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THE SUPERSONIC COMPRESSOR RESEARCH AT THE VON KARMAN INSTITUTE FOR FLUID DYNAMICS

Frans A. E. Breugelmans

von Karman Institute for Fluid Dynamics

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THE SUPERSONIC COMPRESSOR RESEARCH

AT THE

VON KARMAN INSTITUTE FOR FLUID DYNAMICS

F.A.E. Breugelmans

contribution presented at the

1st INTERNATIONAL SYMPOSIUM ON AIR BREATHING ENGINES

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13. ABSTRACT

The basic principles of the supersonic cascade and wheel are described. A series of blade profiles and supersonic wheels developed at the VKI are discussed and the progress due to the research effoec is emphasized.

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Ib

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IC

TABLE OF CONTENTS

Id

SUMMARY

The research effort in the field of high speed compressor has paid-off when one compares the progress made during the last decade.

The higher efficiencies will be obtained by improved rotor flow diffusion and an efficient high turning high speed exit guide vane.

These items are analysed and illustrated in the chapter concerning the supersonic cascade and tne high turning stator.

Finally, some published supersonic stage performances are discussed.

LIST OF FIGURES

INTRODUCTION

In the past, the requirements of good efficiency, wide operating range and flexibility for industrial compressors have caused high speed flows to be carefully avoided due to the additional uncertainties introduced into the design process. However, the economic drive toward larger units with increased specific mass flow have made design objectives harder to achieve with conventional design ideas.

The introduction of high preswirl inlet guide vanes can be cited as a method of limiting the inlet Mach number which cannot be applied indefinitely. As tip speeds and mass flows are increased, supersonic pockets and shock waves will be formed on the blade surfaces even though the relative inlet Mach number is less than unity. Moreover, choking or blockage of the blade channel will occur and limit the operating range of the compressor. Thus, in such circumstances, one would be better off by applying a transonic stage.

In general, one can observe that a decade elapses between the first laboratory prototype and the general application of the principle in the aeronautical field. Another decade passes before some of the principles are incorporated in the industrial designs. The special requirements of flexibility, operating range and efficiency will not always permit the application of all high speed principles in a single design.

The progress made in high speed compressor stages can be illustrated by comparing different designs on the basis of mean relative inlet Mach number. Note that this parameter does not take into account the hub-tip ratio, also an essential item in the degree of difficulty.

Figs 1 and 2 summarize what has been achieved, the design objectives and the latest results on a laboratory prototype : all these data points are for shock in rotor stages.

1. THE SUPERSONIC CASCADE

1.1 Principle

The use of a shock wave has been recognized as a possible and important means of flow diffusion or static pressure increase. The consequence will be a shorter and higher performance compressor.

The shock system must be captured between two blades to produce a periodic system in order to arrive at an infinite cascade or a rotor. This last requirement will be discussed in the following paragraphs.

Six different methods available for supersonic flow diffusion are the normal shock, the external compression, the oblique shocks, the supersonic convergent-divergent diffusor, the partial supersonic diffusion and the pipe flow.

These different possibilities are shown in Fig.3.

The normal shock as such is very difficult to obtain in cascade. The condition just before spill-point at a moderate inlet Mach number is the closest one can get. The channels do not produce a normal shock static pressure ratio or total pressure recovery at that Mach number. Forward facing steps forming the throat were incorporated in the design in order to stabilize the normal shock (Fig.3a).

The oblique shock and convergent-divergent supersonic diffusor ideas are applied in the designs of Figs.3b and 3c. The expansion corners inside the channel are designed to compensate for the shock. However, in cases where the leading edge shock does not hit this corner, important separation will occur. The performance is good only for the particular flow condition being demonstrated in the cascade model.

An interesting possibility of supersonic flow diffusion by turning, based on weak compression and expansion waves is shown in Fig.3d. This has been demonstrated in a single channel cascade and very good agreement with the characteristic method was obtained. The outlet flow is supersonic. It should be observed that only a low or moderate static pressure rise is obtained in the design conditions and that the use of a single channel does not produce the unique inlet flow conditions as described later.

A promising type of supersonic diffusion is the contoured spike or Oswatitsch diffusor. An equivalent model is shown in Fig.3e. Ideally, the flow should come in tangent to the suction surface, be decelerated through a weak compression wave system and produce a normal shock just downstream of the throat. Further diffusion follows in the subsonic divergent part. The finite wedge angle and the slope of the compression surface will limit the lowest Mach number at which this principle will work with correct inlet flow conditions and incidences achieved.

A long and almost constant area channel is another method for supersonic flow diffusion. At low Mach numbers a normal shock is formed in the blade passage. Increasing the Mach number complicates the shock pattern and a multiple shock system, called a "pseudo-shock", is formed. It has, according to the literature, the equivalent performance of a normal shock. Studies done on flow in pipes show that there is a minimum starting length for the shock system and a minimum "pseudo-shock" length for each Mach number. The principle is therefore valid in high solidity, high hub-tip ratio design.

In order to achieve an almost constant area passage in a cascade model, one has to increase the blade thickness as the flow turning progresses. The result is a blunt trailing edge blade as shown in Fig.3f.

1.2 The inlet flow field

At first, it was thought that the supersonic inlet conditions would simplify cascade testing and that a single blade channel would suffice to simulate the infinite cascade.

The periodic inlet flow conditions requires that a well defined inlet flow direction be established. This inlet flow direction depends upon the blade configuration and the inlet Mach number.

This has been shown by Levine and Ferri in Refs. 34 and 35, and recently reemphasized in Ref.33 by Starken and Lichtfuss.

From periodicity considerations, the existence of an unique incidence can be shown for supersonic flow. A shock wave system with finite strength at infinity upstream leads to infinite loss when the flow approaches the supersonic cascade leading edge plane. The shocks are quickly attenuated and approach the Mach line determining the upstream infinity conditions, and finite loss occurs in reality.

This M_∞ line, providing us with the relative flow conditions for a supersonic rotor section, can be computed by the Levine method. The method assumes isentropic flow and applies the continuity and Prandtl-Meyer relation in a given cascade model. For an assumed limiting Mach line, one can compute the M_∞ line by solving through an iteration procedure the relation existing between M_∞, β_{1ss} (suction surface angle), $\frac{S}{R}$ (pitch to suction surface radius of curvature) and θ_E (geometrical angle between tangent at the leading edge and that point in suction surface where the limiting M line originates.

That portion of the suction surface from which the emanating waves or shocks pass the leading edge of the next blade determines the inlet flow field.

The Mach line, which does not intercept the leading edge shock wave of neighbouring blades, determines the inlet flow direction and inlet Mach number, i.e., β_{1_∞} and M_{R_1}.

The periodic flow solution for a flat suction surface will have the incoming flow tangent to the suction surface.

The convex DCA blade surface will have its determining Mach line emanating from a point C, between the leading edge and the limiting Mach line.

An increase of the Mach number will decrease the entrance region and finally a sonic axial component will be reached. The inlet flow direction will vary towards higher β_{1_∞} for DCA blades and constant β_{1_∞} for flat suction surfaces.

The inlet relative flow direction can change if one can influence the shock system. At spill point, for example, a small reduction in mass flow occurs and the shock detaches from the leading edge. This condition is so close to an unstarted supersonic flow that actual measurements in cascade are very hard to perform.

The existence of a unique inlet flow direction simplifies the cascade testing. It is thus concluded that the blade determines the inlet flow conditions.

The constant β_{1_∞} is also the reason for the vertical characteristic of a supersonic compressor and the eventual matching problems between blade rows and stages.

In a finite cascade, one always has a first blade which will not fulfil the periodic onditions and the flow will be deflected through a first shock wave. Thereafter, the adaptation to the repeatable wave pattern can take place.

1.3 The back pressure

The cascade model, as described, will not produce a static pressure increase. The shock wave system has to be forced to produce a static pressure rise by applying a back pressure.

A different shock configuration will be obtained in the cases of low and high back pressures.

The shock wave system in a supersonic cascade is divided into two parts. They are the channel shock, main contributor in the loss and static pressure rise, and the inlet shock degenerating into the M_∞ line under influence of the expansion waves.

Shock wave shape and blade pressure distributions are easily predicted at no back pressure, using the oblique shock relations and the characteristic method. Good agreement with experiment is obtained. Some conditions will result in a turbine operation of the cascade, i.e., $M_2 > M_1$.

At medium and high back pressure positions, the outlet flow becomes sonic or subsonic. The shock strength is increased and serious shock wave - boundary layer interaction appears. A careful prediction of the flow and cascade performance becomes very difficult due to the strong dependency of the shock and flow model on blade shape, cascade geometry and back pressure.

An example of shock pattern-back pressure influence is shown in Fig.4 which is deduced from schlieren observations.

The oblique shocks of the channel are gradually transformed to a normal shock as they move upstream. The straight entrance simplifies the problem as in principle the flow comes in parallel to the suction surface and the shocks which are generated are then only due to the finite thickness of the leading edge. At high back pressures, the shock pattern can be pushed out of the channel and a bow-shock is formed in front of the leading edge. A stagnation streamline shift occurs and the expansion around the nose curves the shock and attenuates it until the M_∞ line is reached. At this condition a small change in axial component will be possible.

The inlet shock is always curved in the case of convex or concave suction surface under the influence of the expansion fan on both sides of the M line which is the asymptote for the shock wave.

1.4 The losses

The overall loss in a high speed cascade can be divided into

1. Shock losses from the channel shock wave and from the bow wave;
2. Profile friction losses;
3. Shock wave induced separation of the boundary layer;
4. Mixing.

These four items are strongly dependent on the static pressure rise through the cascade model and therefore very difficult to separate from the global loss.

The shock wave boundary layer interaction has been studied on simple models and certain rules have been derived with respect to the normal shock static pressure increase and the shock induced separation of the boundary layer. The Pearcey criterion relating the static pressures has not been achieved in cascades, as derived from the blade pressure measurements (see Refs.22-26).

The Pearcy criterion for the shock intensity that produces separation is not valid in cascades where higher pressure gradients exist than on an isolated airfoil. The limit downstream pressure is smaller in cascade than for an isolated airfoil with an identical curvature. The downstream displacement and momentum thicknessess are discussed in Ref.22.

Most of the effort in loss evaluation for supersonic cascades is concentrated on the shock loss evaluation. The original NACA-model presented by Hartmann has proved to be a very useful tool and has the advantage of its simplicity.

The shock wave is divided into two parts by the stagnation streamline : the passage shock and the bow shock. The passage shock has a variable inlet Mach number in case of a non-straight entrance. Therefore an average Mach number has been defined for which the equivalent normal shock losses are computed. The inlet Mach number, called M_∞ in the previous paragraph, is not the one we will find at the blade leading edge. A slightly higher Mach number, according to the inlet flow calculation, will be used together with the suction surface M_{SS} in order to evaluate the average inlet Mach number.

The bow shock losses can be evaluated by integrating local loss coefficient along the shock wave.

The Moeckel theory provides us with the asymptote, the shock location on the stagnation streamline, with respect to the origin of the asymptote, and the slope of the shock wave at each point. This enables the computation oı the total pressure recovery along the shock.

The application of this theory shows that a very fast attenuation of the shock occurs.

The importance of suction surface curvature and solidity, which restricts the total Prandtl-Meyer expansion imposed upon the flow in the blade passage, is illustrated in Fig.5. The normal shock total pressure loss coefficient is presented as a function of the suction surface Mach number M_{SS} according to the model described above.

For a given inlet Mach number M_1 the equivalent normal shock losses increase for higher M_{SS}. Increasing M_{SS} values can be obtained by higher camber angles or lower solidities for a given convex blade cascade. The obvious conclusion from Fig.5 is that one has to decamber the entrance region in order to avoid high shock losses.

Further, one arrives at smaller shock intensities at the suction surface and therefore reduces the interaction with the boundary layer.

