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TECHNICAL REPORT

WSRL-0332-TR

STATIC THRUST AUGMENTATION OF ROCKET MOTORS BY AIR ENTRAINMENT

R.D. IRVINE

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SUMMARY

The effect of duct geometry and primary jet characteristics on the level of thrust augmentation of conventional rocket motor jets, with conical nozzles exhausting into a stationary atmosphere, has been investigated. For a range of straight, cylindrical ducts with bell-mouth inlets, it is found that the duct thrust is primarily a function of duct length and primary thrust. Chamber pressure and nozzle conditions have little effect other than their effect on primary thrust. Further, over a range of length to diameter ratios of 3 to 7, diameter appears to have little effect, leaving thrust augmentation proportional to $LF_o^{-\frac{1}{2}}$ where L is duct length and F_o is the primary thrust.

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1. INTRODUCTION

Rocket motors, with high velocity jets, are least efficient at low speeds, or stationary. If there is a need to use rocket propulsion under these conditions, some gains might be possible by the addition of a non-afterburning, thrust augmenting ejector, which entrains atmospheric air to increase the mass-flow and reduce the final jet velocity, hopefully increasing total momentum flow in the process.

Although much work has been done on augmenters for low pressure-ratio jets, for V/STOL aircraft turbine propulsion, little has been published on non-afterburning augmenters for pressure ratios appropriate to rocket motors(ref.1). In recent years, Weapons Systems Research Laboratory and Aeronautical Research Laboratories have been conducting a cooperative research program into the augmentation of such jets(ref.2,3). Parallel work has been carried out by ARL, using high pressure cold air jets in a wide range of duct geometries, and by WSRL, using rocket motor exhausts, in some of the more promising of the duct geometries.

The tests reported here cover a range of simple, cylindrical ducts, with bell-mouthed inlets. A conventional solid propellant rocket motor, with simple conical nozzle provides a hot gas exhaust down the centreline of the duct. The variables considered include the length and diameter of the duct, the size (throat diameter) of the rocket motor nozzle and the chamber pressure of the rocket motor. The aim of the tests is to characterise the thrust on the duct over a range of parameters, and to provide a comparison with the result on similar ducts obtained by ARL using cold air tests, preferably leading to the ability to predict the level of thrust augmentation of a given duct in a rocket motor exhaust by measurement using only cold-air jet tests.

2. EXPERIMENTAL PROCEDURES

The experimental rocket motor firings were conducted using a vented vessel containing a suitable length of propellant, and exhausting through a short transfer tube and a simple, interchangeable, nozzle.

The vented vessel is a thick-walled steel motor case of diameter 250 mm, with a brass shear disc at the head-end, providing full diameter venting in the case of over-pressurisation. The nozzle end-closure consists of a flange to accept refurbished Murawa sustainer end closures, with a shortened, thick walled transfer pipe, and a Murawa sustainer nozzle. The steel motor case requires no thermal insulation, because of the bulk of the thick steel wall. This enables a quick turn-around and low cost firings to be achieved. The nozzle end-closure is insulated with a high density moulding of asbestos flock-phenolic resin, and the transfer tube is lined with a wound asbestos string/phenolic resin liner. The Murawa nozzle has a molybdenum insert machined to give the simple convergent-divergent form, with a conical divergence of 7½° half-angle. A second nozzle, identical in form but with reduced throat diameter, was machined from molybdenum to add variation of scale. Both nozzles have an expansion area ratio of 4.4 giving a pressure ratio $P_c/P_e = 26$.

The propellant charges were prepared from expired-life Murawa M3 Sustainer charges, containing cast double base propellant giving the following properties:

Flame Temperature:	2400°K
Specific Impulse:	2200 Ns/kg
Molecular Weight of Exhaust Gas:	22
Specific Heat Ratio:	1.24

The chamber pressure was varied during a firing by machining charges to give a regressive burning surface area. Some charges were machined in the form of a 90° conical burning surface on a short cylindrical charge As the burning conical surface intersected the rear face, the surface area decreased. A more gradual reduction in pressure was achieved by manufacturing charges with flat, "cigarette" burning surfaces, but with a gradual taper on the outside of the charge, to achieve the change in burning surface area. These charges were inhibited on the outside surface by casting a rubber inhibitor around them. Comparison of the two types of charges confirmed that there was no measurable error in thrust augmentation measurement due to the more rapid decay in pressure in the conical-face-charges. The charges are illustrated in figure 1. Typical pressure-time curves are illustrated in figure 2.

The ducts considered here all had a simple bell mouth, of radius 0.215 x diameter, and a straight cylindrical duct of varying length, L. Two nozzles, of throat diameter 13.85 mm and 11.0 mm, two duct diameters, 175 mm and 120 mm, and a variety of duct lengths from 390 mm to 1080 mm, were tested over a range of chamber pressures of 7 MPa down to 1.4 MPa. In all cases, the nozzle exit plane was coincident with the start of the cylindrical portion of the duct, and the nozzle centreline was aligned with the centreline of the duct. The arrangement is shown in figures 3 and 4. Pressure tappings were provided in the motor chamber to give chamber pressure, and, in the duct, level with the nozzle exit plane and at some downstream stations, to give duct static pressure.

The thrust of the duct was measured independently of the thrust of the rocket motor. The rocket motor thrust was measured on some occasions, to ensure that it agreed with the calculated thrust, but in general, the primary thrust of the rocket was derived from the chamber pressure. The duct was mounted on rollers and held down under its own weight for most firings, although for later firings a more elaborate flexure rig was developed.

In general, no afterburning occurred within the ducts, although some flashing and associated noise accompanied the firings. In the case of some of the longest ducts, of very high L/D ratio, an afterburning flame attached intermittently to the duct wall, giving rise to large pressure and thrust fluctuations, and some damage to the ducts. However, all these firings have been eliminated from the reported results. Further work remains to be done to clarify the conditions leading to afterburning and its effect on duct thrust.

