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# A Study of Variable Geometry in Advanced Gas Turbines

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To

My Parents

## ABSTRACT

The loss of performance of a gas turbine engine at off-design is primarily due to the rapid drop of the major cycle performance parameters with decrease in power and this may be aggravated by poor component performance. More and more stringent requirements are being put on the performance demanded from gas turbines and if future engines are to exhibit performances superior to those of present day engines, then a means must be found of controlling engine cycle such that the lapse rate of the major cycle parameters with power is reduced. In certain applications, it may be desirable to vary engine cycle with operating conditions in an attempt to re-optimize performance.

Variable geometry in key engine components offers the advantage of either improving the internal performance of a component or re-matching engine cycle to alter the flow-temperature-pressure relationships. Either method has the potential to improve engine performance.

Future gas turbines, more so those for aeronautical applications, will extensively use variable geometry components and therefore, a tool must exist which is capable of evaluating the off-design performance of such engines right from the conceptual stage. With this in mind, a computer program was developed which can simulate the steady state performance of arbitrary gas turbines with or without variable geometry in the gas path components. The program is a thermodynamic component-matching analysis program which uses component performance maps to evaluate the conditions of the gas at the various engine stations.

The program was used to study the performance of a number of cycles incorporating variable geometry and it was concluded that variable geometry can significantly improve the off-design performance of gas turbines.

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## Nomenclature

| | |
|---|---|
| $A$ | Area |
| $A_c$ | Inlet capture area |
| $A_{9max}$ | Maximum nozzle exit area |
| $A_{10}$ | Maximum fuselage cross-sectional area |
| $A_{11}$ | Fuselage cross-sectional area at engine connection point |
| C | Flow velocity |
| CN | Non-dimensional rotational speed relative to design, $(N/\sqrt{T})/(N/\sqrt{T})_{des}$ |
| CV | Velocity coefficient |
| $C_D$ | Drag coefficient |
| $C_{FG}$ | Nozzle gross thrust coefficient, $F_G/F_I$ |
| $C_{FGR}$ | Nozzle gross thrust coefficient of actual nozzle |
| $C_{FGT}$ | Nozzle gross thrust coefficient of test nozzle |
| $C_L$ | Lift coefficient |
| D | Drag |
| $D_R$ | Ram drag |
| ECS | Environmental control system |
| F | Thrust |
| FPR | Fan Pressure Ratio |
| $F_G$ | Uninstalled gross thrust |
| $F_I$ | Ideal gross thrust |
| HP | High pressure |
| IGV | Inlet guide vane |
| ISA | International standard atmosphere |
| LP | Low pressure |
| LPC | Low pressure compressor |
| M | Mach number |
| N | Rotational speed |
| OEI | One engine inoperative |
| OPR | Overall pressure ratio |
| P | Total pressure |

| | |
|---|---|
| PCN | Rotational speed relative to design, $N/N_{des}$ |
| PR | Pressure ratio |
| $P_s$ | Static pressure |
| $PR_{max}$ | Surge pressure ratio |
| $PR_{min}$ | Choking pressure ratio |
| SLS | Sea level static |
| T | Total temperature |
| TFF | Turbine flow function, $W\sqrt{T}/P$ |
| TIT | Turbine inlet temperature |
| $T_M$ | Maximum cycle temperature |
| U | Blade speed |
| V | Velocity |
| VABI | Variable area bypass injector |
| VAT | Variable area turbine |
| VATEMP | Variable area turbine engine matching program |
| VCE | Variable cycle engine |
| VIGV | Variable inlet guide vane |
| VPR | Variable pitch rotor |
| V/STOL | Vertical and short take-off and landing |
| W | Mass flow |
| Z | $= \dfrac{PR - PR_{min}}{PR_{max} - PR_{min}}$ |
| $\Delta F_{\eta R}$ | Thrust loss due to pressure recovery |
| $\Delta H/N^2$ | Turbine work-speed parameter |
| $\Delta H/T$ | Turbine work function |
| $c_{pa}$ | Specific heat capacity at constant pressure for air |
| $c_{pg}$ | Specific heat capacity at constant pressure for gases |
| f | Fuel/air ratio |
| k | Constant |
| m | Mass flow |
| q | Non-dimensional mass flow, $W\sqrt{T}/P$ |
| r | Pressure ratio |
| sfc | Specific fuel consumption |
| t | Temperature ratio |
| $t_4$ | $T_4/T_1$ |

## Greek Symbols

| | |
|---|---|
| $\Delta$ | Change |
| $\psi$ | Stage loading coefficient |
| $\alpha$ | Flow incidence |
| $\gamma$ | Ratio of specific heats, $c_p/c_v$ |
| $\delta$ | Total pressure referred to standard day |
| $\zeta$ | IGV angle setting |
| $\eta$ | Efficiency |
| $\theta$ | Total temperature referred to standard day |
| $\mu$ | Bypass ratio |
| $\rho$ | Density |
| $\phi$ | Flow coefficient |
| $\omega$ | Angular speed |

## Suffices

| | |
|---|---|
| Aft-End | Rear fuselage |
| INL | Inlet |
| INST | Installed |
| POW | Power |
| R | Recovery |
| TRIM | Trim |
| a | Axial |
| c | Compressor |
| cc | Combustion Chamber |
| p | Propulsive |
| t | Turbine |
| th | Thermal |

## Station Numbering (Turbojet only)

| | |
|---|---|
| a | Ambient |
| j | Jet |
| 0 | Free stream |
| 1 | Inlet entry |

| | |
|---|---|
| 2 | Compressor inlet |
| 3 | Compressor delivery |
| 4 | Turbine inlet |
| 5 | Turbine exit |
| 6 | Jet pipe inlet |
| 7 | Nozzle inlet |
| 8 | Nozzle throat |
| 9 | Nozzle exit |

# INTRODUCTION

The gas turbine engine is one of the most efficient and reliable power-plants ever designed and is used to satisfy the propulsion requirements of quite a variety of mechanical systems, in the air, sea, and on land. Its overall performance is usually presented as curves depicting the variation of specific fuel consumption (sfc) with output. The latter is either shaft power or jet thrust while the former is defined as the amount of fuel consumed to produce one unit of output at a given output level. There are other factors which are considered when the performance of a gas turbine is being evaluated and these include the response rate, that is, the time it takes to effect a change of output, and the amount of safety margin on critical components.

At the design point, the major cycle performance parameters and component performance are chosen such that for the required duty, cycle performance is optimized to produce the desired power.[1] The engine is then probably operating most efficiently. When the engine is operating away from its design point, that is, at off-design, the performance of the various components deteriorates and coupled with a reduction in the cycle performance parameters, the overall performance of the cycle deteriorates causing sfc to rise. At certain power levels, the sfc may be unacceptably high and operation of the engine then becomes uneconomical.

The power developed by an engine is related to the airflow swallowed and the maximum cycle temperature commonly called turbine inlet temperature (TIT) and therefore, the power is a maximum at the point where both airflow and TIT are highest. For any given set of operating conditions defined by ambient conditions and, in the case of aircraft engines, Mach number as well, the maximum power is limited by either TIT or rotor speed and these in turn are limited by materials and stress considerations. So, if the power requirement at any given operating conditions happens to be greater than the maximum power that can be produced by the engine, then the load demands are to be reduced which could result in a loss of operating revenue for and/or annoying discomfort to either the user or customer or both.

Also, there are applications where a rapid response is desirable in cases of emergency where a catastrophe is being avoided. Under these circumstances, power change must occur quickly and safely, and therefore, a knowledge of the dynamic response of the engine is required during the

1. Hereafter, the term 'power' refers to engine output unless otherwise implied.

design phase so that a control system can be designed to give the required response rate and safety margins.

From the foregoing, it is clear that the operation of a gas turbine engine involves meeting the load demands not only economically but also safely.

The physics of operation of the gas turbine involve satisfying the various conditions of compatibility between the components. At the design point, the components are all well matched and at off-design, in order to satisfy the compatibility equations, it is possible that the components may be badly matched such that one or more components are operating in regions of low efficiency leading to poor engine performance. Loss of thrust or an increase in sfc or both may be the end result.

A reason for poor matching of the components at off-design is that the area at inlet to the various components is determined from the flow conditions at the design point and operation with these areas at off-design restricts the range in which the operating point may fall on the component performance maps, so if these areas can be varied with flow conditions a better matching may be obtained. It will be shown in Chapter 3 that the drop in the major cycle parameters with reduction of power is mainly responsible for the deterioration in cycle performance as power reduces. Therefore, if an engine can be controlled such that the lapse rate of the major cycle parameters with power is reduced, then cycle performance can be improved.

Unlike marine and land based gas turbines, aircraft gas turbines operate over a wide range of compressor inlet temperatures and for jet engines sfc does not only depend on thermal efficiency which decreases with reduction of thrust but also propulsive efficiency which increases with reduction of thrust. There is also the problem of installation losses which cause the thrust developed by an engine when installed in the aircraft to be markedly reduced from that produced when on a test rig. It is known that the reduction in airflow with decrease in thrust increases the installation losses and therefore a higher airflow is desirable which is also beneficial to sfc as propulsive efficiency is improved.

The thrust, F, of a fully expanded jet engine is defined as

$$F = m(V_j - V_a) \qquad (1.1)$$

in the usual notation. The specific thrust is defined as the thrust per unit mass flow which from (1.1) is equal to the difference of jet and flight velocities, and is a direct measure of the linear dimensions or size of an engine. For supersonic propulsion, a high specific thrust is desirable in order to keep weight and frontal area down and hence, drag. A high specific thrust results in a low propulsive efficiency which is detrimental to good subsonic performance. Therefore, a compromise has to be struck in the selection of the cycle parameters for engines capable of supersonic propulsion. An ideal engine would be one whose cycle can be varied to suit any given flight conditions. This requires active control of the cycle parameters which can only be achieved by the use of variable geometry within the engine. By effective control of bypass ratio, $\mu$, these variable cycle engines (VCE) can re-optimize their cycle parameters to give an overall performance that is suited for the given flight conditions.

The trend in propulsion demands placed on gas turbines is that these engines should be capable of delivering a wider range of power more efficiently and reliably at reduced size and with increased maximum power capability. The transient performance is also expected to be improved. If future engines are to give performance levels much more advanced than those of present day engines, then some degree of flexibility has to be incorporated into the design of the engines such that the operating point of the critical components can be controlled to fall in regions of the component performance maps which will give an improvement in overall engine performance. This improvement will be the result of increasing the thermal or propulsive efficiency, or of increasing power further by increasing either spool speed or TIT when the other has reached a maximum, or a combination of these.

Variable geometry has been used in compressors and propelling nozzles for many years in an attempt to improve the matching of gas turbine components at off-design. Variable geometry in compressors is used passively to improve the matching of the compressor stages whereas a variable area propelling nozzle affects the matching of the entire engine by being able to move the operating line on a low pressure (LP) compressor or fan thereby permitting a greater excursion on the map. There is further potential for improved engine performance by the use of variable geometry in a turbine.

Turbines are normally designed to operate choked over a wide range of operating conditions, that is, the mass flow function at inlet is a maximum represented mathematically as

$$\frac{W\sqrt{T}}{A\ P} = \text{Const} \qquad (1.2)$$

With the fixed geometry turbine, $W\sqrt{T}/P$ is a constant and therefore when T changes to change power, P and, most probably, W change to satisfy (1.2), but with A varying, the variable geometry turbine gives the engine an extra degree of freedom which can be used to keep the major cycle parameters at a higher level than what conventional fixed geometry engines can give with the result that engine performance is improved.

During the design and development of a gas turbine, in order to reduce development time and prevent possible cost overruns, accurate and efficient analytical tools are needed to aid the performance analyst in his quest for the best cycle to match the load demands. Computer programs for cycle analysis have been a valuable tool to the performance engineer when optimizing his cycle or predicting the performance of the engine at off-design. However, the vast majority of these programs have the capability to only simulate engines with fixed geometry components.

A program called "TURBOMATCH" exists at Cranfield which can simulate the design point and steady state performance at off-design of any gas turbine type. The only variable geometry capability is that of nozzle area change. It was desired that a program be developed with variable geometry capability in as many gas path components as is possible. It was decided to use the Turbocode on which Turbomatch was built due to its flexibility. As a result, a computer program called VATEMP was developed around Turbomatch with increased simulation capability. The installed performance of aircraft gas turbine engines can also be evaluated.

The layout of the rest of the thesis is as follows. Chapter 2 surveys the literature on gas turbine performance with a view to presenting the proposals that have been put forward for future development of gas turbines with the aim of obtaining performances more advanced than what present day engines can obtain. Some of the potential uses of the variable geometry compressor and turbine to improve engine performance are highlighted in Chapter 3. The reader is introduced to the problem of evaluating the installed performance of aircraft gas turbines in Chapter 4 and a model is presented from which both the uninstalled and installed thrusts can be obtained.

In Chapter 5, variable cycle engines are studied with special attention being paid to the problem of supersonic propulsion. The computer program that was developed is briefly described in Chapter 6. The application of the program is studied in Chapters 7 and 8 for shaft power and jet engine cycles, respectively, while Chapter 9 discusses some of the factors that are to be taken into consideration when variable geometry is used in compressors and turbines. A summary of the findings of the investigations carried out in this thesis is presented in Chapter 10.

# CHAPTER 2

## DEVELOPMENT AND PERFORMANCE IMPROVEMENT TRENDS

The phenomenal development of the aircraft gas turbine engine from its infancy five decades ago to today's highly efficient and compact powerplant can be hailed as one the major technological achievements of the post war era. The excessive demands generated by an aircraft system to efficiently cover the vast and varied flight envelope encountered have placed stringent requirements on propulsion system performance which has led to the advancement of the aircraft gas turbine far ahead of its marine and land based counterparts.

It has been argued in certain quarters that aircraft gas turbine technology is a "sunset" technology that has reached its improvement plateau, and that further improvement would not yield a good return on further financial investment in research [41].[1] It should be remembered that though the demands placed by an aircraft on its powerplant have been responsible for exhausting the technological state-of-the-art, they are also responsible for creating new technologies which in return have set the pace for the further development of advanced aircraft systems. Similar arguments can be presented for non-aeronautical gas turbines in the light of their required duties so therefore, there is the need for continued investigation into possible improvements that can be made, from the point of view of both engine and component performances.

This argument could furthermore be compounded when one takes a look at the requirements placed on aircraft engines for military use.

Fighter aircraft are required to operate over a wide range of flight conditions. In the past, the philosophy of all fighter operation required that their powerplants be optimized for a given role. The present trend in fighter logistics is to deploy these aircraft flexibly to cover both offensive and defensive roles effectively. In addition to the traditional characteristics such as manoeuvrability, range, and payload, postulated mission requirements of future fighter aircraft will include high levels of supersonic persistence, reduced observables or stealth, and short takeoff and landing capability for the reasons cited

1. The numbers in square brackets denote references at the end of the thesis.

in [20,26]. Present day fighters cannot achieve these new mission elements satisfactorily using state-of-the-art technology engines without incurring severe penalties in mission performance and aircraft size, weight, and cost. In order to realize these new operational goals without any trade-offs or compromises, new advanced technology engines will have to be designed which offer the opportunity of tailoring their performance and design characteristics to mission requirements right from the outset to produce an aircraft system within weight and cost constraints [20] and with excellent airframe/engine integration characteristics.

The arguments presented in [19,41,62,63] and the mission studies conducted in [20,23,42] suggest that the way to go to meet the propulsion requirements of future fighter aircraft is to use engines that have the capability to change their cycle to match any current flight conditions. These variable cycle engines would incorporate some variable geometry features which will enable them to change the primary cycle parameters which are bypass ratio, overall pressure ratio (OPR), and turbine inlet temperature, to optimum values whenever desired. Cycle and mission studies that have been conducted, [3,12,13,18,56,58,59], show that similar arguments can be presented for supersonic transport engines. These studies have yielded engine cycles which possess not only the mixed mission requirements for both subsonic and supersonic cruise, but also balance these requirements with those of reduced noise and low exhaust emissions.

The primary performance variables of a variable cycle engine should be judiciously chosen such that the ensuing cycle gives the best compromise between the optimum cycles for subsonic and supersonic operation. This may suggest that severe performance penalties would be incurred when the engine is operating in either flight leg. Fighter aircraft are fortunate in that an engine design can satisfy several mission requirements as fighter missions are not too sensitive to engine cycle variations. Data is presented in [42] which shows that while the optimum engine cycle changes with mission requirements, the penalty associated with a non-optimum engine cycle is small for a large range of cycle variables. Thus a VCE could give an acceptable performance in both legs of its flight mission.

The high bypass turbofan is used to power subsonic transport aircraft due to its low specific fuel consumption characteristics at cruise. However, loss of fan performance, especially at takeoff, could contribute to a larger less efficient core which has an impact on cruise performance. Variable geometry components can be employed to improve the performance and flexibility of such engines and as reported in [49], some engines that were studied demonstrated improved performance over conventional engines at takeoff, cruise, and landing, by the use of variable geometry in critical components.

Another suitable application of VCEs to subsonic flight is in aircraft systems where a constant source of high bleed air is required during flight. Reference 40 reports the results obtained from a large turbofan engine which was modified to give it variable cycle capability and as was concluded, such cycles could provide a viable solution to the propulsion requirements of powered lift aircraft.

With shaft power cycles, the justification for the use of variable geometry need not be elaborated on as in the case of the aircraft gas turbine since these engines do not have to operate over a wide range of inlet temperatures. However, in this case, the thermal efficiency, $\eta_{th}$, gives a direct indication of how effectively the engine is being used as a propulsion unit and since $\eta_{th}$ falls rapidly with decreasing power, any cycle change or improvement that may lower the efficiency lapse rate is worth looking into.

In an attempt to improve the part load performance and handling characteristics of regenerative gas turbines for road vehicular operation in particular, variable geometry has been used in the compressors and turbines of this class of engines. Large power augmentation can be obtained by the use of variable inlet guide vanes (VIGV) thereby permitting the use of a smaller engine which gives lower losses at part power operation resulting in improved fuel economy in addition to providing better acceleration characteristics. The variable geometry turbine can also significantly improve the part power performance of turboshaft engines and as shown in [29], sfc can be held down, even at its design point level, over a wide power range. Other operating advantages can also be obtained such as excellent starting, acceleration, and braking [45].

An application of variable geometry in which great interest has been developed of late is that of using compressor variable geometry to control the flow into a fan or compressor so as to modulate output at constant rotational speed. VIGVs and variable compressor stators have been in extensive use in gas turbines for quite a number of years but are chiefly employed to control the stability of compressors at low and intermediate corrected speeds. Controlling output at constant speed is highly desirable for vertical and short take-off and landing (V/STOL) aircraft control during takeoff, landing, hover, and emergency, for fast response. References 16, 43, and 61 report some work in this area for fixed wing aircraft while Mann [25] recently investigated the possibility of using a variable geometry compressor in conjunction with other possible power modulation devices to control the power of helicopter turboshafts. Other work carried out in this area have found out that with turbofans, VIGVs positioned to span only the outer annulus can result in a saving in engine weight and fuel consumption while at the same time providing the desired response.

Rig tests have been conducted on variable area turbines as have tests on engines incorporating such turbines since the early 1950s [6,17,30,45,52] but variable geometry turbine technology has not caught on chiefly because of the hostile environment in which these turbines will have to operate. However, the interest in supersonic propulsion that was rekindled in the United States in the early 1970s and the quest for improved subsonic performance of aircraft engines have led to further investigations into the potential use of variable area turbines (VAT) in aircraft powerplants. It was greatly feared that varying turbine vanes would result in loss of performance which might negate any possible benefits which might be gained from improved matching characteristics of the VAT. A twin-spool turbofan engine with a variable area low pressure turbine was assembled and tested at both sea level static (SLS) and altitude conditions. These tests [8,57] dispelled all fears that a significant loss in performance would occur when a VAT is used in an engine.

The encouraging results obtained from these tests prompted the US Armed Forces to support developmental work on both high pressure (HP) and LP turbines and these have found their way into technology demonstrator engines such as the Advanced Turbine Engine Gas Generator and the Joint Technology Demonstrator Engine. The tough goals that the US Air Force have set for their next fighter aircraft scheduled to enter service in the mid 1990s demand a propulsion system that will set benchmarks in all aspects of gas turbine technology. The two candidate engines competing for this arduous task are undergoing rigorous full scale tests at the present and it is suspected that one or more turbines in these engines have variable geometry capability. Thus, it is expected that variable geometry turbines will be in extensive use by the turn of the century.

The situation with industrial and other non-aeronautical gas turbines is less documented. Westinghouse Canada have produced two industrial gas turbines, the CW 352 and CW 182, with variable geometry in the power turbine. By constantly rematching the power turbine with the gas generator, a saving in thermal efficiency of two and a half percentage points can be obtained over that with fixed geometry [15].

From the above discussion, it is clear that gas turbine technology is moving into a new era to efficiently meet the diverse power needs that will be required in the future. The potential performance benefits that can be attained with variable geometry components have caused a rethink of cycle selection procedures amongst propulsion engineers and as a result, radical cycles will emerge in the future which will make the selection procedure of present day cycles look facile by comparison.