1.5 The cascade characteristic

As has been discussed, the incidence variation is of no consideration in high speed cascade testing. The back pressure is the additional parameter.

The inlet Mach number does not change up to the moment of arrival at the spill point when shock expulsion occurs.

Fig.6 represents some performance characteristics of a high speed cascade. The static pressure ratio and outlet Mach number are presented as a function of the inlet Mach number; the total loss coefficient or recovery ratio is given as a function of the back pressure.

The actual loss data of Fig.7 illustrates the wide variety of results. There are relatively few cascade models which achieve the normal shock loss values for sonic or subsonic exit Mach numbers.

2. THE HIGH TURNING CASCADE

The second problem of a supersonic compressor stage dealt with the high subsonic high swirling exit flow of the rotor.

2.1 The stator problem

In order to illustrate the stator problem, an example of the rotor exit flow conditions has been chosen. These data are obtained by a radial equilibrium calculation.

The supersonic stage has a 0.77 inlet hub-tip ratio and M_{1_R} = 1.55. The rotor and stator are the VKI designs FB-IV. Design total pressure ratio is 2.2/1. (Fig.8).

The very high pressure ratio designs for shock in rotor models require flow angles of the order of 50° to 65° from axial and absolute flow Mach numbers between 0.70 and 1.05.

Here the problem of the maximum inlet Mach number at choking conditions arises.

2.2 The operating range

The maximum achievable inlet Mach number depends upon the blade shape, the cascade geometry and the inlet flow conditions.

The operating range of a blade section decreases drastically with inlet Mach number (Fig.10). An obvious conclusion is that an incorrect blade choice can make the design objectives unattainable and an incorrect blade staggering has the same effect.

This requires an extreme precision in the work distribution (exact evaluation of incidence and deviation angles, axial velocity ratio), an extreme precision in the entropy or loss gradient term of the radial equilibrium equation, a good evaluation of the curvature terms and the blockage effects throughout the compressor stage.

The profiles with a rounded nose have an important range of operation for low Mach numbers, while sharp leading edge are rather indicated for higher speeds.

Some high turning, high subsonic cascades were investigated at VKI and the performances are presented in Fig.11.

The upper diagram combines loss and turning in a single parameter $\frac{\omega}{\theta}$ or percent total pressure losses per degree turning. The average flow turning of the different cascades is around 47° to 55°.

3. THE SUPERSONIC STAGE

Supersonic compressor research at VKI has been sponsored in part by the Aerospace Research Laboratories (ARL - WPAFB), Dayton, Ohio through different grants and contracts over the years 1963-1972. The author gratefully acknowledges the support of Mr. E. Johnson (ARL-WPAFB).

3.1 Description

The geometrical characteristics and some of the design parameters of the supersonic stages discussed herein are presented in Table I. Full details of these transonic and supersonic stages can be found in the references.

Numbers 1, 2, 3 : Fig.9
These supersonic wheels were designed and investigated at VKI under a USAF grant and contract. The model was intended to investigate the pipe flow diffusion in a wheel. The different configurations were designed in order to control the exit flow, work distribution and trailing edge thickness effect. Nr 2 has been combined with a stator, Fig.12.

Number 4 :
This is the DCA model of a series of four MCA wheels designed and tested by G.E. under NASA contract. The designs were made for the investigation of the influence of blade shape in the supersonic portion of the flow field in a transonic compressor.

Number 5 : fig.13
This is a low hub-tip ratio VKI design under USAF contract. The intention of this prototype is to explore the high Mach number, high mass flow and high pressure ratio combination in a single supersonic stage.

Number 6

A VKI design for small mass flow.

Number 7

This design by ARL-WPAFB is a medium hub-tip ratio, high pressure ratio stage. Limit blade loading is used in the wheel and exit guide vanes.

Number 8

This model represents the Curtiss Wright small mass flow supersonic compressor stage with inlet guide vanes.

Number 9

This number is for t . NASA rotor 5 used for casing treatment studies.

These few examples of published designs cover the period of 1965 to 1972. They were chosen in order to illustrate the research effort in high-speed compressor stages for small gas turbines and in front stage with very high mass flow capability and pressure ratio of the future.

3.2 Performance of a supersonic stage

The performance of a stage will be discussed by comparing different wheels and illustrating the high turning stator problem.

The overall performance of the transonic and supersonic wheels, described in Table I, are shown in Figs.14-15. The rotor tip relative inlet Mach number has been chosen as the parameter for the comparison of the total pressure ratio (Fig.14). The low hub-tip ratio designs arrive at the same average total pressure ratio but at much higher tip speeds. An increased discrepancy between tip and hub values will be the result of such designs.

The adiabatic efficiency (Fig.15) shows very promising prospects for the high speed compressor, providing one can develop an efficient exit guide vane.

A comparison of the data on R 32-I and II indicated an improved flow at the blade tip, but a stalled hub section for R 32-II. Compare the M_2 - α_2 distribution in Fig.16. A series of holes, connecteu to the cavity formed by the inner stator casing improved the exit total pressure profile as was found in 1967 (Fig.17).

The design speed data on the G.E.-2D and NASA-5 rotors illustrate the variable total pressure ratio result with a well marked influence of the part span shroud. Rotor exit Mach numbers of 0.70 to 0.80 and swirl angles of 45° to 50° are observed.

The problem of high turning blades with small aspect ratio crops up in small, high-performance compressors. The idea of fighting the secondary flow by partial slots has been developed at VKI and applied on the stage FB-I of Fig.12. The results of these investigations are presented in Fig.18. The remarkable improvement for the slotted configuration is illustrated.

Other investigators did not observe an equivalent improvement.

The performance of the Curtiss Wright small mass flow supersonic stages is also presented in this figure. An adiabatic efficiency of close to 80% can be expected for stage total pressure ratios of 2,0/1.

Other performance data on supersonic stages are given in Figs.1 and 2 and prove the progressmade.

CONCLUSION

The supersonic stage has improved to a competitive level for some applications.

High pressure ratios (close to 3,0/1) and adiabatic efficiencies (close to 80%) may be expected for a low hub-tip ratio design.

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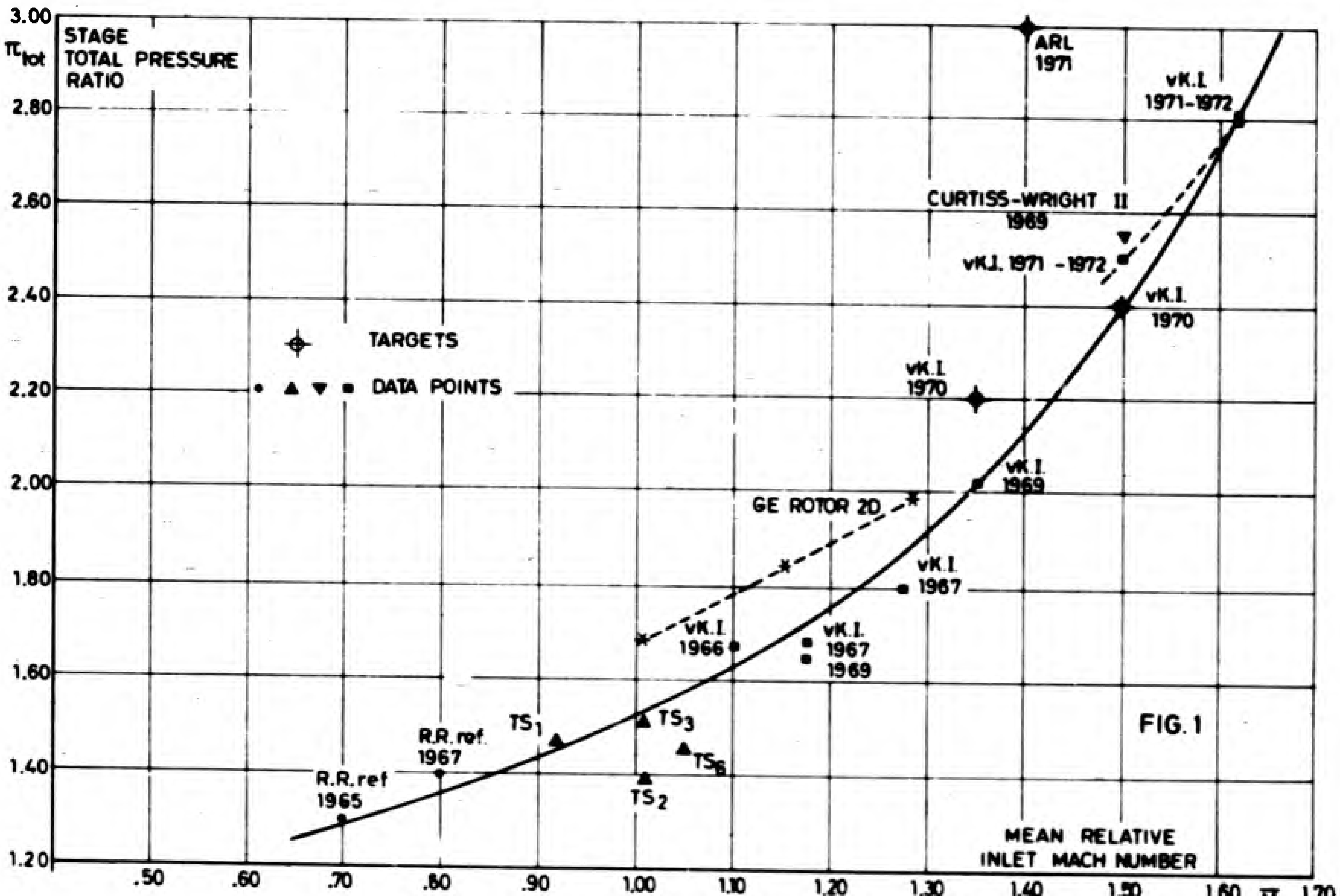
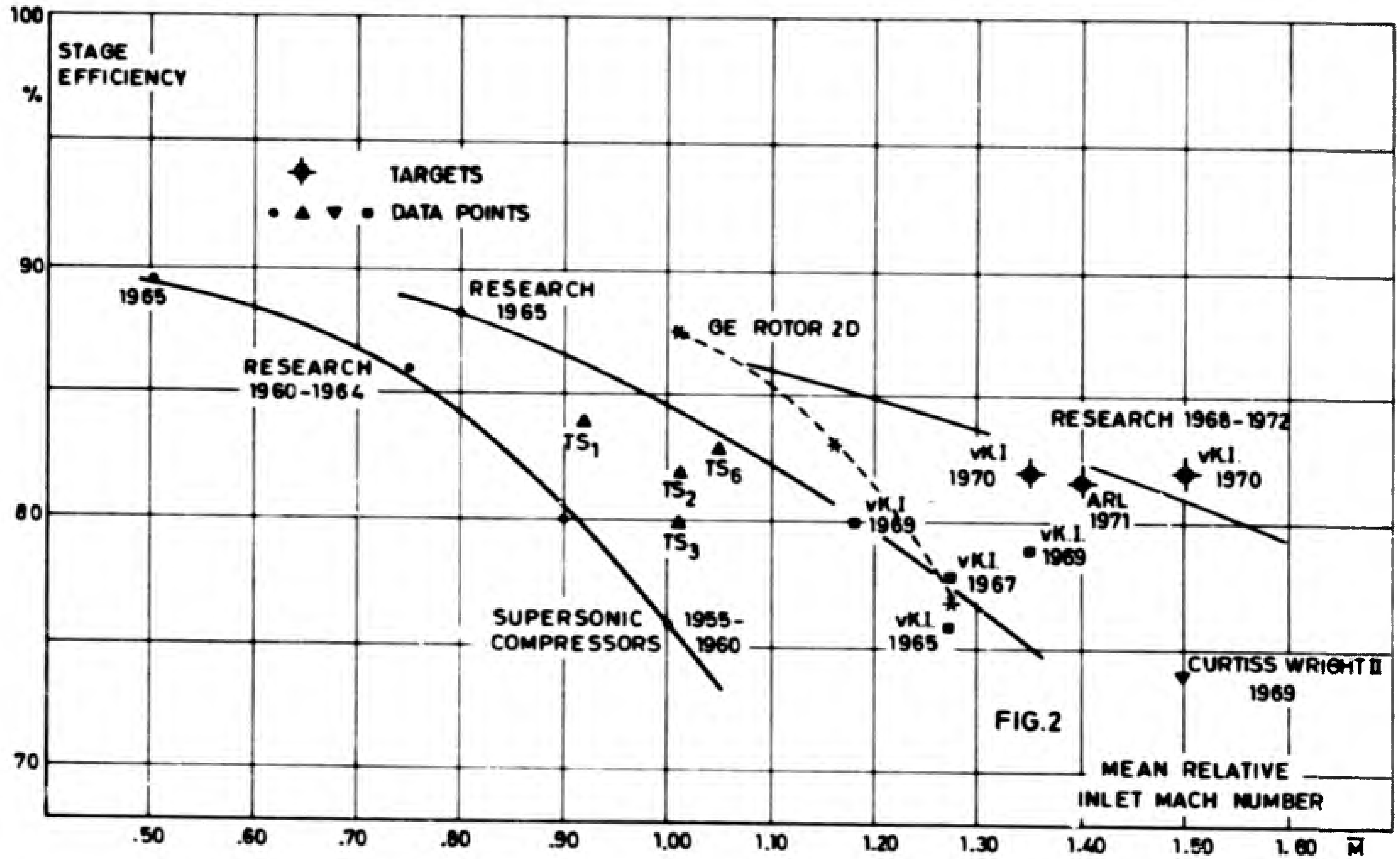