3. EXPERIMENTAL RESULTS

Typical results of thrust augmentation ratio (the ratio of duct thrust to primary thrust) against motor chamber pressure are shown in figures 5(a) and 5(b). It can be seen that above a pressure ratio $P_c/P_a = 15$, the results form a system of roughly parallel curves, with the augmentation ratio being a decreasing function of chamber pressure. The augmentation ratio is also an increasing function of duct length, for constant duct diameter. chamber pressure and nozzle size, and a decreasing function of nozzle throat diameter for constant duct size and chamber pressure. This family of parallel curves can be transformed to a graph of thrust augmentation ratio against the function $LF_o^{-\frac{1}{2}}$, where F_o is the primary jet thrust and L the length of the duct. The curves plotted in figure 6 show the significant and surprising result that the thrust augmentation ratio is approximately proportional to $LF_o^{-\frac{1}{2}}$ within the range of parameters tested in this study. In particular, this relationship holds over a wide range of primary thrusts, for L/D greater than 3 and less than 7, and for nozzles under or over expanded between $P_a/P_e = 0.4$ and $P_a/P_e = 2.0$. In this range, the augumentation ratio is not sensitive to duct diameter. The relationship fails when L/D is reduced to 2.2, and when L/D is increased to 8.6, in both cases by the thrust augmentation being lower than predicted by the straight line. The thrust augmentation ratio fluctuates above and below this function when the nozzle is grossly overexpanded, $P_a/P_e > 2$, possibly associated with nozzle flow separation.

These results are plotted in a different form in figure 7. $F_d/L\,F_o^{\frac{1}{2}}$ is seen to be fairly insensitive to chamber pressure, although the curves do exhibit detectable maxima near the point of correct nozzle expansion, $P_c/P_a = 25$. Apparently the effect of varying chamber pressure on augmenter thrust can be approximately expressed in terms of its effect on primary jet thrust.

4. DISCUSSION

The mixing between the rocket exhaust and the entrained air is a very complex process, and it is beyond the scope of this paper to attempt to model the system, although the simplicity of the results holds hope that a relatively straight-forward model might be made to predict the experimental results in the region of interest. Consequently this discussion will be restricted to the limitations of applicability of the equation

$$F_d = \alpha L F_o^{\frac{1}{2}} \qquad (1)$$

where α is a constant, and to the applicability of this equation to the cold air measurements of ARL.

Firstly, we noted in Section 3 that the independence of the duct thrust on diameter broke down at length to diameter ratios as low as 2.2 and under, and of 8 and greater. For the small diameter ducts, L/D > 8, we have a very large value of inlet pressure drop ΔP. This leads to a large velocity (at least for large values of the primary nozzle thrust, F_o), and we might expect turbulent separation losses in the secondary flow. If this were the whole answer, however, we would expect the curve of F_d/F_o versus $L\,F_o^{-\frac{1}{2}}$ to return to the linear result, for values of F_o small enough to give ΔP values equal to those occurring for lesser values of L/D. That this is not the case is obvious from figure 6, where the curve for L/D = 8.6 is below the curves for L/D = 6.5 for all values of $L\,F_o^{-\frac{1}{2}}$ tested. An alternative explanation would be that tne central core of the exhaust plume spreads conically from the nozzle, and tends to intersect any duct at a point about 6 to 8 diameters downstream, causing a large increase in duct drag for L/D ratios greater than this value. Thus we would expect an increase in ΔP, due to increasing entrainment, but no proportional increase in duct thrust. This is just what we do see in this case, although the amount of evidence is small and the scatter large.

For large diameter ducts, with L/D = 2.2, we clearly have no excess drag terms to account for. ΔP is always small, and is essentially proportional to F_d, as it would be for isentropic flow with no losses. A factor which may become important in such low L/D ratio ducts is the breakdown of the assumptions of one dimensional flow. For such ducts, it is no longer even approximately true that the flow in the duct is parallel with the axis of the duct or uniform across the duct. This would have a tendancy to reduce the duct thrust for a given mass flow.

The empirically derived equation (1) assumes the air entrainment is independent of nozzle over or under-expansion. This is a reasonable assumption, since the exit velocity is relatively independent of expansion ratio (over the modest range of pressure ratios considered here) and the jet is free to expand or contract to the ambient pressure after leaving the nozzle. We might expect a new mixing mode to develop when the nozzle flow separates within the nozzle, but this is outside the range of pressures of interest.

The assumption is upheld by the results plotted in figure 7. It can be seen that, over a range of chamber pressures from 1.5 MPa to 7.0 MPa, there is little variation in $F_d/LF_o^{\frac{1}{2}}$ apart from a slight increase near the point of correct expansion, at 2.8 MPa. Since our experiments were conducted with only a single nozzle expansion ratio, the independence of entrainment with chamber pressure is demonstrated only at one expansion ratio. It would be valuable to confirm this by experiments with at least one nozzle of significantly different area ratio, with chamber pressures varied about the correct nominal pressure.

The nature of the exhausting gas has not been considered so far, but it would be valuable to relate this work to the more extensive test results obtained using cold air jets, at ARL(ref.2,3). The differences between the jets involve temperature, specific heat ratio, and molecular weight (see Table 1). All other conditions are identical.

TABLE 1. PROPERTIES OF EXHAUST JET FOR ROCKET MOTOR TESTS (WSRL) AND COLD AIR TESTS (ARL)

	Rocket Motor Gas	Cold Air
Chamber Pressure	1.5 MPa - 7.0 MPa	1.5 MPa - 6.0 MPa
Nozzle Area Ratio	4.4	4.1
Nozzle Pressure Ratio	28	34
Stagnation Temperature	2400°K	290°K
Specific Heat Ratio	1.24	1.40
Molecular Weight	22	29

If there is any correspondence with the results for rocket motor exhaust jets described here, the same relationship,

$$F_d = \alpha L F_o^{\frac{1}{2}} \qquad (2)$$

would be expected but with a different value of α, and, perhaps different limits to the range of L/D in which (2) is valid. Figure 9 is a plot of F_d/F_o against $L F_o^{-\frac{1}{2}}$ for a range of values of L/D from 4.5 to 9.1, for a range of chamber pressures from 1.5 MPa to 6 MPa, and for five different nozzle areas. The solid line represents a best fit straight line for the previously considered hot gas results, taken from figure 6.

Although the agreement with equation (2) is not as striking as figure 6, it can be seen that, for L/D > 6, a straight line results at high pressure (low values of $LF_o^{-\frac{1}{2}}$), followed by a break upwards, at a critical value of L/D_o. As it happens, a single duct diameter was used for all the cold air tests. In searching for a common denominator for the break point at each value of L/D, it was noticed that the break point occurred at nearly the same value of primary thrust. We can see this in figure 8 where a transformed version of equation (2) is plotted, F_d against F_o. Here, the cold air results are compatible with aquation (2) provided L/D >6, and the primary thrust was greater than 200 N, for the given duct diameter of 0.111 m. Below this critical value of primary thrust, there appears to be a significant change in the mechanism of air entrainment, causing large fluctuations in the level of thrust augmentation.

The lower limit on L/D of about 6, is much higher (about twice as high) than for rocket motor jets, and the upper limit is higher than the highest tested, 9.1. This could be due to the hotter jet spreading at a greater rate than the cold jet, amd interfering with the duct at an earlier point downstream. It would appear to rule out the possibility of non-one-dimensional flow. At L/D ratios of 5, such effects should have vanished.

