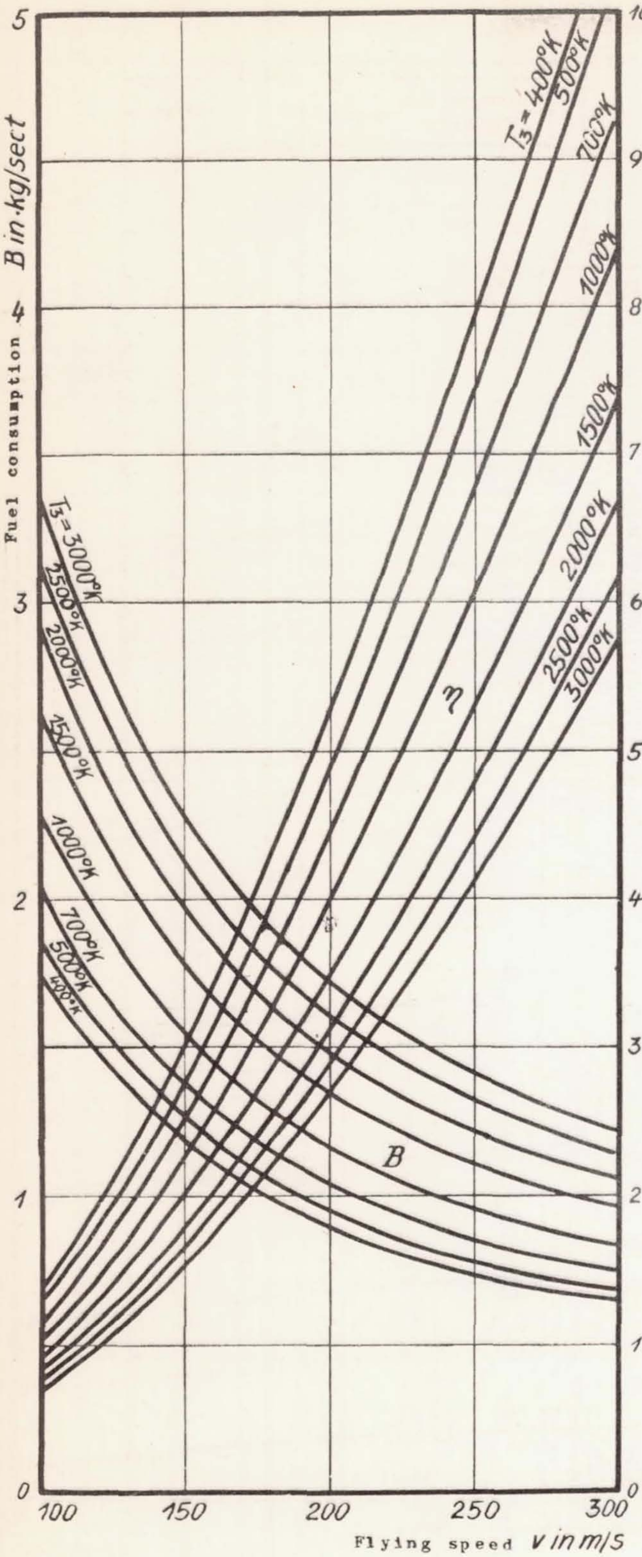

Figure 2.- Efficiency and fuel consumption of jet tube at T_1 = 288°K (near sea level) by approximation.

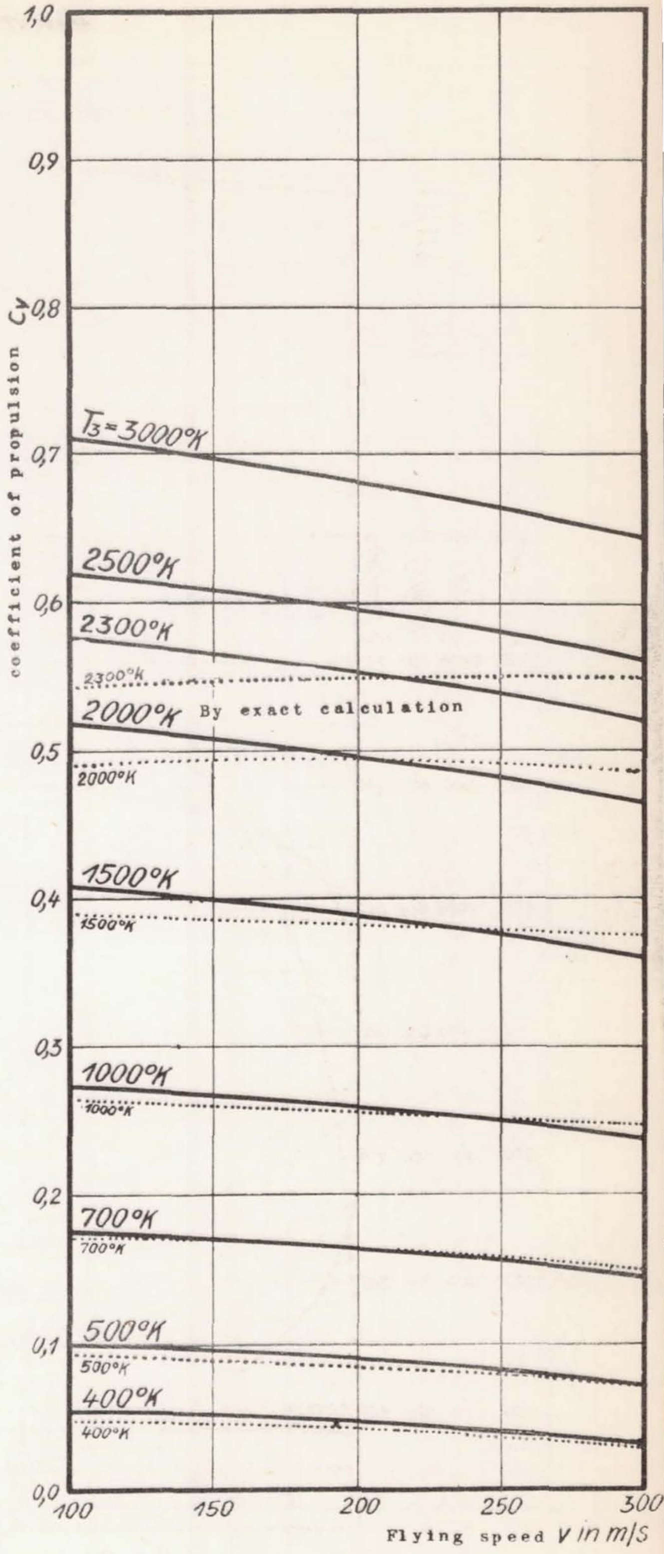

Figure 3.- Propulsive efficiency of jet tube at T_1 = 288°K (sea level) by approximate method (solid curve) and by exact calculation (dotted line).

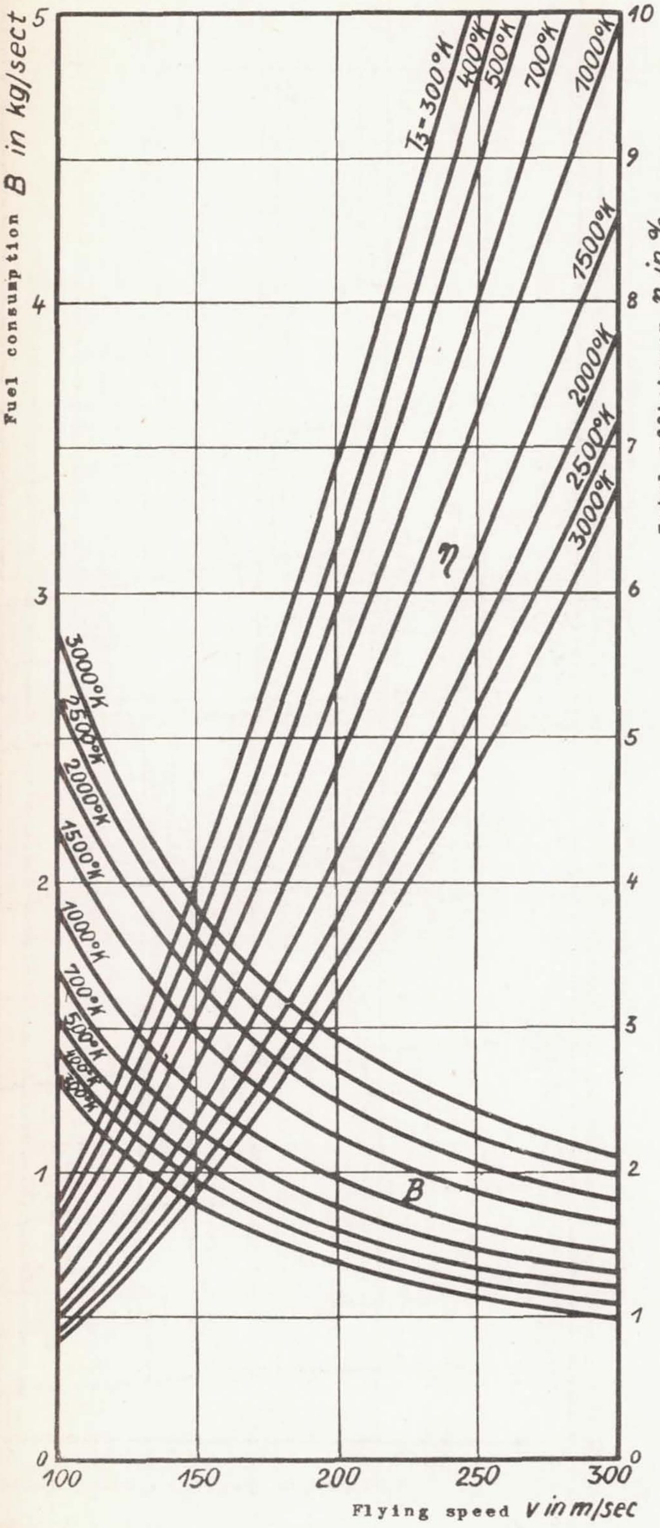

Figure 4.- Efficiency and fuel consumption of jet tube at T_1 = 216.5 K (stratosphere) by approximate method.

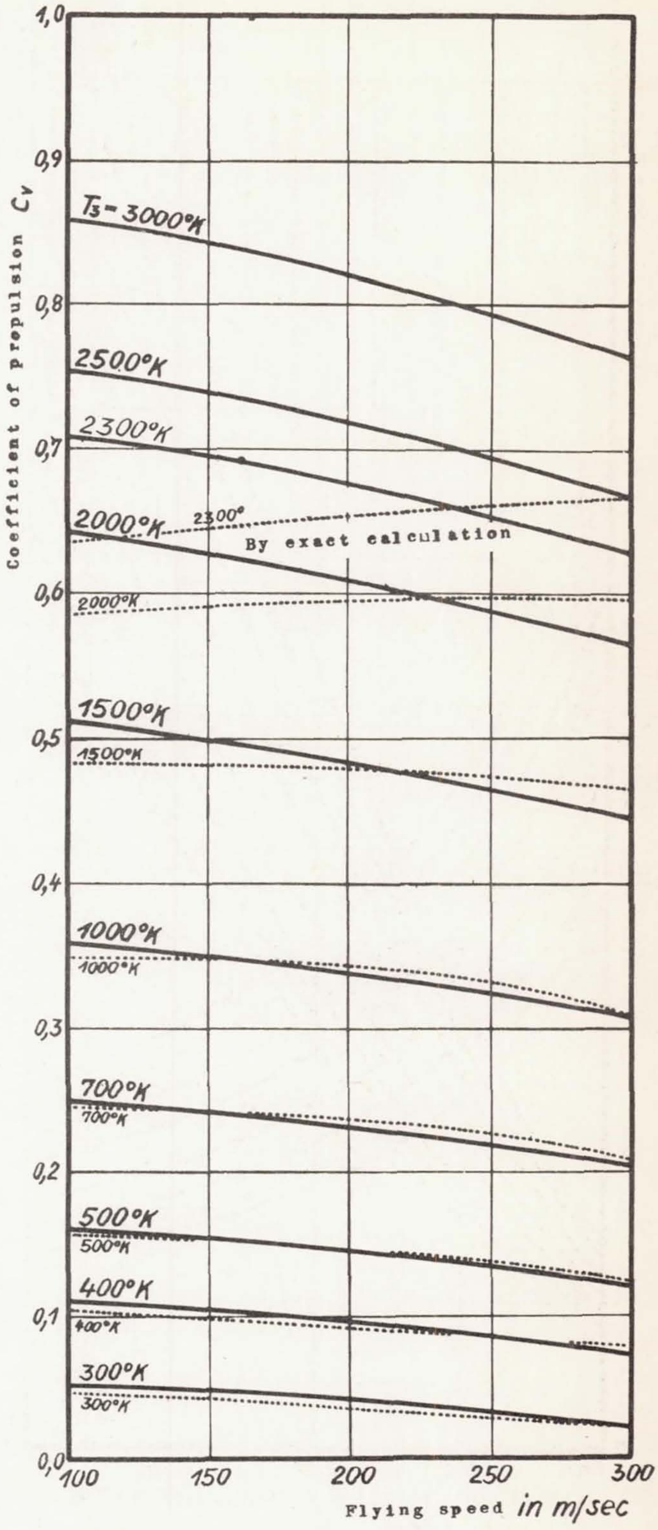

Figure 5.- Propulsive efficiency of jet tube at T_1 = 216.5 K (stratosphere) by approximation (solid line) and by exact calculation (dotted lines).

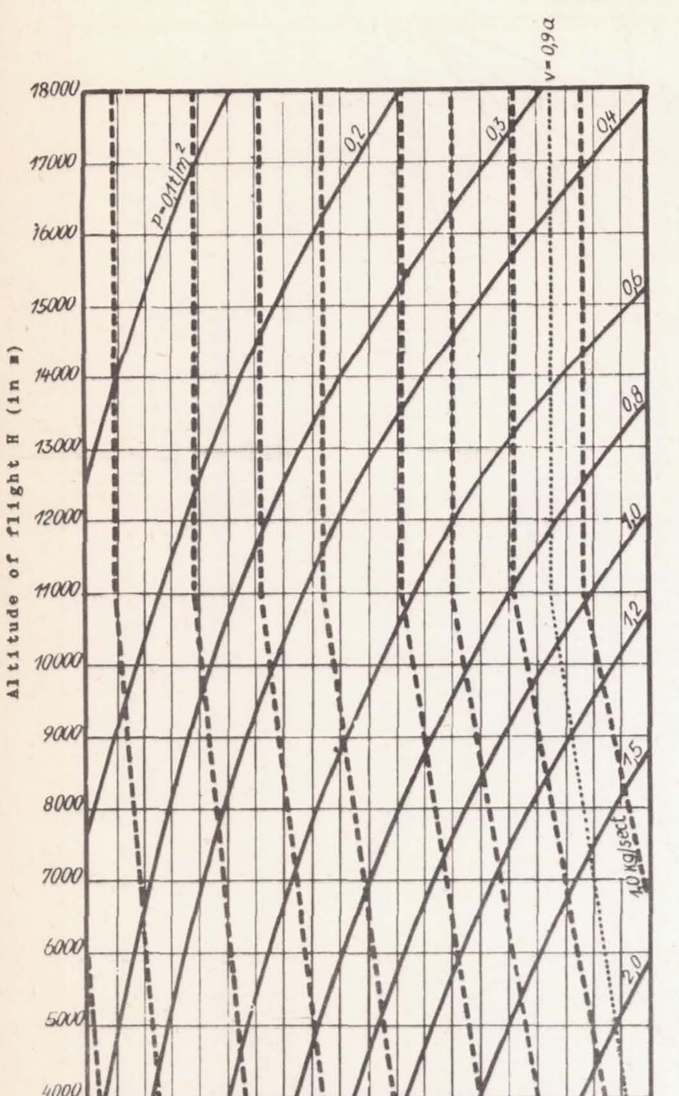

Figure 6.- Maximum thrust per m^2 of maximum cross-sectional area P/F_2 (t/m^2) and the respective fuel consumption B (Kg/sect) for stoichiometric hydrocarbon combustion.