3.00
π_{tot}
STAGE
TOTAL PRESSURE
RATIO
2.80
2.60
2.40
2.20
2.00
1.80
1.60
1.40
1.20
.50
.60
.70
.80
.90
1.00
1.10
1.20
1.30
1.40
1.50
1.60
$\overline{M}$
1.70
TARGETS
DATA POINTS
ARL
1971
vK.I.
1971-1972
CURTISS-WRIGHT II
1969
vK.I. 1971 -1972
vK.I.
1970
vK.I.
1970
vK.I.
1969
GE ROTOR 2D
vK.I.
1967
vK.I.
1966
vK.I.
1967
1969
TS1
TS3
TS6
TS2
R.R.ref.
1967
R.R.ref
1965
MEAN RELATIVE
INLET MACH NUMBER

FIG. 1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90
80
70
TARGETS
DATA POINTS
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RESEARCH
1960-1964
RESEARCH
1965
GE ROTOR 2D
RESEARCH 1968-1972
TS_1
TS_2
TS_3
TS_6
SUPERSONIC
COMPRESSORS
1955-
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vK.I.
1969
vK.I.
1967
vK.I.
1965
vK.I.
1970
vK.I.
1969
ARL
1971
vK.I.
1970
CURTISS WRIGHT II
1969
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1.50
1.60
$\bar{M}$