It is interesting that the value of the constant α, for the cold air case, is not significantly different from its value for rocket motor jets. Although it is impossible to say, at this stage, whether the process involved is remarkably insensitive to exhaust temperature, and exhaust gas properties, or whether coincidence has balanced out competing changes in this case, evidence from the work of Simonson and Schmeer(ref.1) strongly supports the idea that such a constant is independent of exhaust conditions. Their jet had a stagnation temperature of about 1300°K and a specific heat ratio of 1.31. Figure 10 plots their results for similar simple ducts and the dashed line represents the mean line from figure 6. Despite having a grossly overexpanded nozzle (area ratio of 15 and a pressure ratio of 25 to 45), and having exhaust properties mid-way between the WSRL hot jet and the ARL cold jet, the curves still lie strikingly close to the mean line of the WSRL results. Their results cover a restricted range of primary thrust, and only one value of duct length and diameter, but still provide support for equation (2).

However, there are many unknowns in the process, and correspondence between the hot-gas and cold air cases is not well established. The main difficulties are:

(i) The anomalous results for low primary jet thrust in the cold air jet, and in particular, the sudden onset at a threshold thrust, do not appear at all in the hot gas results. The variation in this threshold thrust with duct diameter is not known.

(ii) Although the optimum ejector performance for a given length is comparable between the hot and cold jets, the optimum value of diameter for a given length is quite different, by about a factor of 2.

5. CONCLUSIONS AND RECOMMENDATIONS

Test firings of rocket motor jets through simple cylindrical ducts with bell mouth entries have shown that duct thrusts of up to 35% of primary thrust are possible, and that, for a range of duct length to diameter ratios of 3.9 to 6.5, the duct thrust is proportional to the length of the duct, and to the square root of the primary thrust. In the range tested, duct diameter and nozzle under or over expansion had little influence.

Some agreement with the results of cold air tests at ARL has been obtained. It would appear that cold air tests could be a useful and easy way to test augmenter ducts, provided the region of the cold air test is away from that area which gives anomalous results. However, further work is needed to establish the cause of the anomalous cold air results, or at least define the conditions which are needed to avoid such results.

Further work is needed to clarify the results at very high length to diameter ratios, since for a given primary thrust, greatest augmentation is achieved at long lengths, and to reduce duct weight, the smallest diameter would be used. The effect of different nozzle expansion ratios to the one used here should also be investigated, although it is not expected that this effect will be significant. It has been reported earlier(ref.3) that varying nozzle expansion ratio for a cold air jet has no appreciable effect on thrust augmentation.

Fisher(ref.2,3) has suggested introducing a lobed nozzle, to increase the mixing of the jet, and has reported significant increases in duct thrust. Some limited tests have been carried out on rocket effluxes to confirm that an improvement does result. Further work is recommended to quantify the improvement, and optimise the nozzle shape.

The effect of afterburning in the duct, and the control of afterburning remains an unresolved problem. This should be investigated, using propellants representative of likely rocket motor propellants of interest.

6. ACKNOWLEDGEMENTS

Much credit is due to Mr Sam Fisher of Aeronautical Research Laboratories, who supplied the results of ARL cold-air experiments and who provided much advice and assistance.

NOTATION

D	duct diameter
F_d	duct thrust
F_o	primary thrust
L	duct length (measured from start of cylindrical section)
L/D	length/diameter ratio
P_a	ambient pressure
P_c	chamber pressure
P_e	exhaust pressure
ΔP	inlet pressure drop = $P_a - P_{inlet}$

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2	Fisher, S.A.	"Thrust Augmenting Ejectors for High Pressure Ratio Propulsive Jets". Proc. 7th Australasian Conference on Hydraulics and Fluid Mechanics, August 1980
3	Fisher, S.A. and Irvine, R.D.	"Air Augmentation of Rockets for Low Speed Application". Proc. 5th International Symposium on Air Breathing Engines, February 1981