# CHAPTER 3

## VARIABLE GEOMETRY COMPRESSORS AND TURBINES

The amount of energy that can be added to the air entering a gas turbine combustor is proportional to the mass of air consumed, and for efficient combustion, the pressure of the delivered air should be quite high. The compressor should be able to provide a maximum of high pressure air with minimum temperature rise while at the same time keeping within stable operation if a high overall engine performance is to be obtained. Therefore, the compression system is a key element in all gas turbine operations.

The compressor type commonly used in large gas turbines is the axial flow compressor due to its inherently high efficiency and small frontal area for a given airflow compared with those of the centrifugal type. Some of these compressors have variable inlet guide vanes and stators which permit the flow angles entering the downstream stage to change with rotational speed in order to improve their off-design performance. These can also be used to control the flow into the compressor at fixed rotational speed to improve engine response.

The axial flow turbine is used in the vast majority of gas turbine engines to drive one or more compressors and accessories, or to produce power to drive a load as in the case with shaft engines. The nozzle guide vane area is a critical part of engine design as its size greatly impacts the surging, acceleration, and sfc characteristics of the engine [47]. The turbine is normally designed with the guide vanes choked over a wide range of operating conditions and this puts a restriction on the excursions that can be made on the compressor characteristics which in turn limits engine operational flexibility. Since a duty of the guide vanes is to direct the flow onto the turbine blades at the correct angle to produce a large component of force in the plane of the rotor, then ideally, the vane angles should vary with flow conditions but are normally kept fixed due to mechanical complexity. A variable geometry turbine can improve the matching characteristics of some of the engine components.

In order to realize what potential performance benefits that can be obtained from engines which employ variable geometry compressors and turbines, the operating characteristics of these components should first of all be examined with a view to identifying the reasons for the loss of performance at off-design.

## 3.1 Justification for Use of Variable Geometry Compressors

An axial flow compressor is made up of several stages, each with its own characteristics; therefore, its performance characteristics greatly depend on the characteristics of the individual stages and their relative matching.

There are two important parameters which affect the performance of axial compressors. These are the non-dimensional airflow and speed. Though an axial compressor can attain high efficiency, its inherently narrow range of operation poses some engine operational difficulties. As can be seen from Figure 3.1, the useful flow range at any given speed may be limited by either a rapid change in pressure ratio with small changes of flow, or a marked drop in efficiency with changes of flow. There is also the possibility of unstable operation, that is, surging, when the flow is decreased beyond a certain point.

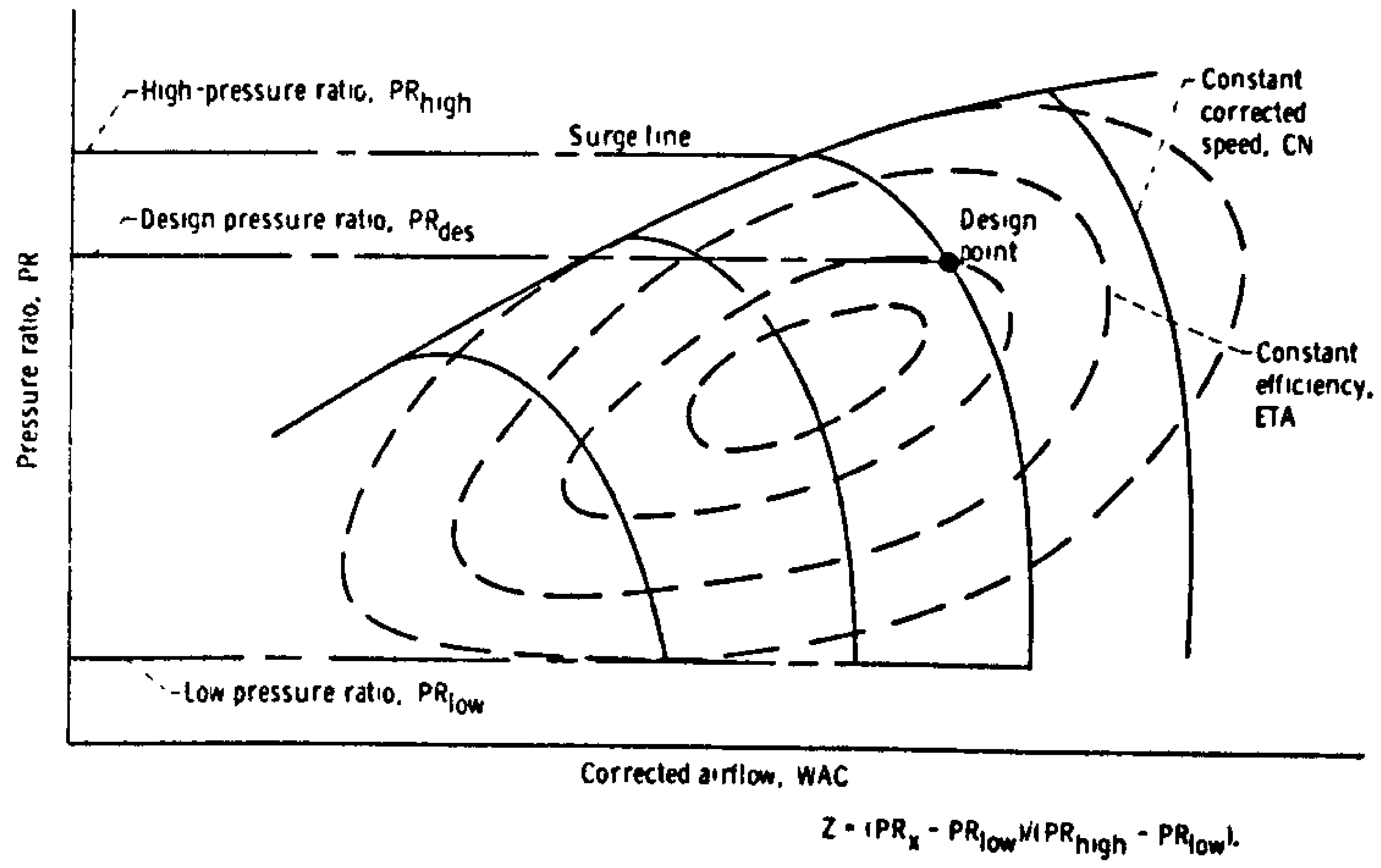



Fig. 3.1 Typical Compressor Characteristics

The useful speed range is also limited by a marked drop in efficiency with changes in speed or by a large difference between the peak efficiency flow and the required flow at speeds appreciably different from the designed [54]. If the operating line intersects or runs quite close to the surge line as is possible at low and intermediate speeds, the engine will not be able to deliver the maximum power without some remedial action being taken. A closer look at stage performance at off-design will give an insight into the problem of performance degradation at off-design to which a solution can then be sought.

### Stage Performance

A compressor stage characteristic, Fig. 3.2, can be expressed in terms of a stage loading coefficient, $\psi$, a flow coefficient, $\phi$, and either a pressure coefficient or stage efficiency [7,54]. At the design point, the flow directions are all correct and the stage efficiency is highest. Since the lift coefficient of a blade row is a function of the incidence of the flow on the blading, for efficient operation, the rotor should be able to maintain a high lift coefficient over a wide incidence range, Fig. 3.3.

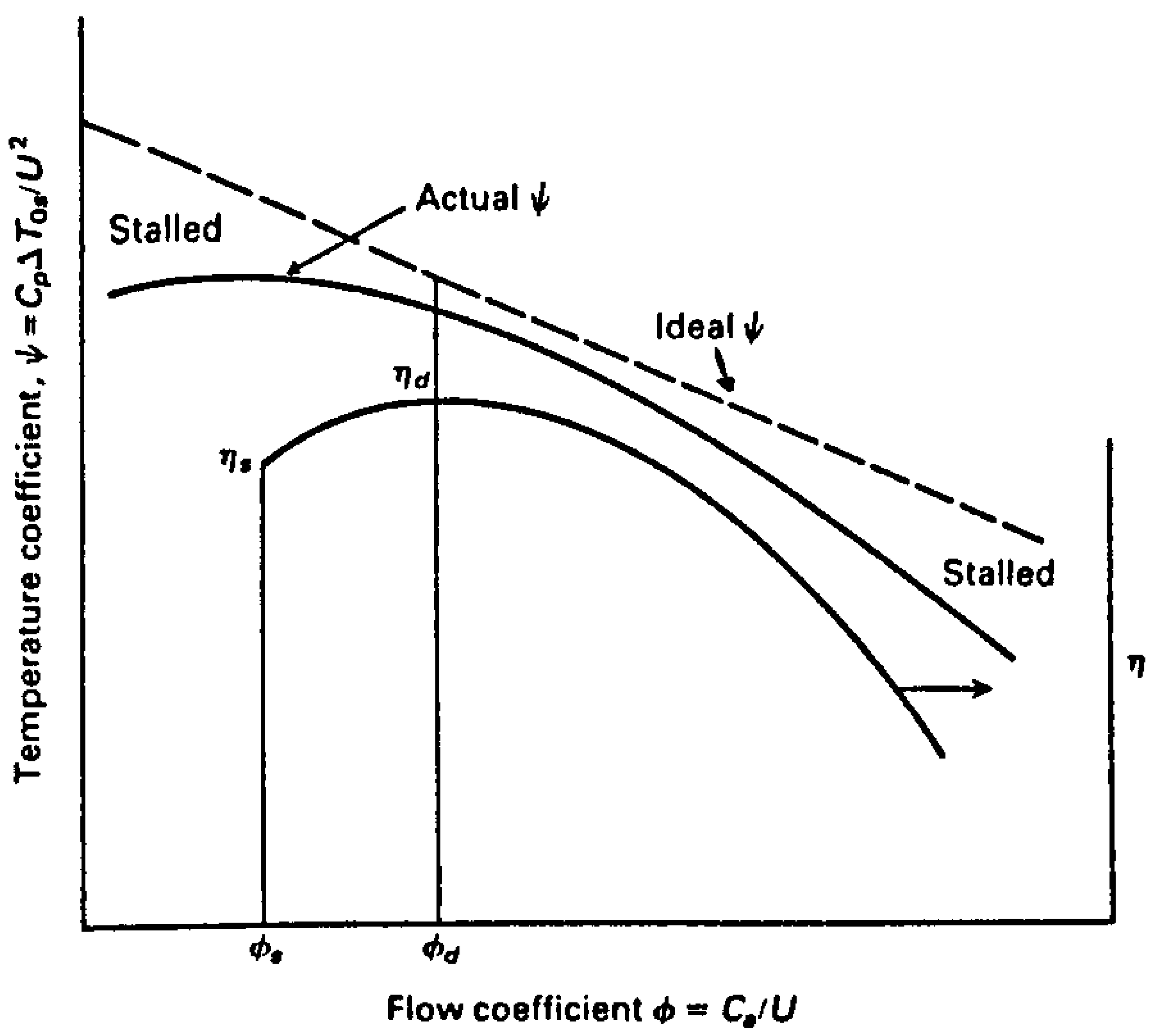



Fig. 3.2 Representation of compressor stage performance [7]

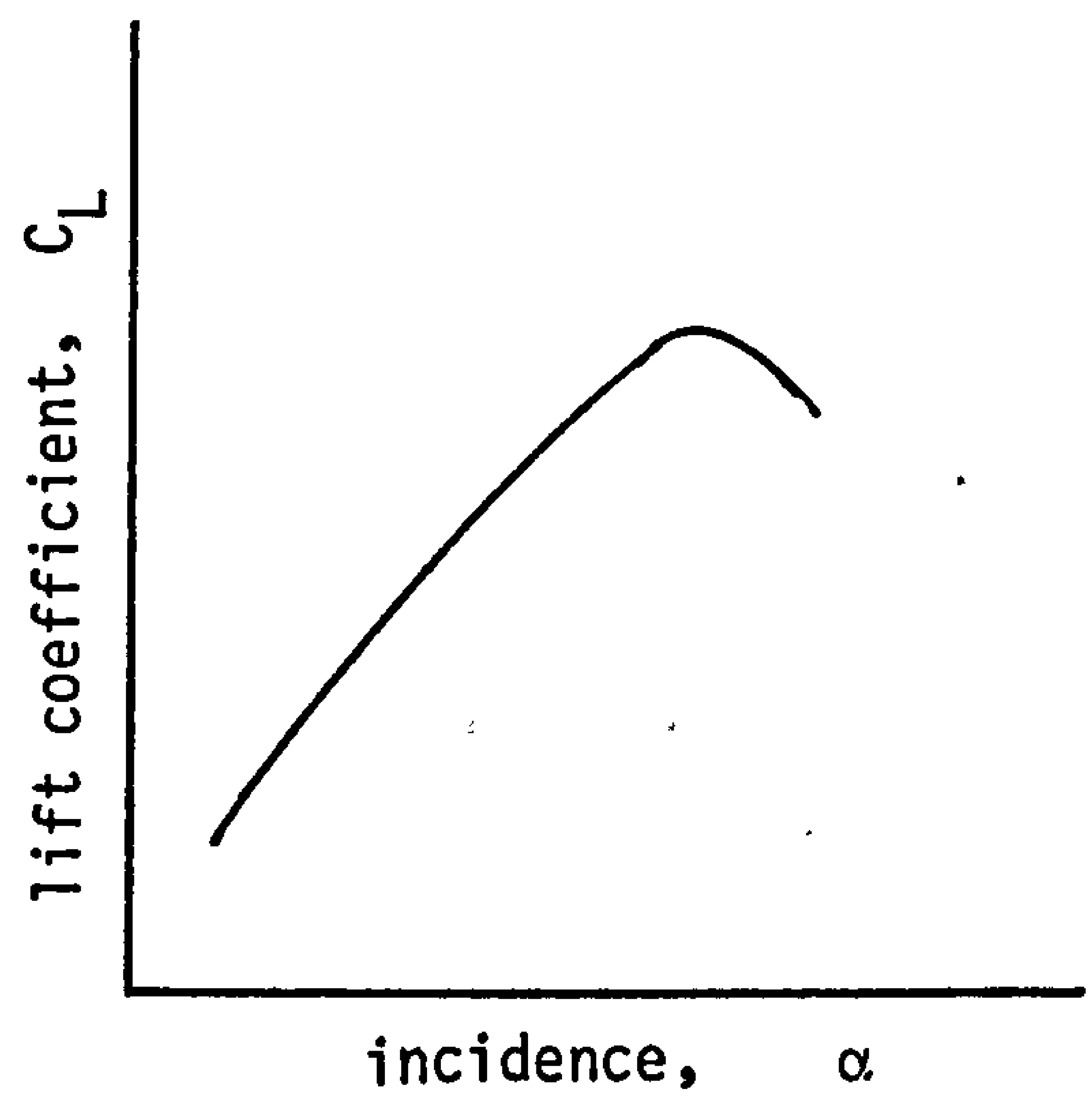



Fig. 3.3 Aerofoil section lift coefficient

As the compressor operates away from its design point, the flow and/or speed changes, causing a change in density distribution in the various stages. The effect of flow on incidence for a typical stage is exemplified in Fig. 3.4. The angles of attack of both stator and rotor blade rows increase with decreasing flow, and vice versa. This phenomenon occurs in all stages and since the effect of density change is cumulative as the flow moves downstream, multistage compressors easily degrade in performance at off-design. A change in compressor speed alters the density ratio into a stage and will therefore alter the axial velocity distribution from the designed.

It is therefore possible that incidence changes may cause a stage to stall at positive incidence or choke at negative incidence both resulting in a loss of stage performance which can limit compressor operation. The reason for an operating line running quite close to or intersecting a surge line at medium and low speeds is that at these loadings, several of the front stages, more so the inlet stage, are stalling, causing a drop in compressor pressure ratio. Therefore, it is desirable that at normal running conditions, each stage should operate in the high efficiency region without running into stall or choke.

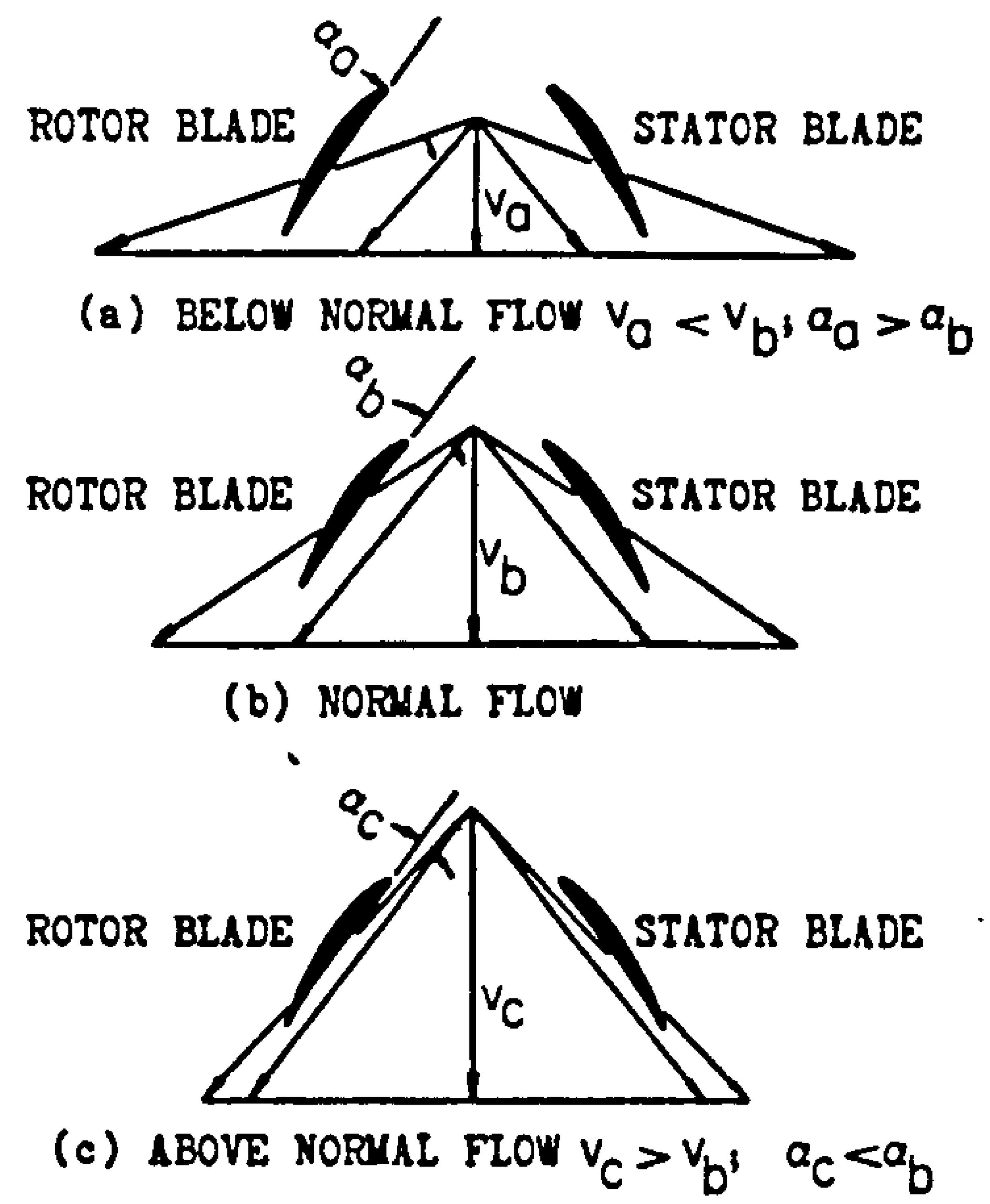



Fig. 3.4 Effect of flow on angle of attack for a typical compressor stage [54]

IGV and Stator Adjustment

From the above discussion, it is clear that if the setting of some of the blade rows is changed with operating conditions, the restriction on compressor performance may be either lifted or relaxed. Due to mechanical complexity, rotor blade rows are not normally adjusted. Experiments that have been carried out on some single stage fans have shown that depending on the required duty, rotor row adjustment may give a better performance aerodynamically than does stator row adjustment [16]. However, rotor stalling can be prevented by adjusting the vane row just ahead of the rotor row. Fig. 3.5 shows that adjusting a stationary blade row changes the incidence on both stationary and rotating blade rows causing a change in lift coefficient on both blade rows. Such an adjustment should be accomplished such that the lift coefficient on each row is kept within reasonable limits, and, in particular, below the stalling point of the blades.

Readjusting the IGV settings alone could give a substantial improvement in compressor performance but excessive angle change should be avoided [54]. IGV adjustment alone may not be enough to raise the surge line appreciably but if a few of the stator stages are also added to the exercise, safe operation of the compressor at low speeds can be made possible.