FIG.2

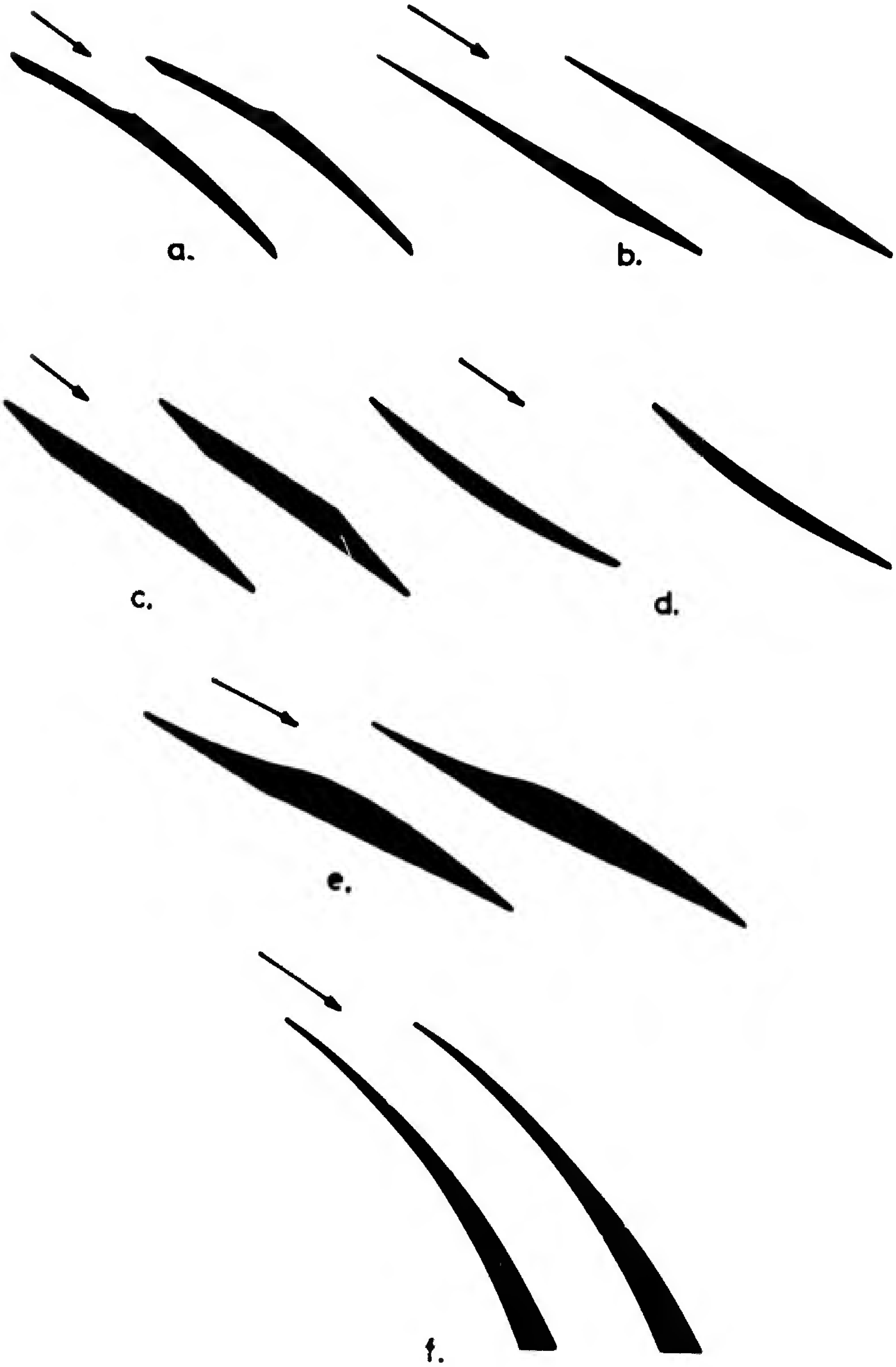


FIG.3

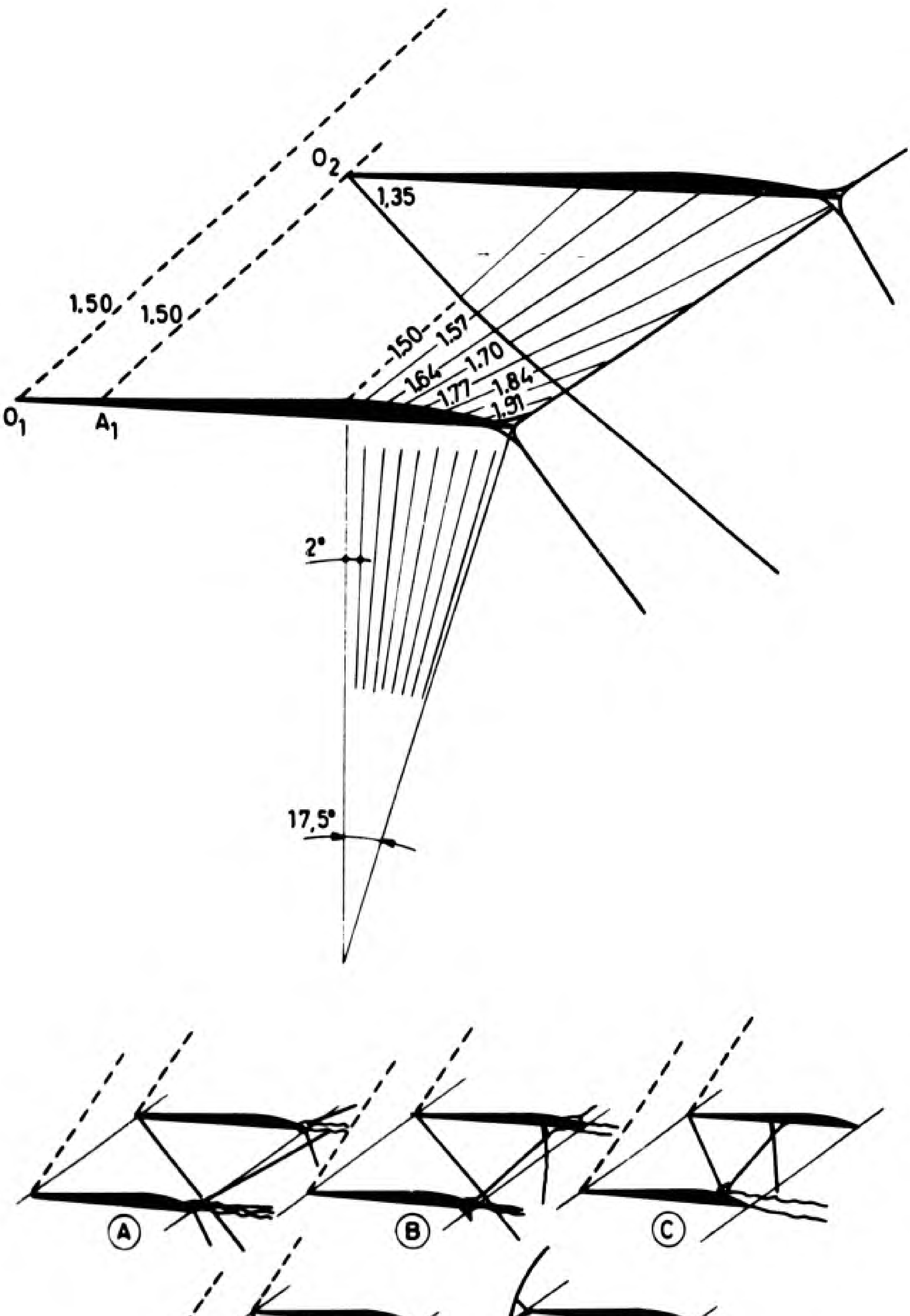
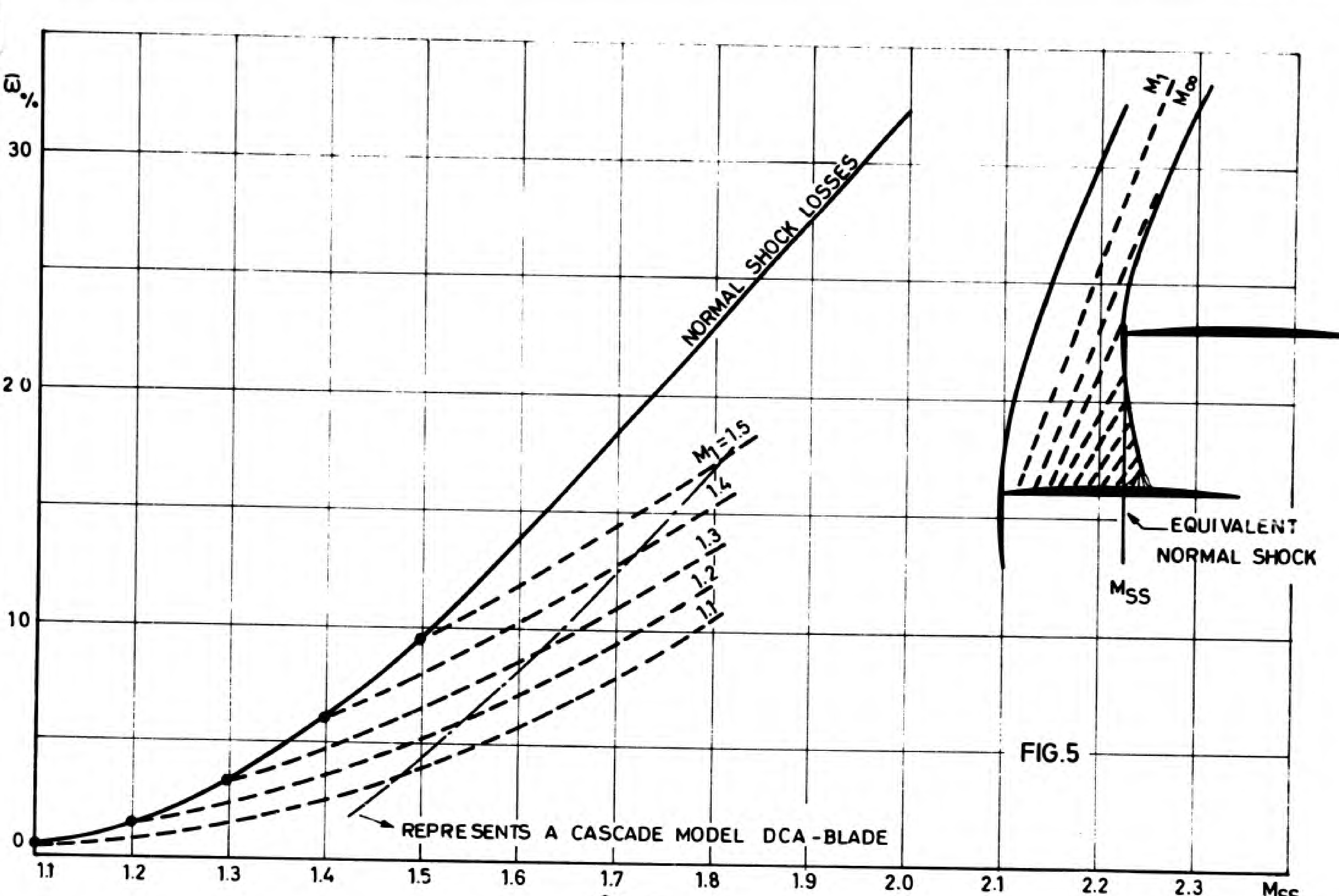
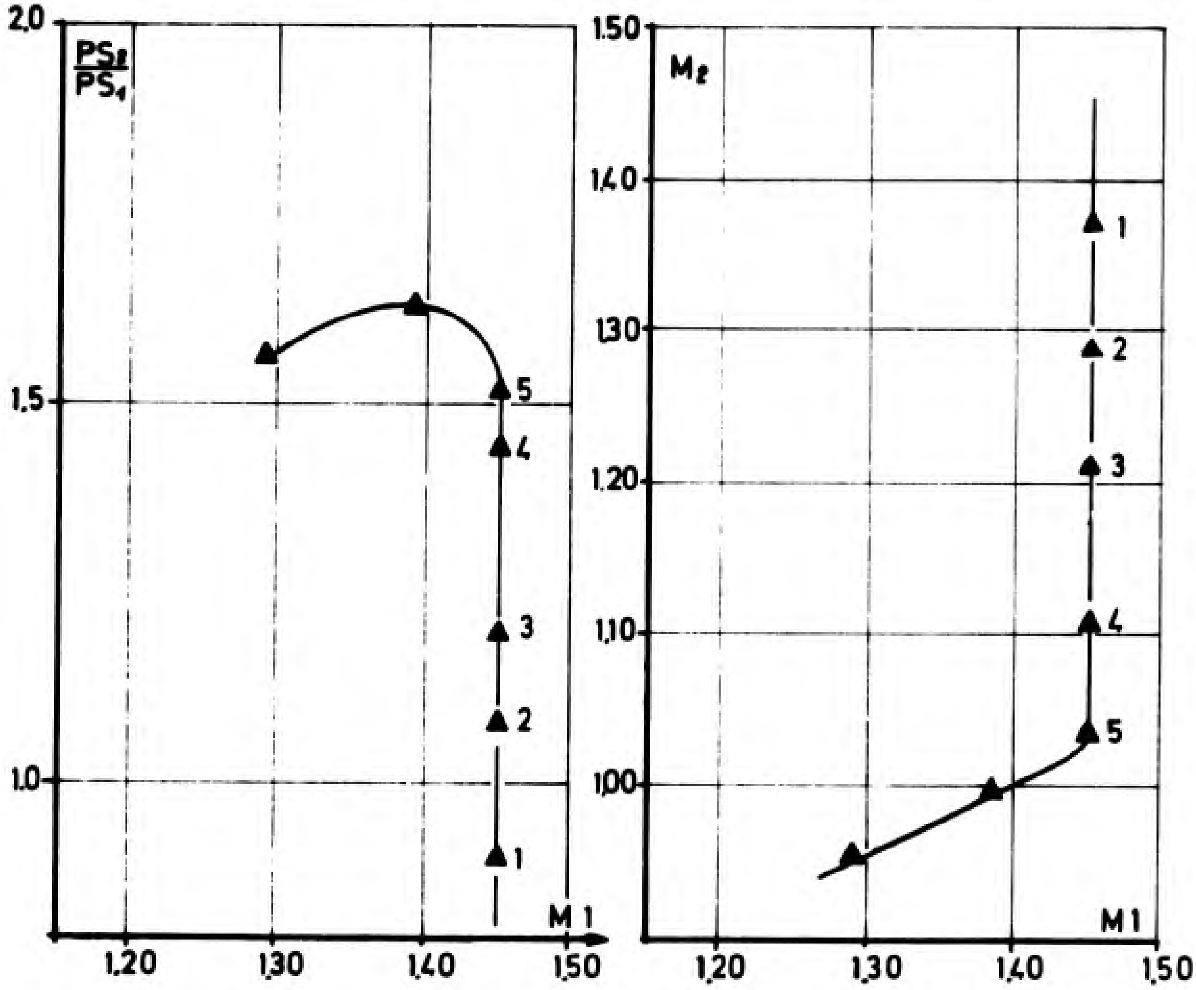


FIG.4

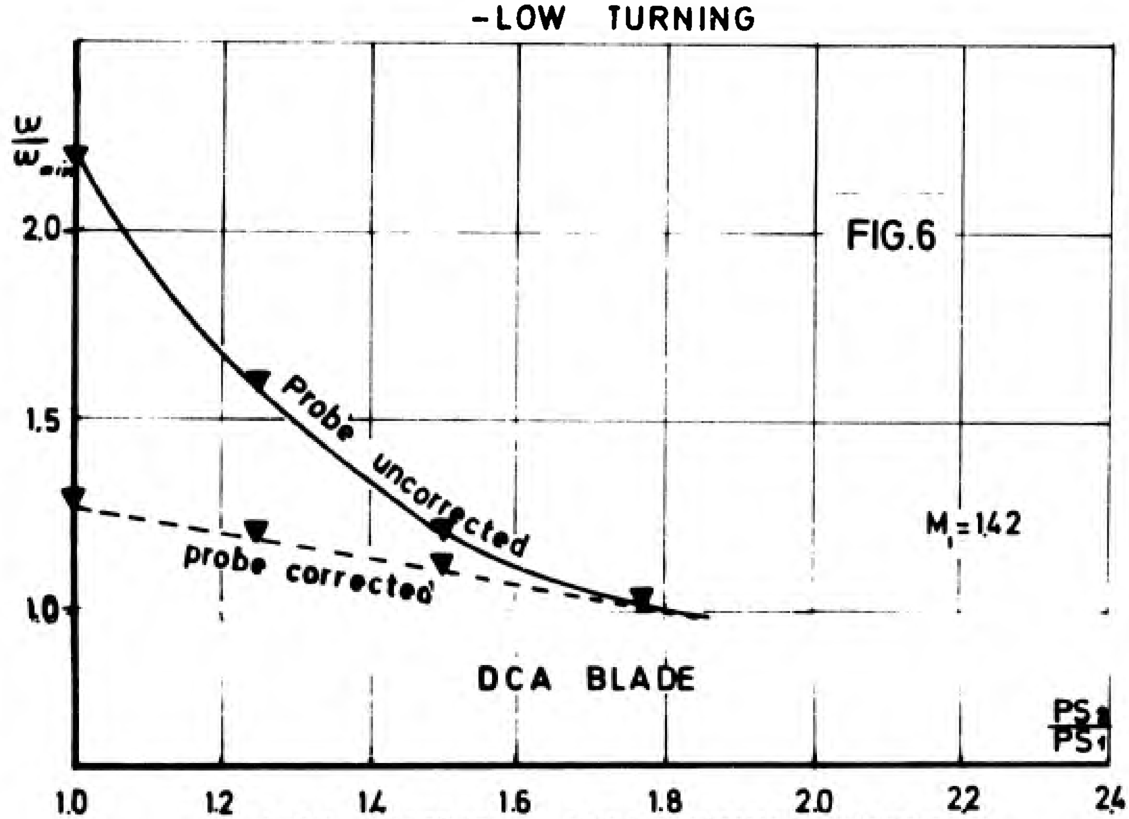


$\bar{\omega}$ %
30
20
10
0
NORMAL SHOCK LOSSES
$M_1 = 1.5$
1.4
1.3
1.2
1.1
REPRESENTS A CASCADE MODEL DCA-BLADE
M_1
M_∞
EQUIVALENT NORMAL SHOCK
M_{SS}
1.1
1.2
1.3
1.4
1.5
1.6
1.7
1.8
1.9
2.0
2.1
2.2
2.3
M_{SS}