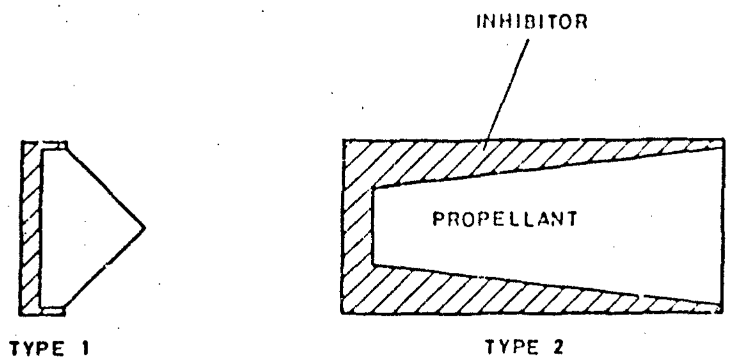

Figure 1. Charge geometries used

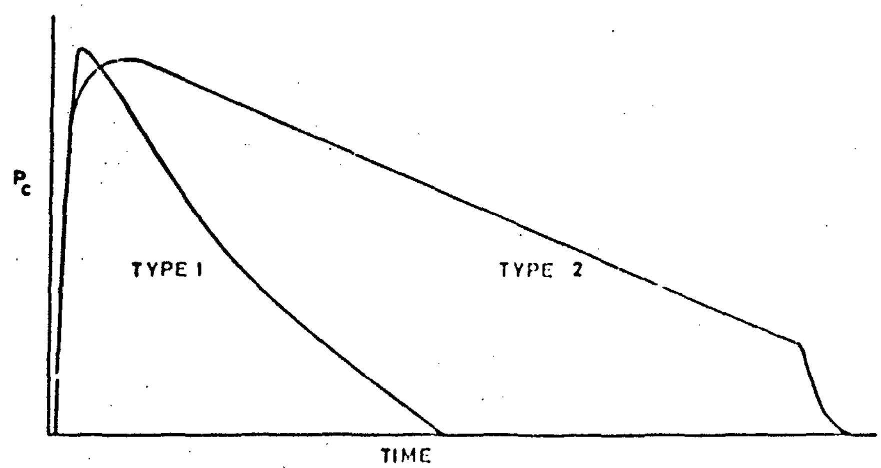

Figure 2. Typical pressure - time curves

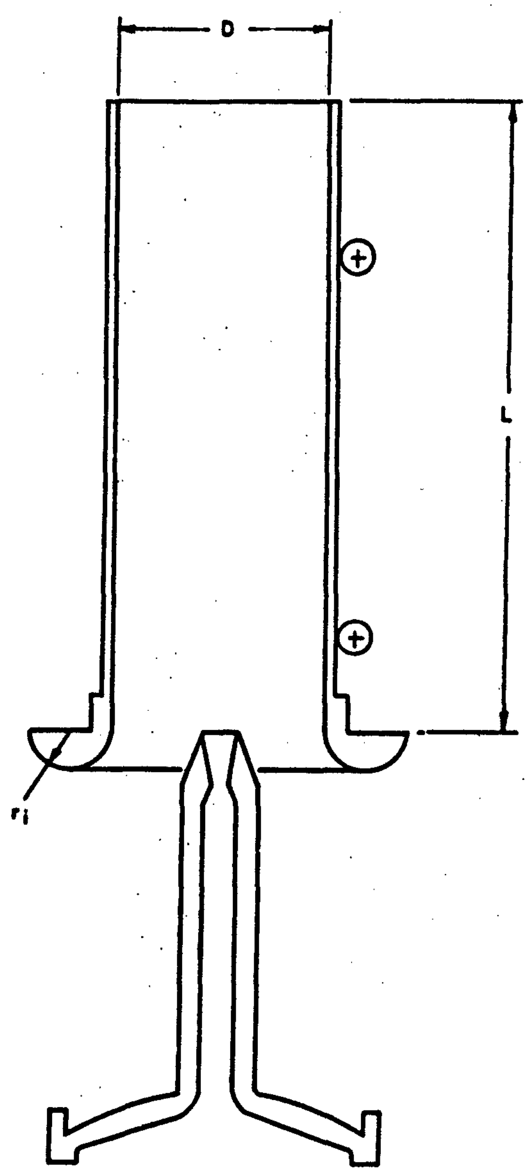