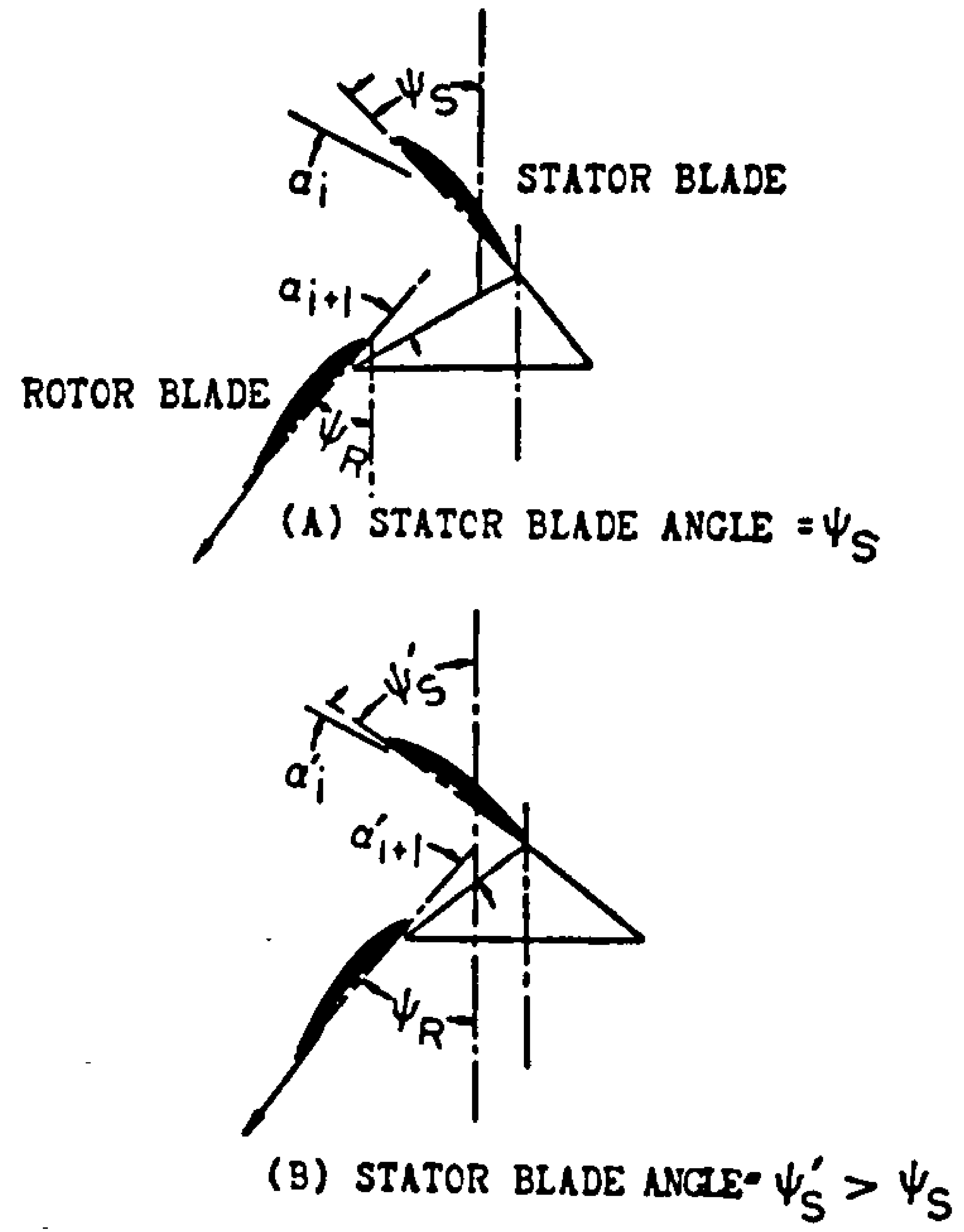



Fig. 3.5 Effect of stator blade angle change on angle of attack of stage rows [54]

There are certain operating conditions where it may be desirable to modulate output at constant rotational speed [25,60]; the rotors must therefore be able to accept a wide range of flows efficiently. Steinke and Crouse [55] have shown that by using variable cambered IGVs, the inlet rotor row can be made to accept the required flow range while keeping within the low loss incidence range. This has the potential of giving an engine an extra degree of freedom.

## 3.2 The Variable Area Turbine

The restriction put on the matching characteristics between a compressor and a turbine by a fixed geometry turbine is partly responsible for the poor part load performance of gas turbine engines. Any improvement in off-design performance would require some degree of flexibility within the engine, but the specified mode of operation will determine whether this flexibility should be provided by the compressor or turbine or both.

Since variable geometry in compressors is deployed passively, it can only affect the internal performance of the compressor in question. On the other hand, variable geometry in turbines affects the entire engine cycle as it is active in nature, so therefore, there is some scope for cycle improvement at off-design by the use of such turbines.

The three most important variables which determine the off-design characteristics of any gas turbine are

1. Airflow,
2. Turbine Inlet Temperature, and
3. Overall Pressure Ratio.

Airflow and turbine inlet temperature primarily determine the output of the engine whilst TIT and overall pressure ratio dictate the level of cycle efficiency and hence, sfc. Component efficiencies do affect both output and sfc but these are of a second order nature. The components of any gas turbine engine match such that a specified combination of these three variables gives a unique combination of output and sfc, but the type of engine and/or control method may impose a restriction on the possible combination of these three variables at off-design.

Conventional engines with fixed geometry turbines must reduce fuel flow and hence, turbine temperature, to reduce power. With shaft power cycles, sfc rises with decreasing power whereas with jet engine cycles the beneficial effects of increased propulsive efficiency as thrust decreases may or may not outweigh the detrimental effects of reduced thermal efficiency with decreased thrust, therefore, one cannot conclude whether or not sfc increases with reduction in thrust.

$$\eta_{th} = \frac{\left\{\eta_c \eta_t (T_M/T_1) - r_c^{\varepsilon}\right\}\left\{r_c^{\varepsilon} - 1\right\}}{\left\{\eta_c (T_M/T_1 - 1) - (r_c^{\varepsilon} - 1)\right\} r_c^{\varepsilon}} \qquad 3.1$$

where $\varepsilon = \frac{\gamma - 1}{\gamma}$

Equation 3.1 gives an expression for the Joule thermal efficiency for a simple shaft power cycle. An expression for the regenerative cycle can also be written although a bit complicated [29]. If it is assumed that component efficiencies remain fixed as the engine is throttled back, the thermal efficiency can be plotted against pressure ratio for both simple and regenerative cycles as shown in Fig. 3.6. It is clearly seen that turbine inlet temperature greatly affects the thermal efficiency of both cycles whereas pressure ratio has a pronounced effect only on the thermal efficiency of the simple cycle. If pressure ratio, or more so, TIT, can be held at its design value or at a level greater than that given by conventional cycles, then it can be expected that the off-design performance of a shaft engine can be improved.

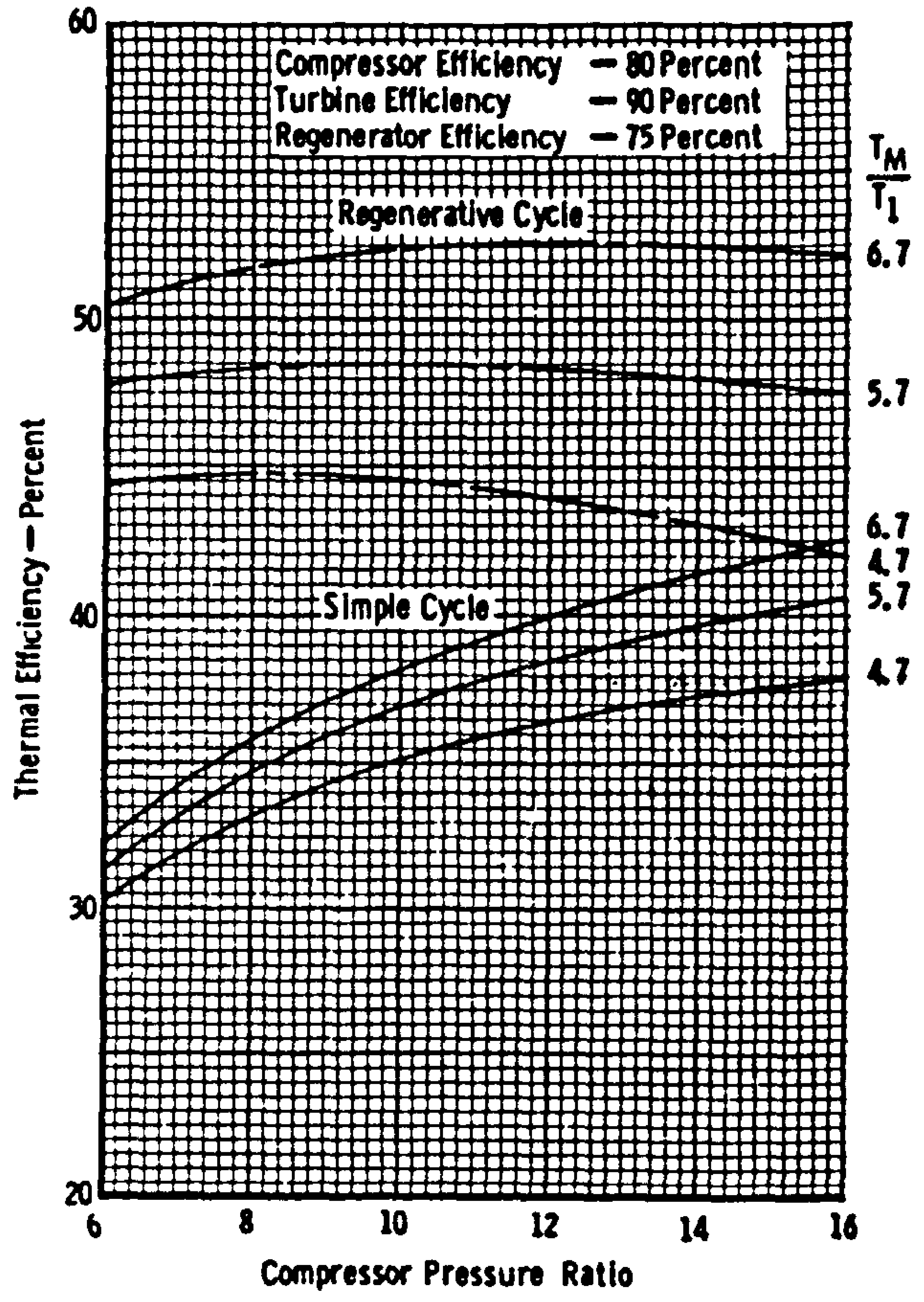



Fig. 3.6 Effect of pressure ratio and TIT on cycle efficiency [29]

A jet engine is not readily amenable to the sort of cycle analysis as that done for the shaft engine in order to show the effect of the deterioration in both pressure ratio and TIT on cycle performance, but an examination of the compatibility equations for certain components will give an indication of the magnitude of the problems encountered at off-design due to the fall off of the major cycle performance variables with reduction of thrust.

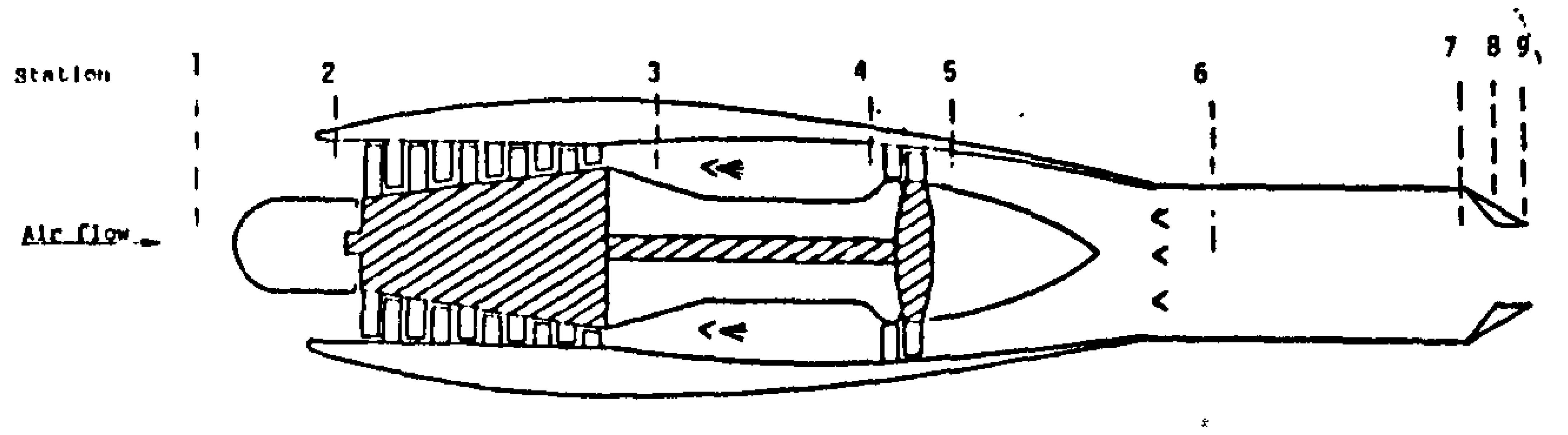



Fig. 3.7 Jet engine schematic

Figure 3.7 is a schematic for a single-spool jet engine. Mass balance at station 4 yields

$$W_4 = \frac{W_4\sqrt{T_4}}{A_4P_4} \frac{A_4}{\sqrt{T_4/T_1}} \frac{1}{\sqrt{T_1}} \frac{P_4}{P_3} \frac{P_3}{P_2} \frac{P_2}{P_1} P_1 \qquad 3.2$$

Neglecting any mass of air bled off from the compressor,

$$W_4 = (1+f)W_2 = q_4 \frac{A_4}{\sqrt{t_4}} \frac{P_1}{\sqrt{T_1}} r_c\, r_R \left[1 - \frac{\Delta P_{cc}}{P_3}\right] \qquad 3.3$$

For fixed flight conditions, we have, neglecting small changes in combustor pressure loss and fuel flow with power setting, and assuming a choked turbine,

$$W_2 \propto \frac{A_4\, r_c}{\sqrt{t_4}} \qquad 3.4$$

Compressor and turbine work balance gives,

$$\eta_m(1+f)\frac{c_{pg}}{c_{pa}}W_2\, \frac{T_4}{T_1} \frac{T_1}{T_2}\left[1- \frac{T_5}{T_4}\right] = \left[\frac{T_3}{T_2} - 1\right]W_2 \qquad 3.5$$

assuming constant specific heats for both the cold and hot ends. Eq. 3.5 can be re-written as

$$t_c = 1 + kt_4(1 - 1/t_t) \qquad 3.6$$

where k is a constant. It can be shown that if the propelling nozzle area is choked,

$$t_t = \text{constant} \qquad 3.7$$

and since

$$r_c = r_c(t_c, \eta_c, \gamma_c) \qquad 3.8$$

it can be deduced from eqs 3.6 and 3.8 that as turbine inlet temperature decreases, compressor pressure ratio decreases and the effect of this on mass flow outweighs that of a decrease of TIT to give a decrease of mass flow as temperature is reduced to decrease thrust.

The reduction in cycle pressure ratio will reduce the thermal efficiency whereas the propulsive efficiency will increase, but to a lower level, as mass flow decreases, compared to that which would have been obtained from a higher mass flow. Also, the reduction in mass flow will cause an increase in both the spillage and boattail drags associated with the inlet and nozzle respectively, as the airflow demanded by the engine is much lower than which the inlet can handle, and the nozzle area will have to close down to accommodate the reduced mass flow thereby providing more aft-end projected area. The reduction in both airflow and pressure ratio causes the compressor to operate over a wide range on its characteristics with poor efficiency at some points. Therefore, it is important to hold both airflow and pressure ratio at high levels while thrust is modulated with temperature.

The variable geometry turbine, within certain limitations, can alter the flow-temperature-pressure ratio matching characteristics of a gas turbine such that any of these variables, or a combination, can be controlled to improve cycle performance. There are other operating advantages of the variable area turbine which make it an attractive element for engine performance control.

### 3.3 Potential Operating Advantages of Variable Geometry

#### 3.3.1 Compressor

Variable geometry in compressors is used for surge control but is finding increasing use in other areas of engine performance control. As is shown in Fig. 3.8, the running line may be quite close to the surge line at low speeds at the nominal angle setting but by closing the vanes, the surge margin can be raised thereby increasing the margin for safe operation. It is possible that even though the surge margin may be adequate for safe engine operation, the compressor may be operating at low efficiency levels. The vane angles could be reset to give better efficiencies.

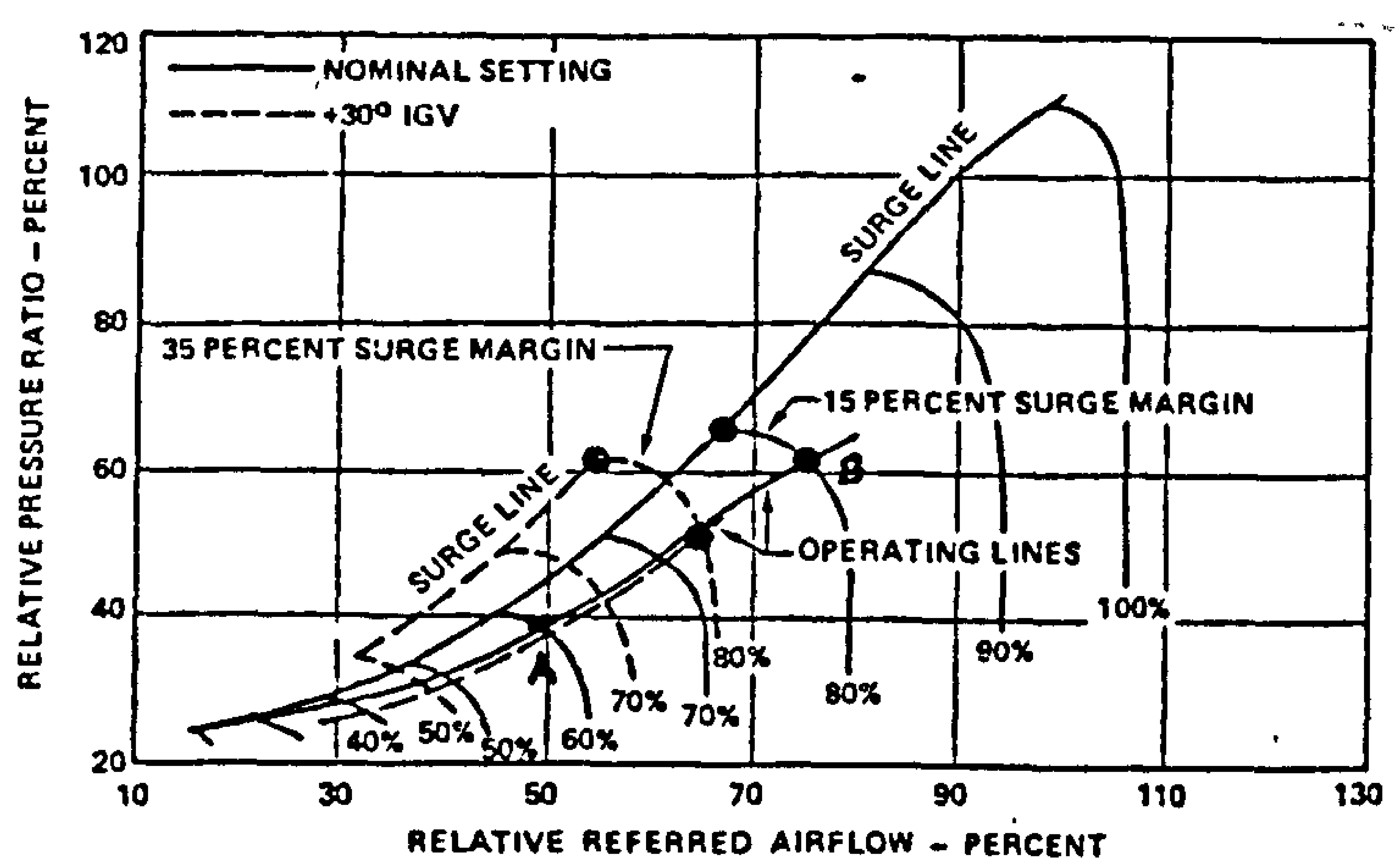



Fig. 3.8 Effect of IGV angle on surge margin [57]

An advantage of variable geometry compressors is that of reduced engine acceleration time. Since engine operating line is unaffected by compressor variable geometry, an acceleration from A to B, say, on Fig. 3.8 which would involve a 20 percent speed change at nominal vane angle setting can be accomplished at a lower speed change by using positive preswirl at point A to raise the initial speed, and negative preswirl at point B to lower the resulting speed. This has been the primary use of VIGVs in automotive gas turbines. This schedule of vane angles will give a higher surge margin at A and a lower margin at B compared with those obtained at the nominal angle setting.