FIG.5



INFLUENCE OF $\bar{PS}_2/PS_1$ ON $\bar{M}_1$ AND $\bar{M}_2$ FOR A TYPICAL - LOW SOLIDITY CASCADE - LOW TURNING



LOW SOLIDITY - LOW TURNING CASCADE

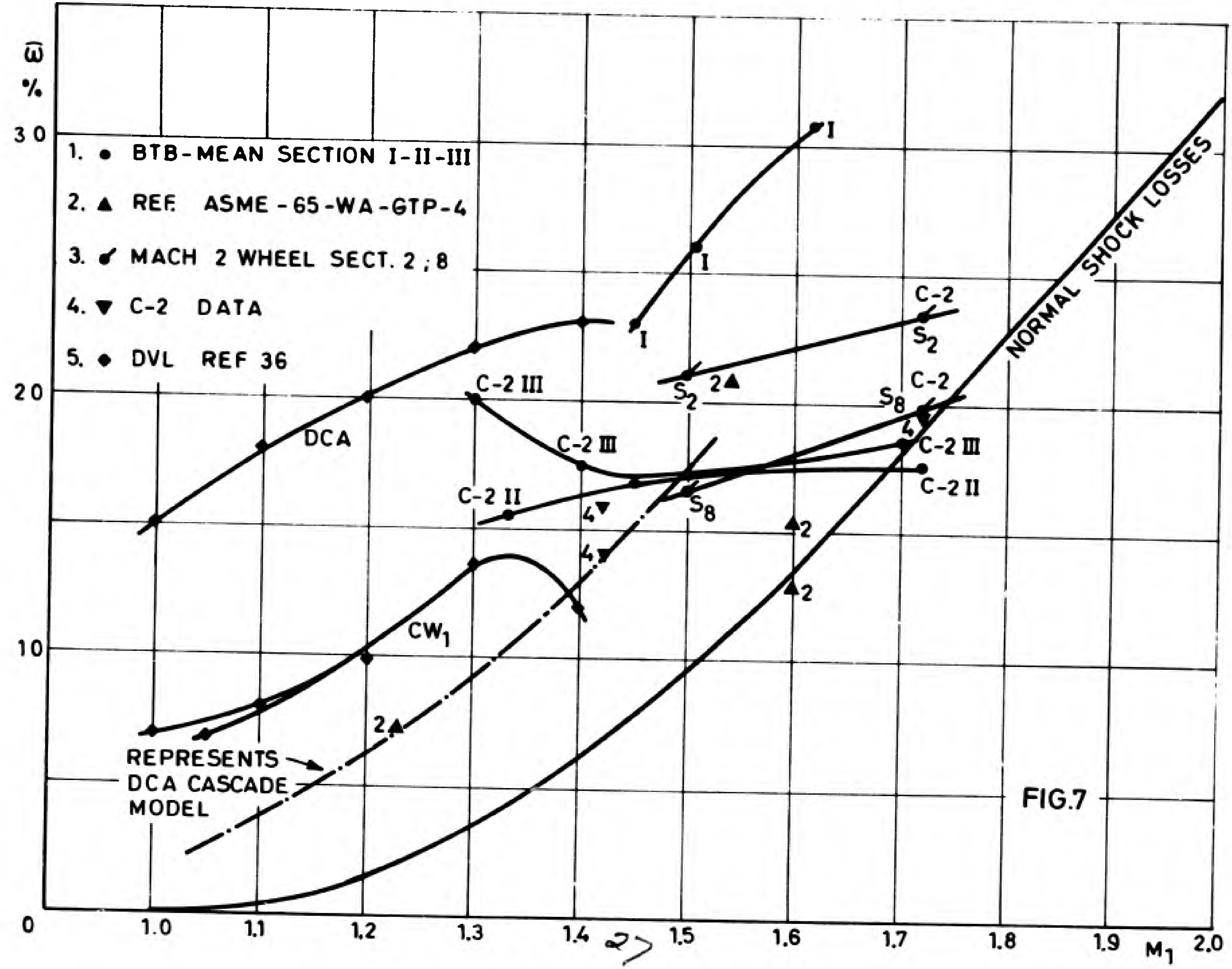
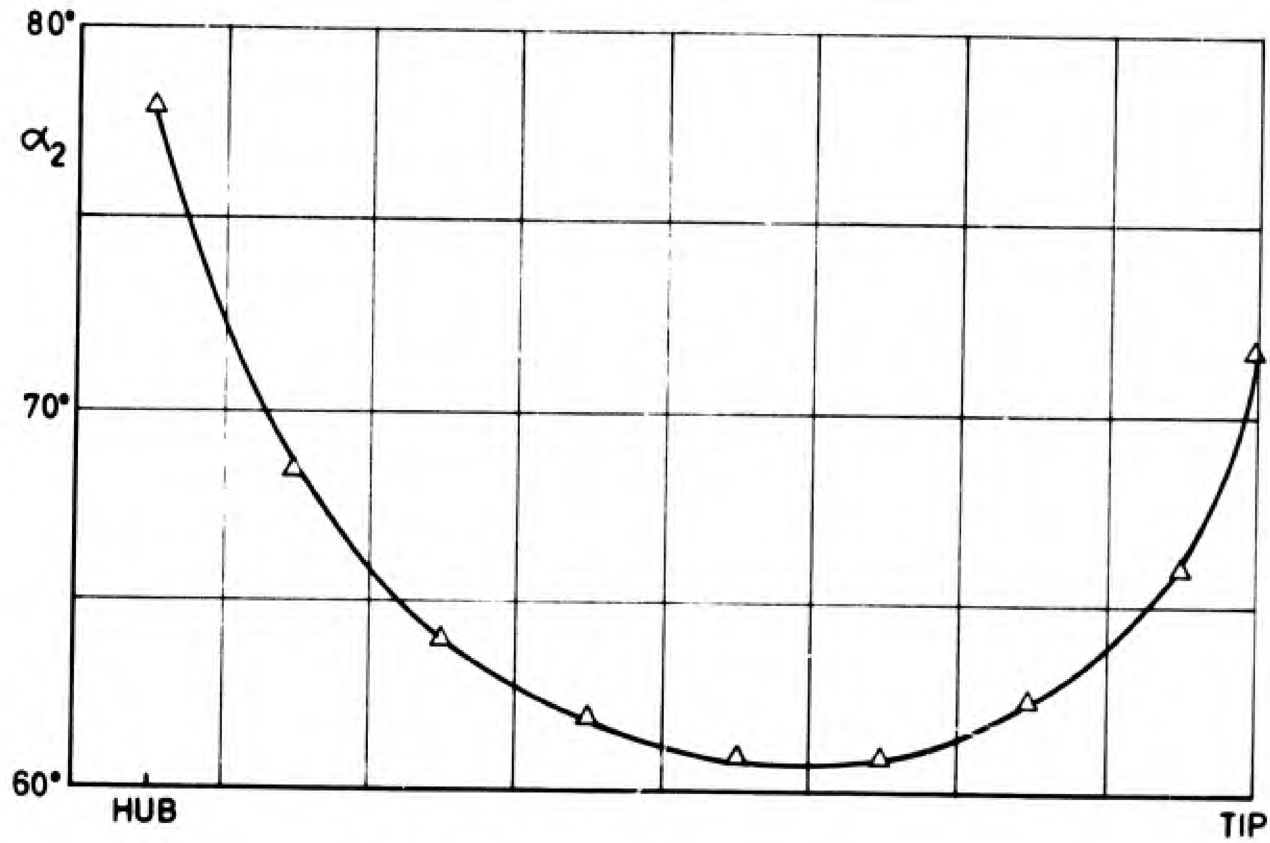

$\bar{\omega}$ %
30
20
10
0
1.0
1.1
1.2
1.3
1.4
1.5
1.6
1.7
1.8
1.9
2.0
M_1
1. ● BTB-MEAN SECTION I-II-III
2. ▲ REF. ASME-65-WA-GTP-4
3. MACH 2 WHEEL SECT. 2 ; 8
4. ▼ C-2 DATA
5. ◆ DVL REF 36
NORMAL SHOCK LOSSES
DCA
CW_1
C-2 III
C-2 II
C-2
S_2
S_8
REPRESENTS DCA CASCADE MODEL

FIG.7

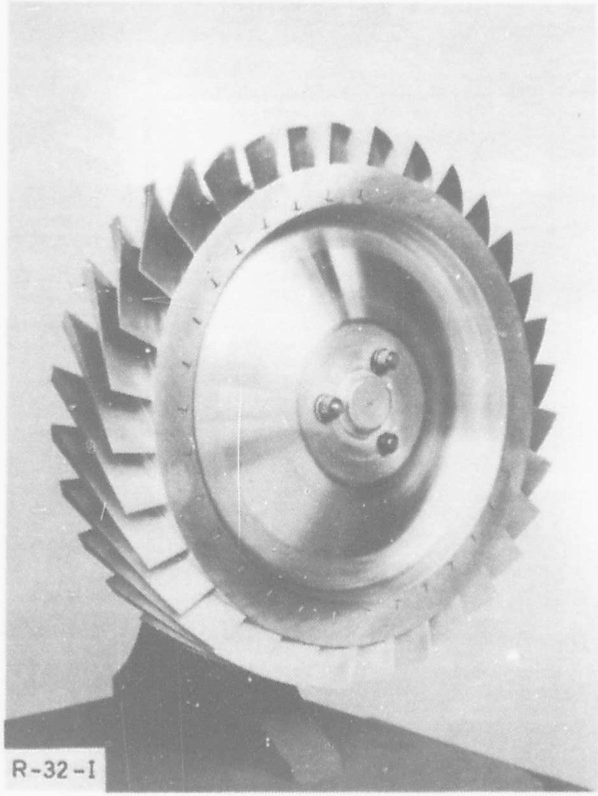
ROTOR vK.I.-F.B-IV

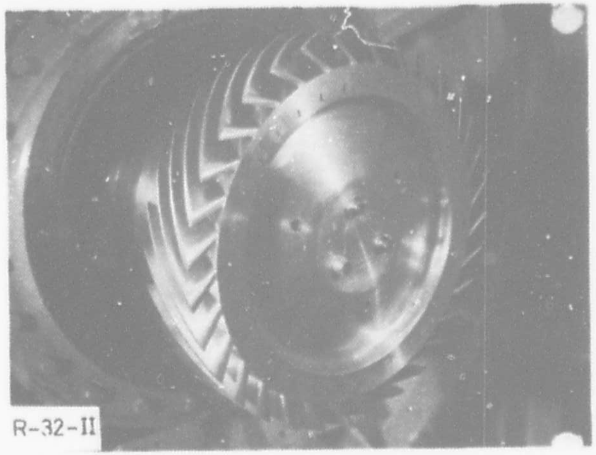


DESIGN DATA AT STATOR INLET.