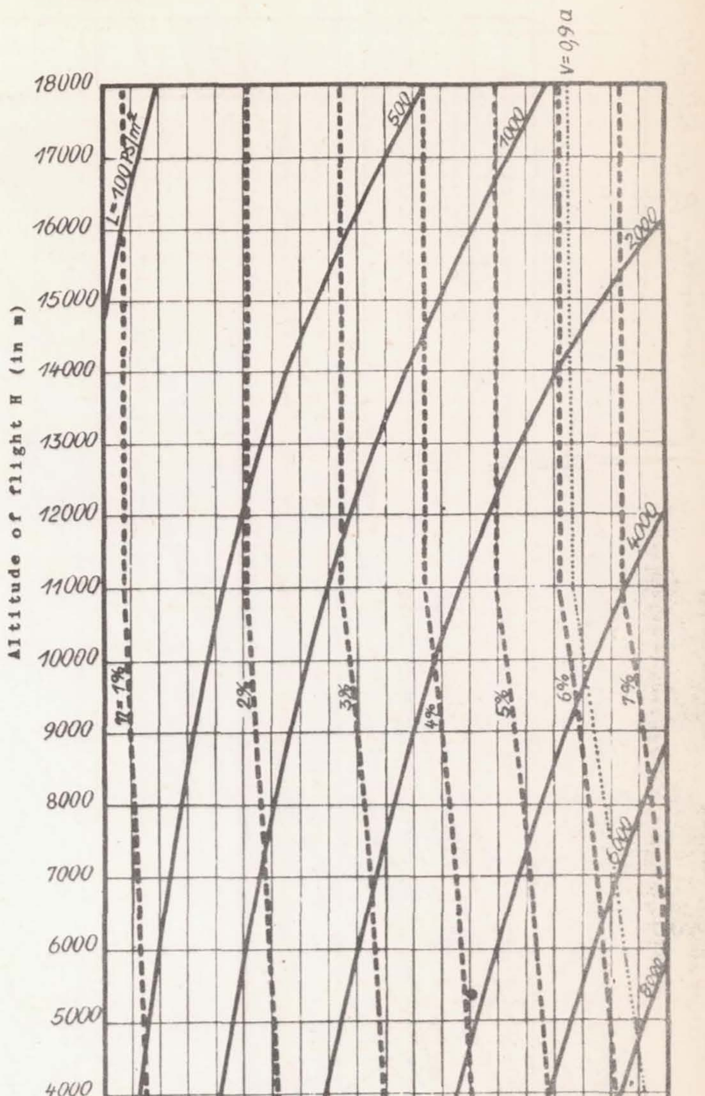

Figure 7.- Maximum thrust horsepower per m^2 of maximum cross-sectional area L/F_2 (HP/m^2) and the respective efficiency η for stoichiometric hydrocarbon combustion.

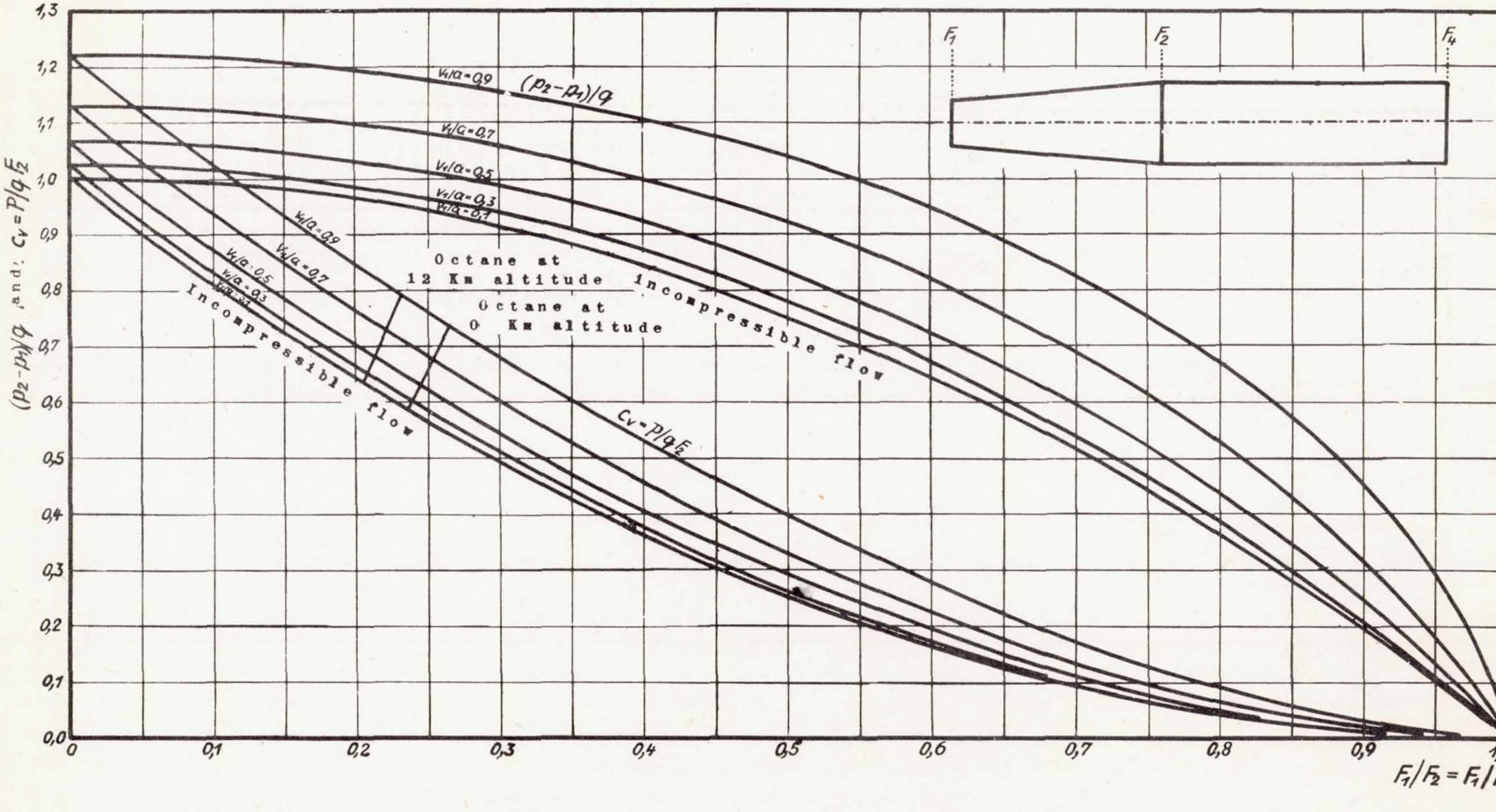

Figure 8.- Jet tube without discharge nozzle; pressure wire in diffuser and coefficient of propulsion c_v.

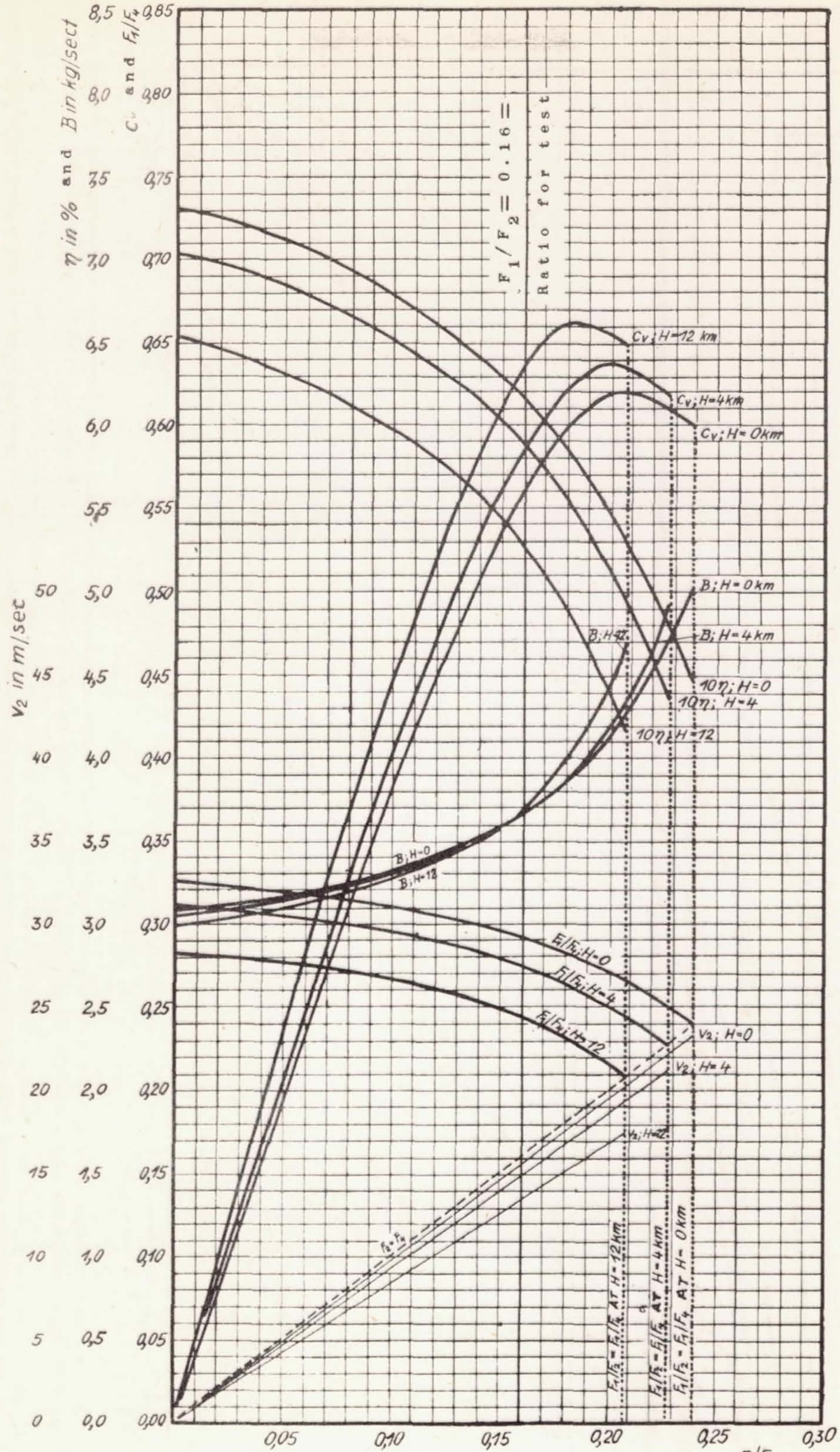

Figure 9.- Coefficient of propulsion c_v(-), efficiency (%), cross-section ratio F_1/F_4(-) and fuel consumption B (Kg/sec)) plotted against ratio F_1/F_2 for v/a = 0.3 and stoichiometric octane-air combustion at 0.4 and 12 Km altitude.

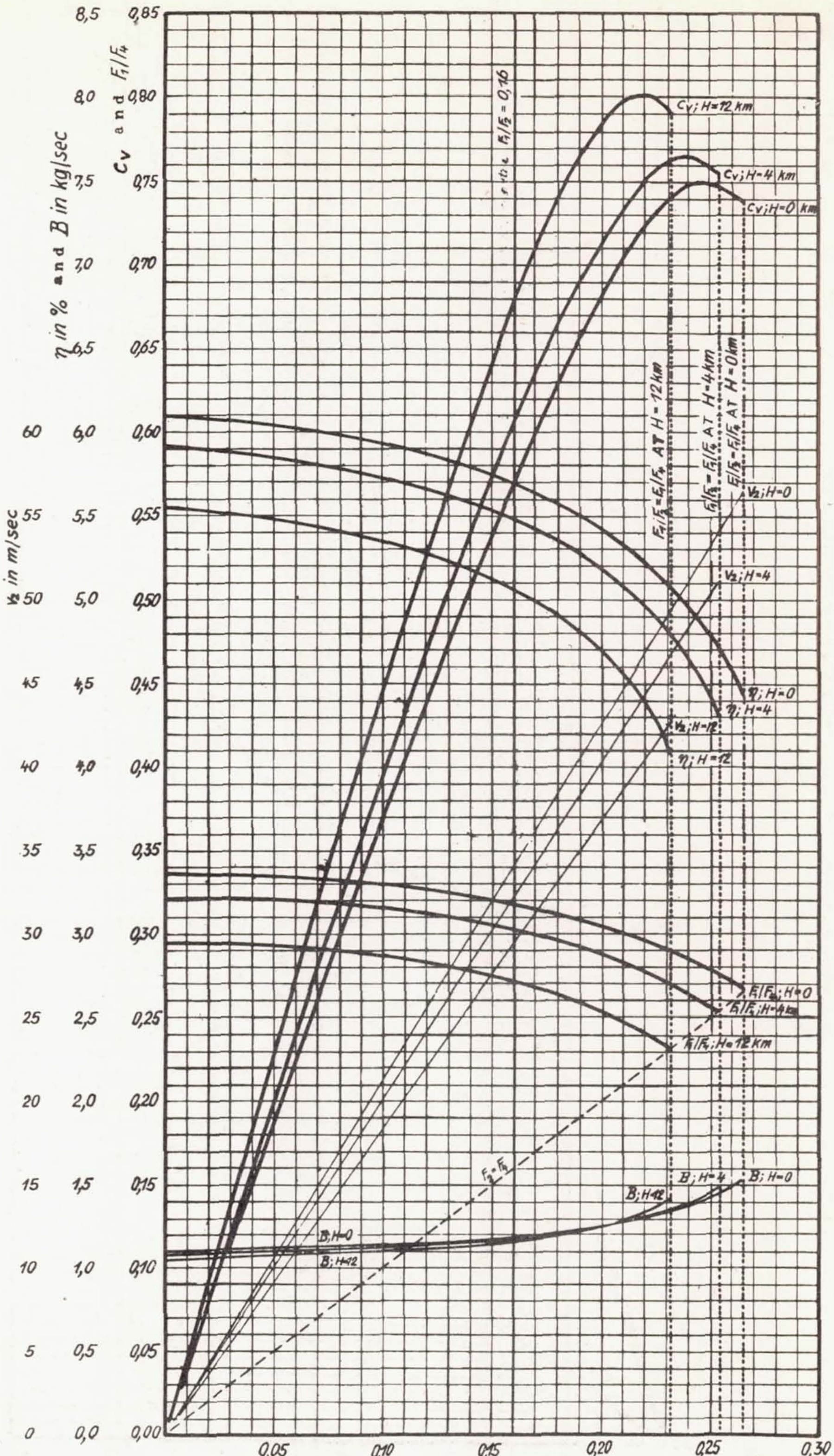

Figure 10.- Coefficient of propulsion c_v(-), efficiency (%), cross-section ratio F_1/F_4(-) and fuel consumption B (Kg/sec) plotted against ratio F_1F_2 for v/a = 0.9 and stoichiometric octane-air combustion at 0.4 and 12 Km altitude.