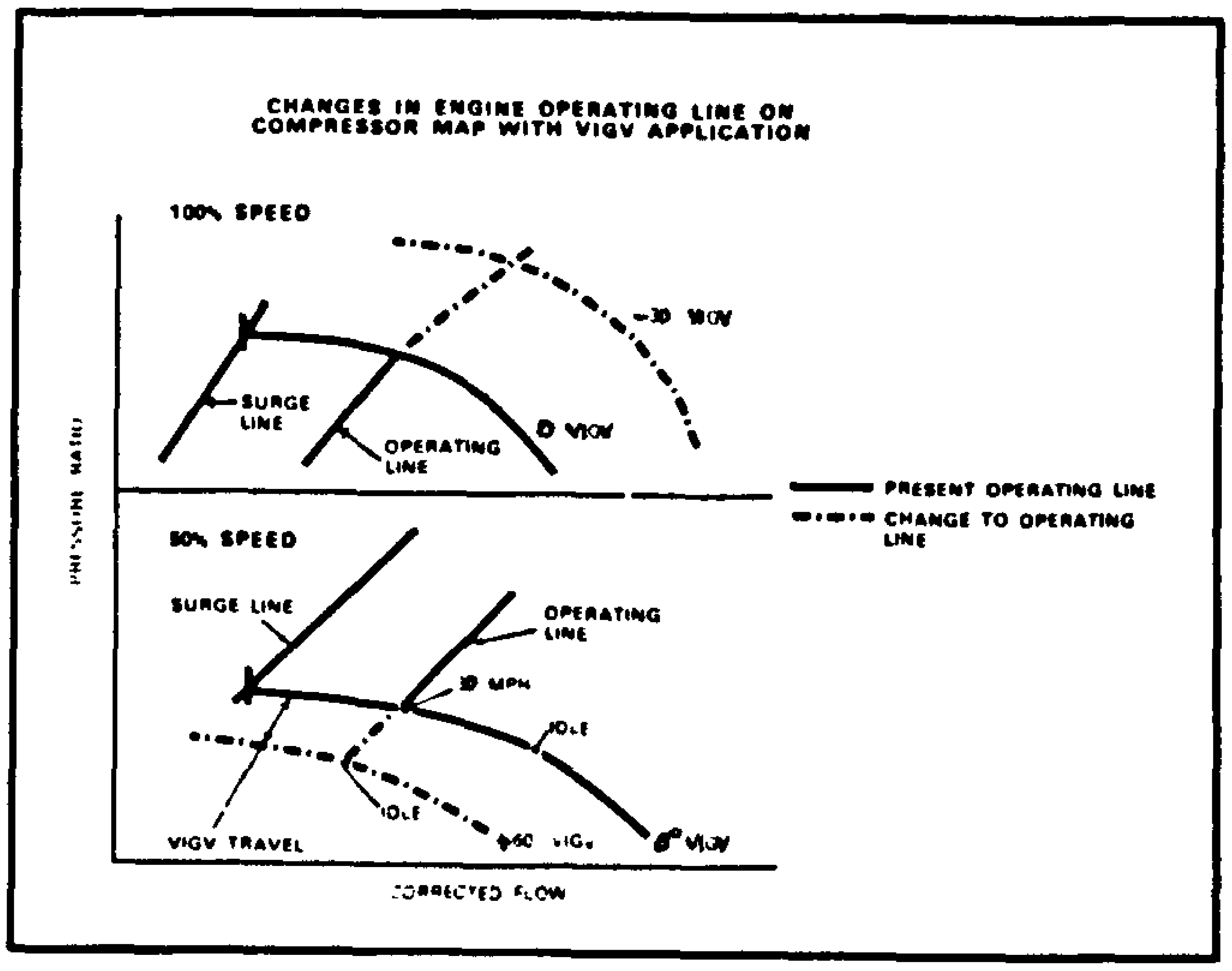
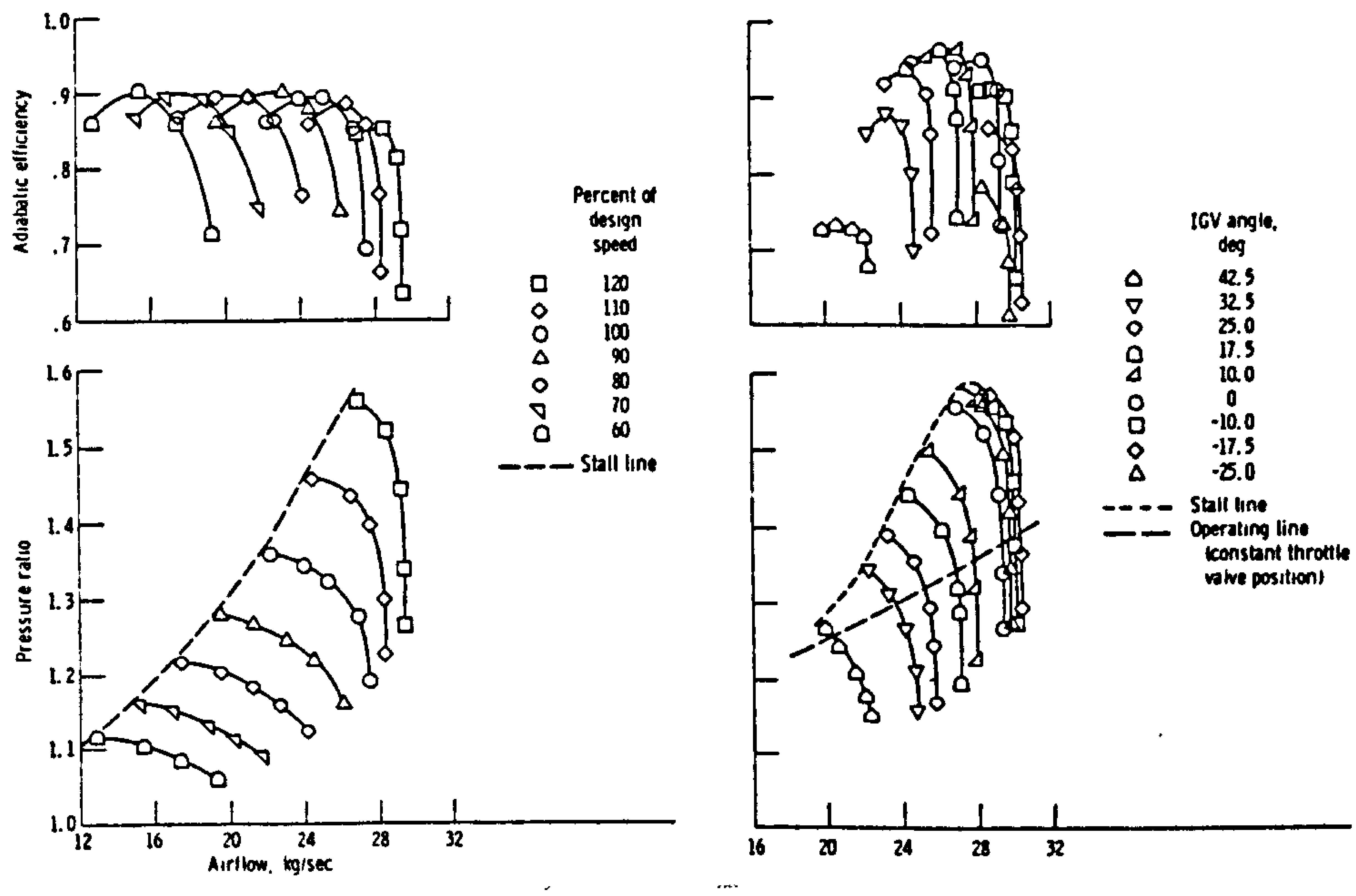



Fig. 3.9 Control of operating line by use of compressor geometry change [38]

As was mentioned in the previous chapter, compressor variable geometry can be used to augment power or improve sfc of automobile gas turbines by changing the swirl in the flow through the compressor. This is illustrated in Fig. 3.9 which shows partial operating characteristics at 50 and 100 percent speeds. By opening the vanes at maximum speed, compressor pressure ratio and airflow can be increased with an attendant increase in power. The operating advantage stems from the fact that the losses at part power are lower for a smaller engine than for a larger one. Therefore, by sizing the engine for a lower power, fuel economy is improved in the vehicle driving range.

At idle speed, fuel could be economized if the vanes are closed to give a lower pressure ratio and airflow. At 50 percent speed, power is reduced from the baseline operating value to the idle value by reducing turbine inlet temperature with an accompanying increase in airflow. Provided both turbine and compressor efficiencies are maintained, the engine idles at a lower fuel flow rate with positive preswirl.

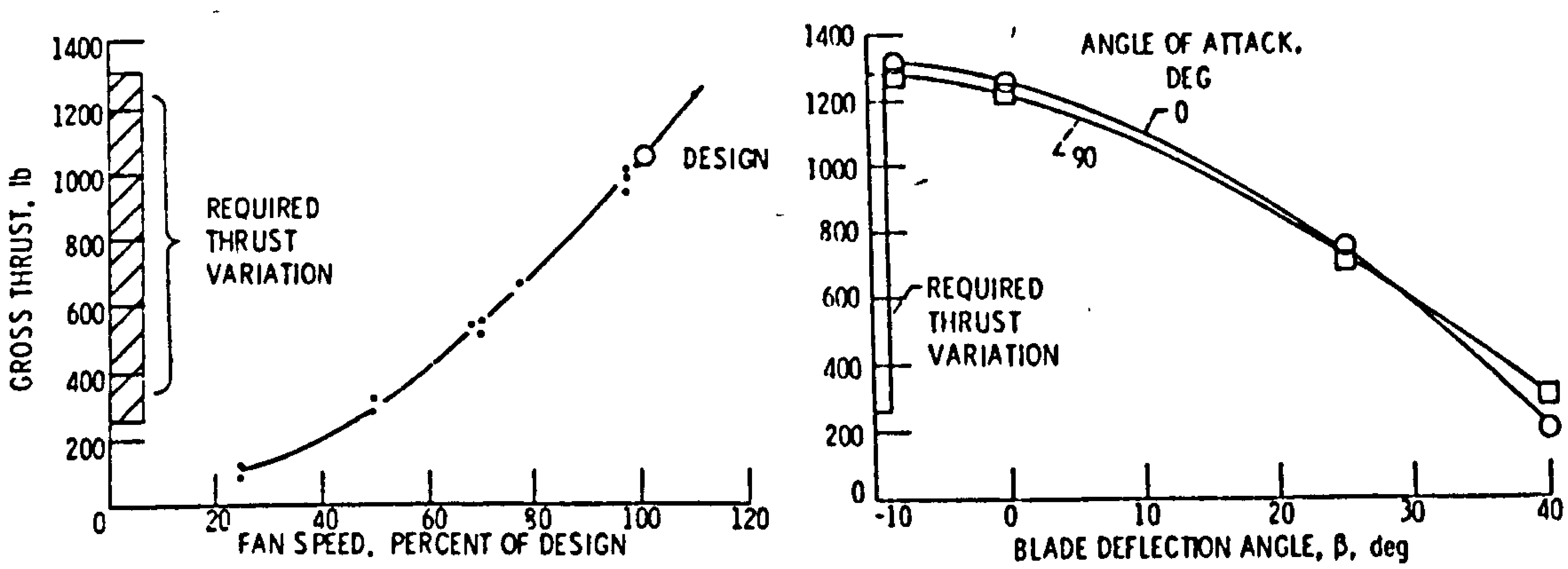
Variable geometry compressors can also be used to modulate thrust or power while maintaining engine speed constant. Fig. 3.10a shows the characteristics of a VIGV fan at an IGV setting of 0 degree for a range of speeds whilst the characteristics for the same fan are plotted in Fig. 3.10b for a range of IGV angles at 120 percent of design speed. A




(a) vane angles fixed (b) speed fixed

Fig. 3.10 Fan characteristics [31]

possible running line is also shown. The thrust requirement for a V/STOL aircraft on the approach to landing is shown in Fig. 3.11. The necessary speed variation to produce this thrust at fixed vane angle is shown in Fig. 3.11a, whilst Fig. 3.11b shows the required range of vane angles (at two values of inlet angle of attack) to meet the thrust demand at constant speed.




(a) speed control (b) VIGV control

Fig. 3.11 Thrust modulation obtained by speed and IGV control [61]

### 3.3.2 Turbine

Since a change of turbine area affects the engine cycle, there may be a restriction on the amount of area change used as limits on variables such as surge margin, spool speed, etc, on other components may be exceeded. Also, a turbine shares the work to drive the compression system with other turbines in a multi-spool engine, and in jet engines the nozzle is also involved in the pressure ratio split. Therefore, employing a variable area turbine may require one or more of the other components in the expansion system to vary its area as well. None the less, the advantages to be gained with a VAT could be enormous.

Inlet-Engine Matching

The inlet of an aircraft engine has an airflow characteristic independent of that of the engine and when the two are installed in the aircraft, they will match efficiently at some operating point whereas a mismatch at other points could lead to excessive drag on the aircraft with poor system performance. This is most critical for high Mach aircraft especially when flying at subsonic speeds where the engine has been throttled back considerably.

With high speed aircraft, the inlet would normally be sized to pass the maximum airflow encountered at altitude, and this is most likely to occur at a high Mach number and most probably with the afterburner on. The nozzle area will be quite large and therefore aft-end drag will be minimal. At the low power settings required for subsonic cruise, the inlet air handling capability exceeds engine demand and therefore some air is spilled around the intake resulting in a large increase in drag. Also, the exhaust nozzle area,if variable, will have to close down due to the lower exhaust flow, giving rise to a large aft-end drag.

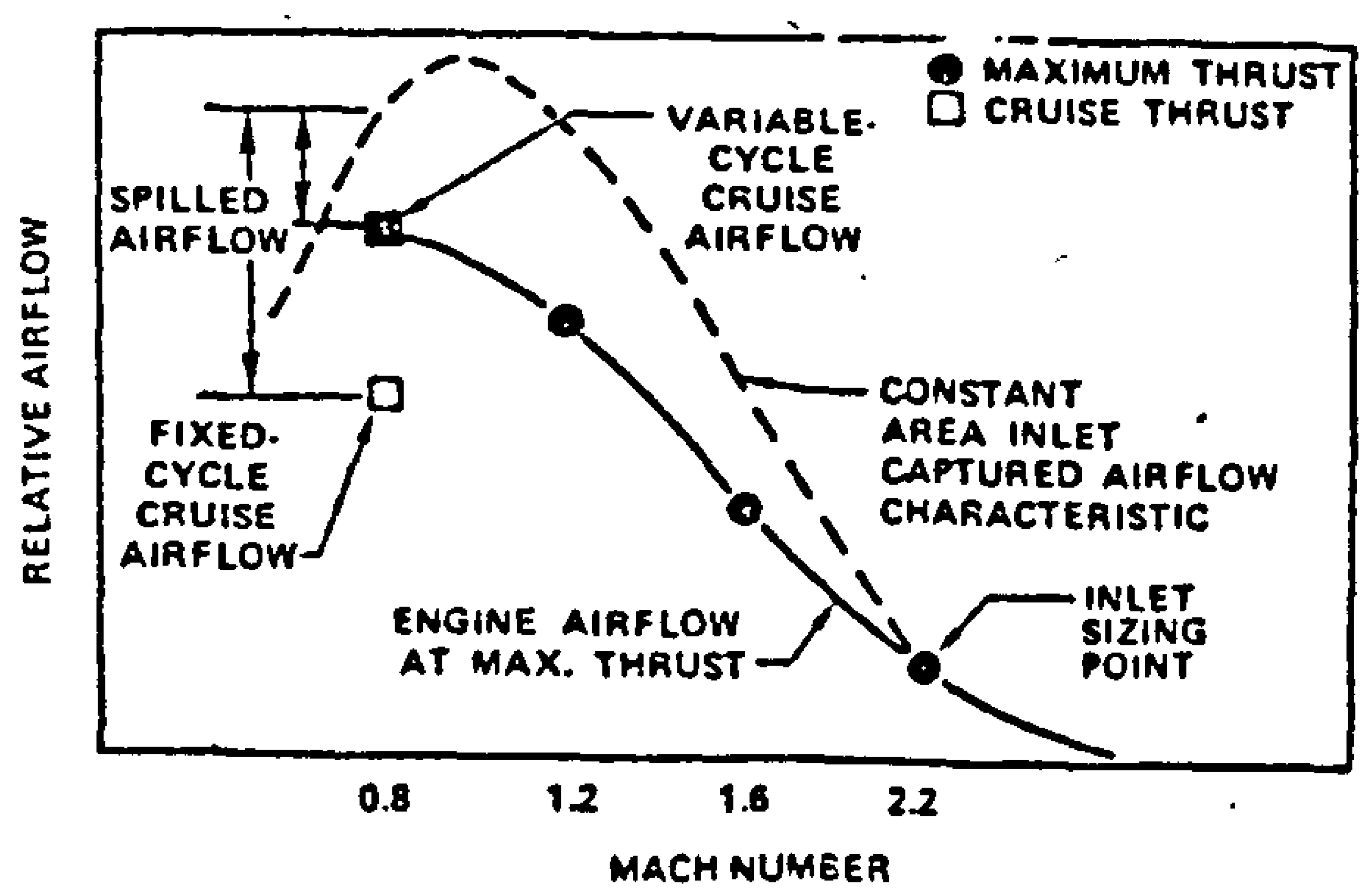



Fig. 3.12 Inlet-engine airflow characteristics [57]

Figure 3.12 shows the problem of inlet-engine incompatibility. Spillage drag can be eliminated if the fan or LP compressor can be controlled with the aid of a VAT to swallow all the airflow passed by the inlet, as shown by the dashed line, or can be reduced if the engine airflow is scheduled to follow the solid line. In both cases, the aft-end drag is reduced as the exit area is larger to pass a higher mass flow. This problem of inlet-engine mismatch is solved in present day aircraft systems by the use of complicated variable geometry inlets. A variable area turbine coupled with a simpler inlet could be more attractive.

## Surge Control

Another promising operating advantage of the VAT is that of surge control. Variable geometry compressors are used extensively in current engines to raise the surge line at low and medium spool speeds but at the expense of a reduction in airflow at a given spool speed, Fig. 3.8. If low speed operation is caused by high Mach flight, then the

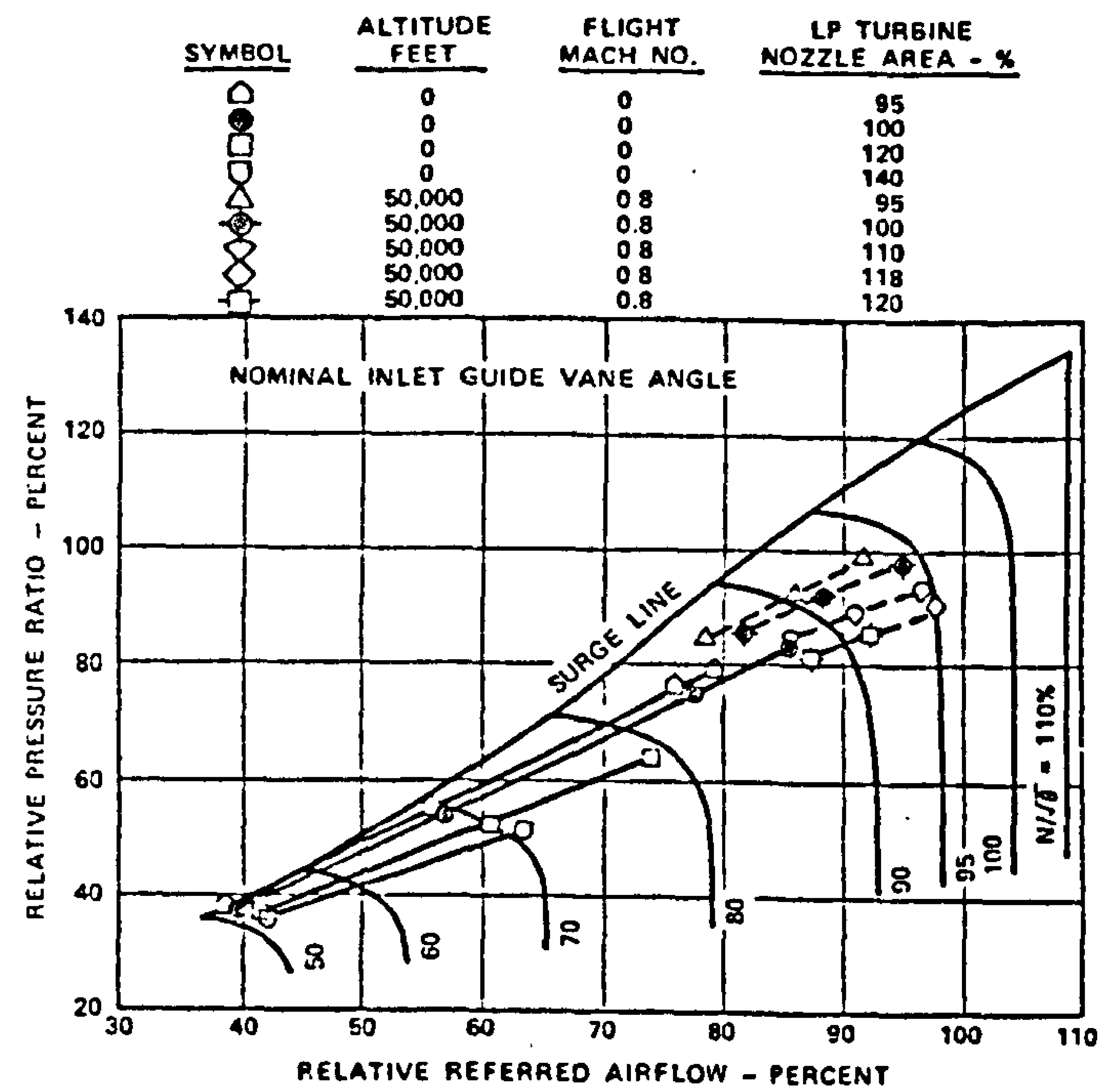



Fig. 3.13 Variable area turbine control of operating line [57]

available power is reduced. On the other hand, a VAT can be used to lower the operating line with an accompanying increase in mass flow at fixed rotational speed, Fig. 3.13. As shown in Fig. 3.14, the VAT is an equally strong candidate for surge control as is the variable geometry compressor. Since the deployment of both compressor and turbine variable geometry to control surge could be restricted by limits on some variables, a combination of variable geometries in these two components could be used effectively for surge control such that no limits are encountered.