Figure 3. Layout of rocket motor and duct

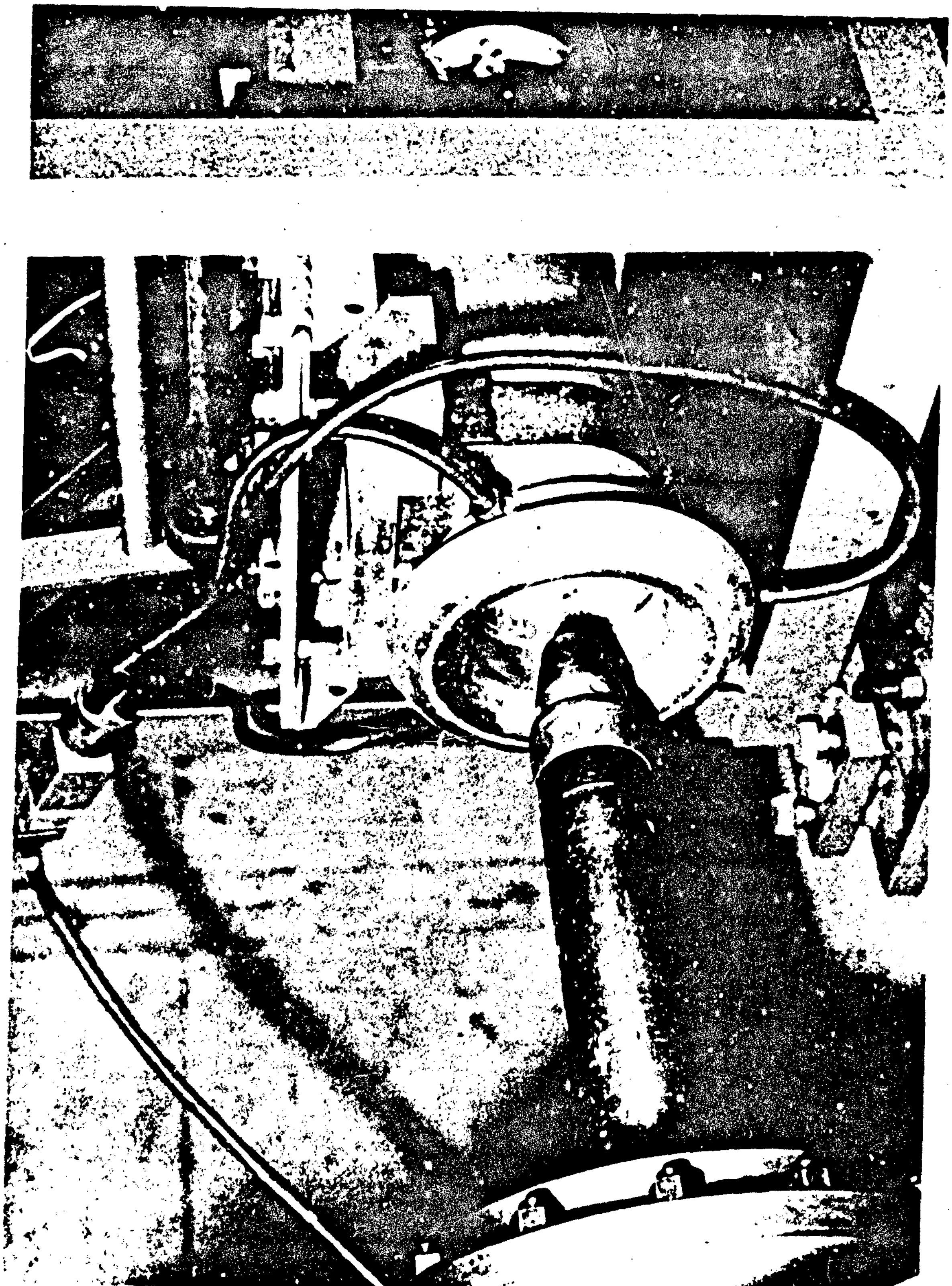

Figure 4. Ducted motor in static firing site

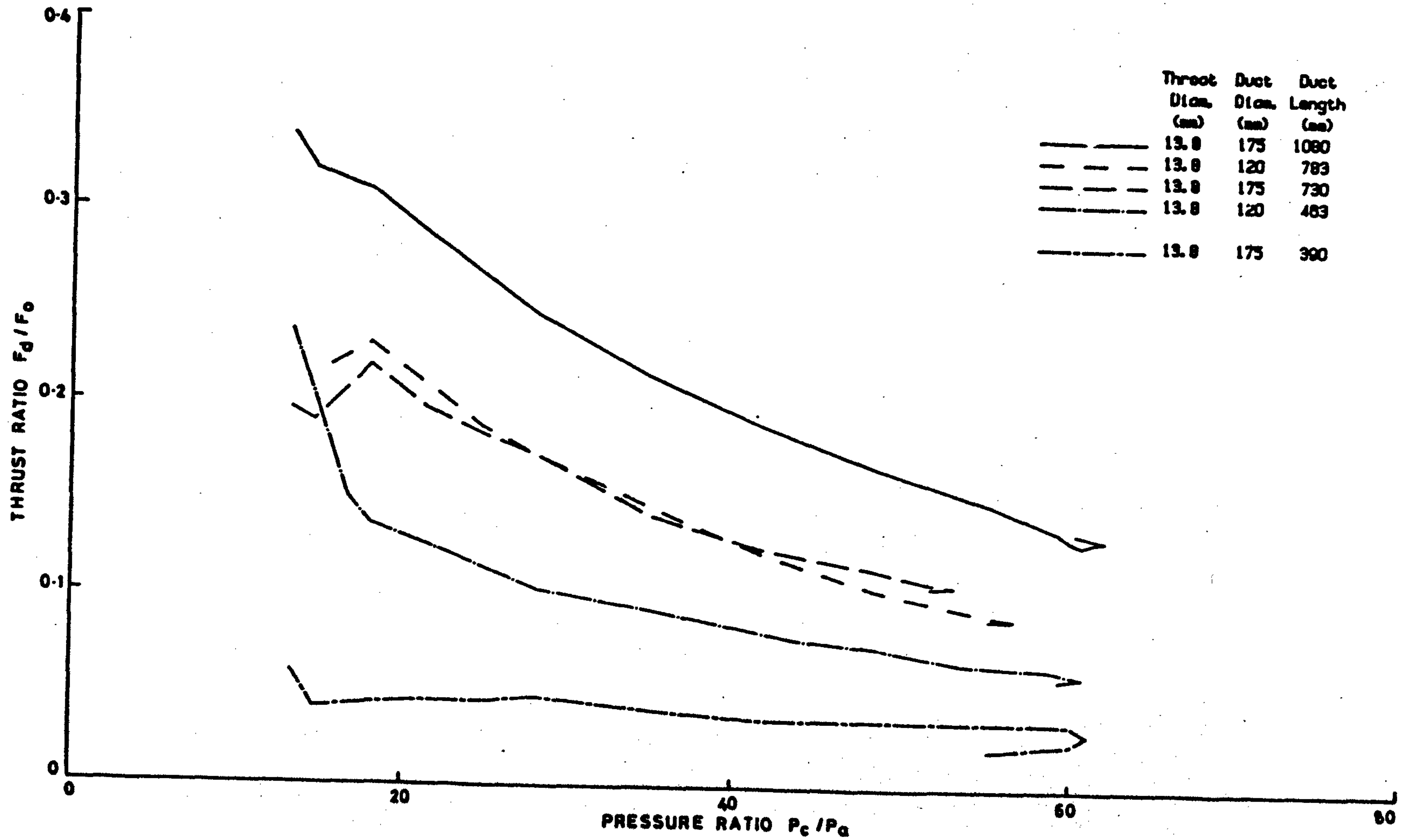

Figure 5(a). Thrust ratio versus pressure ratio