FIG. 8 AXIAL COMPRESSOR STAGE $\pi_{tot} = 2.2/1$

H/T=0.77





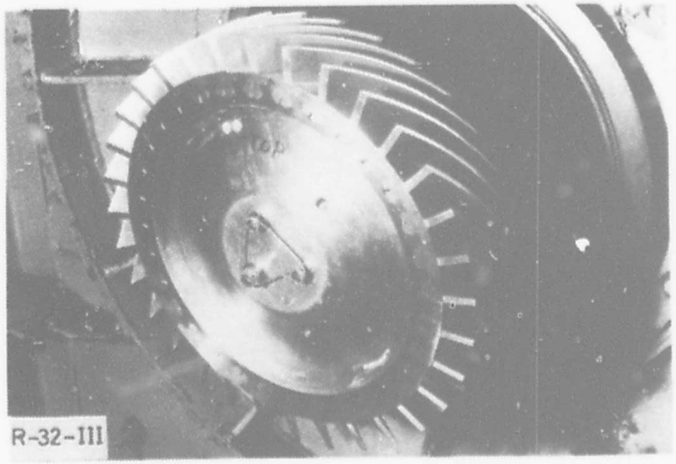
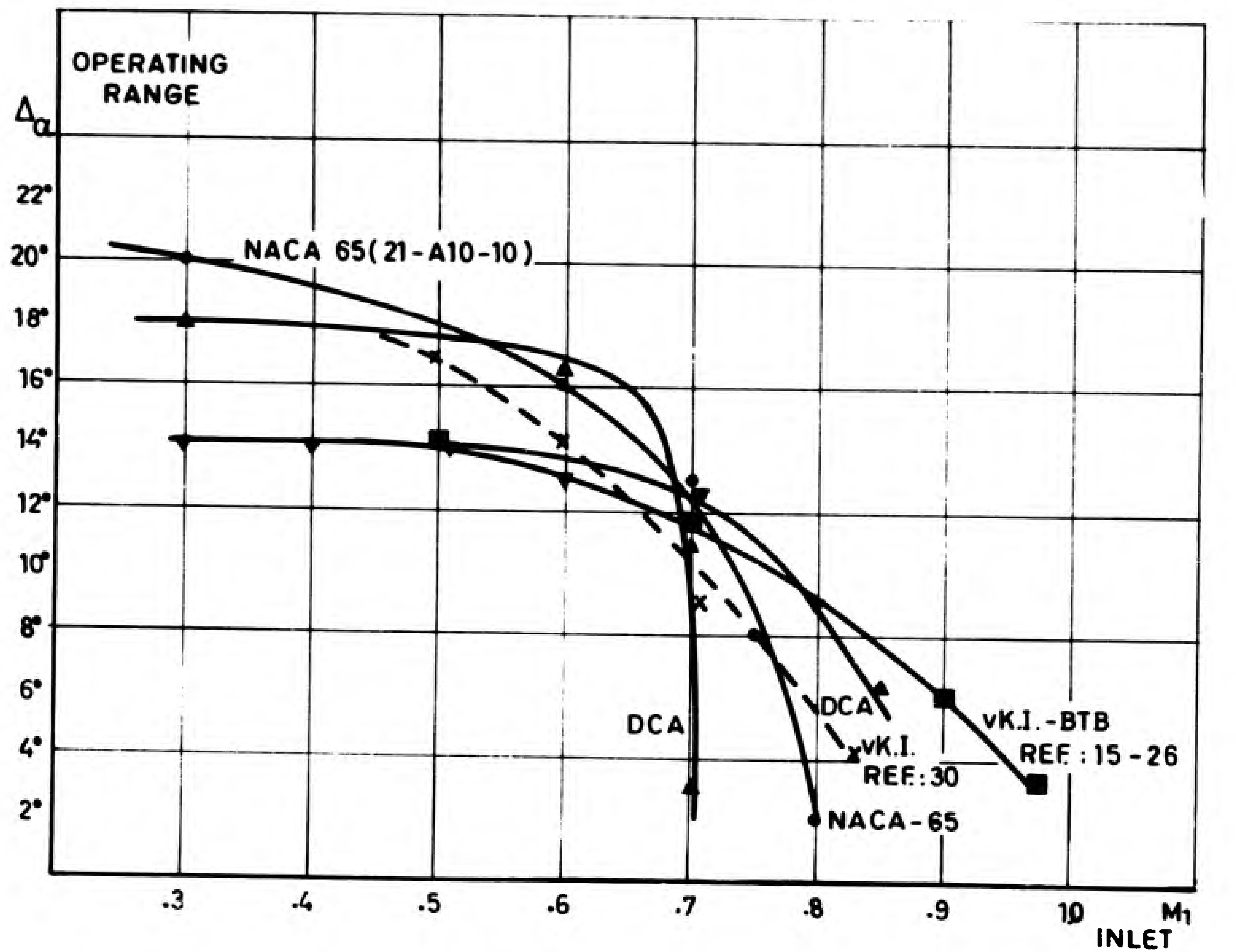
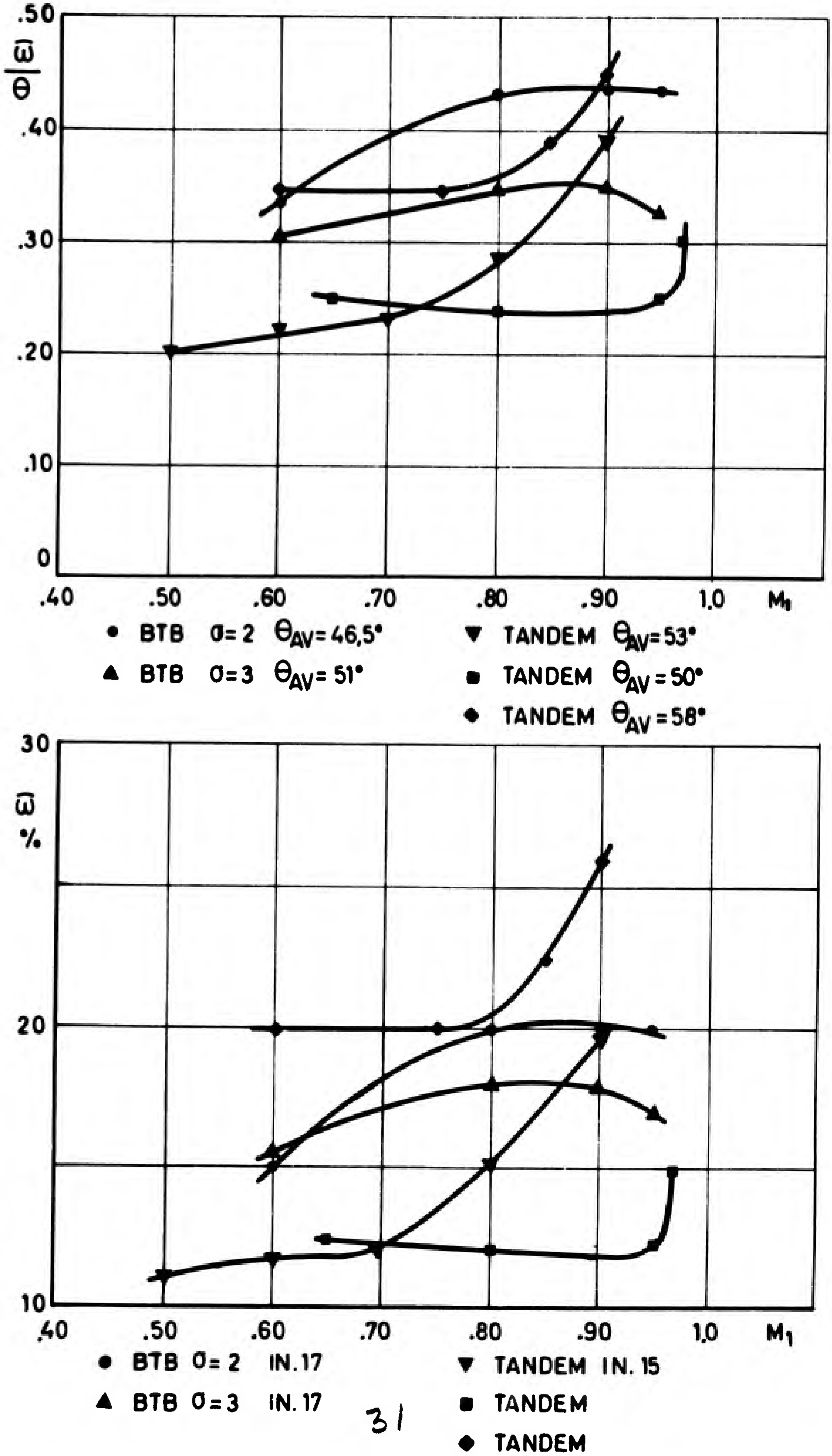


FIG.9



OPERATING
RANGE
Δ_α
22°
20°
18°
16°
14°
12°
10°
8°
6°
4°
2°
NACA 65(21-A10-10)
DCA
DCA
vK.I.
REF.:30
vK.I.-BTB
REF.:15-26
NACA-65
.3
.4
.5
.6
.7
.8
.9
1.0
M_1
INLET
MACH NUMBER

FIG.10



$\overline{\omega}/\theta$
.50
.40
.30
.20
.10
0
.40 .50 .60 .70 .80 .90 1.0 M_1
● BTB $\sigma = 2$ $\theta_{AV} = 46{,}5°$
▲ BTB $\sigma = 3$ $\theta_{AV} = 51°$
▼ TANDEM $\theta_{AV} = 53°$
■ TANDEM $\theta_{AV} = 50°$
◆ TANDEM $\theta_{AV} = 58°$
$\overline{\omega}$ %
30
20
10
.40 .50 .60 .70 .80 .90 1.0 M_1
● BTB $\sigma = 2$ IN. 17
▲ BTB $\sigma = 3$ IN. 17
▼ TANDEM IN. 15
■ TANDEM
◆ TANDEM

FIG.11

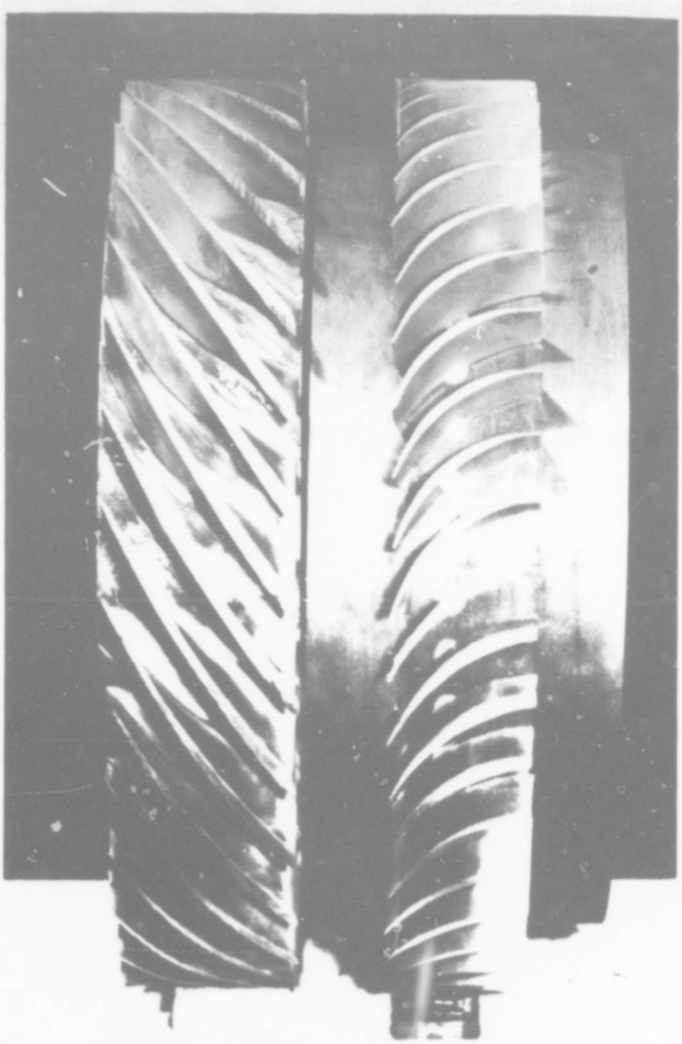
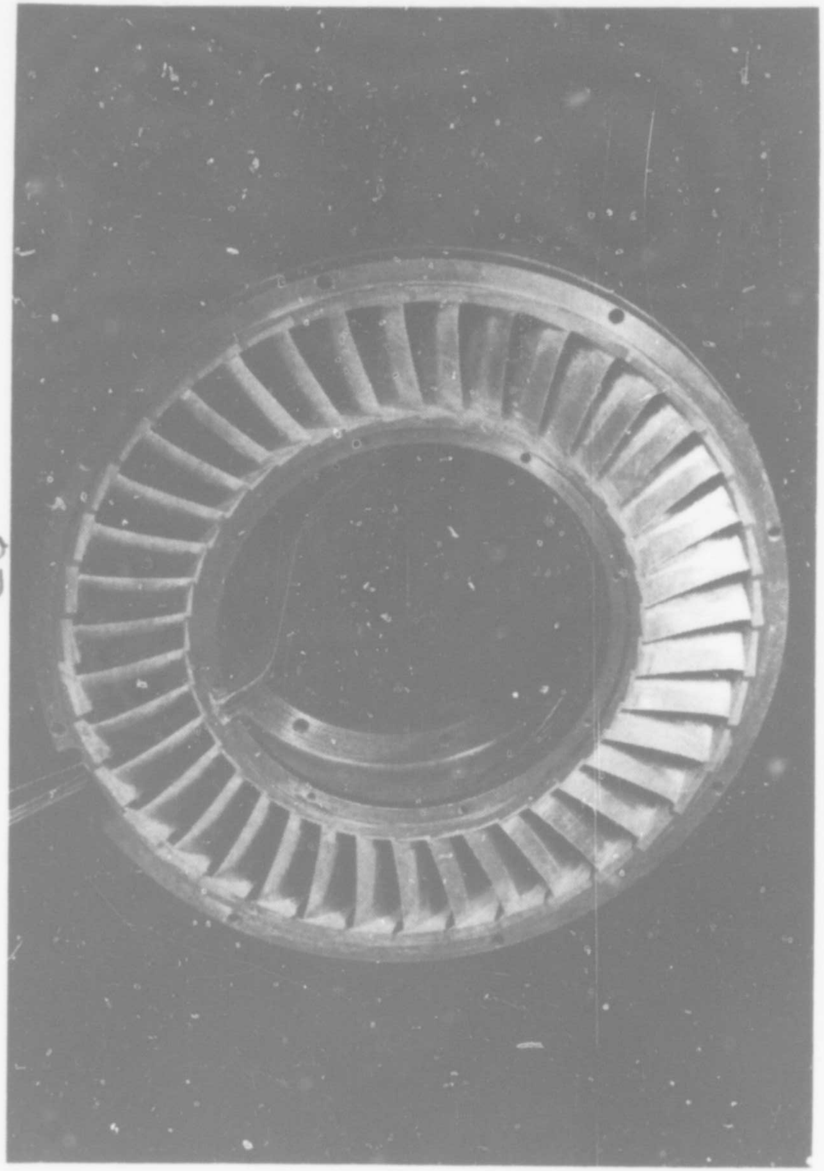