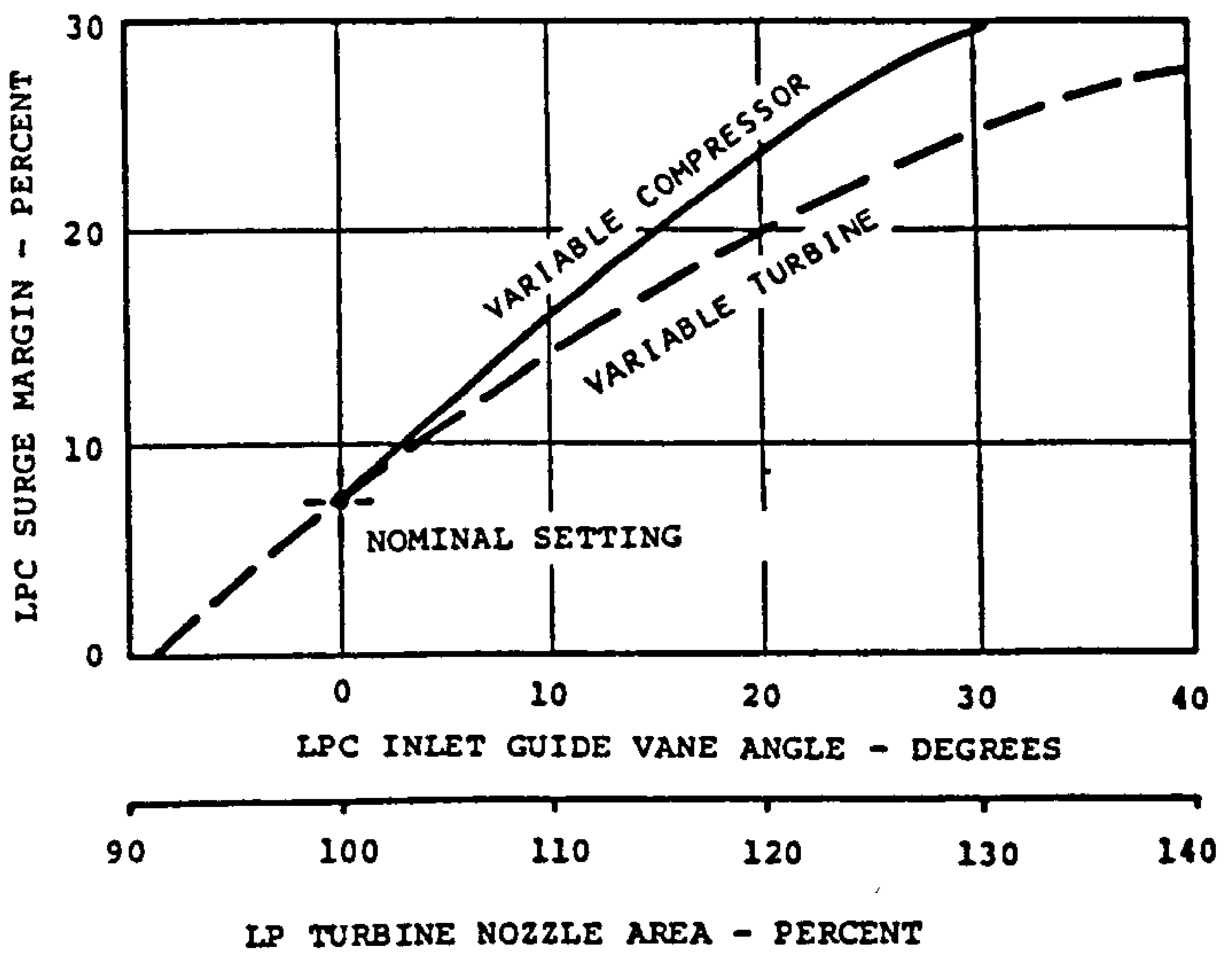



Fig. 3.14 Comparison of surge margins obtained by compressor and turbine geometry change [8]

## Extended Performance

It may not be possible for a gas turbine engine to deliver more thrust or power at certain operating conditions as a result of one or more components reaching an operating limit. The limits encountered are usually spool speed and TIT. Fig 3.15 is an illustration of possible restrictions

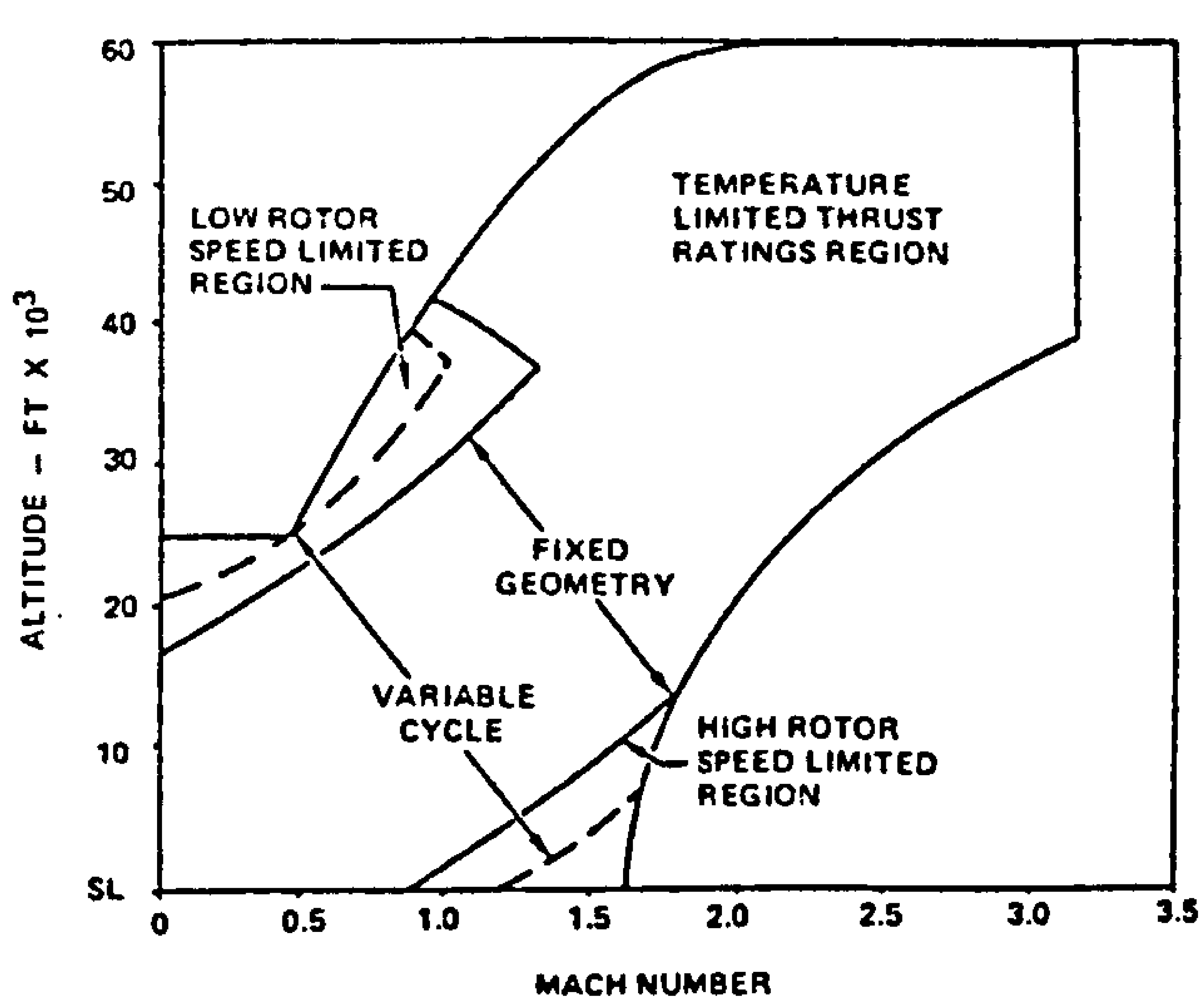



Fig. 3.15 Extension of flight corridor by operating at engine limits [57]

put on engine performance for a particular flight envelope. Since a nozzle, either propelling or turbine, immediately downstream of a compressor turbine controls turbine work

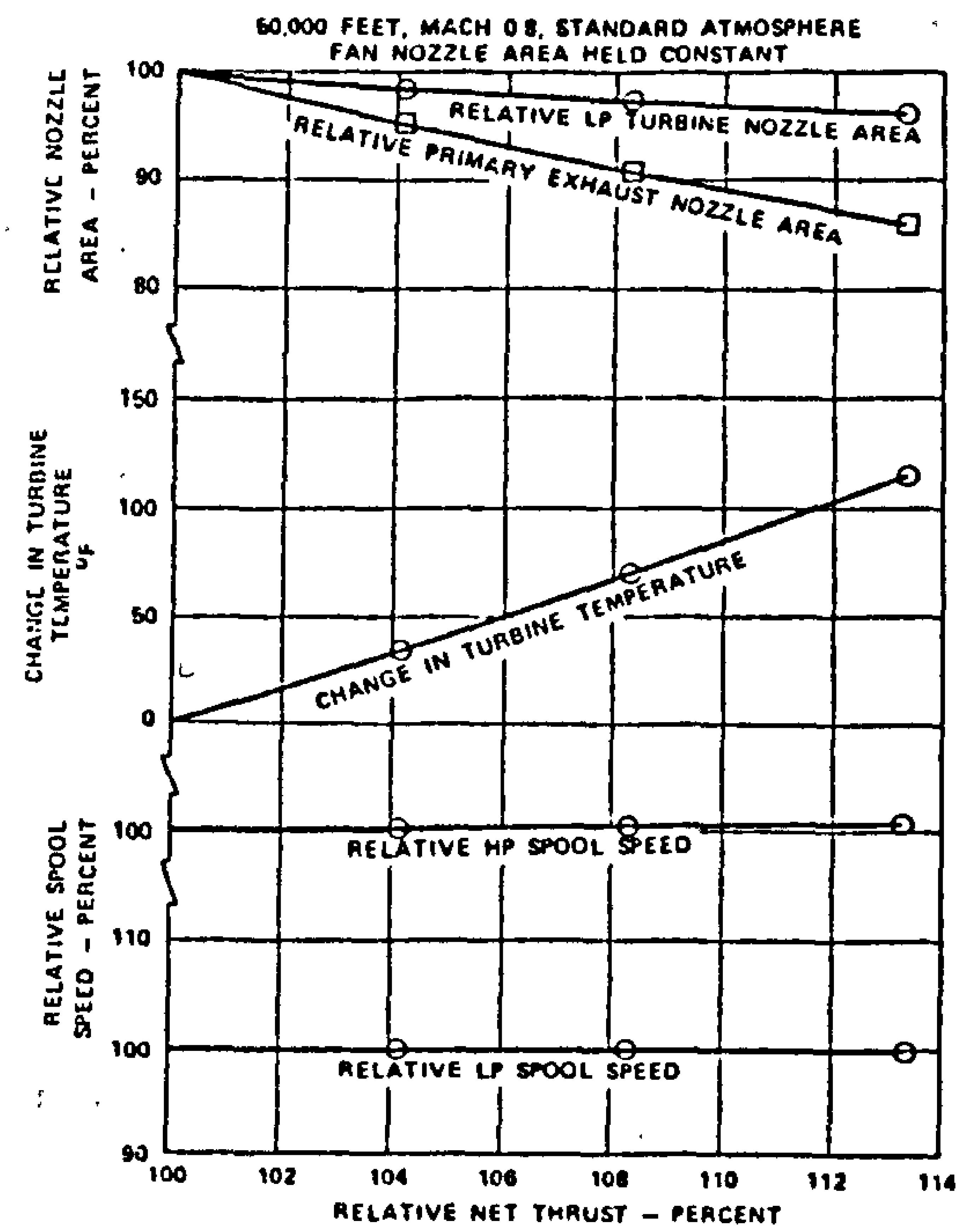



Fig. 3.16 Potential thrust increase by deployment of variable geometry [57]

function, a VAT can be used to improve engine performance by increasing one or more variables while keeping the limiting variable at bay. Fig. 3.16 shows that by using variable geometry in the LP turbine and core nozzle of a twin-spool turbofan, thrust can be increased by up to 13 percent by increasing turbine inlet temperature while maintaining the LP spool at its limiting value.

From the above discussions it can be seen that variable geometry can significantly improve the off-design performance of gas turbine engines.

# CHAPTER 4

# INSTALLED PERFORMANCE

A jet engine manufacturer sells thrust to his customer whilst the airframe manufacturer tries to minimize airframe drag. The engine manufacturer would quote the amount of gross thrust his engine will deliver when tested at static conditions at either sea level or altitude, whilst the airframe drag would be obtained from tests conducted on a wind tunnel model. Therefore, when an engine is installed in or integrated with an airframe, thrust minus drag is of primary interest as this parameter partly defines the overall performance of the aircraft.

The magnitude of engine thrust as installed has been found to be substantially lower than that obtained in the test cell. This is so as losses are introduced when the engine is installed in the aircraft. The sources of these installation losses will become apparent as the chapter progresses. The magnitude of these losses can greatly impact the choice of the cycle selected for a given duty and as these losses become increasingly important as the engine operates further away from the design point, a better understanding of their magnitude and characteristics is mandatory if future high performance aircraft are to achieve their planned performance goals. Therefore, a method has to be developed whereby these losses can be accurately predicted in order to effectively integrate engine and airframe to meet the specified mission requirements. Before describing the method by which the installed thrust is arrived at, the problem of force identification and accounting is first presented with a view to enlightening the reader with the difficulty of attaining a specified thrust requirement.

## 4.1 Sources of Performance Loss

The successful development of any high performance aircraft system greatly depends on how well the propulsion system installation corrections are accounted for, predicted, and measured right from conceptual studies through proposal, evaluation, wind tunnel and flight tests.

During the conceptual and proposal phases where many cycles are studied and compared, it is quite important to be able to predict the installation losses accurately as it is necessary to remove the installation effects to understand what is required of the basic cycle to meet a given set of requirements [5]. If unrealistic or incomplete loss coefficients are used in this process, it becomes difficult or almost impossible for the inlet or nozzle designer to successfully produce a component that will match with the badly chosen cycle to obtain the required system performance. Even a good engine installed in a good airframe could give an inferior or non-competitive aircraft if not properly installed.

It is highly unlikely that there would be installed performance data for the particular combination of inlet, engine, nozzle, and airframe being considered and therefore, the integration analysis would have to be derived from independently generated sets of data for each constituent element. The predictions range from extrapolation of existing data to large scale model testing [44] and the problem is compounded by the fact that the installation losses vary considerably throughout the flight envelope in both overall magnitude and relative importance. Existing data will only be useful if a similar component or engine or airframe installed in a similar manner exists as for the proposed component or engine or airframe. Testing will only occur after the engine and airframe have been chosen and will be used to further develop the integration technique.

The installation losses are primarily caused by inlet and exhaust inefficiencies and the mutual interaction between the engine and airframe. Other factors include power bleed from engine and safety margins of one form or another. Reference 14 lists all possible causes for the change of thrust from the uninstalled to the installed.

The causes for the loss of performance of the inlet and nozzle will be explained later on but as can be seen from Fig. 4.1, the fact that a simple intake and nozzle, a bellmouth and a convergent nozzle, say, may be used for static tests whereas these components could be complex when installed accounts for a difference in uninstalled and installed thrusts. Also, the tests are carried out at static conditions whereas the installed thrust may be obtained at forward speed where the external flow over the airframe may affect the internal performance of the engine.

Power may be extracted from the engine in the form of compressed air for various purposes such as providing conditioned air for the crew and/or cabin, driving pneumatic actuators, powered lift, boundary layer control, etc. This air is lost to the cycle causing a loss of thrust. Also, shaft power may be removed from the engine to provide mechanical and electrical power for driving equipment such as pumps and radar. These powers extracted also account for part of the difference between uninstalled and installed thrusts.

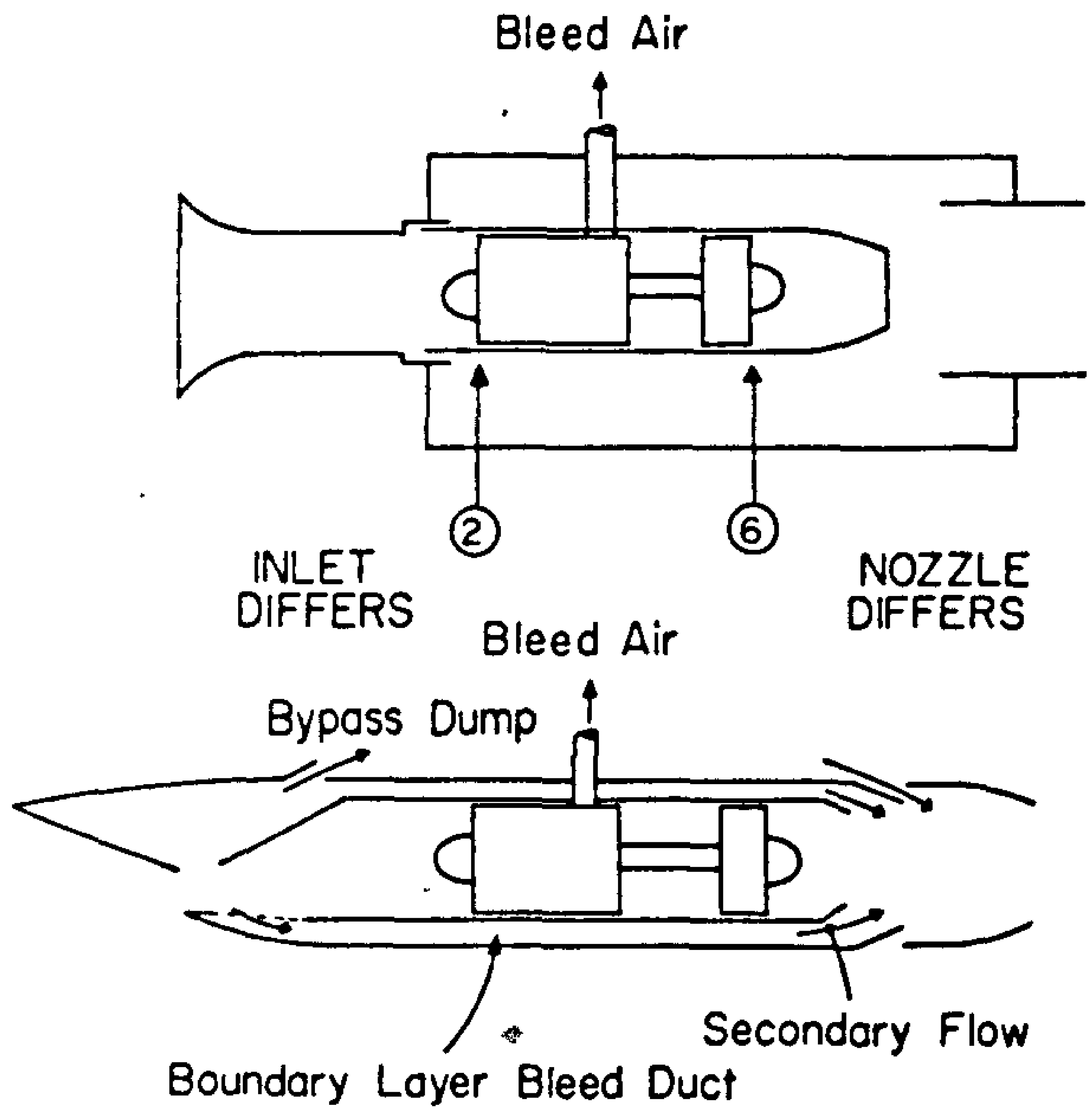



Fig. 4.1 Possible uninstalled and installed engine configurations [14]

Inlet distortion may be another source of installation loss. Distorted flow may be caused by turbulent air or the ingestion of foreign gases such as steam from a catapult launch or exhaust gases from a rocket. These are absent when the engine is in the test cell and as a result adequate stall margin may have to be built into the compressor to avoid engine surge.

## 4.2 Accounting of Propulsion Forces

An accurate performance prediction of an integrated aircraft can only be achieved if the aerodynamic and propulsion forces are properly accounted for, and in order to account for these forces, they must be defined precisely and consistently in all phases of the aircraft development process. The concept of force accounting has been used for several years and has proved to be an invaluable tool in the quest for better prediction of overall aircraft and propulsion system performance.