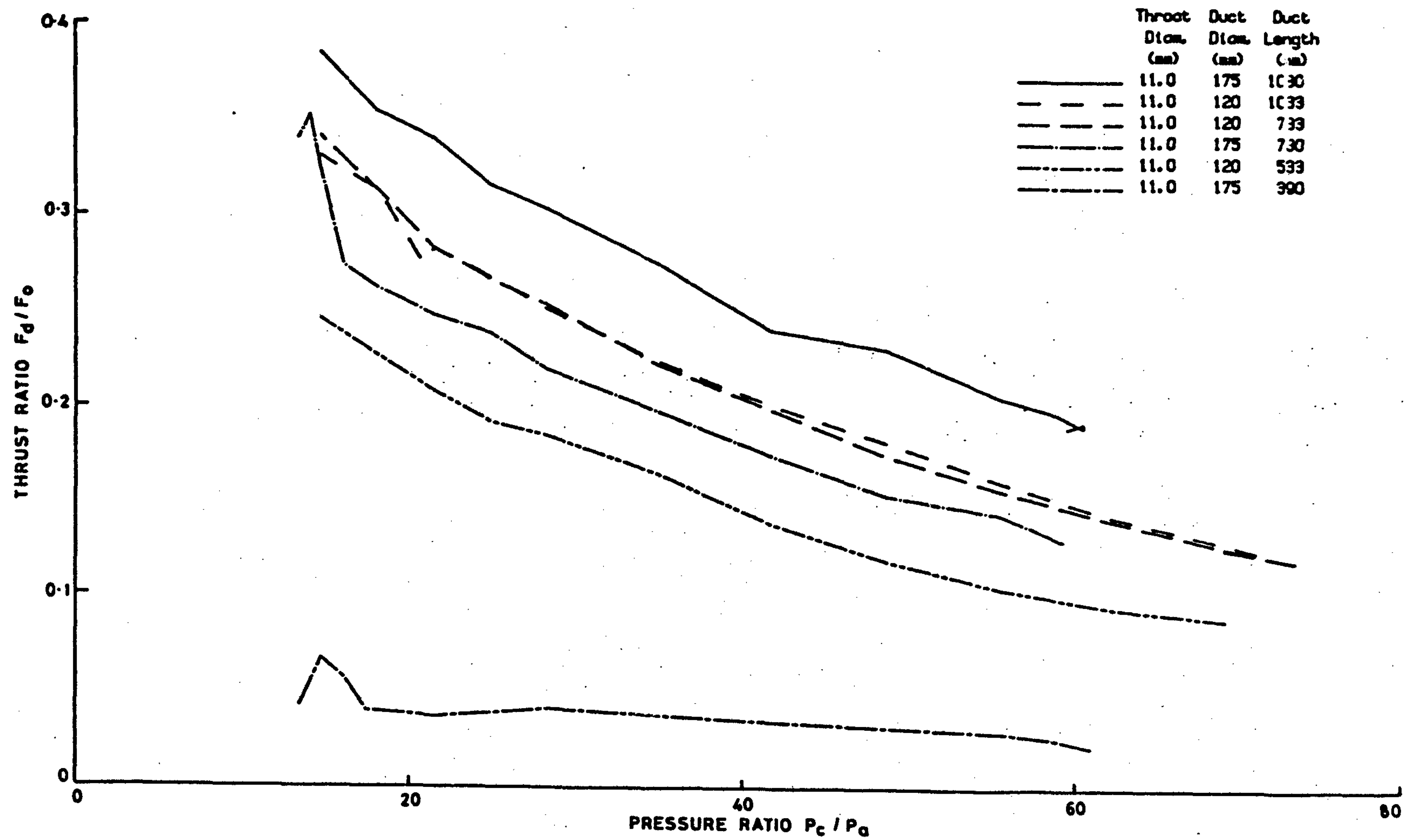

Figure 5(b). Thrust ratio versus pressure ratio

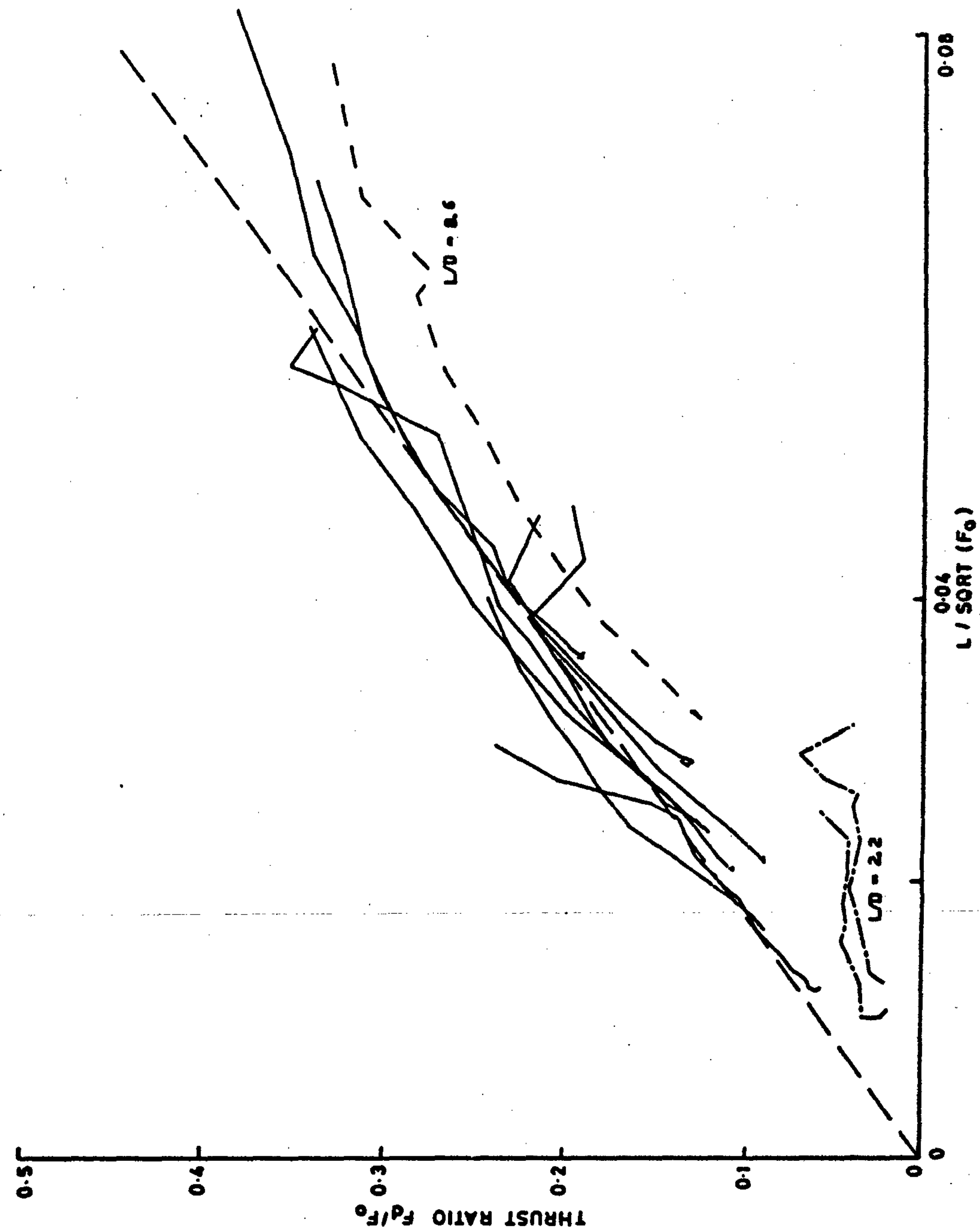

Figure 6. Thrust ratio versus L/SQRT (F_o)