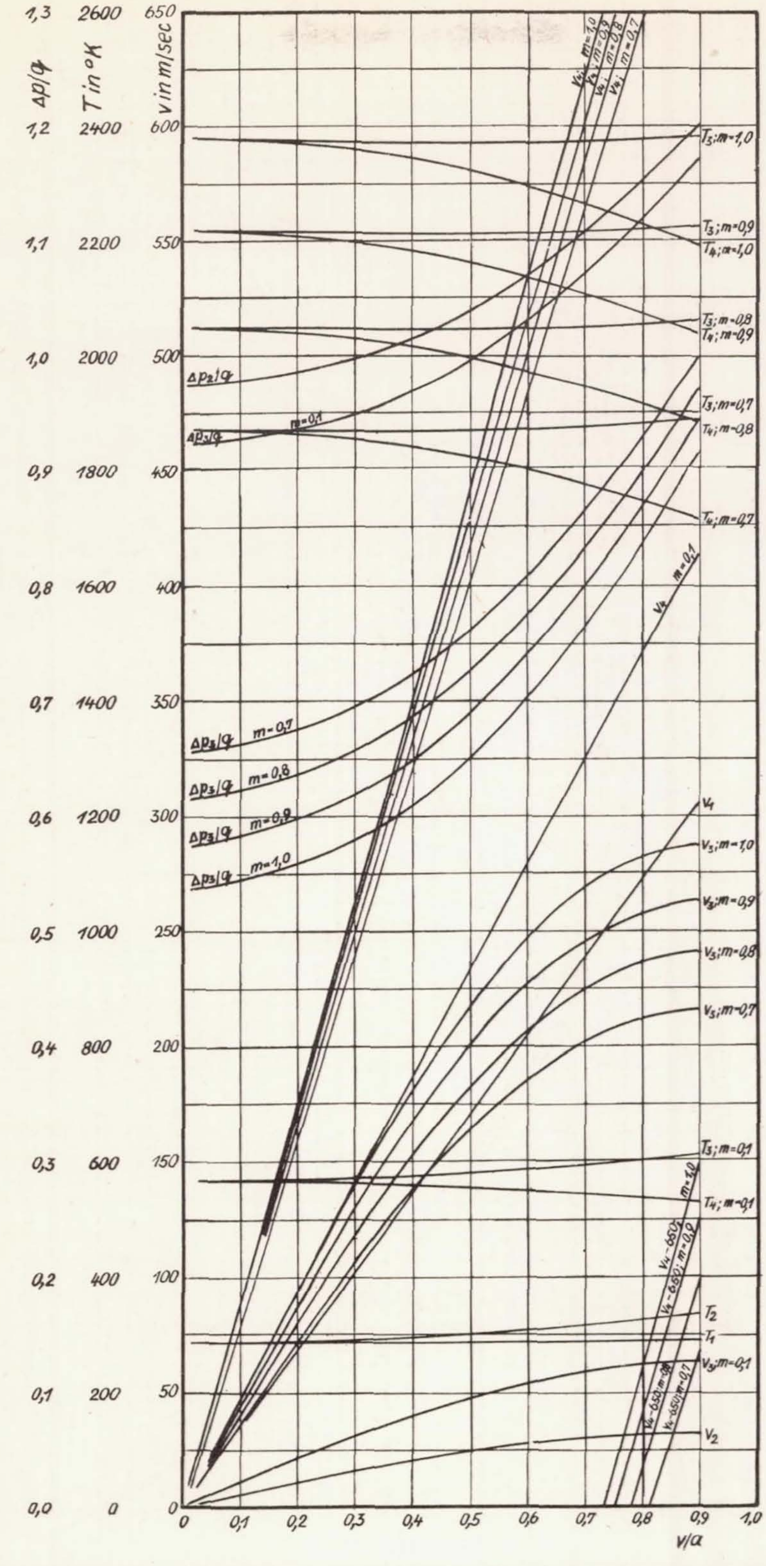

Figure 11.- Phase quantities $\Delta p/q$, T and v in the four maximum cross sections F_1, F_2, F_3, and F_4 of a jet tube with the dimensions $F_1/F_2 = 0.16$ plotted against the flying speed v/a at 0 Km flight level and with respect to the fuel volume factor m.

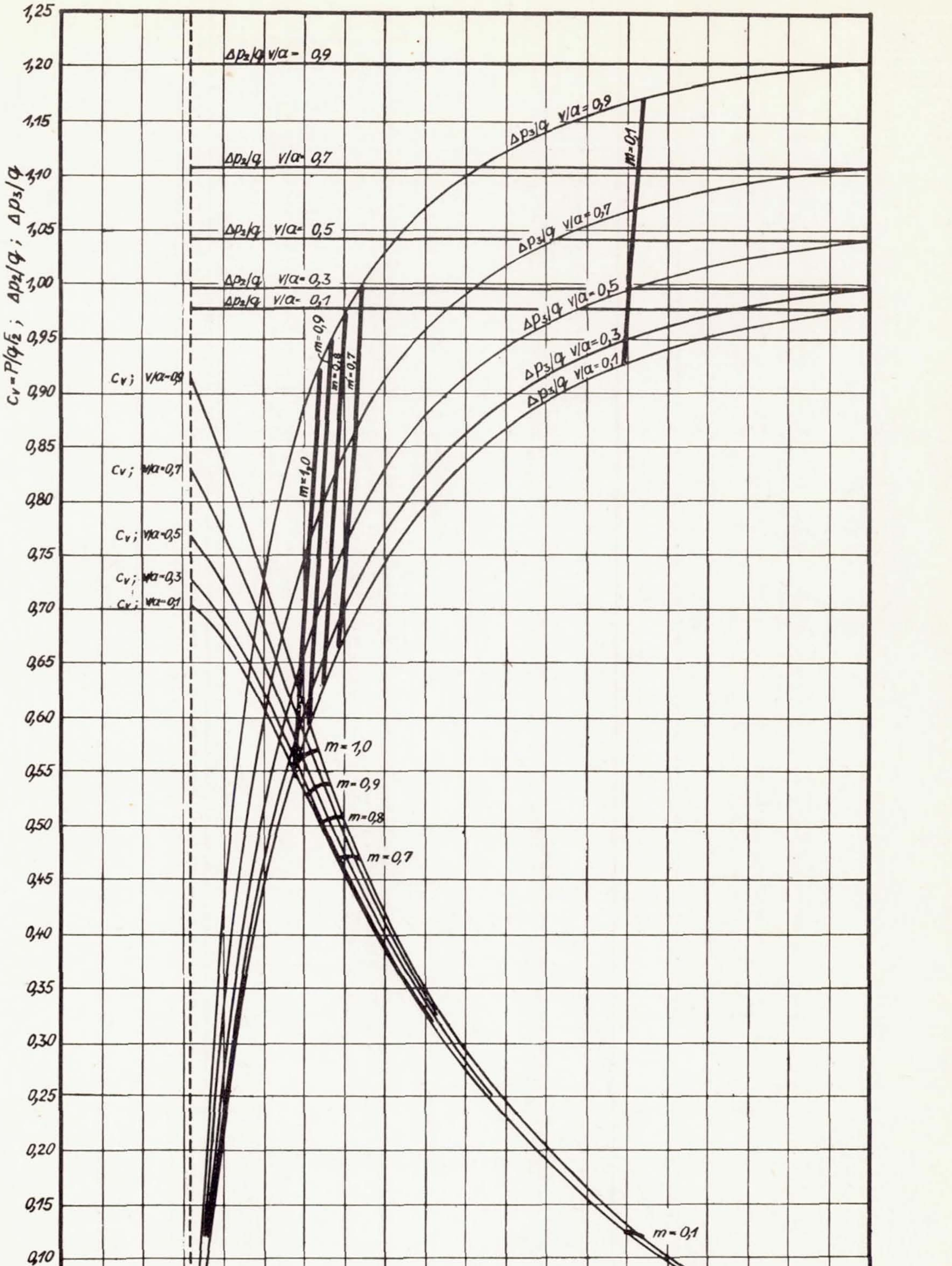

Figure 12. Coefficients c_v, excess pressures $\Delta p_2^2/q$ and $\Delta p_3/q$ at 0 Km height for a jet tube with the dimensions $F_1/F_2 = 0.16$ for various hydrocarbon-air mixtures plotted against the cross section ratio F_1/F_4.

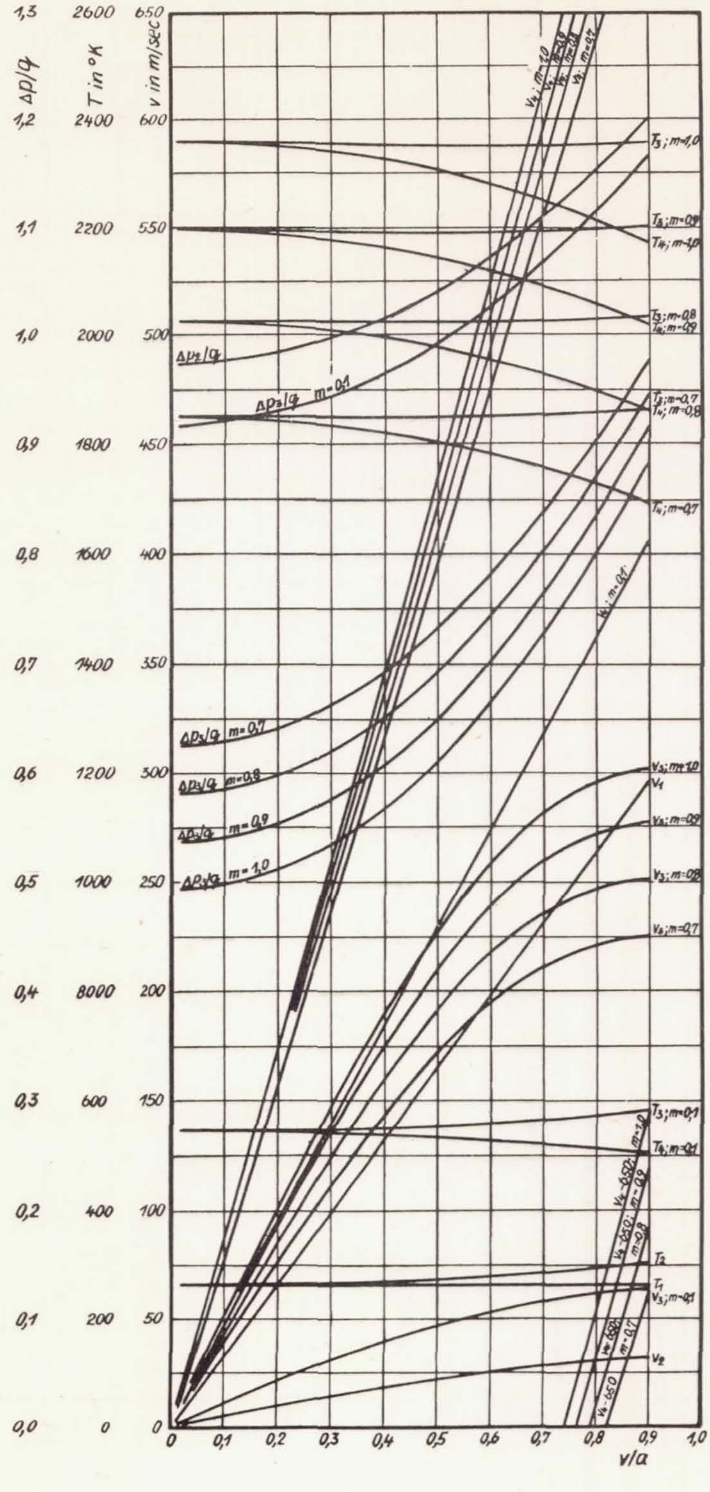

Figure 13.- Phase quantities $\Delta p/g$, T and v in the four maximum cross sections F_1, F_2, F_3, F_4, of a jet tube with the dimensions $F_1/F_2 = 0.16$ plotted against flying speed v/a at 4 Km flight altitude and fuel-volume factor m.

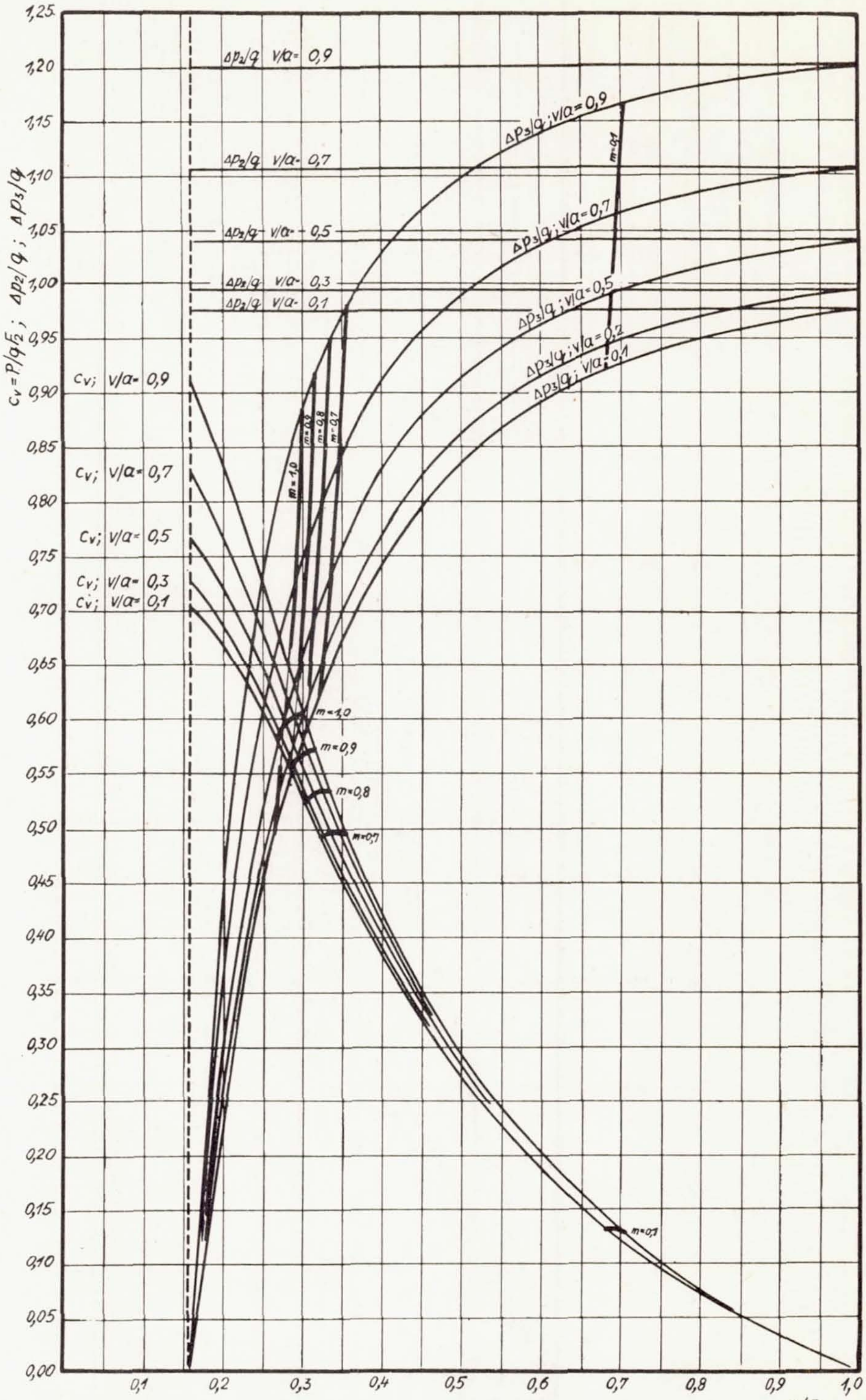

Figure 14.- Coefficients c_v, excess pressures $\Delta p_2/q$ and $\Delta p_3/q$ at 4 Km flight altitude for a jet tube with the dimensions $F_1/F_2 = 0.16$ for different hydrocarbon-air mixtures plotted against cross section ratio F_1/F_4.