FIG.12



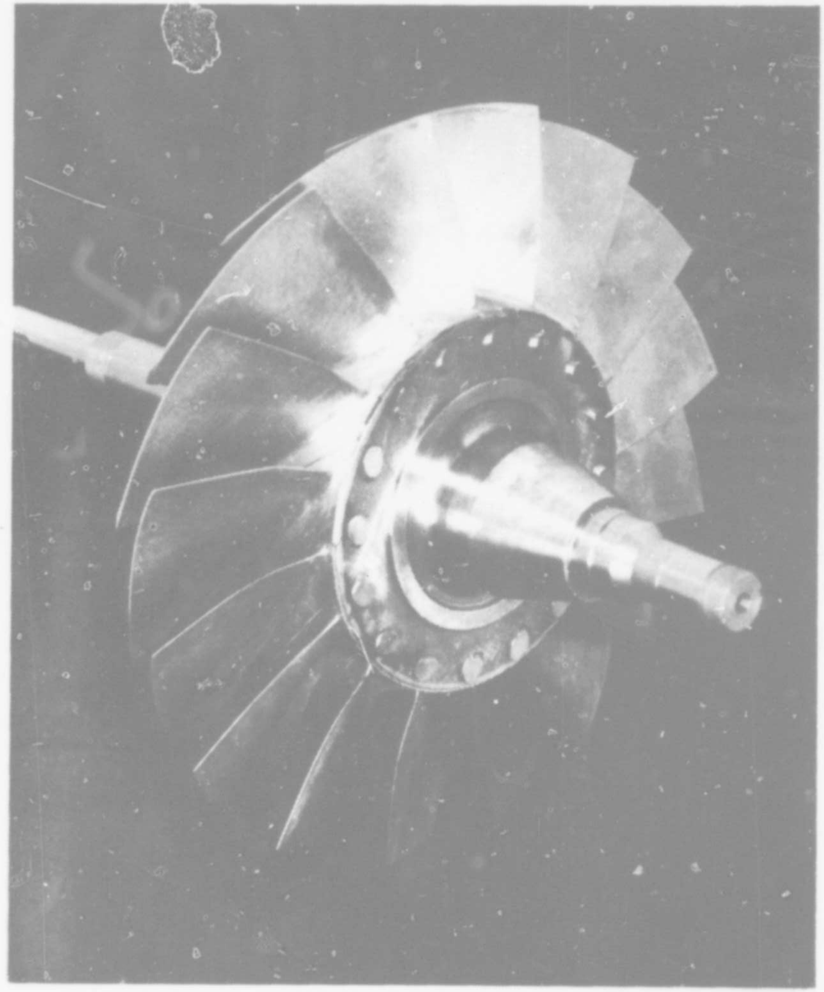
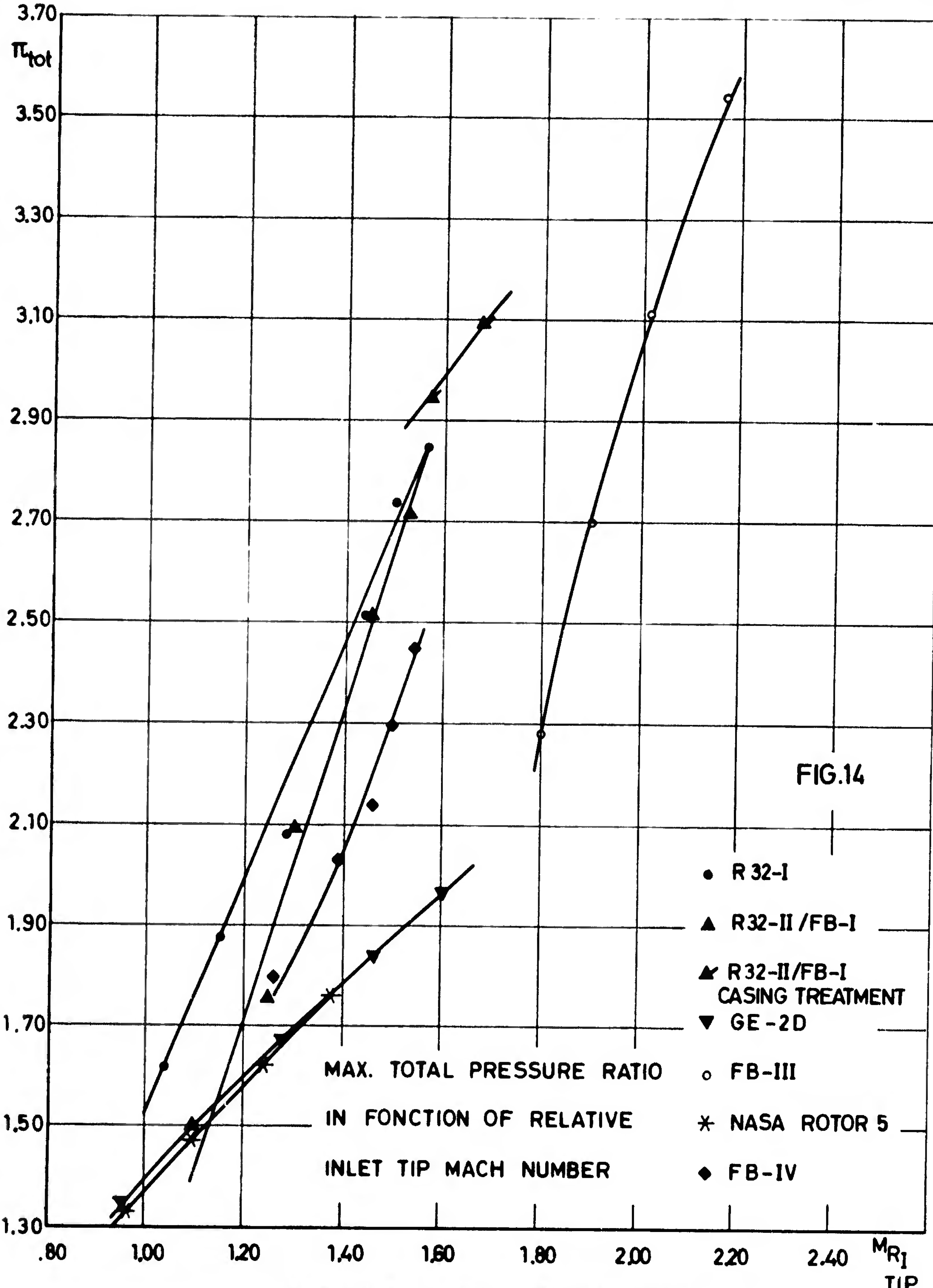


FIG.13



SUPERSONIC ROTOR PERFORMANCE

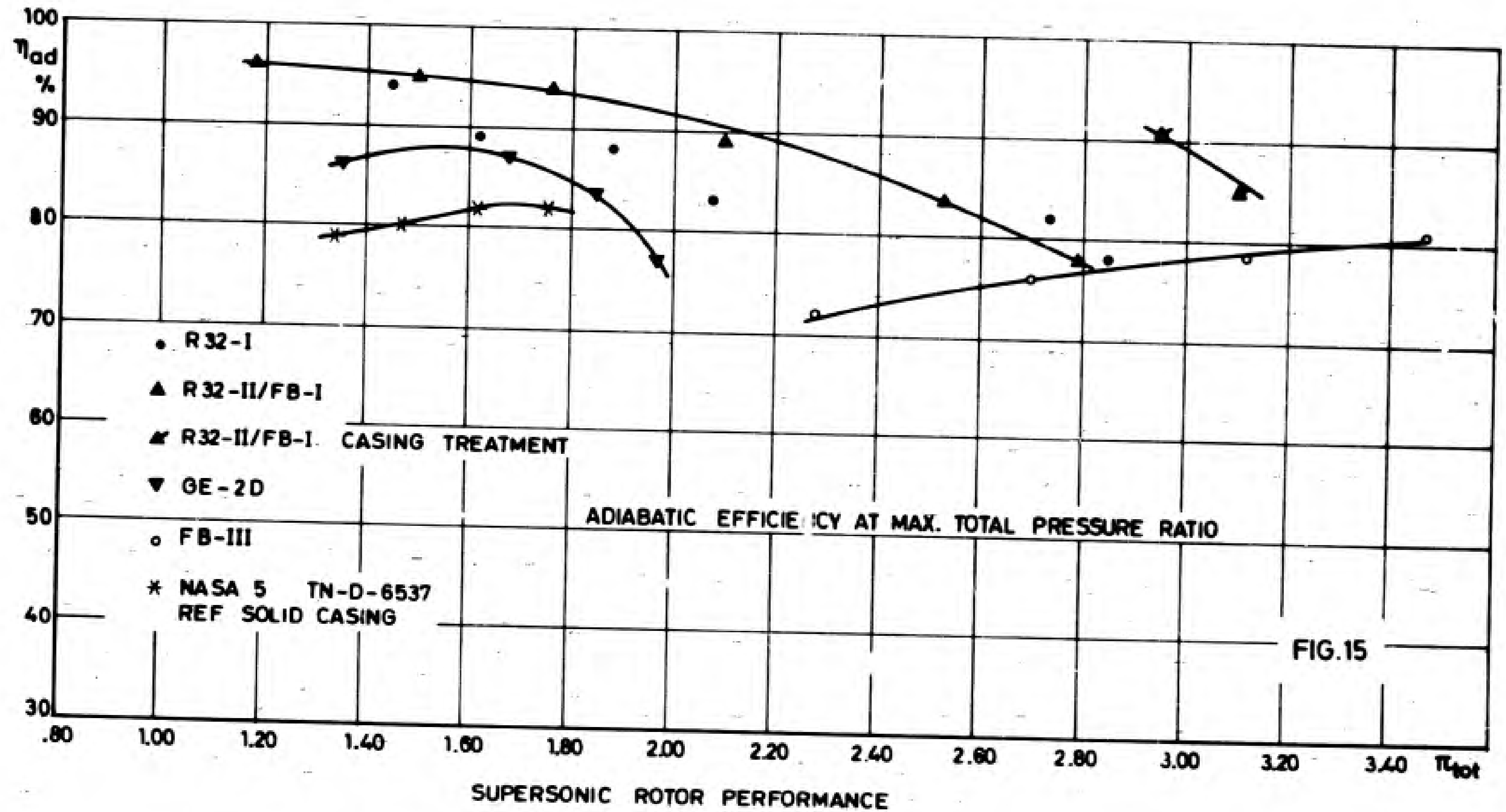
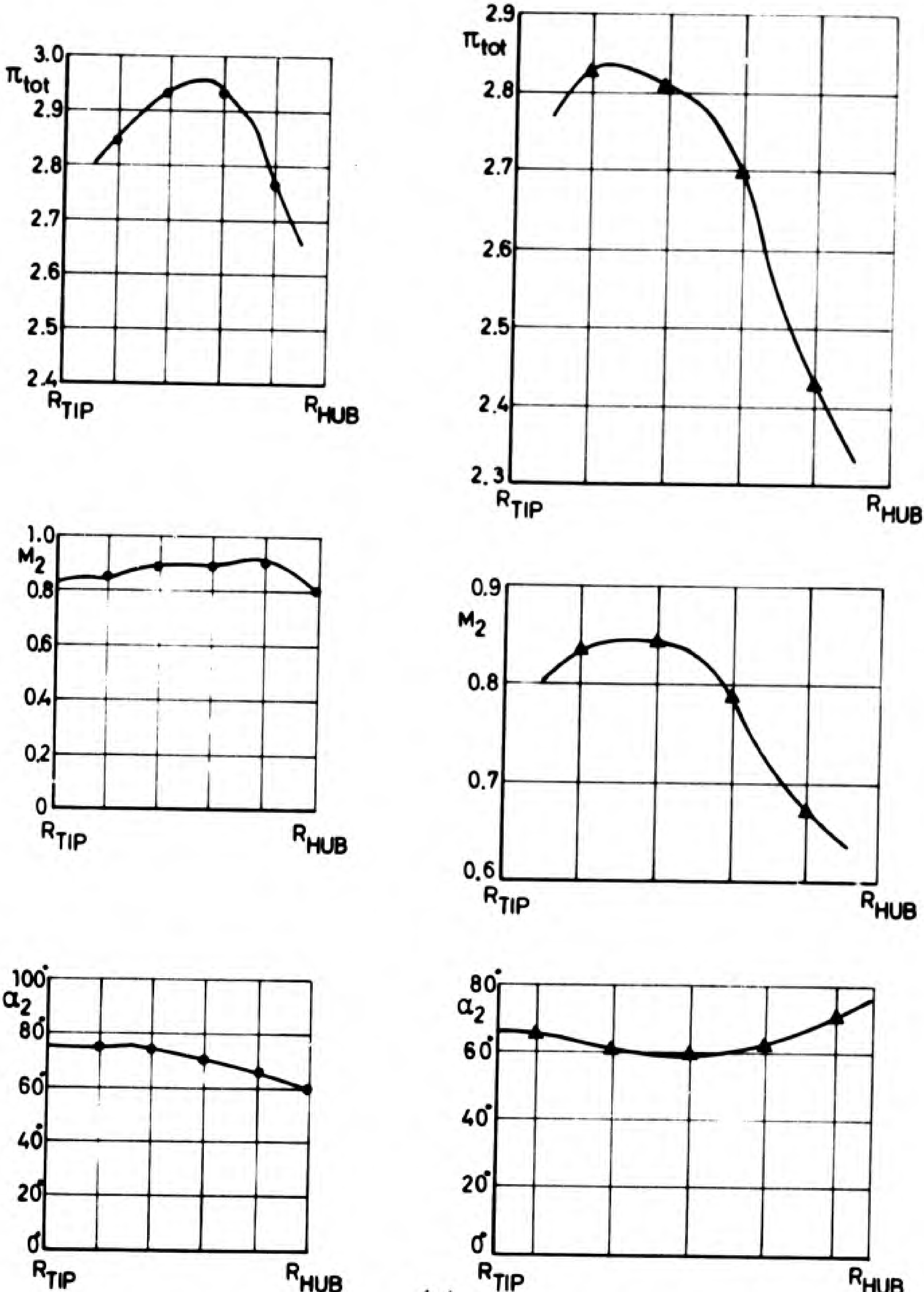