Force accounting can be simply defined as the system used to define, relate, and integrate the aerodynamic and propulsion forces acting on an aircraft. Because it is impossible to determine in one calculation or test the various forces acting on an aircraft system that includes the actual nozzle and inlet, force accounting gives the engineer the flexibility of determining these forces separately from other sources during or after the optimization of the individual components. Therefore, a well defined force accounting system is necessary to ensure that the performances predicted for each of the various elements, that is, inlet, nozzle, turbomachinery, and airframe, are properly integrated to give an accurate prediction of overall system performance [4].

The concept of force accounting is simple and its application should not pose any difficulties, but there are immense problems encountered when information about aircraft performance is being communicated between organizations such as engine and airframe companies. The root of these problems lies in the fact that force accounting is a concept and not a definition and therefore there are no international or national standards that one can use when carrying out the necessary calculations. As a result, it is difficult to compare the performances of similar configurations obtained from different sources as the nomenclature and data formats vary from one organization to another.

Problems may be caused when one tries to evaluate subsystem performance due to a lack of standardized reference conditions for separating propulsion and aerodynamic forces. The performances of the various elements are not readily visible as the procedure for selecting split planes is not customary. Also, if wind tunnel tests are not conducted at their proper reference conditions, then all performance results obtained are meaningless. Therefore it is important that continuity be maintained in tracking aircraft performance throughout the development process if substandard performance is to be avoided once the aircraft leaves the production line. Hence, in addition to the requirement for predicting accurately the overall thrust-minus-drag of an aircraft system, a force accounting system should effectively provide visibility of the individual elements and subsystems with consistent definitions throughout the entire aircraft development program.

### 4.3 Definition of Propulsion Forces

Two of the items that most frequently cause confusion and differences of opinion concerning force accounting methods are definition of forces and thrust/drag split [4]. The accounting procedure or bookkeeping (of forces) methods should clearly define the division of responsibility between the engine and airframe manufacturers. For a given flight condition defined by altitude and flight Mach number, a reference condition should be defined and operation away from the reference condition will result in thrust and drag increments. Nozzle pressure and area ratios and aircraft attitude are the three most commonly used variables that define the reference condition. The change of forces at conditions other than the reference can be distributed by simply adding those forces that result from throttle movement to thrust, and to the aircraft drag polar, those changes caused by control surface movement.

The internal forces and those parts of the external forces that arise from throttle movement, that is, installation loss drags, form the throttle dependent forces. The internal force comprises the uninstalled engine net thrust and the internal installation losses. The uninstalled engine net thrust is defined as the difference between the nozzle static gross thrust and engine streamtube ram drag. The static gross thrust is the uninstalled or test cell gross thrust for the reference nozzle at the operating total pressure ratio and the actual nozzle mass flow rate. The engine streamtube is that part of the airflow captured by the inlet which takes part in the engine cycle. This includes the airflow at the engine face and whatever airflow that bypasses the engine but is re-introduced in the jet pipe or nozzle. The excess airflow captured by the engine and ducted overboard through bleed or bypass systems is not part of the engine streamtube.

The internal installation loss is defined as the change in internal force arising from the change in inlet and nozzle internal performances and the change in bleed power requirements when the engine is installed. This accounts for the effects of inlet pressure recovery and nozzle thrust coefficient for the real inlet and exhaust nozzle, respectively, and also for the bleed air and power required by the airframe.

The throttle dependent external force is the difference between the installed propulsive thrust and the internal force as defined above. This accounts for all external installation losses caused by the interference of the external flow over the aircraft with the engine inlet and exhaust flows, the change in momentum of the excess air captured by the inlet, and the change in the trim drag associated with operating the propulsion system at conditions other than the reference.

The total throttle-dependent force acting on an aircraft at a specified flight condition and angle of attack can be obtained by summing up all relevant internal and external forces to obtain

$$F_{INST} = F_G - D_R - \Delta F_{\eta R} - F_G\left(1 - \frac{C_{FGR}}{C_{FGT}}\right) - \Delta F_{POW} - \Delta D_{INL} - \Delta D_{AFT\text{-}END} - \Delta D_{TRIM} \qquad 4.1$$

where,

| | | |
|---|---|---|
| $F_{INST}$ | = | Installed propulsive thrust |
| $F_G$ | = | Uninstalled gross thrust |
| $D_R$ | = | Ram drag |
| $\Delta F_{nR}$ | = | Thrust loss due to pressure recovery |
| $F_G(1-C_{FGR}/C_{FGT})$ | = | Thrust loss due to nozzle internal performance |
| $\Delta F_{POW}$ | = | Thrust loss due to airbleed and power offtake |
| $\Delta D_{INL}$ | = | External drag increment between operating reference and operating conditions, due to inlet |
| $\Delta D_{AFT-END}$ | = | External drag increment between operating reference and operating conditions, due to exhaust system |
| $\Delta D_{TRIM}$ | = | Change in trim drag associated with operating the propulsion system at conditions away from the referenced |
| $C_{FGR}$ | = | Gross thrust coefficient of the installed nozzle |
| $C_{FGT}$ | = | Gross thrust coefficient of the test nozzle |

Fig. 4.2 shows a schematic of an installed engine. The possible areas of thrust loss are also shown. If the drag from the trimmed drag polar is added to Eq. 4.1, the total force acting on an aircraft in level flight along the flight direction is obtained. The thrust loss due to trimming the aircraft as a result of throttle movement is usually very small and therefore can be neglected.

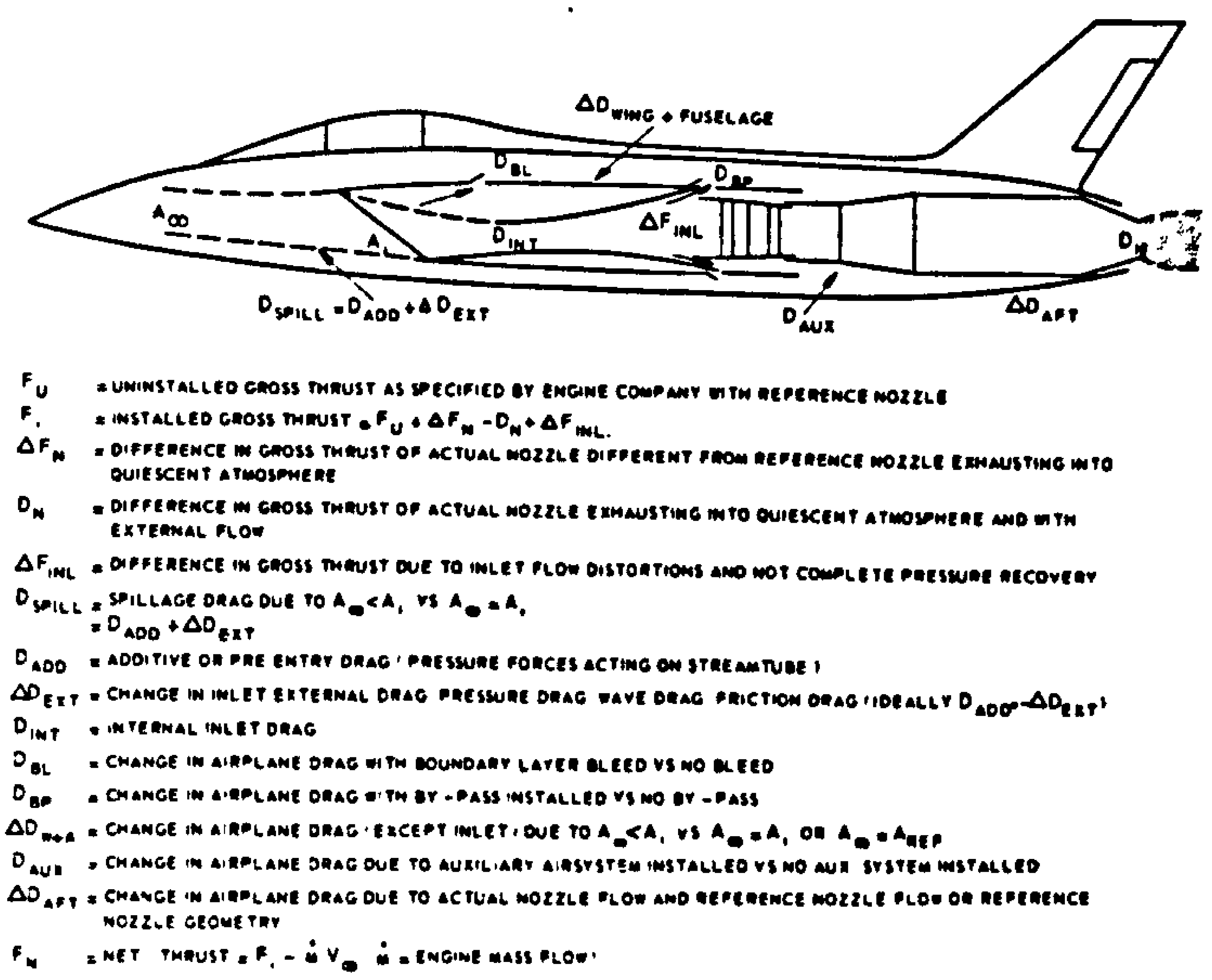



Fig. 4.2 Aero/propulsion forces on an aircraft

Equation 4.1 is an expression for the installed propulsive thrust but is not standardized. A possible reason for the lack of standardized definitions is that the installation losses are configuration dependent, that is, they depend on the type of aircraft and the manner in which the engine is installed. As a result, there are many thrust and drag terms which are not universally defined. Individual aircraft companies have developed their own methods for defining and integrating aero/propulsion forces based on accumulated experience. Some force accounting methods may be more suitable than others for a particular configuration; so are the test techniques. For any given airframe/engine configuration, there is a unique value for the total propulsive force and therefore what is important is that any force accounting procedure adopted should predict this value accurately.

## 4.4 Reference Conditions

As with the definition of forces, it is difficult to select suitable reference conditions when accounting for the aero/propulsion forces. This is so as the reference conditions are configuration dependent and what is convenient for analytical studies may not be convenient for wind tunnel tests. Also the reference conditions will depend on the visibility required, that is, element, subsystem, or system.

Reference 4 defines some possible reference conditions. The operating reference conditions would yield Eq. 4.1 as the propulsion system force and are therefore the conditions to which by definition the drag polar corresponds. Therefore, they are representative of realistic flight conditions. These conditions correspond to a specified engine power setting, usually the maximum.

## 4.5 Inlet and Exhaust Performance

The bulk of the installation losses arise from the fact that the installed inlet and nozzle are not normally used with the engine when in the test cell. The complexities that may be added to these components also introduce losses in the system and therefore, it becomes difficult if not impossible to simulate the flow conditions at the engine face and jet pipe. Also, since there is flow over both inlet and nozzle when in flight and not when in the test cell, the mutual interaction between the engine and external flow alters both the internal and external performances of these components, more so the nozzle. For analytical studies, especially for preliminary design analysis, the performances of inlets and nozzle can be obtained from maps or characteristics from which component performance can be envisaged in response to changes in geometric or thermodynamic variables. Inlet and nozzle performance representation will be discussed next in the light of evaluating the various loss components. Fig. 4.3 shows the various facets of inlet and nozzle performance prediction to be carried out to arrive at propulsion system performance.

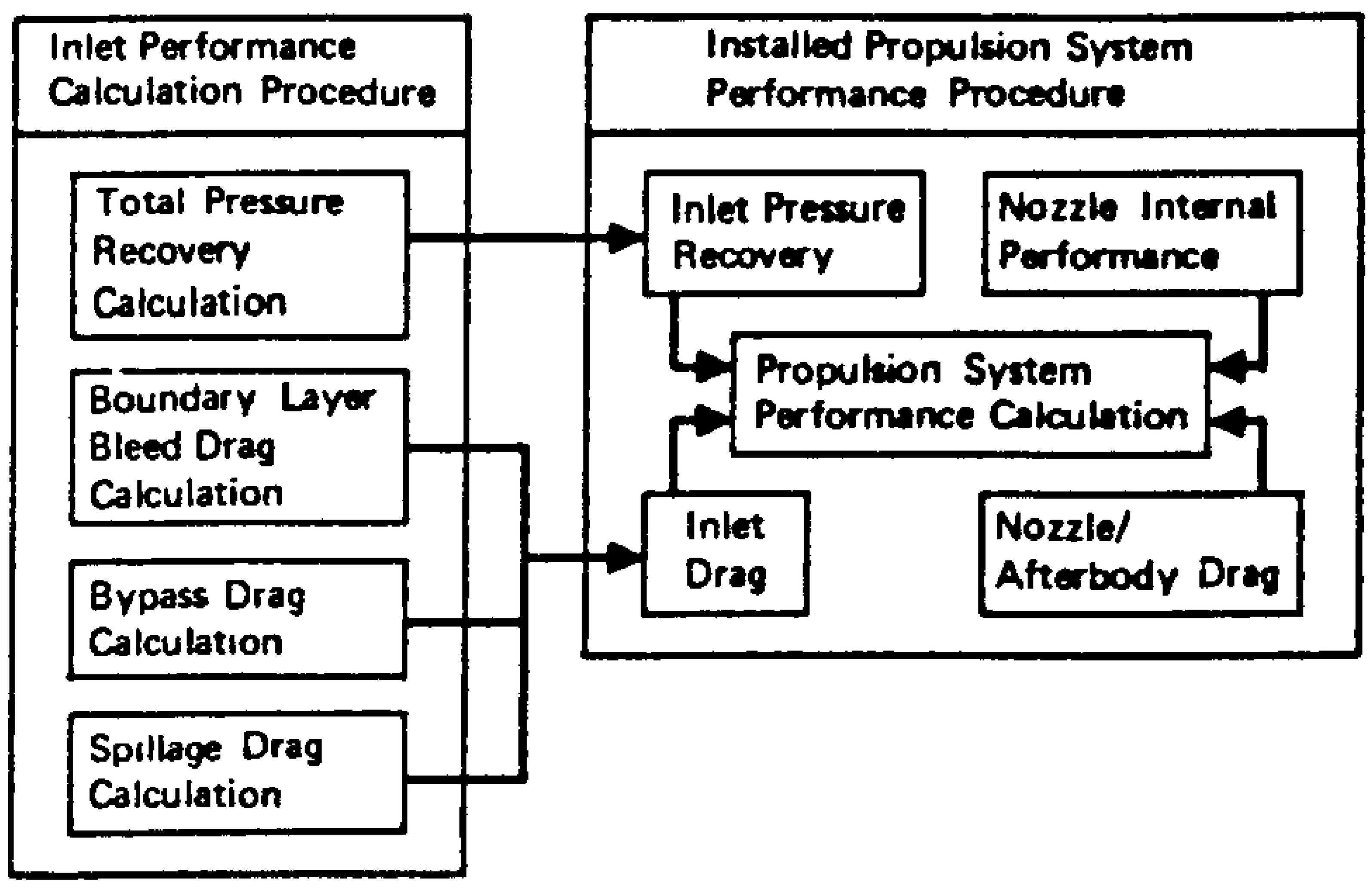



Fig. 4.3 Installed performance procedure [4]

Inlet Performance

Inlet performance comprises pressure recovery and throttle dependent inlet drag. Inlet pressure recovery is a function of inlet geometry, subsonic diffuser efficiency, and pressure losses for any shock system present. Pressure recovery is normally presented as a function of Mach number but is also a function of the mass flow ratio into the air induction system.

Since inlet performance is obtained from isolated tests, a convenient way to represent the performance due to inlet recovery is to plot pressure recovery as a function of area ratio for different Mach numbers as shown in Fig. 4.4a (Item 2). However, the inlet has to be matched with the compressor and therefore it is more convenient to represent inlet performance in terms of the non-dimensional mass flow at engine face as shown in Fig. 4.4a (Item 3). In the case where test data are not available for the particular inlet type, pressure recovery can be obtained from theoretical and semi-empirical procedures using such data as those appearing in Fig. 4.4a (Item 1). All prime variables must be taken into consideration.

Inlet drag is comprised of

1. Spillage drag,
2. Bleed drag,
3. Bypass drag, and
4. Secondary drag.

Spillage drag includes the additive and forebody drags. The additive or pre-entry drag is due to the desire to use freestream conditions to define ram drag whilst forebody drag considers the effects of friction on the forward intake structure, cowl, nacelle, etc. The bleed drag accounts for the change in momentum of that part of the excess air captured by the inlet but which is bled-off to control the boundary layer in order to promote efficient

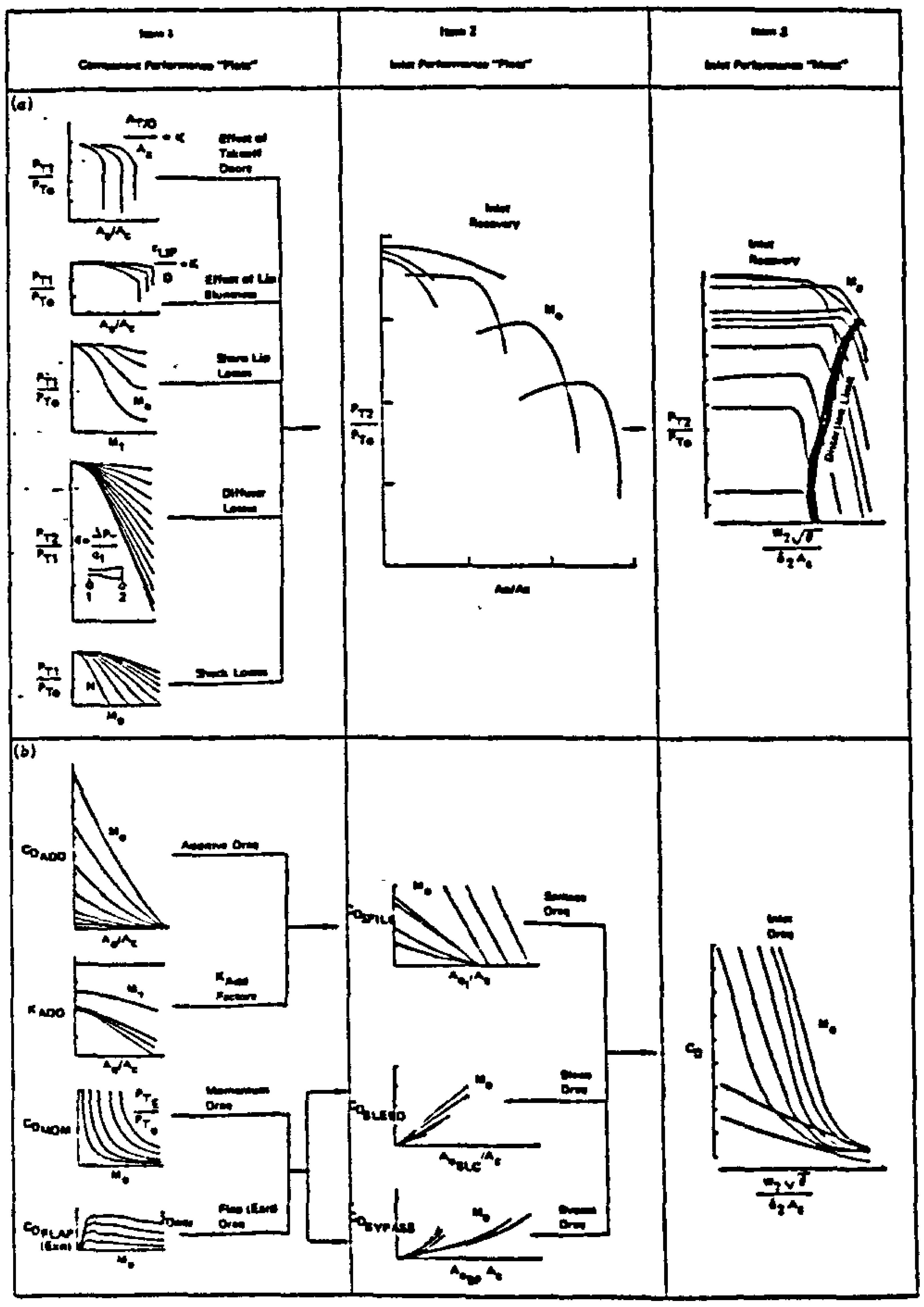


Fig. 4.4 Inlet performance representation [4]

diffusion and uniform flow to the engine. Also some of the air swallowed by the inlet may be dumped overboard to aid in maintaining the position of the normal shock and the drag resulting from the change in momentum of this air is termed the bypass drag. A part of the inlet airflow may be needed for services not connected with the engine cycle, for example, cooling. The loss of momentum of this airflow results in thrust loss which is accounted for by the secondary drag which at times is lumped with bypass drag.

The throttle dependent inlet drag is defined as

$$\Delta D_{INL} = 1/_2 \rho V_o^2 A_c C_{DINL} \qquad 4.2$$

The inlet capture area is determined from the maximum airflow requirement of the engine. This maximum airflow occurs at static conditions and at cruise, operation of the capture area so obtained may result in excessive spillage. Therefore, it may be necessary to size the inlet at altitude with auxiliary doors providing the extra airflow at takeoff. (With supersonic inlets, a compromise is usually struck to give acceptable performance in both flight legs.) The drag coefficient can be represented as shown in Fig. 4.4b (Item 3). An example of airflow area variation from that of the capture area and the associated drag distribution are shown in Fig. 4.5 for a supersonic inlet. It can be seen that variable geometry would be needed to improve the performance of this inlet.