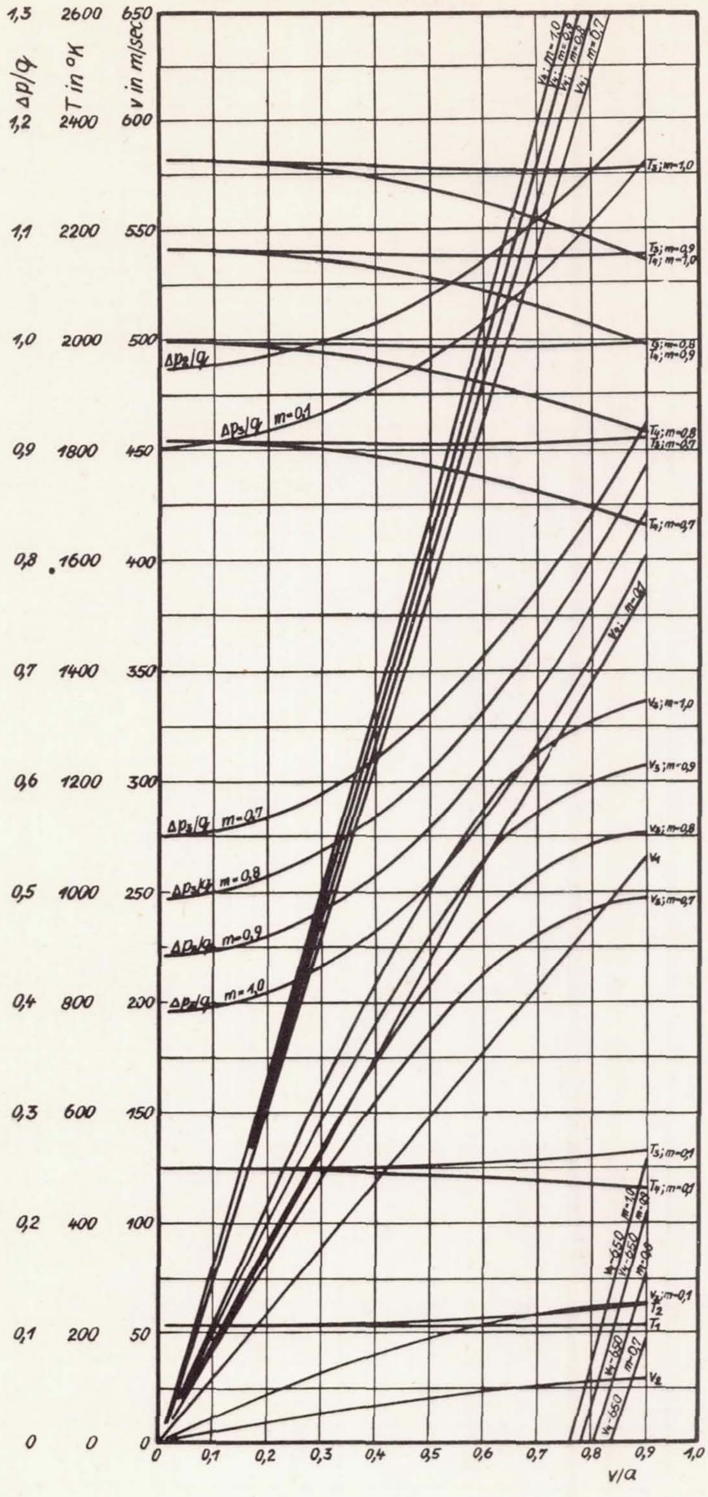

Figure 15.- Quantities $\Delta p/q$, T and v in four maximum cross sections F_1, F_2, F_3, F_4, of a jet tube with the dimensions $F_1/F_2 = 0.16$ plotted against the flying speed v/a at 12 Km altitude and fuel volume factor m.

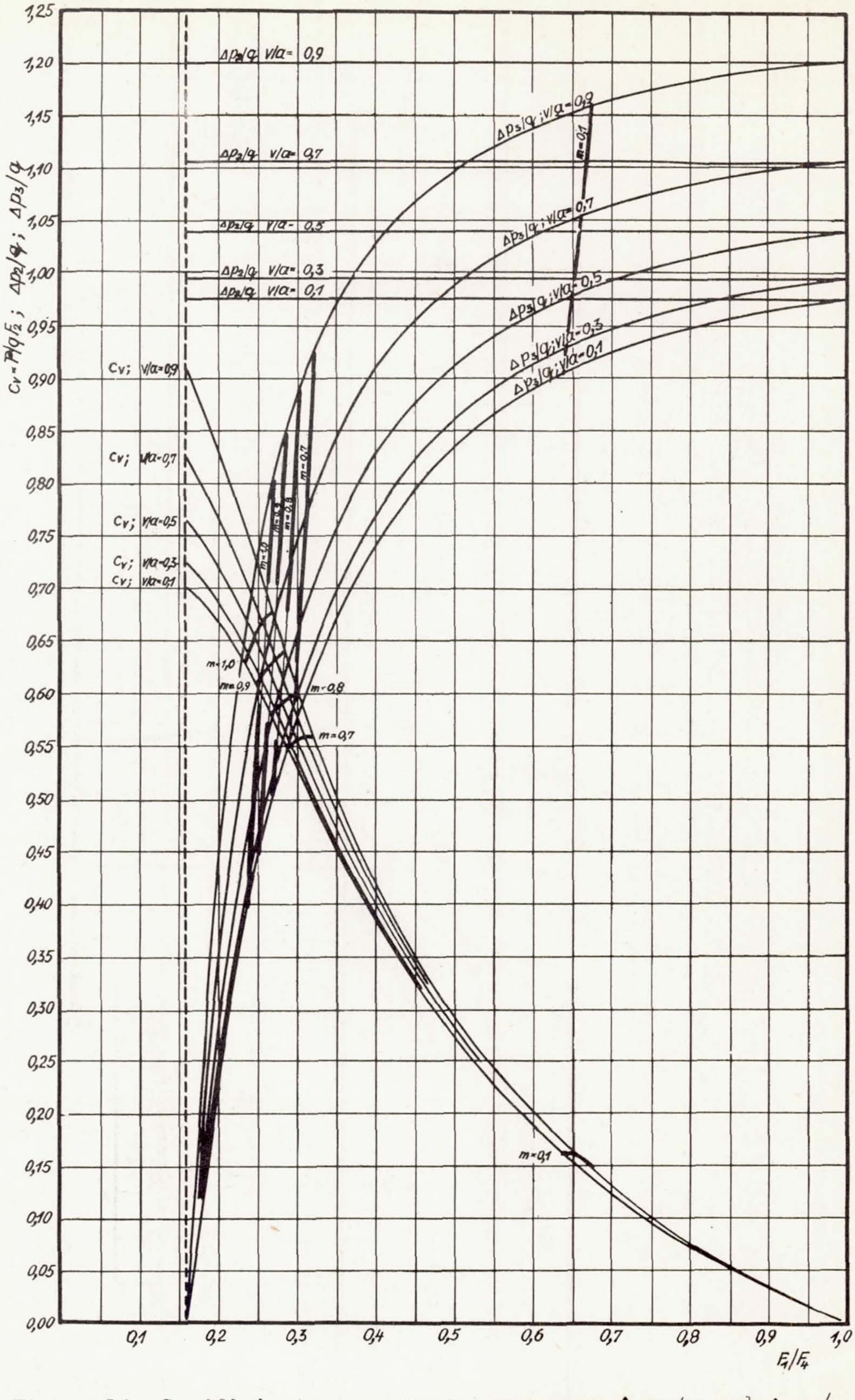

Figure 16. Coefficients c_v, excess pressures $\Delta p_2/q$ and $\Delta p_3/q$ at 12 Km altitude for a jet tube with the dimensions F_1/F_2=0.16 for various hydrocarbon-air mixtures plotted against the cross-section ratio F_1/F_4.

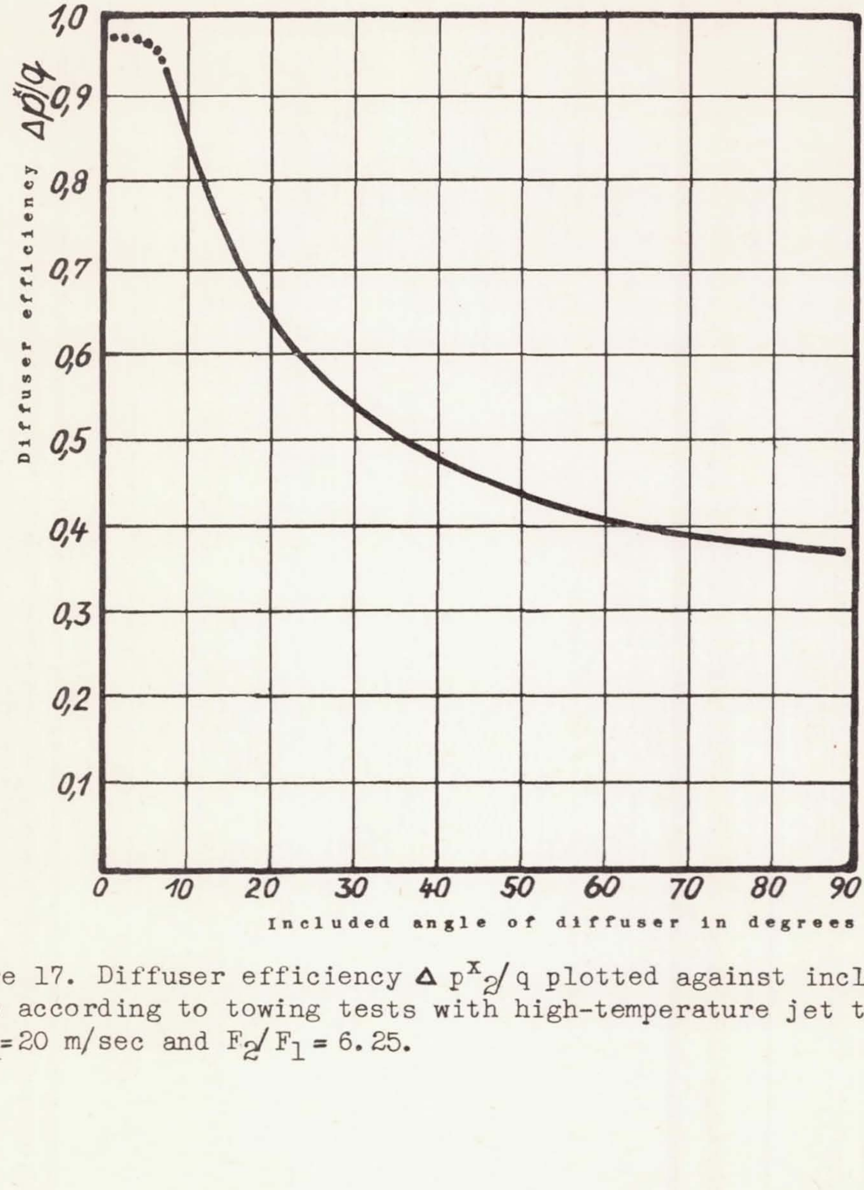

Figure 17. Diffuser efficiency $\Delta p^x{}_2/q$ plotted against included angle according to towing tests with high-temperature jet tubes at $v_1 = 20$ m/sec and $F_2/F_1 = 6.25$.

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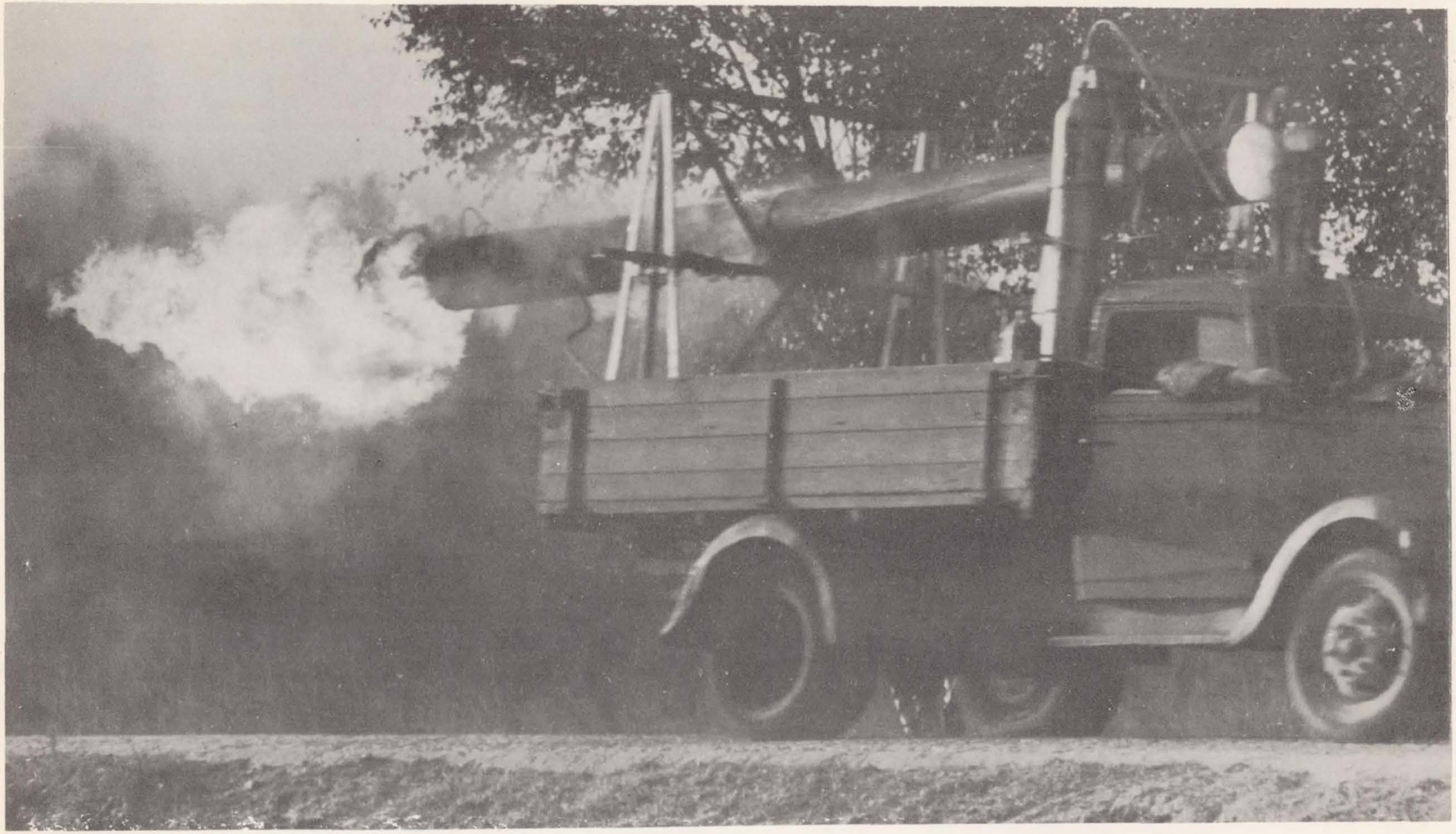

Figure 18.- Road towing test with Lorin combustion chamber for studying the injection and combustion processes.