100
η_{ad} %
90
80
70
60
50
40
30
.80
1.00
1.20
1.40
1.60
1.80
2.00
2.20
2.40
2.60
2.80
3.00
3.20
3.40
π_{tot}
• R 32-I
▲ R 32-II/FB-I
R32-II/FB-I CASING TREATMENT
▼ GE-2D
o FB-III
* NASA 5 TN-D-6537
REF SOLID CASING
ADIABATIC EFFICIENCY AT MAX. TOTAL PRESSURE RATIO
SUPERSONIC ROTOR PERFORMANCE

FIG.15



R 32 - I 92,3 %

R 32 - II - F.B. - I 101 %

FIG.16

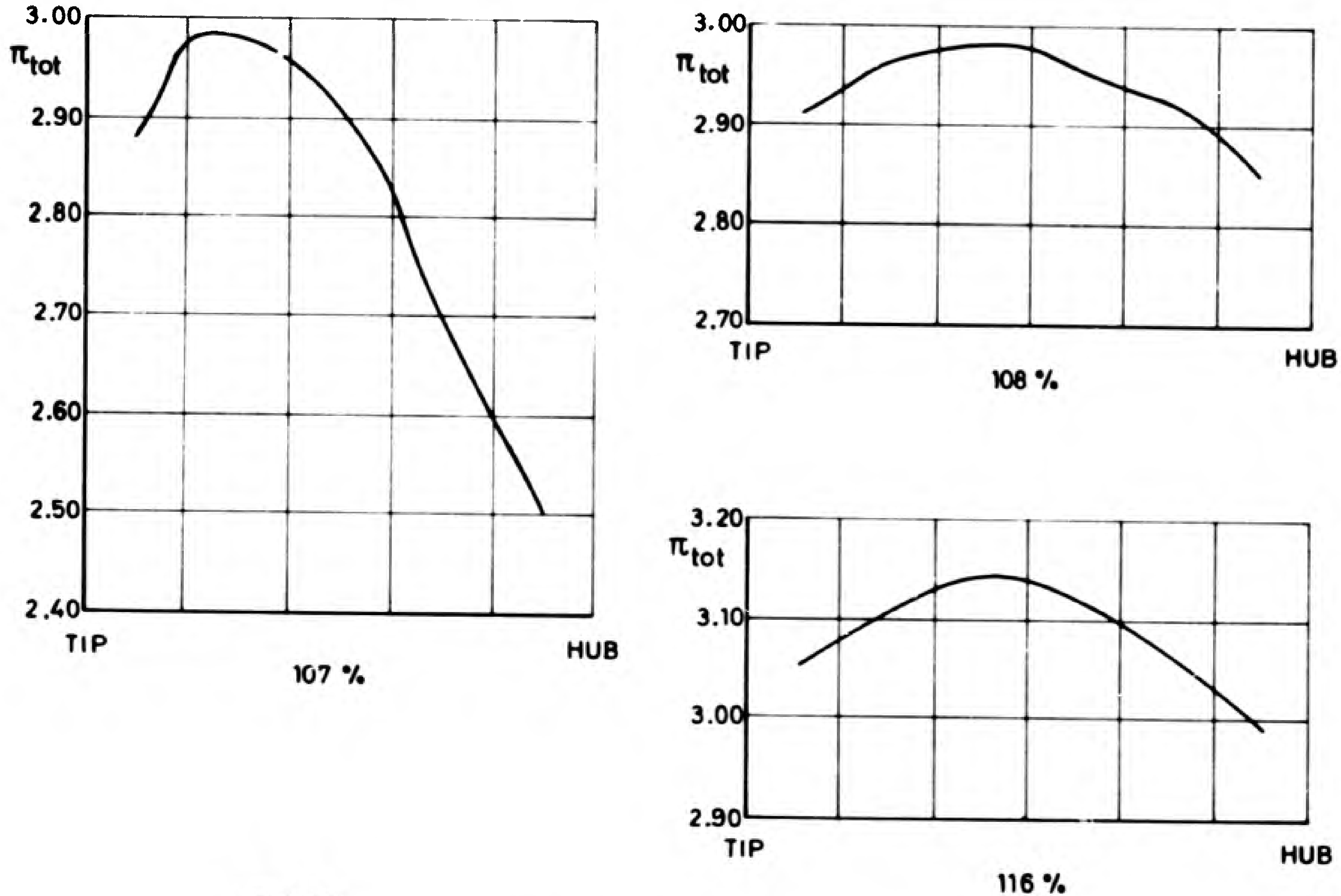


FIG.17 INNER CASING TREATMENT (1967)

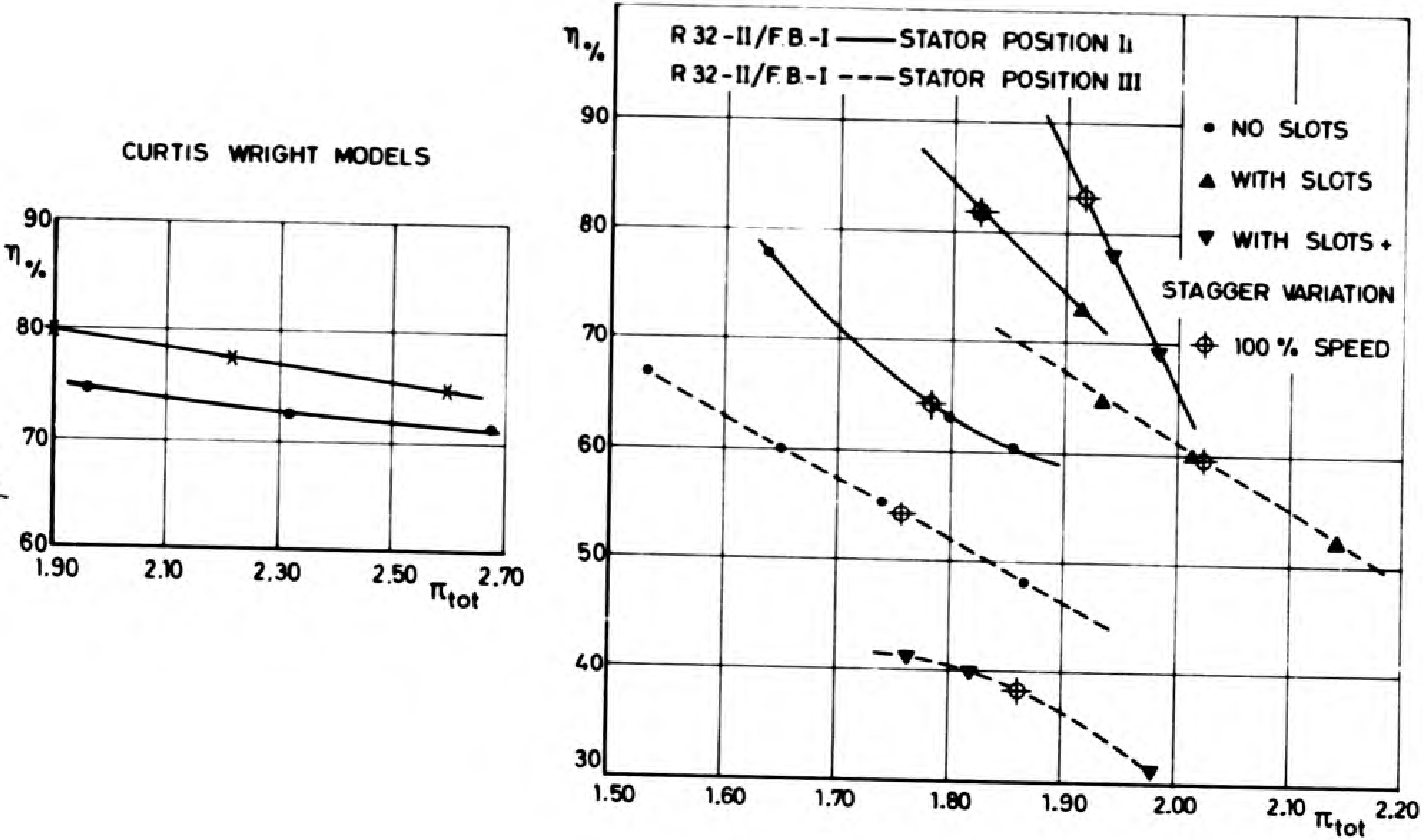


FIG.18 THE SUPERSONIC STAGE PERFORMANCE