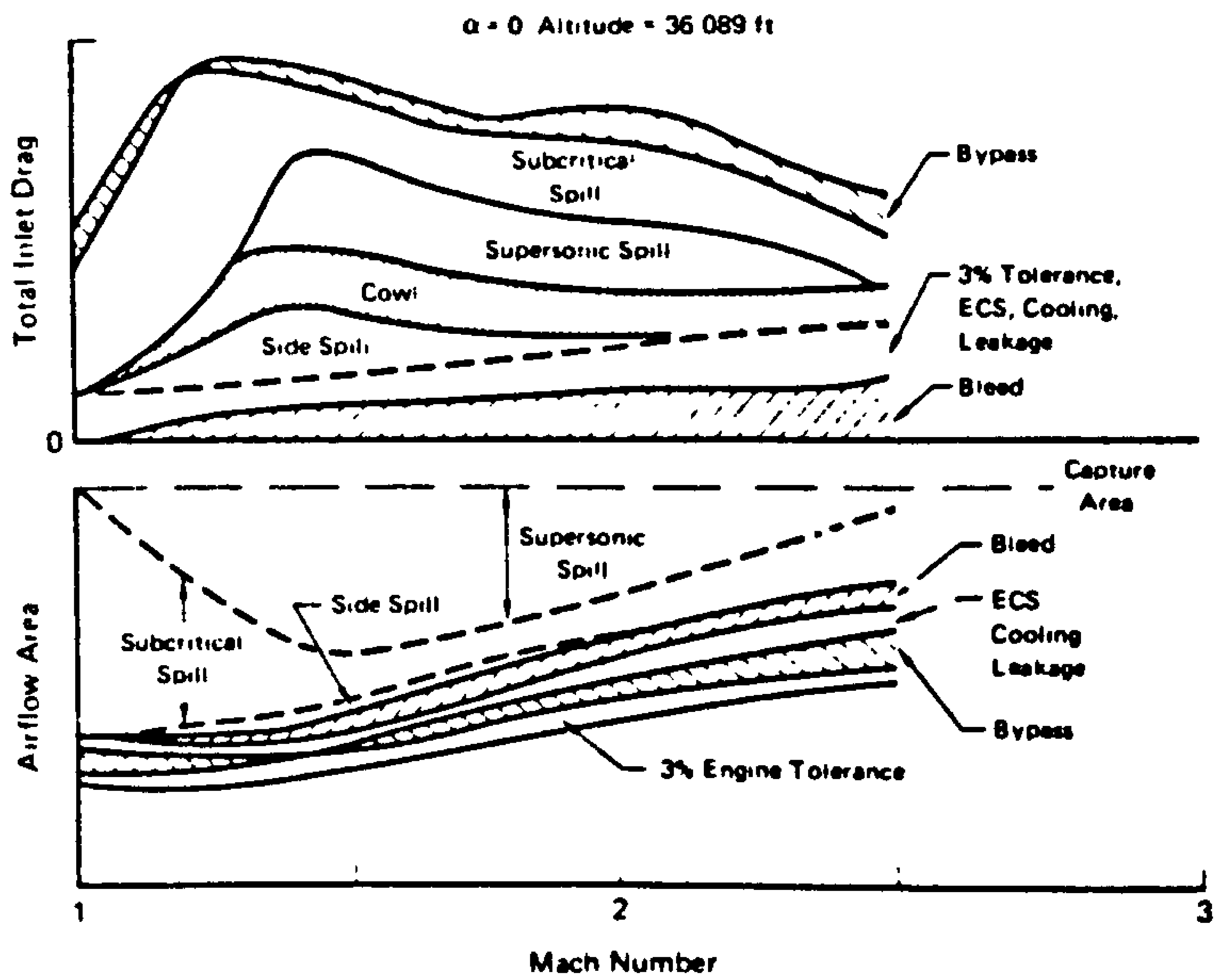



Fig. 4.5 Inlet-engine airflow matching considerations [68]

## Aft - End Characteristics

The performance of the exhaust system is dependent on the afterbody geometry and the thermodynamic properties of the exhaust gases. It is separated into an internal performance governed by nozzle area ratio and flap angle and an external drag which is usually interwoven with the airframe drag.

As was mentioned earlier, the propelling nozzle in the test engine may be different from the installed, and therefore a correction is to be done to account for the difference in nozzle gross thrust coefficients or internal performances. The gross thrust coefficient, $C_{FG}$, is defined as the ratio of actual gross thrust to the ideal gross thrust, that is,

$$C_{FG} = F_G/F_I \qquad 4.3$$

and therefore provides a link between the installed and uninstalled engine gross thrusts. Since cycle analyses are one dimensional, $C_{FG}$ mixes one dimensional concepts with three dimensional flow and therefore the pressure and temperature profiles in the jet pipe will have to be either mass or area weighted to obtain $C_{FG}$.

The gross thrust coefficient takes into account losses due to internal wall divergence, flow separation, friction, leakage, shock, over- and under-expansion. For the same ideal and actual mass flows, $C_{FG}$ reduces to CV, the velocity coefficient, and is normally represented in terms

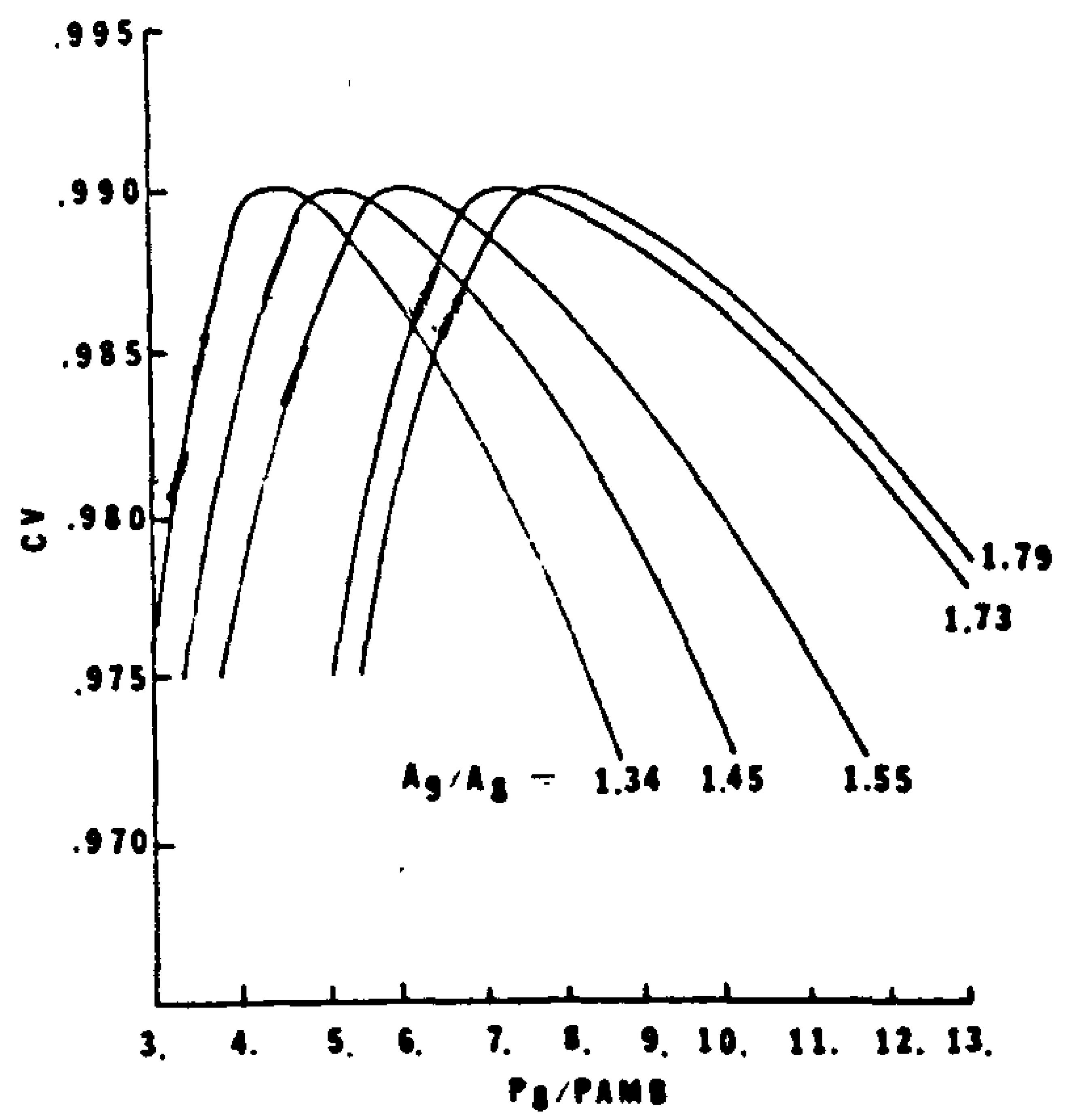



Fig. 4.6 Nozzle internal performance map [27]

of the nozzle area and pressure ratios as shown in Fig. 4.6, using the station numbering in Fig. 4.7. The peaks in the curve represent fully expanded flow conditions. However, this thrust or velocity coefficient has to be corrected as a result of the mutual interaction of the exhaust and external flows.

The aft-end performance also includes corrections due to

1. External flow,
2. Afterbody or boattail,
3. Blunt base, and
4. Exhaust interference.

Fig. 4.8 shows the various areas that account for aft-end drag. The exhaust plume of the nozzle in the test cell is free to occupy a volume in space whereas in flight, the nozzle flow interacts with the external flow which changes both the internal and external performances of the afterbody. Some types of nozzle are more influenced by the

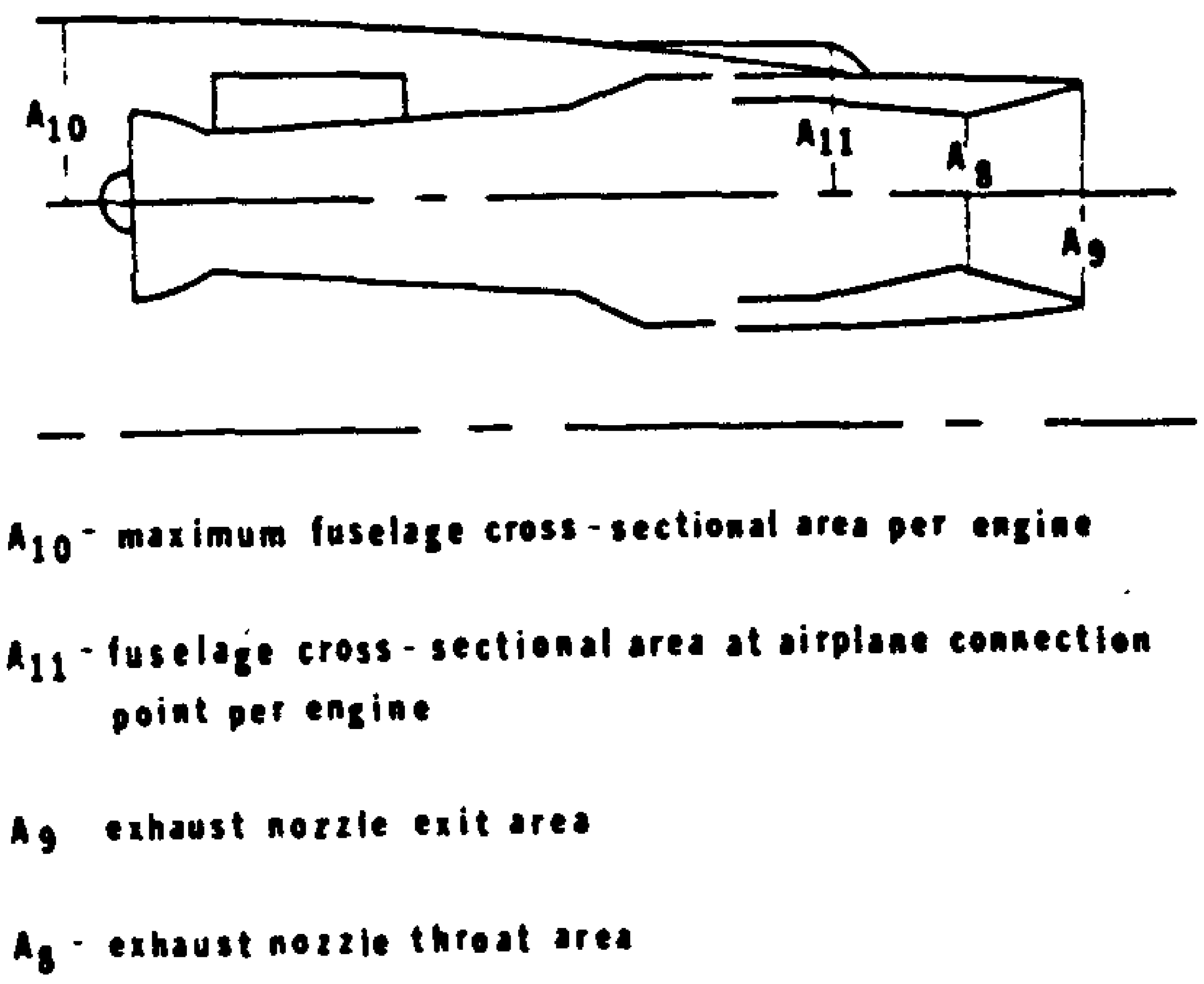



Fig. 4.7 Afterbody nomenclature adopted [27]

external flow than others. The effect of external flow on four types of nozzle is illustrated in Fig. 4.9. A convergent-divergent nozzle is immune to external

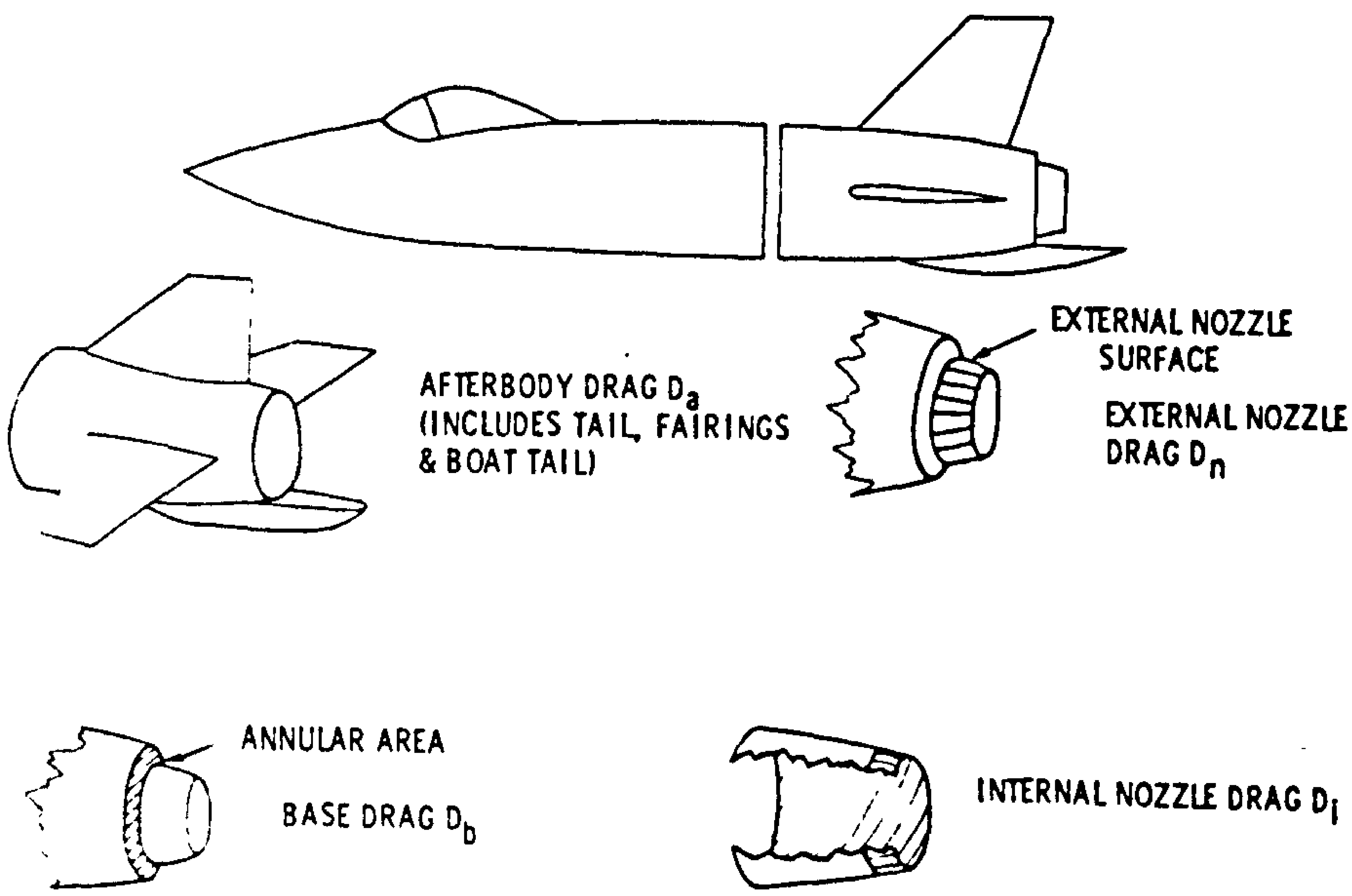



Fig. 4.8 Areas for aft-end drag definitions

influences whereas corrections have to be made in the case of a convergent nozzle, for example. The effect of such a correction on gross thrust coefficient is shown in Fig. 4.10 for a typical plug nozzle.

The drag on the boattail and horizontal stabilizers form the afterbody or boattail drag. The boattail is the surface which reduces the area from the forebody-afterbody split to that at nozzle exit. It is necessary to include the drag

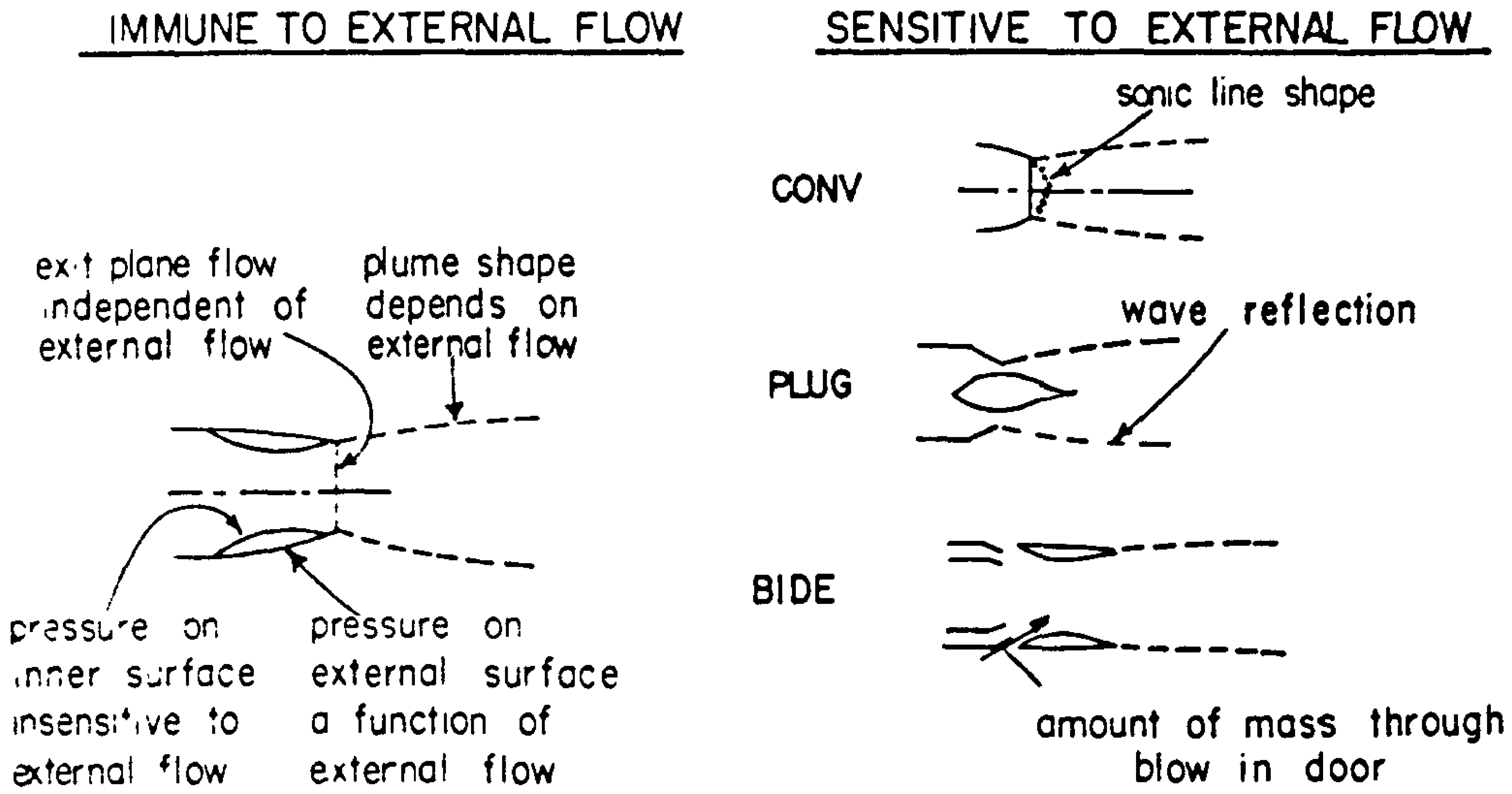



Fig. 4.9 Influence of external flow on internal performance of dynamic nozzles [14]