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Figure 19.- Road towing test with Lorin combustion chamber of 800-mm diameter and 2400°K discharge temperature for studying the injection and combustion processes.

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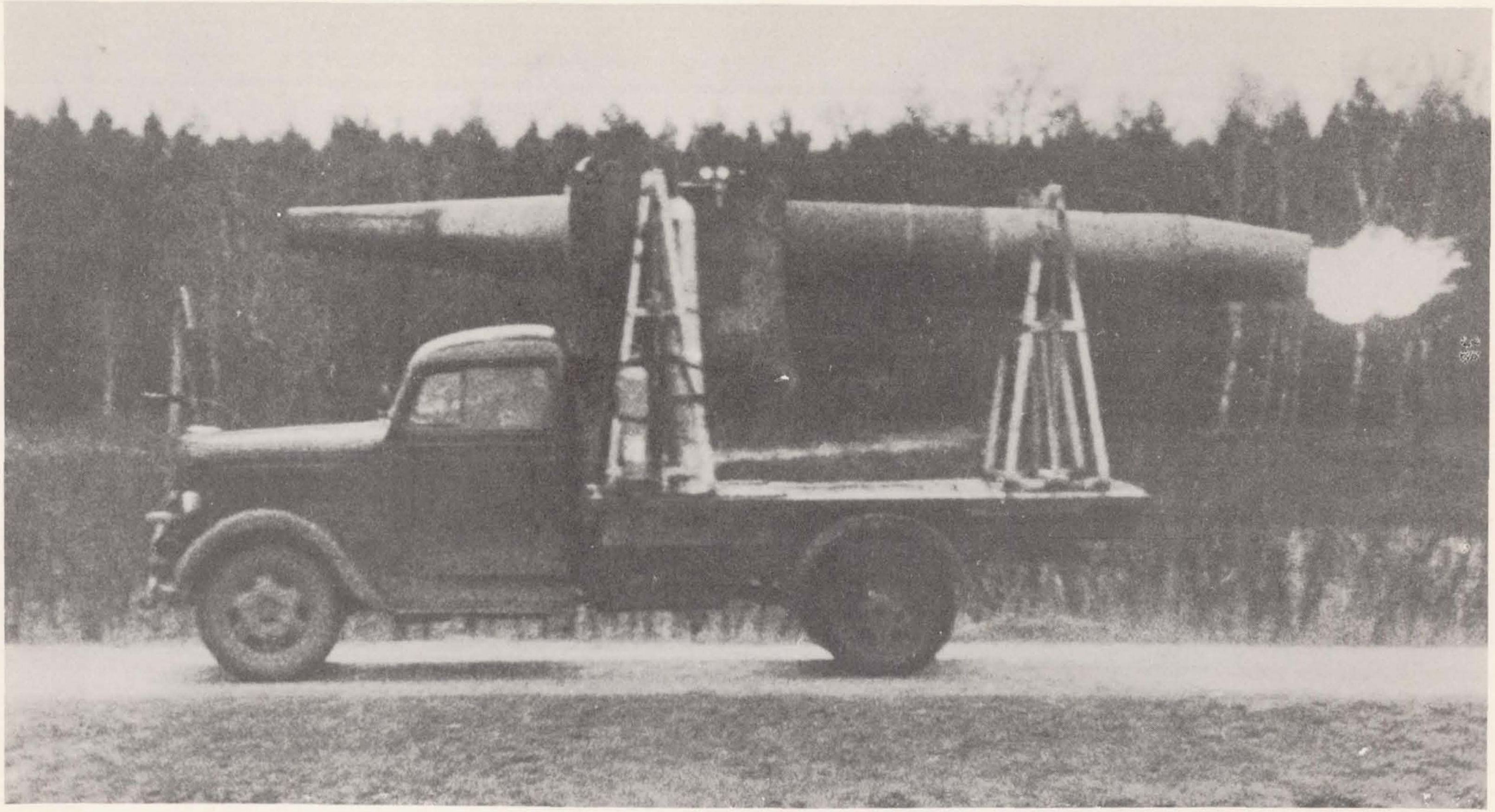

Figure 20.- Road towing test with high-temperature jet tube for measuring the diffuser efficiencies.

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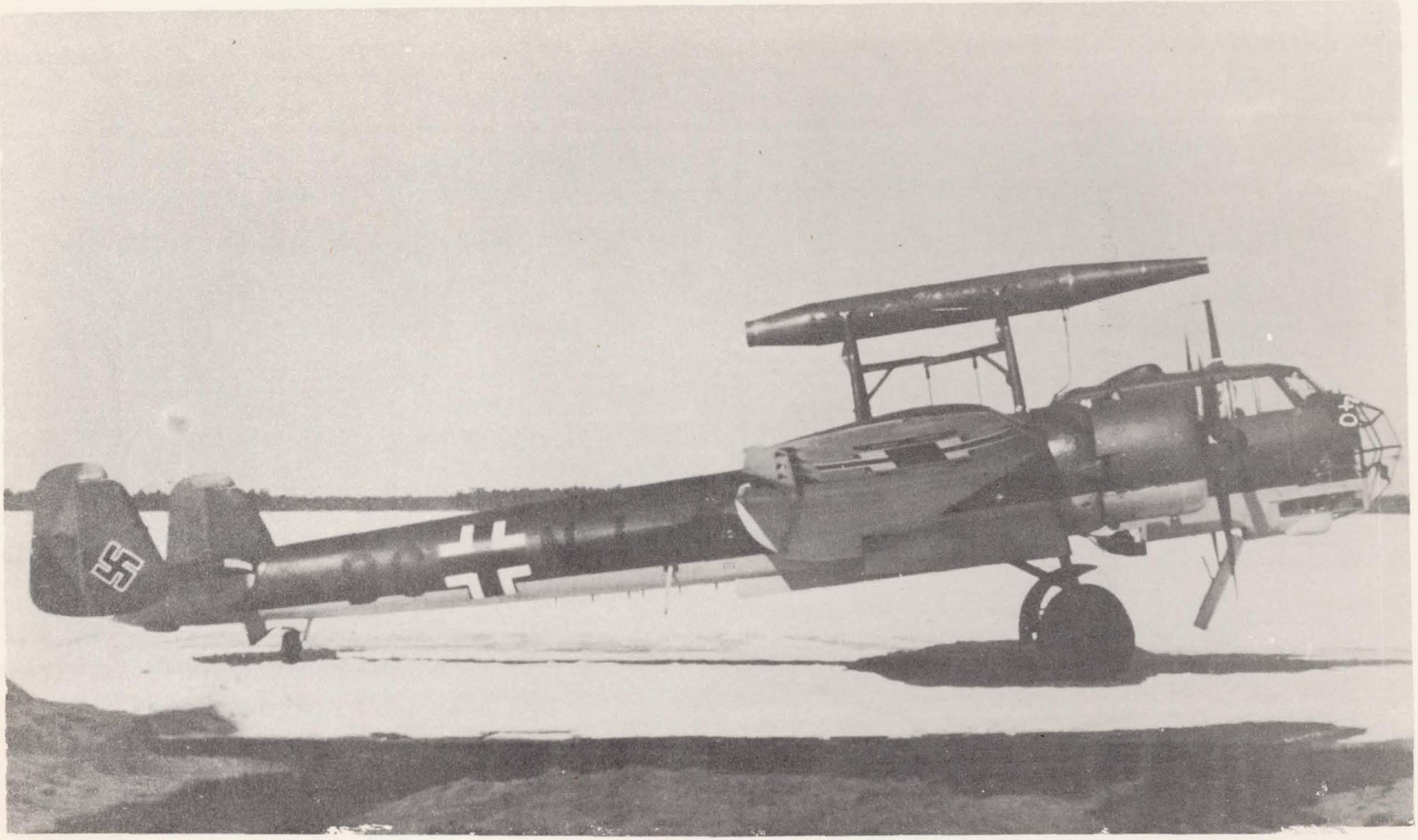

Figure 21.- 2400-hp jet tube on the Do 17Z as experimental carrier.

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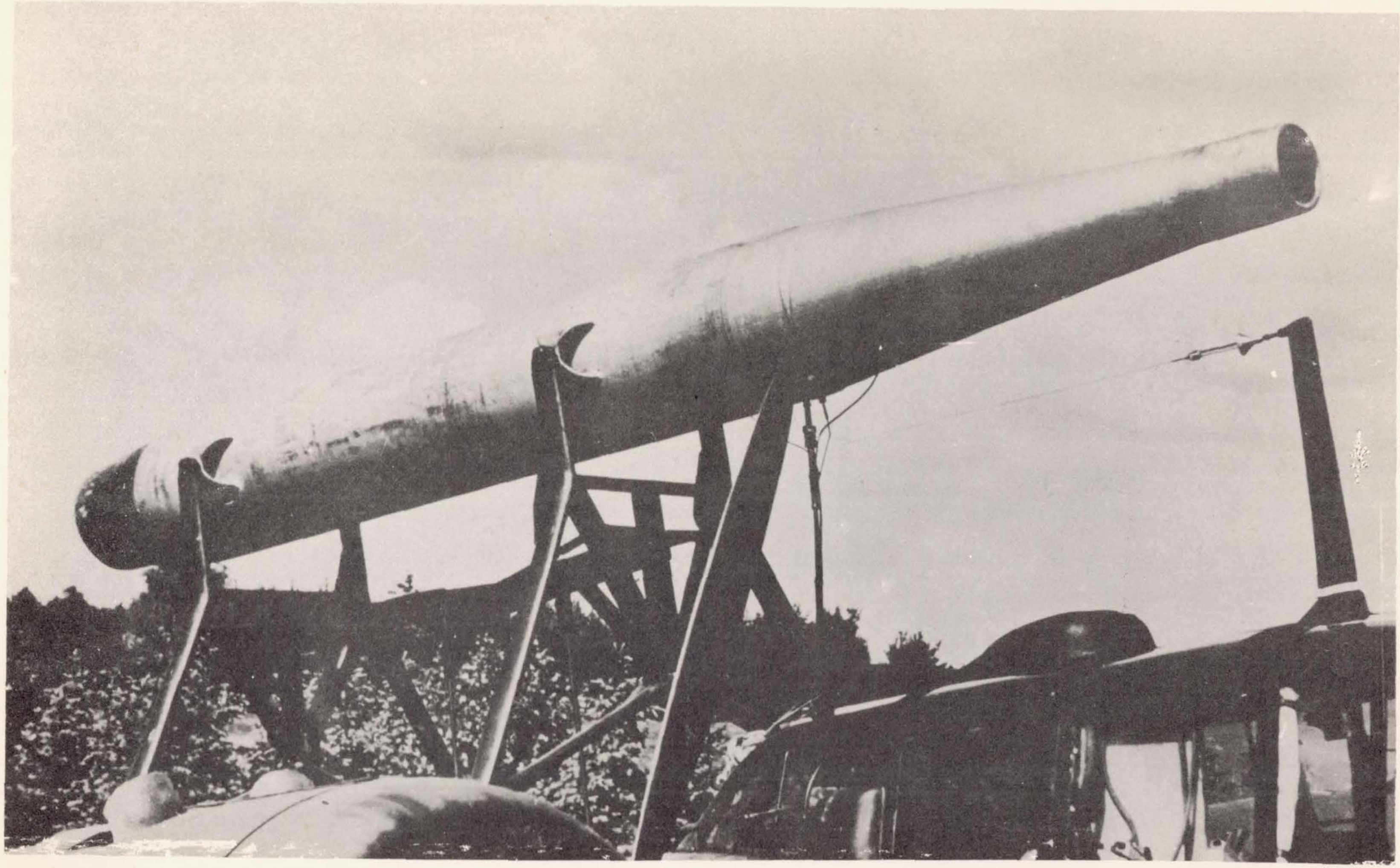

Figure 22.- 2400-hp jet tube and thrust recording setup on the Do. 17Z.

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Figure 23. 2400hp jet tube on the Do 17Z, flight in cold state.

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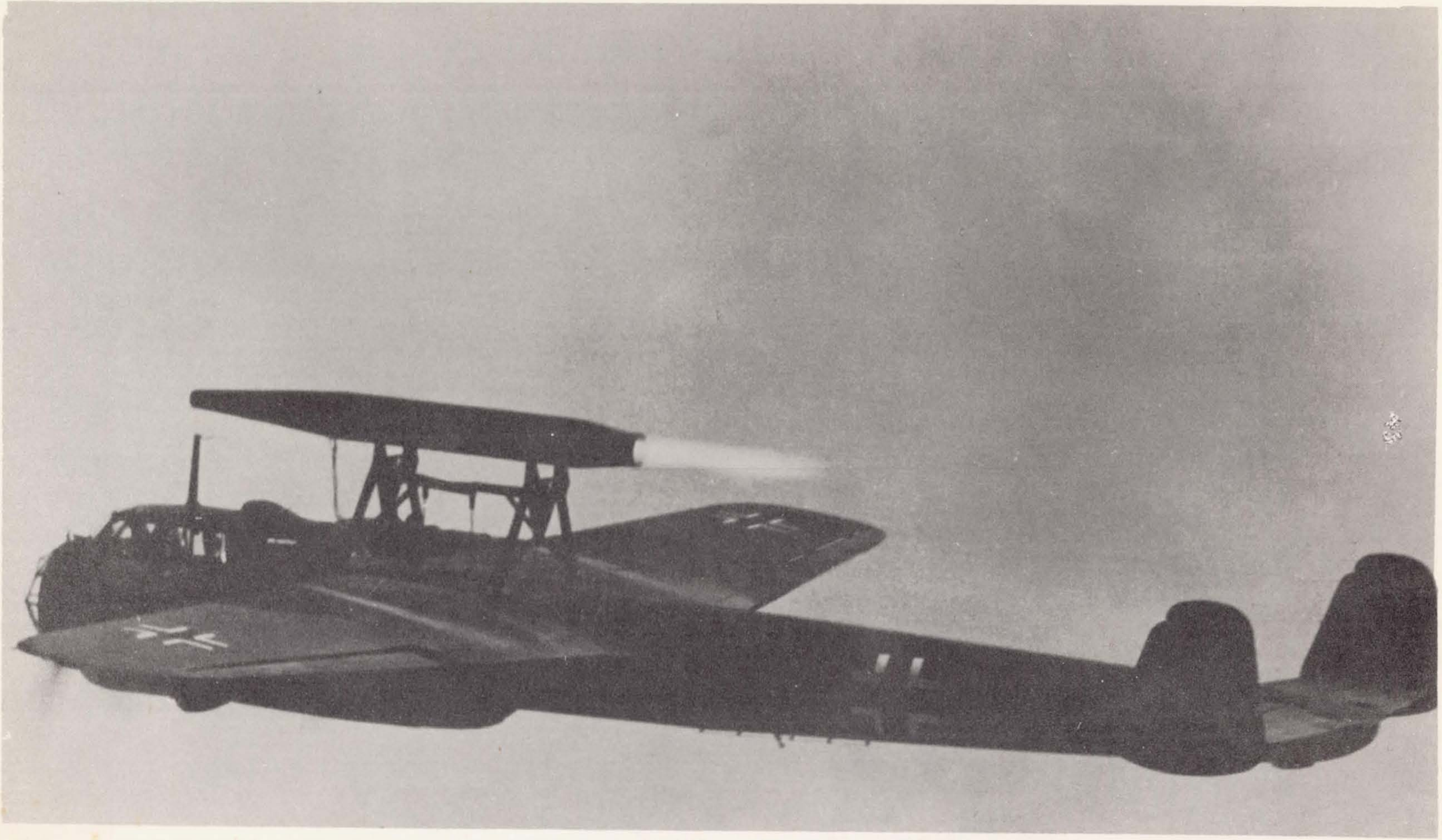

Figure 24. 2400hp jet tube with v=85m/sec. in operation.