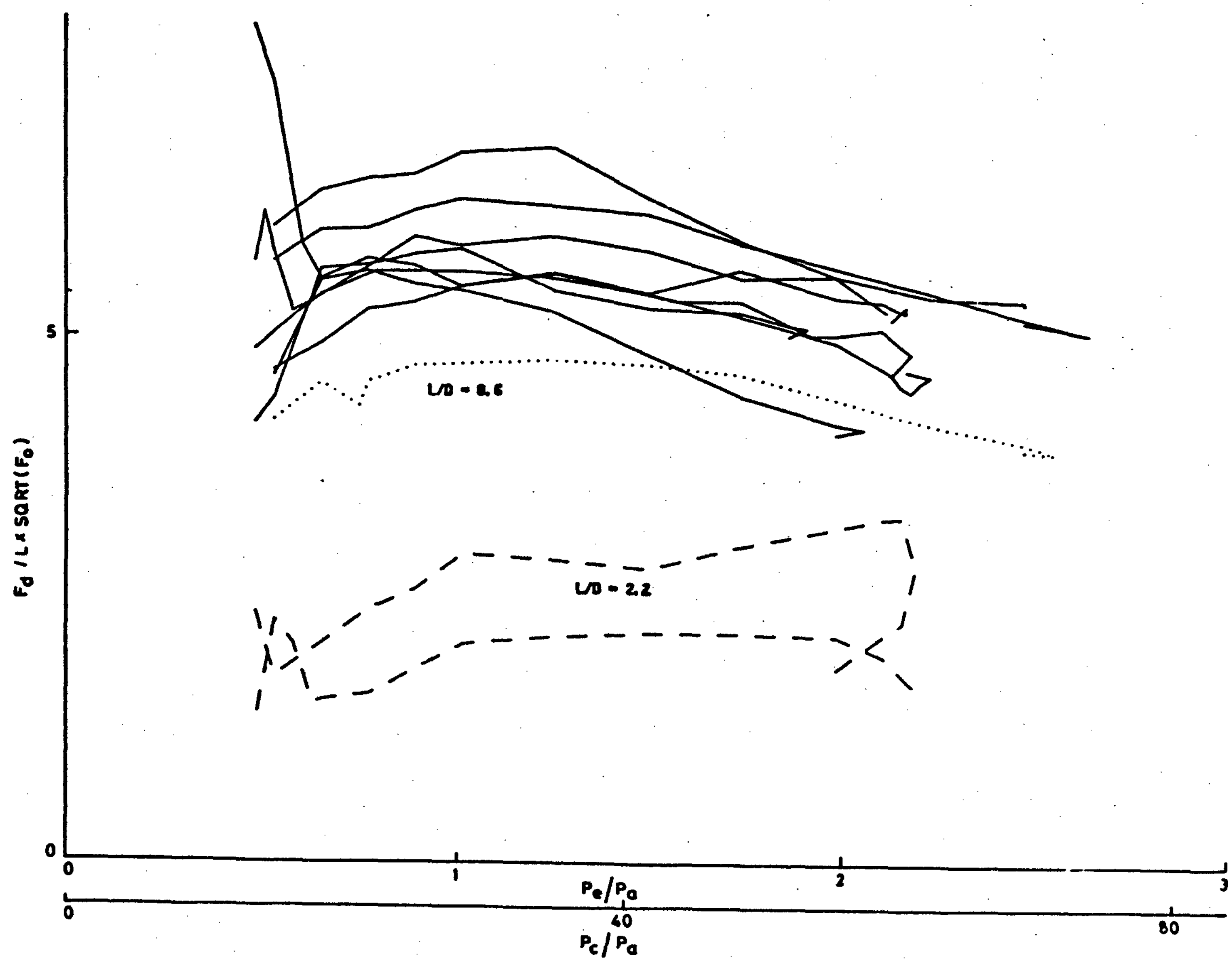

Figure 7. F_d/L.SQRT (F_o) versus pressure ratio

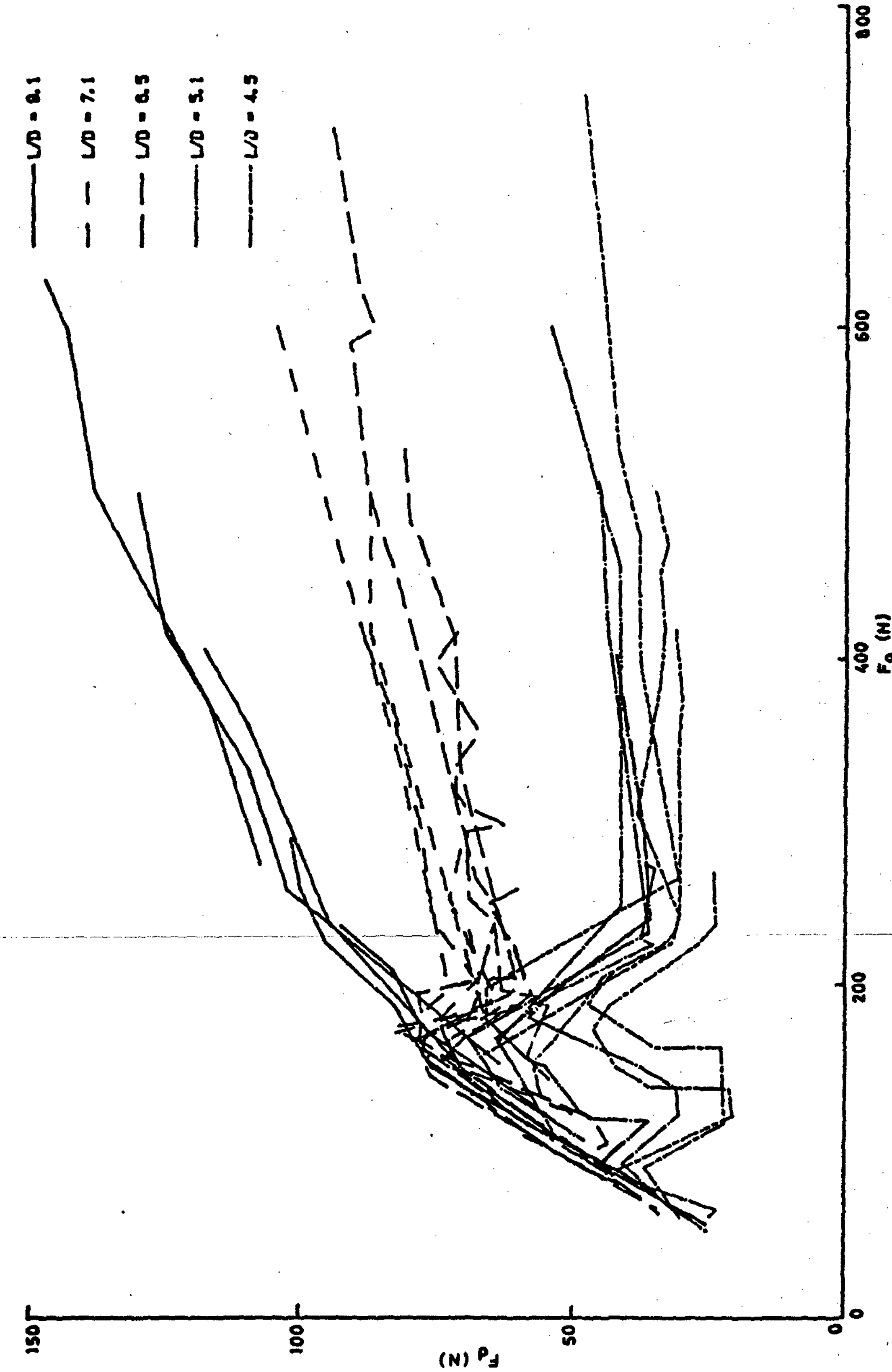

Figure 8. Cold air results (ARL data)