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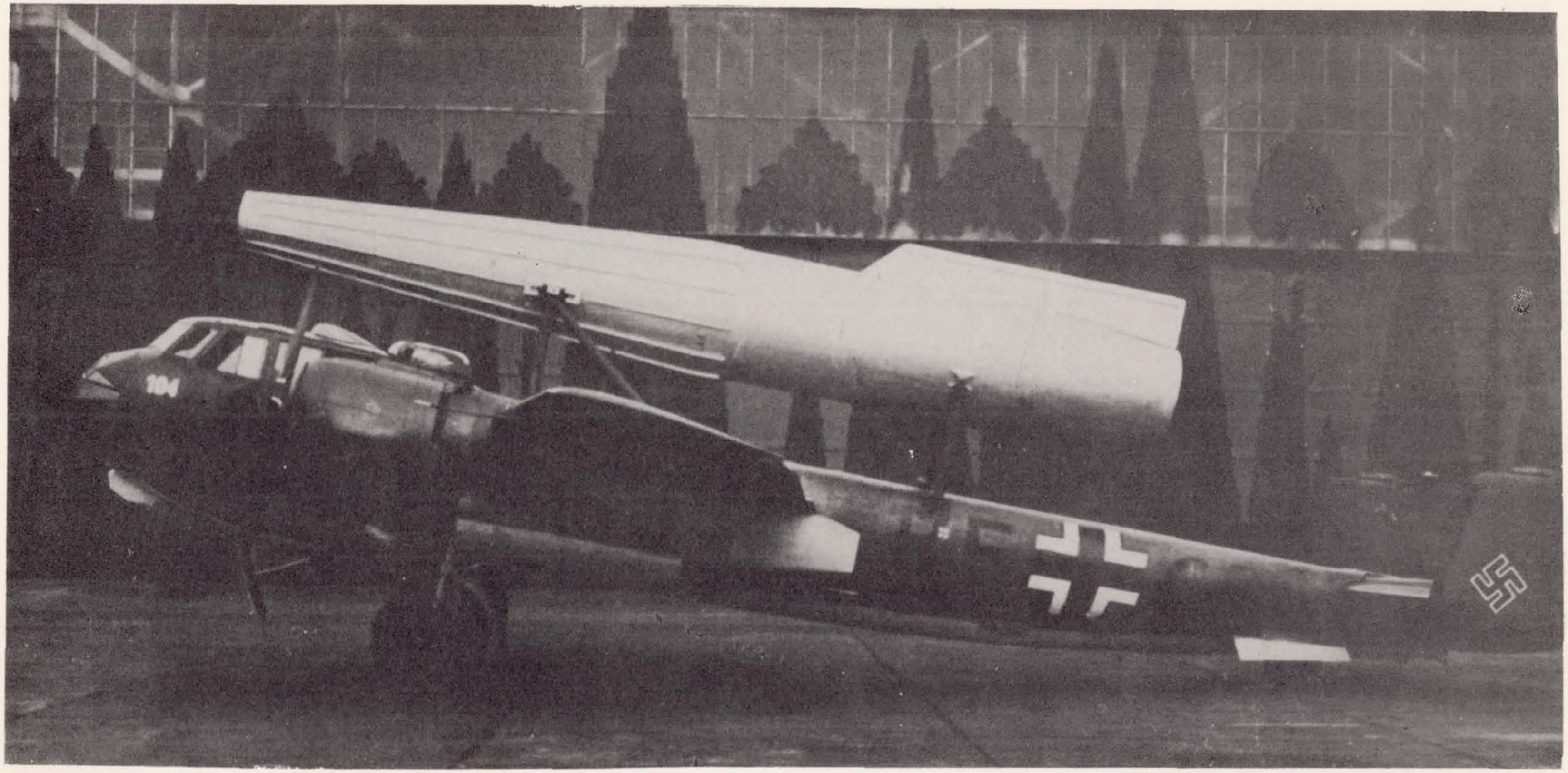

Figure 25.- Side view of 20,000-hp jet tube mounted on the Do. 217E2 as test carrier.

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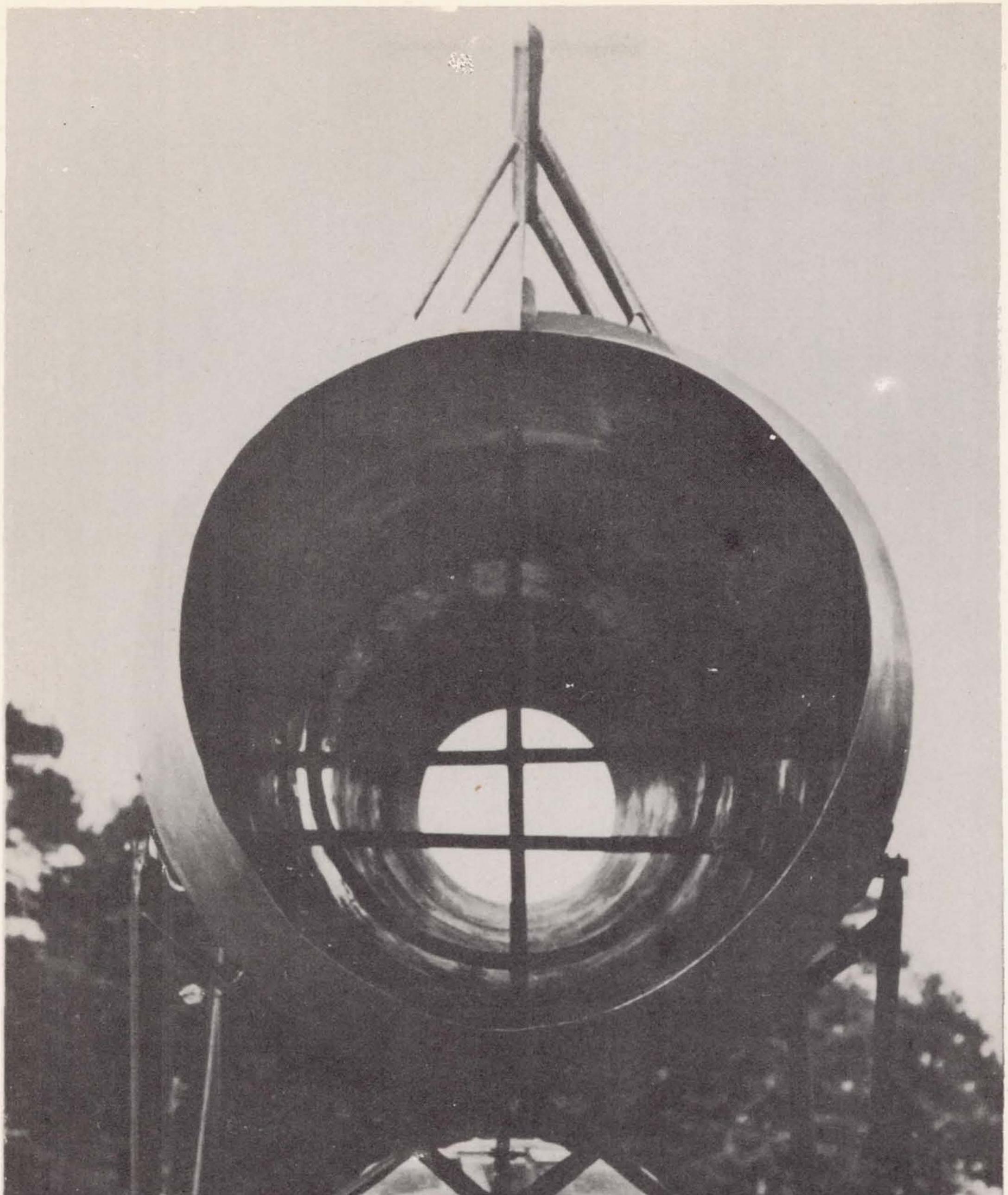

Figure 26.- Interior view showing injection cross from rear. 20,000-hp jet tube mounted on Do. 217E2.

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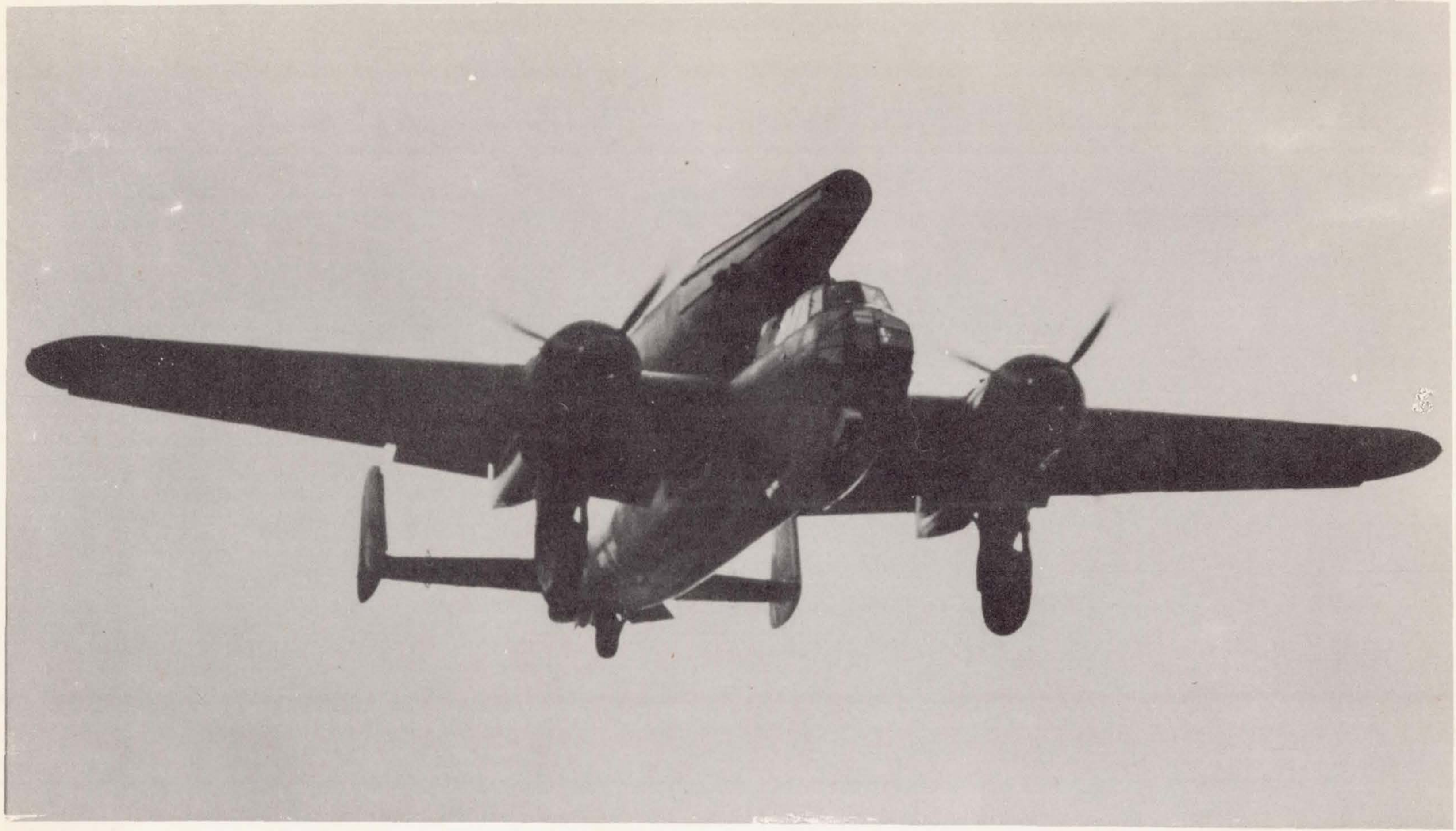

Figure 27.- 20,000-hp jet tube on the Do 217E2, flying in cold state.

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Figure 28.- 20,000-hp jet tube on the Do 217E7 with v = 130 m/sec
in operation.

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Figure 29.- 20,000-hp jet tube on the Do 217E7 with $v = 200$ m/sec in operation.

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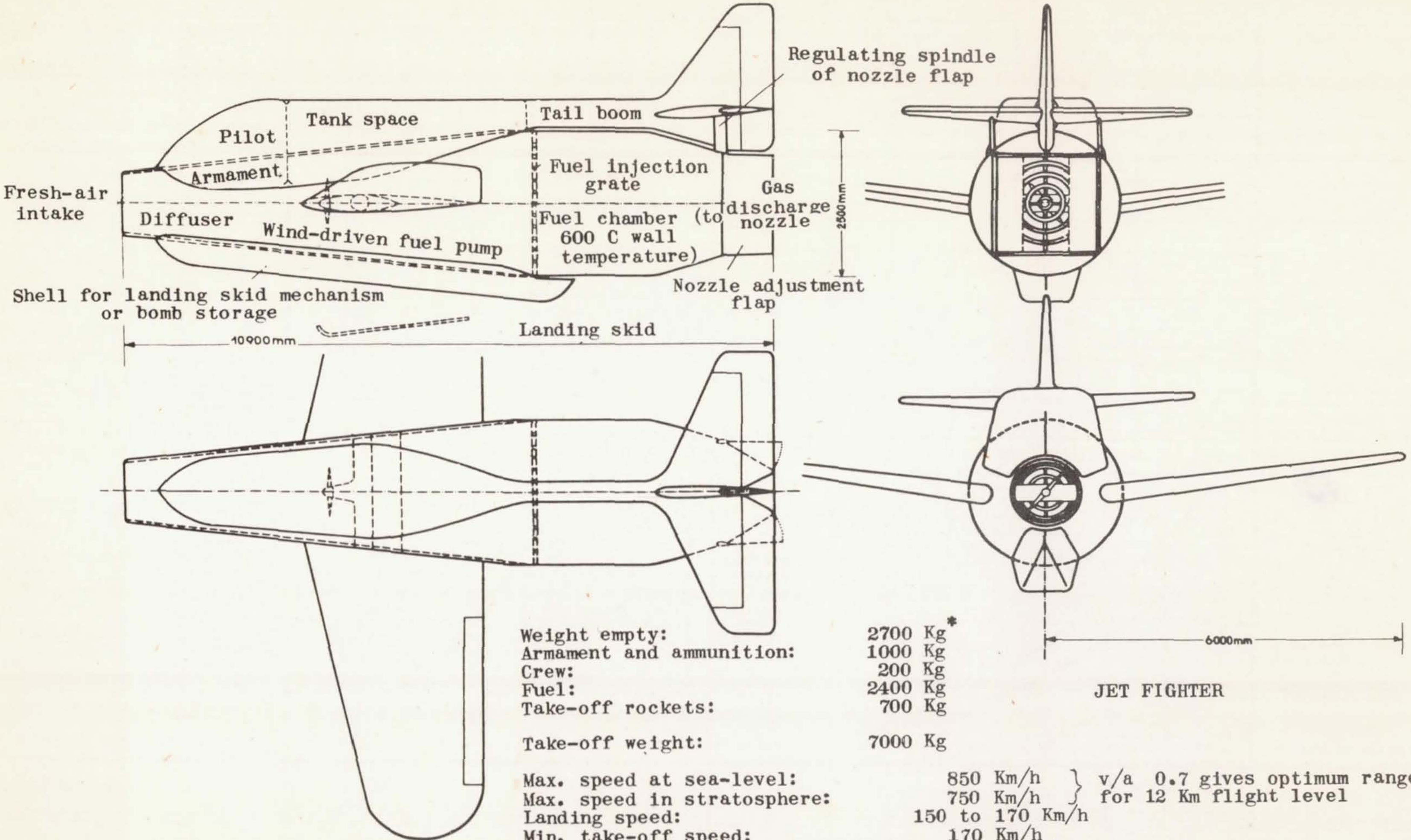

Weight empty:	2700 Kg*
Armament and ammunition:	1000 Kg
Crew:	200 Kg
Fuel:	2400 Kg
Take-off rockets:	700 Kg
Take-off weight:	7000 Kg

Max. speed at sea-level:	850 Km/h	v/a 0.7 gives optimum range for 12 Km flight level
Max. speed in stratosphere:	750 Km/h	
Landing speed:	150 to 170 Km/h	
Min. take-off speed:	170 Km/h	
Stalling speed, sea level:	13 to 15 m/sec	
Take-off and climb to 12 Km:	$2\frac{1}{2}$ min.	Max. duration at 12 Km, 50 min.
Max. length of flight:	800 Km.	Max. output: 30,000 hp.

* Weight used in preliminary test on the Ju 288 or its equivalent.

Figure 30.- Over-all arrangement drawing of a jet fighter with 60,000-hp jet unit.

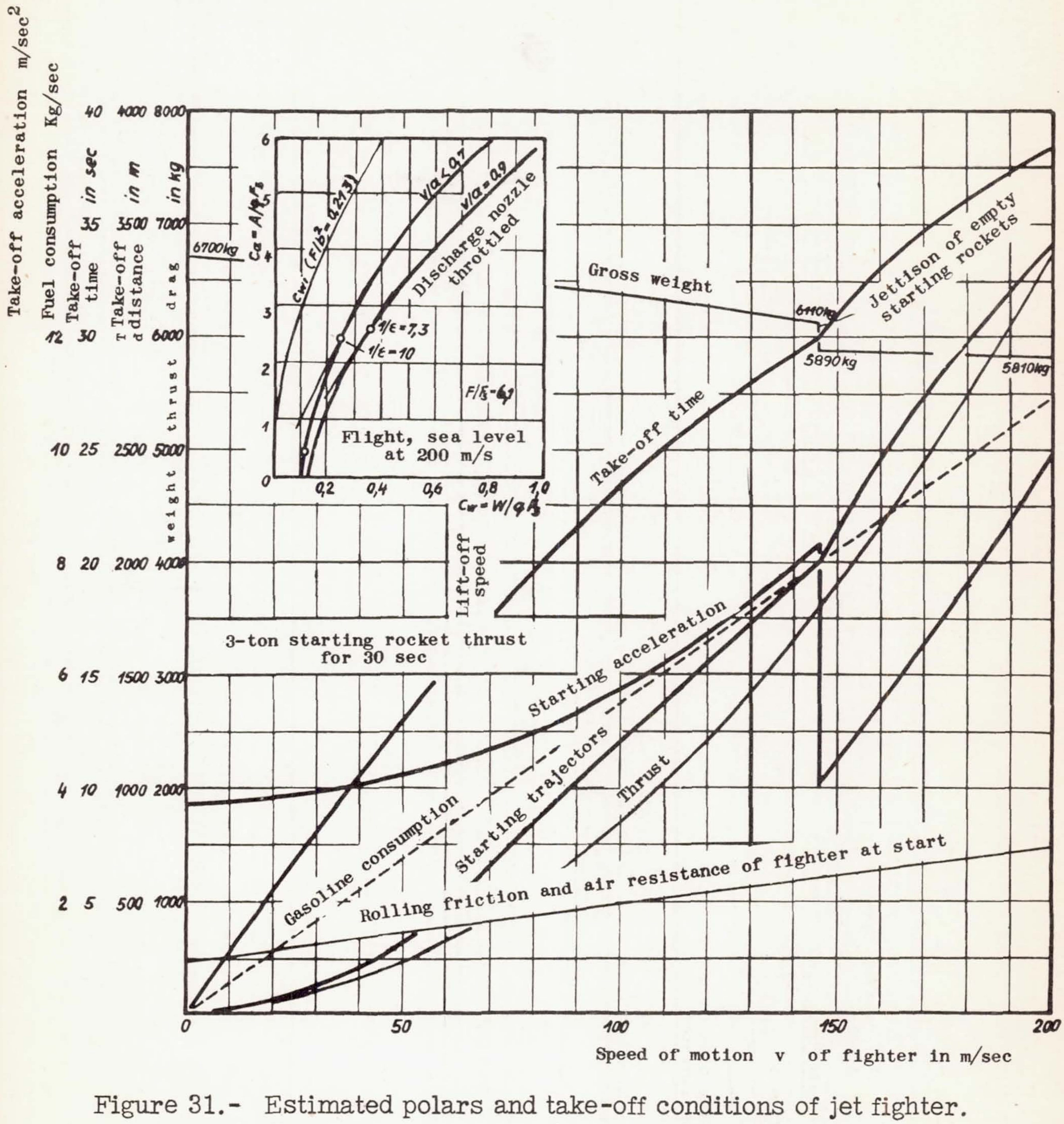

Figure 31.- Estimated polars and take-off conditions of jet fighter.

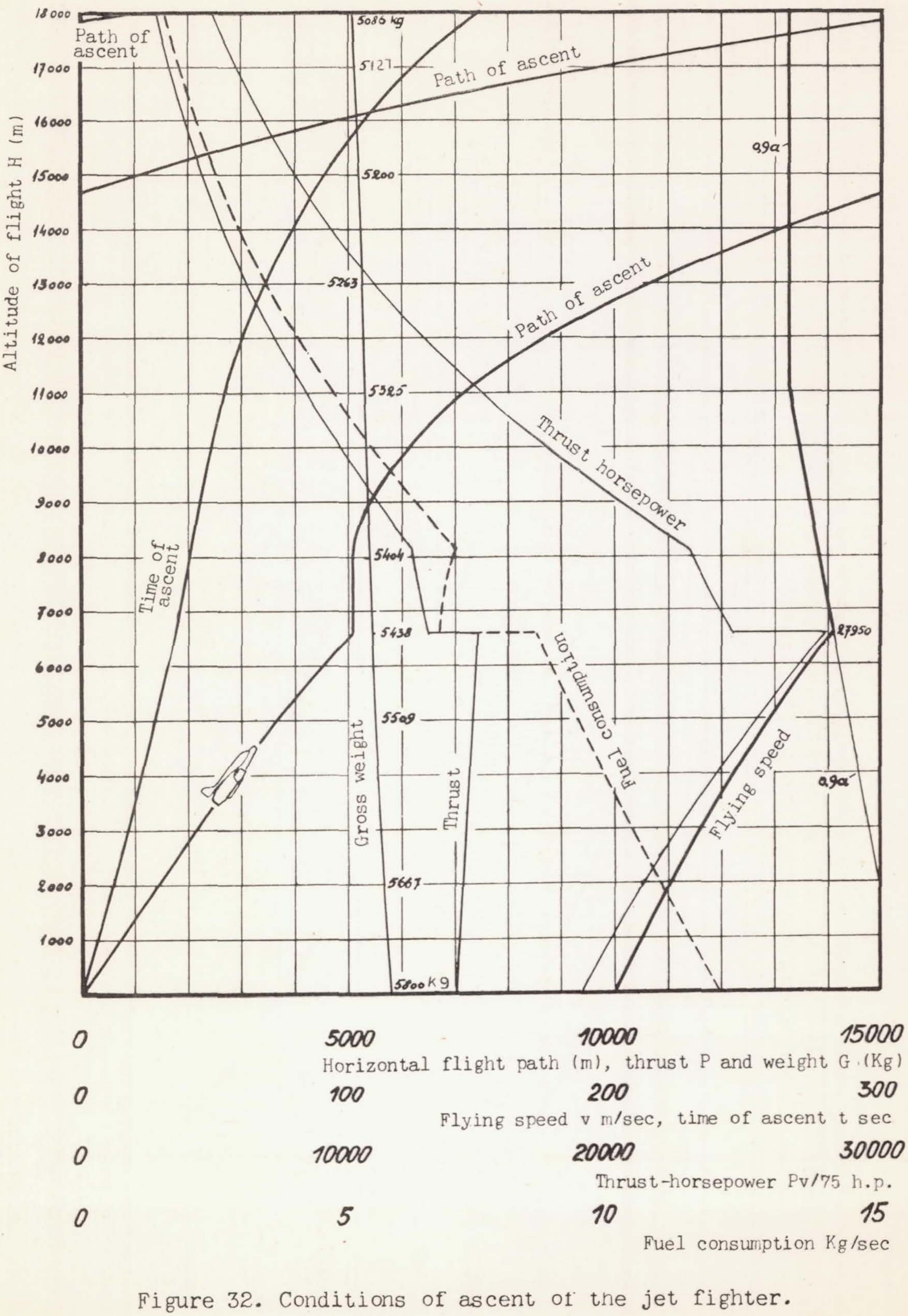

Figure 32. Conditions of ascent of the jet fighter.

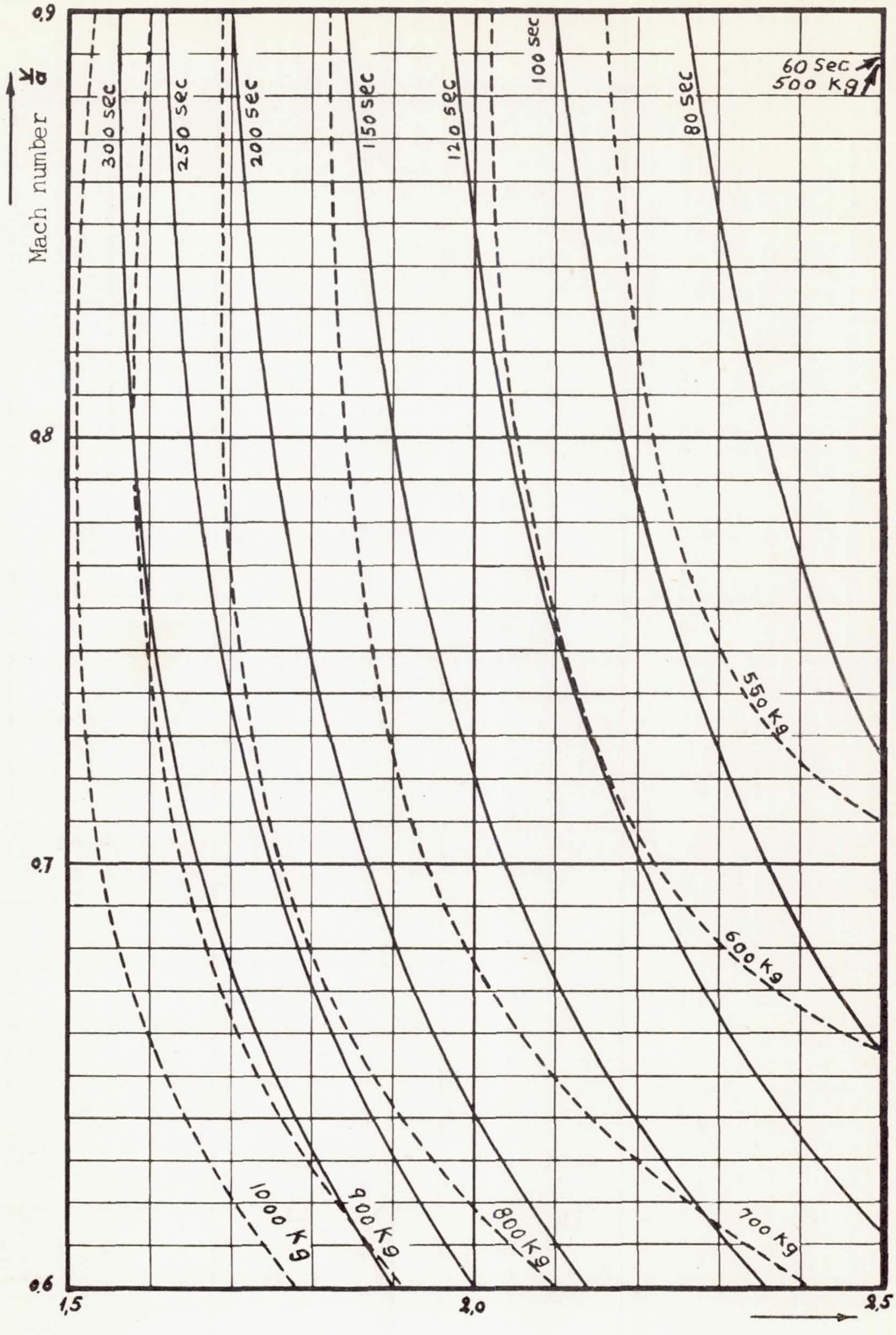

Figure 32a.- Actual time of ascent of jet fighter to 12,000 m altitude and relative fuel consumption plotted against flying speed v and tube diameter d_2.