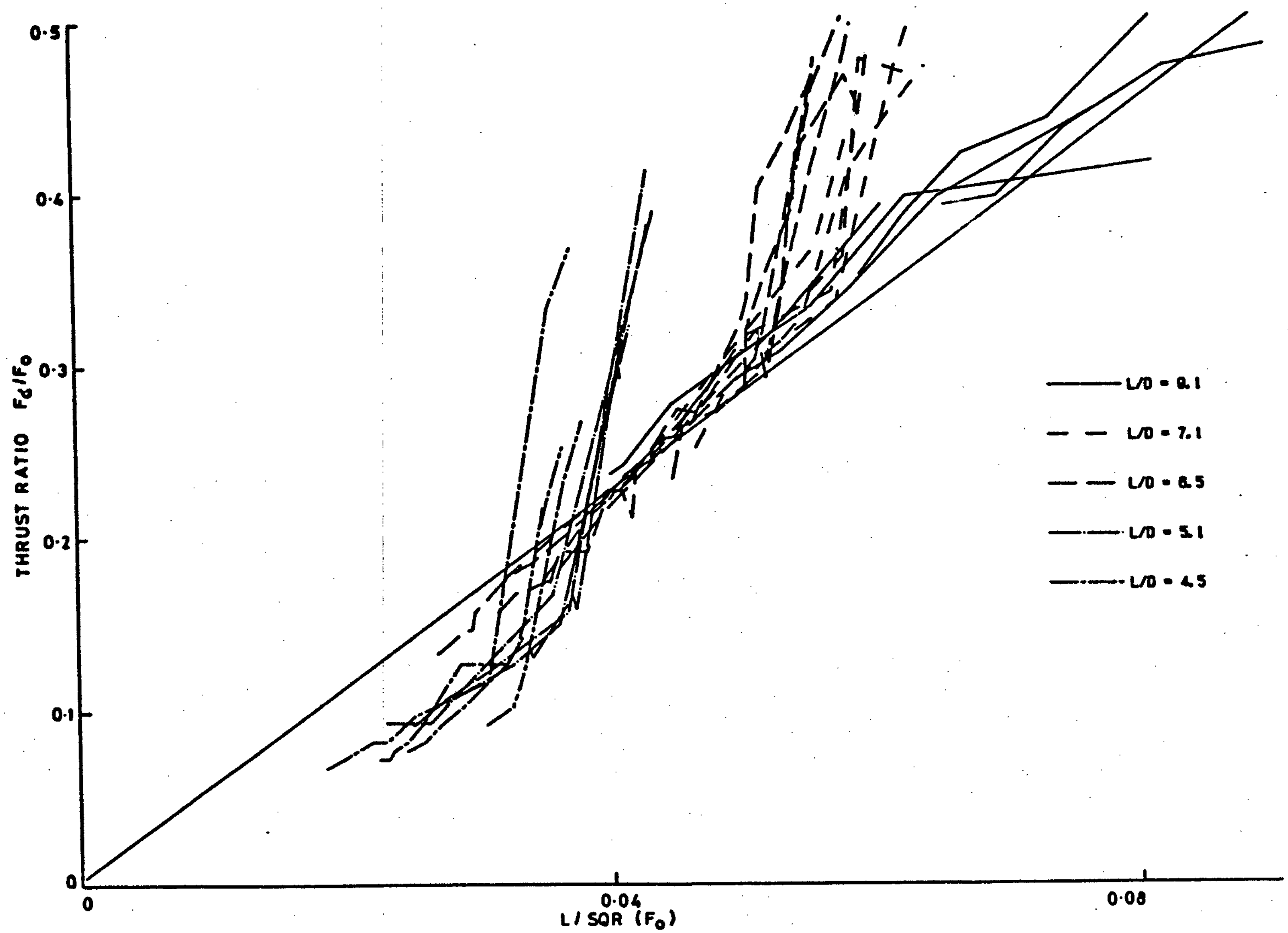

Figure 9. Cold air thrust ratio versus L/SQR(F) (ARL data)

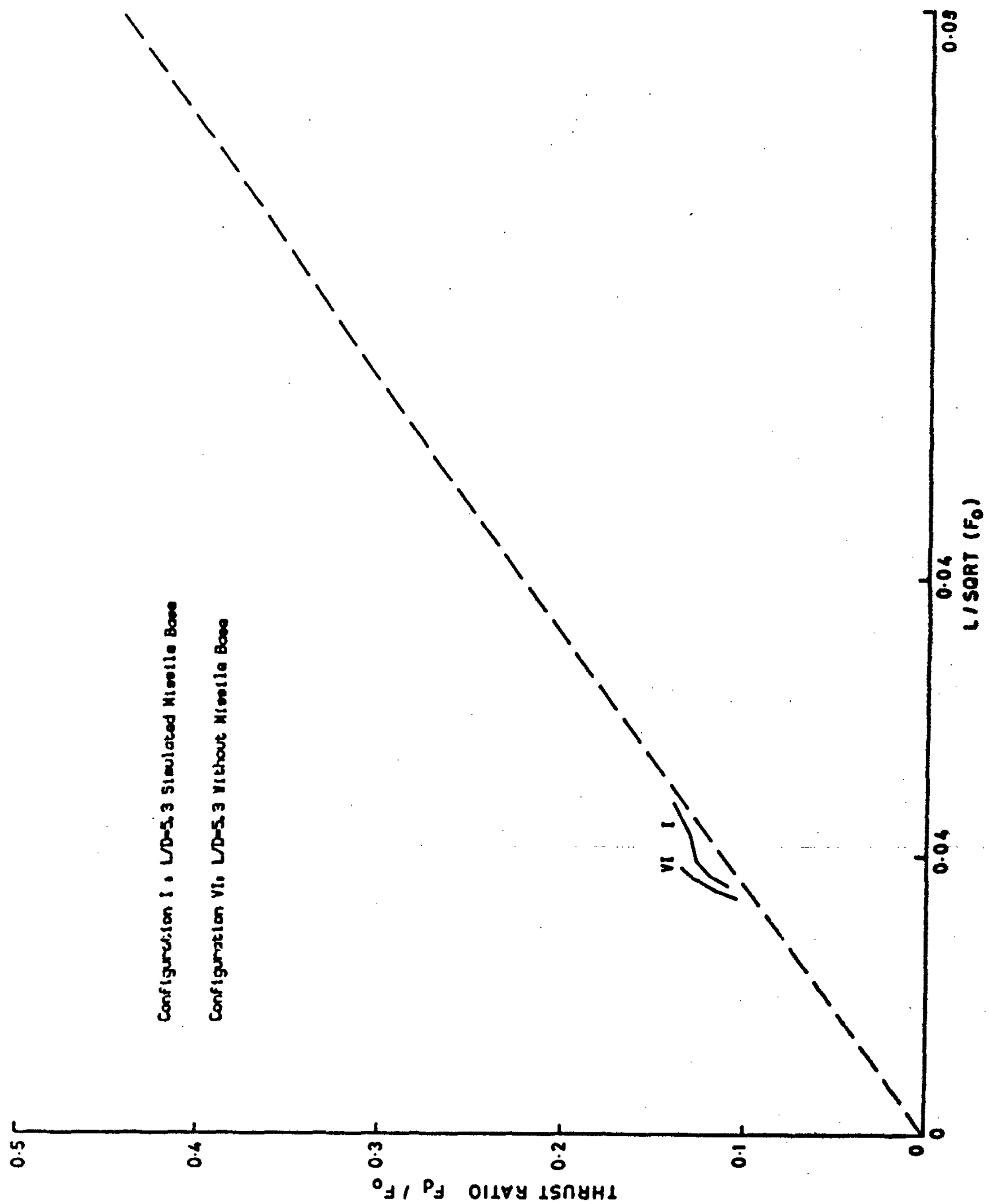

Figure 10. Results of Simonson and Schmeer(ref.1)

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