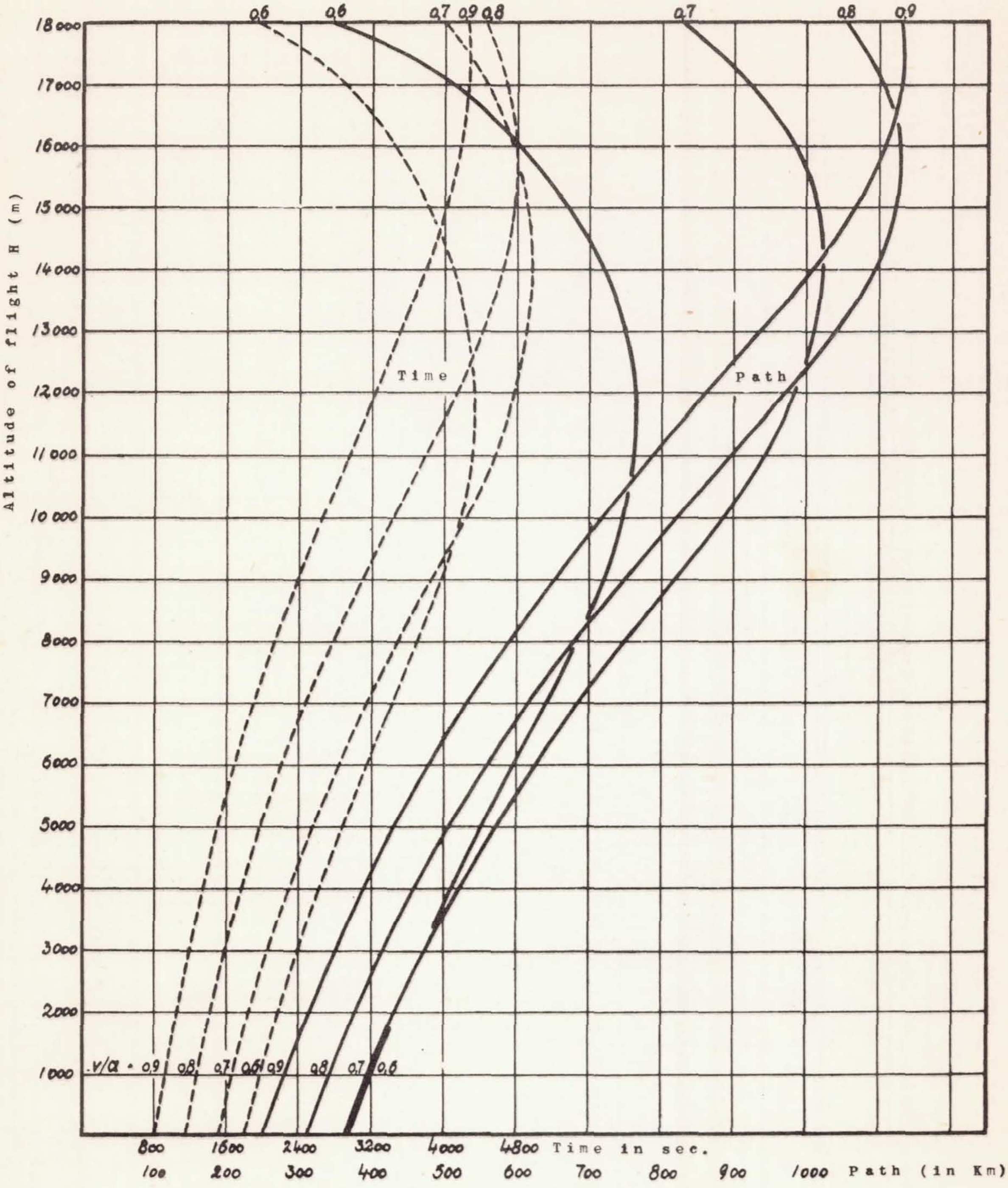

Figure 33. Conditions of horizontal flight.

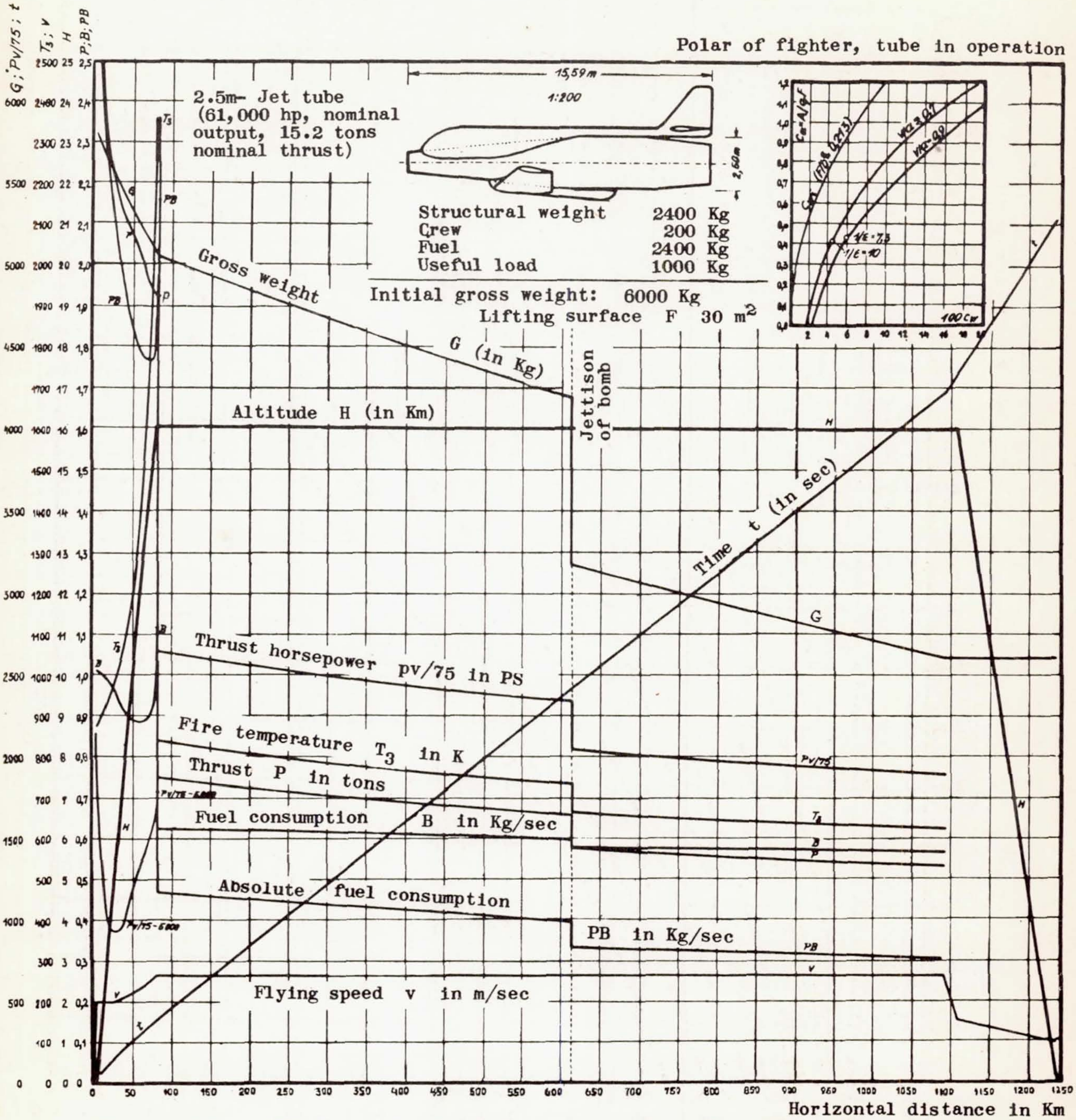

Figure 34.- Flight path of jet fighter used as level bomber gives at 1000 Kg useful load and 16 Km altitude about 430 Km depth of penetration.

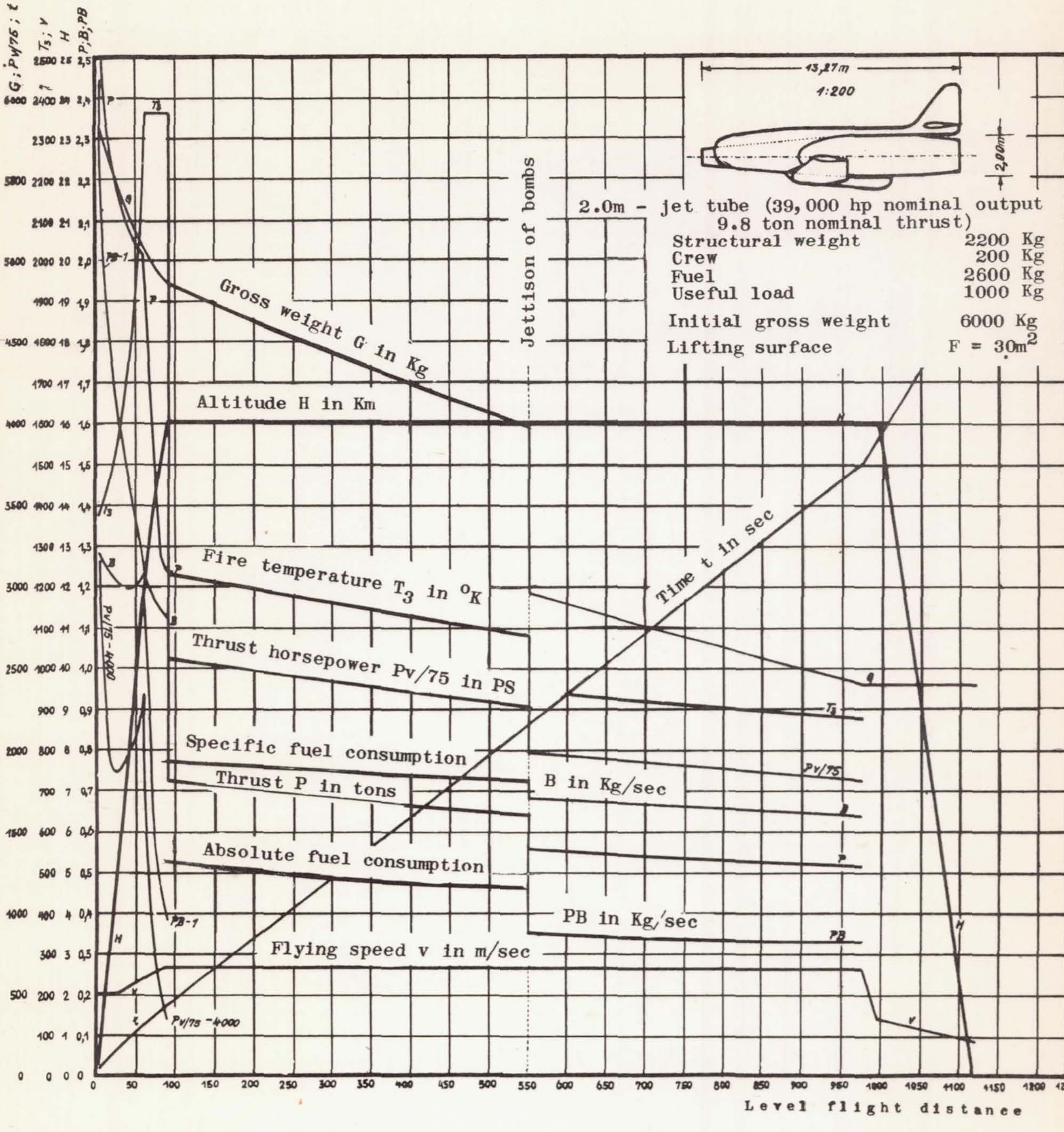

Figure 35.- Flight path of a jet bomber with 4000-hp jet tube gives at 1000 Kg useful load and 16 Km altitude about 390 Km depth of penetration.

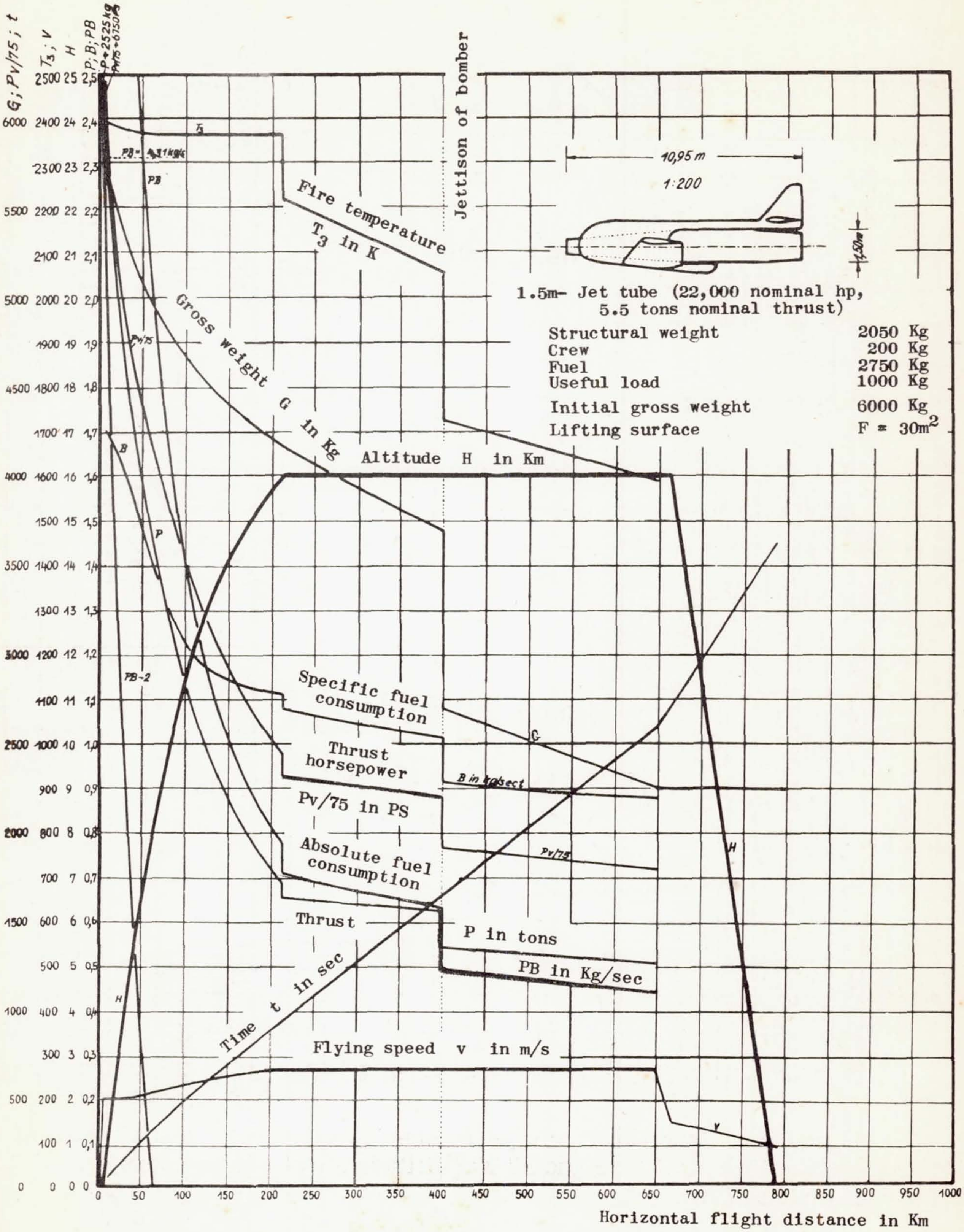

Figure 36.- Flight path of a jet bomber with 20,000-hp jet tube gives at 1000 Kg useful load and 16 Km altitude about 270 Km depth of penetration.