aft of the maximum fuselage cross-sectional area location

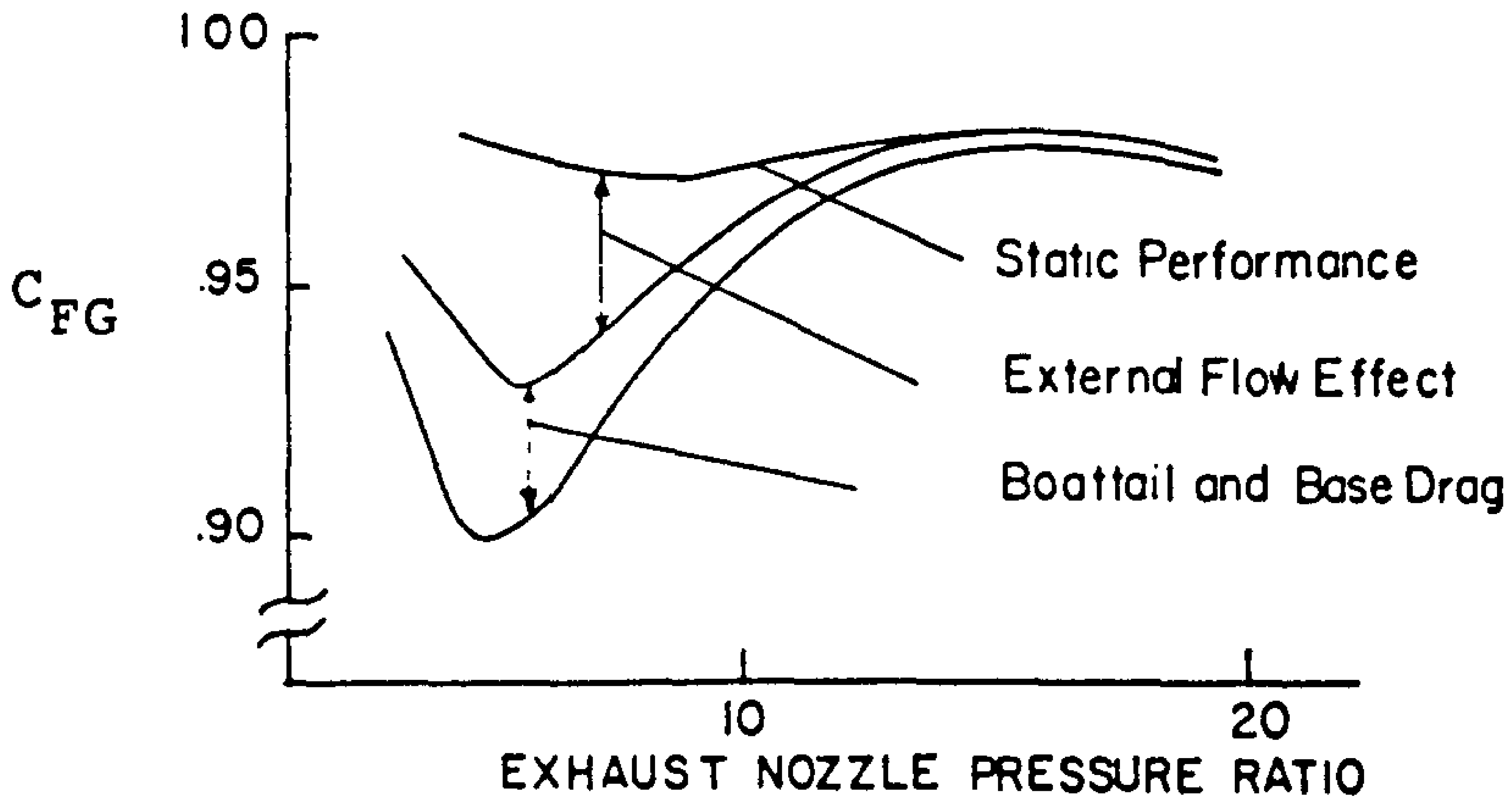



Fig. 4.10 Effect of installation losses on nozzle thrust coefficient [14]

if proper accounting of engine installation effects is to be achieved. Nozzles of afterburning engines do have some surface area exposed to the external flow in order to be free to change area ratio; therefore, there is an additional drag termed nozzle external drag which has to be accounted for. This drag is normally included in the boattail drag.

If the streamlines over any part of the afterbody do not follow the body contour, then the flow is described as a base flow and there is a drag, base drag, associated with it. A typical base area would be the area of any annulus surrounding the nozzle exit. Such a base drag can be lumped with the afterbody drag, but there are blunt bases such as the area between nozzles in a twin engine installation that are best accounted for otherwise. Such base drags can be included in the exhaust interference drag.

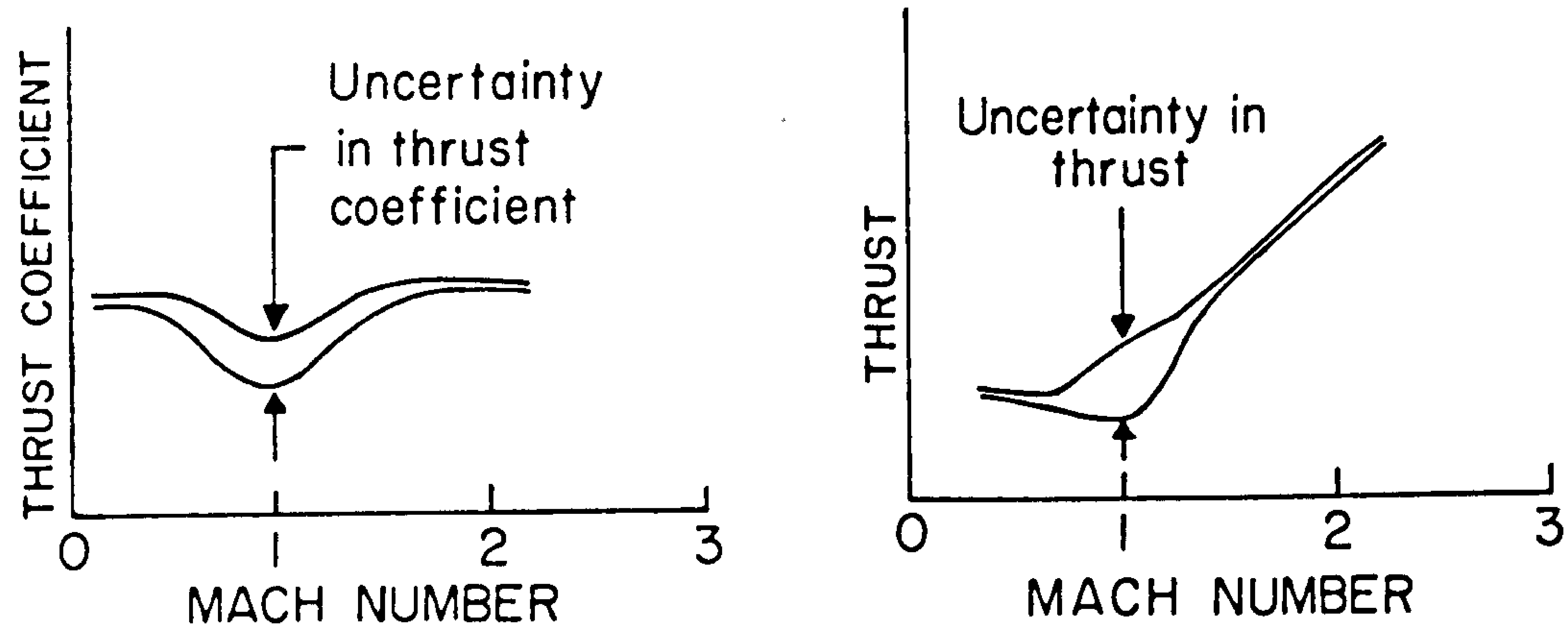



Fig. 4.11 Thrust loss due to uncertainty in $C_{FG}$ [14]

The corrections for boattail drag, etc, are usually reflected as a change in the gross thrust coefficient, Fig. 4.10. The effect of loss of $C_{FG}$ can be significant for a supersonic aircraft especially in the transonic region and may cause an uncertainty in the value of the installed thrust, Fig. 4.11. Figure 4.12 shows that nozzle performance is quite critical for supersonic cruise aircraft. Reference 28 gives an empirical expression for nozzle thrust coefficient in terms of the nozzle geometry and exhaust gas thermodynamic properties. This could be useful when evaluating the performance of one or more engines in the preliminary design stage.

The external drag due to the interference of the exhaust flow with the external flow results from the fact that changes in engine operating conditions change the exhaust plume geometry thereby modifying the flow over the airframe with an attendant increase in drag. This drag comprises the pressure and frictional drags. References 4, 28, and 44 describe how these drag coefficients can be obtained taking into account the shape of the afterbody. Using the nomenclature of Fig. 4.7, the throttle dependent external aft-end drag can be written as

$$\Delta D_{AFT-END} = {}^{1}/_{2}\rho V_o^{2} A_{10} C_{DAFT-END} \qquad 4.4$$

A representative aft-end drag coefficient is shown in Fig. 4.13 as a function of the ratio of nozzle exit to ambient static pressure and the ratio of nozzle exit to maximum fuselage area at a Mach number of 0.9. The effects of the exhaust plume on afterbody drag are accounted for in the pressure ratio term whilst the area ratio takes into account fuselage closure effects on drag [27]. The figure clearly shows that a larger exhaust exit area will reduce drag as will an under-expanded exhaust plume as this will

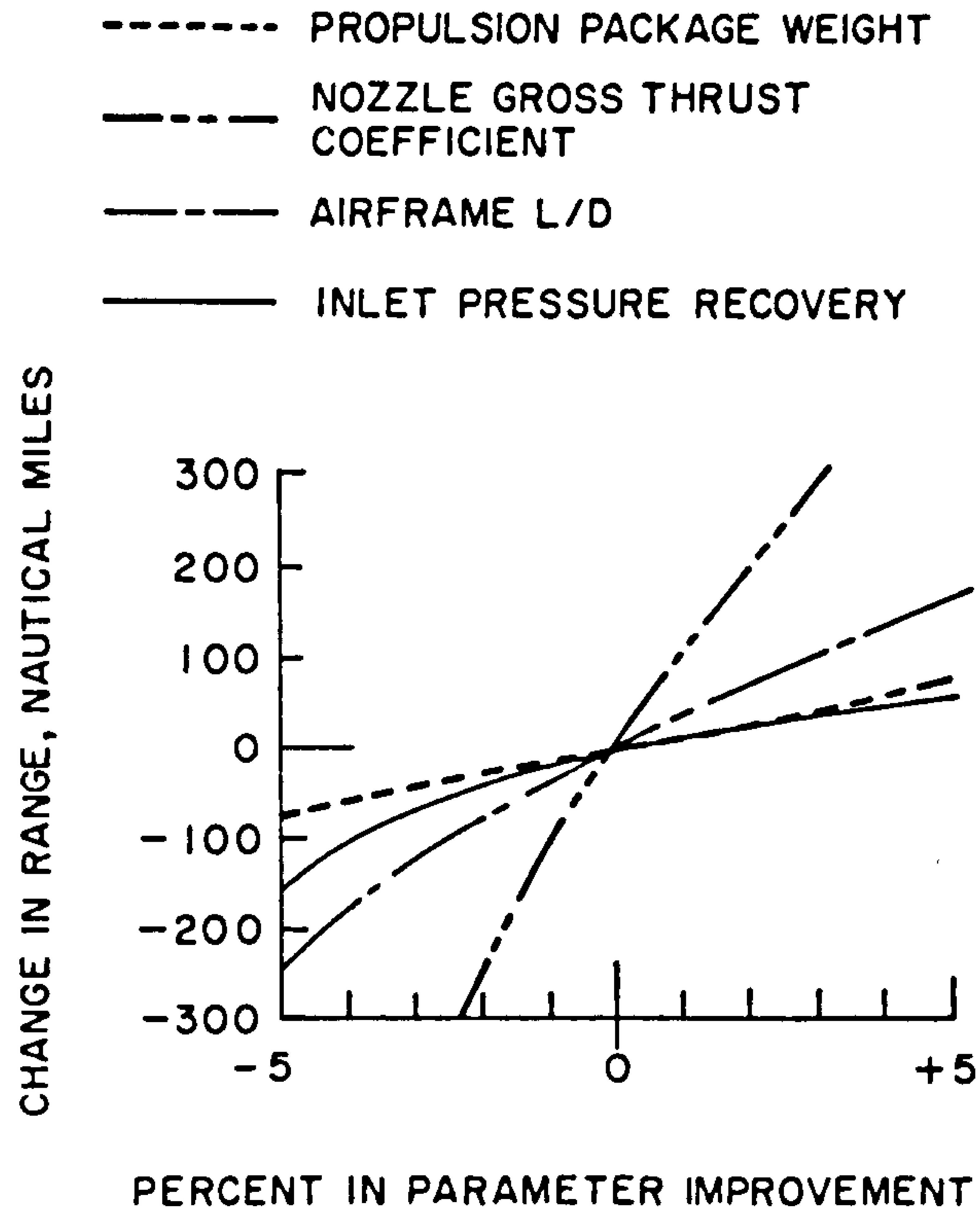



Fig. 4.12 Performance sensitivity of propulsion system installation [14]

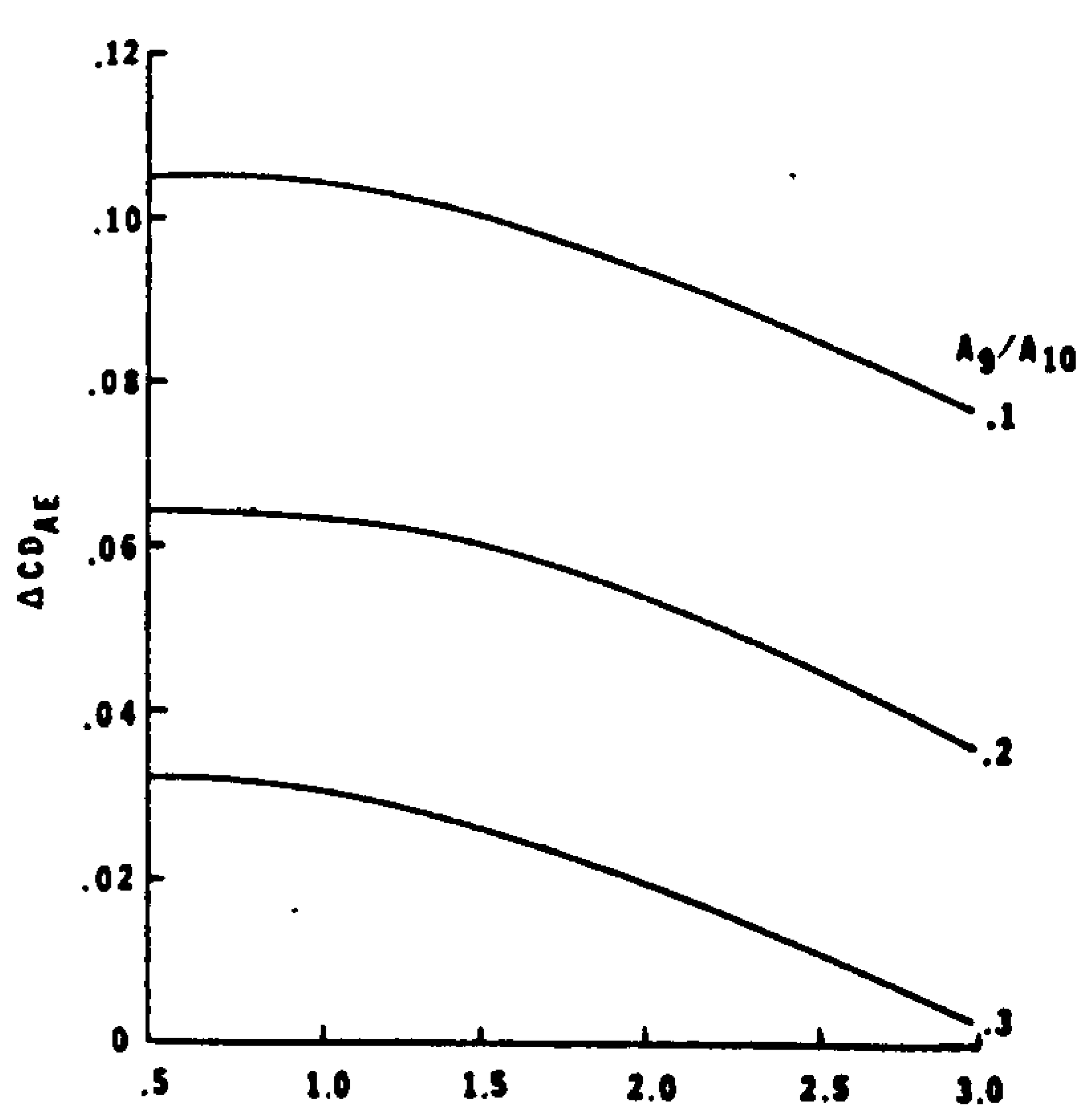



Fig. 4.13 Typical aft-end performance map [27]

tend to turn the external flow thereby compressing it.

Afterbody configuration and engine operating conditions strongly impact exhaust system performance and therefore Fig. 4.13 should not be taken as generalized.

## 4.6 Closure

In order to choose an engine cycle which will meet the diverse goals of an aircraft mission, propulsion system losses need to be defined, identified, and predicted accurately during the preliminary design phase. As the development of both the airframe and engine progresses, test data gathered should be scrutinized with a view to providing information which will be used to refine the force accounting procedure while aiming to integrate the various elements well and at the same time maintaining element visibility.

The data used in the integration procedure should cover the entire flight envelope and should consider all parameters which are of paramount importance. An important arm of the integration process is that of information flow between the engine and airframe manufacturers. A good handling of information transfer is desirable if an excellent aircraft is to be produced from a good airframe and a good engine.

# CHAPTER 5

## Variable Cycle Engines

There are two types of jet engine used for aircraft propulsion. These are the turbojet and the turbofan engines. The latter falls into two categories - the mixed flow and the unmixed or separated flow. The turbojet is characterized by high specific thrust, favourable sfc with reheat, and excellent handling qualities whilst the turbofan has the advantage of favourable dry sfc. The turbojet is therefore suited for high Mach number flight, and the turbofan, for low Mach or subsonic flight.

As was mentioned in Chapter 2, the logistic deployment of future fighter aircraft calls for aircraft with effective capability in both offensive and defensive roles. For persistent air superiority, the flight envelope of these multi - role or mixed mission combat aircraft will have to be stretched further and further thereby requiring propulsion systems to operate more and more off-design. The demand that will be placed on future military propulsion systems can be reduced to improved performance in terms of installed thrust, fuel burn, and stable operation, in all segments of the flight envelope.

Increased thrust loading is required for high manoeuvrability, fast acceleration, and supersonic cruise, whereas efficient low thrust is needed for extended subsonic cruise and loiter. The propulsion system usually chosen to meet the widely varying thrust demands of multi mission fighter aircraft is the mixed flow afterburning turbofan. The inherently high propulsive efficiency of the fan causes this engine to produce efficient low thrust whilst high thrust is delivered by the heavy use of afterburning. If future aircraft are to meet the goal of extended supersonic cruise at dry power or with minimum augmentation, then a cycle approaching that of a turbojet will be required for the supersonic phase.

Because of the conflicting requirements placed on the propulsion system in both legs of a supersonic aircraft speed range, it is logical that if improved performance is to be obtained from future propulsion systems over that of present day engines, then the advantages of both the turbofan and turbojet engines should be built into one engine design. This means that the engine should have the capability of tailoring its performance to match the current required duty. The selection of such an engine cycle, a variable cycle engine, poses one of the greatest technical challenges to gas turbine performance engineers in the search for engines to meet future aircraft propulsion demands.

If the propulsion system is to deliver the required performance throughout the entire flight envelope, then the components should continuously re-match such that each component operates at or quite close to its optimum operating point, and the interface of operation between two components should be optimized. Two such interfaces for which this requirement must hold are compressor/turbine and inlet/engine.

## 5.1 Influence of Turbine Area on Engine Matching

When a gas turbine engine is operating steadily at off-design, the various compatibility equations are maintained and as a result, the components match such that each component operates along a working line. The components at the "hot end" have a controlling influence on the engine cycle and may be used to alter or shift the working line of some of the other components. Therefore, a turbine has an important role to play when determining the performance levels that can be obtained at off-design.

As noted in Chapter 3, a critical area of turbine design is the turbine nozzle area. Turbines are normally designed to operate choked over most part of their operating range. This characteristic imposes severe restrictions in the operation of a compressor coupled to the same shaft, and hence, the engine. It was shown in Chapter 3 that for a single spool fixed geometry turbojet, when turbine inlet temperature decreases, the compressor and turbine match

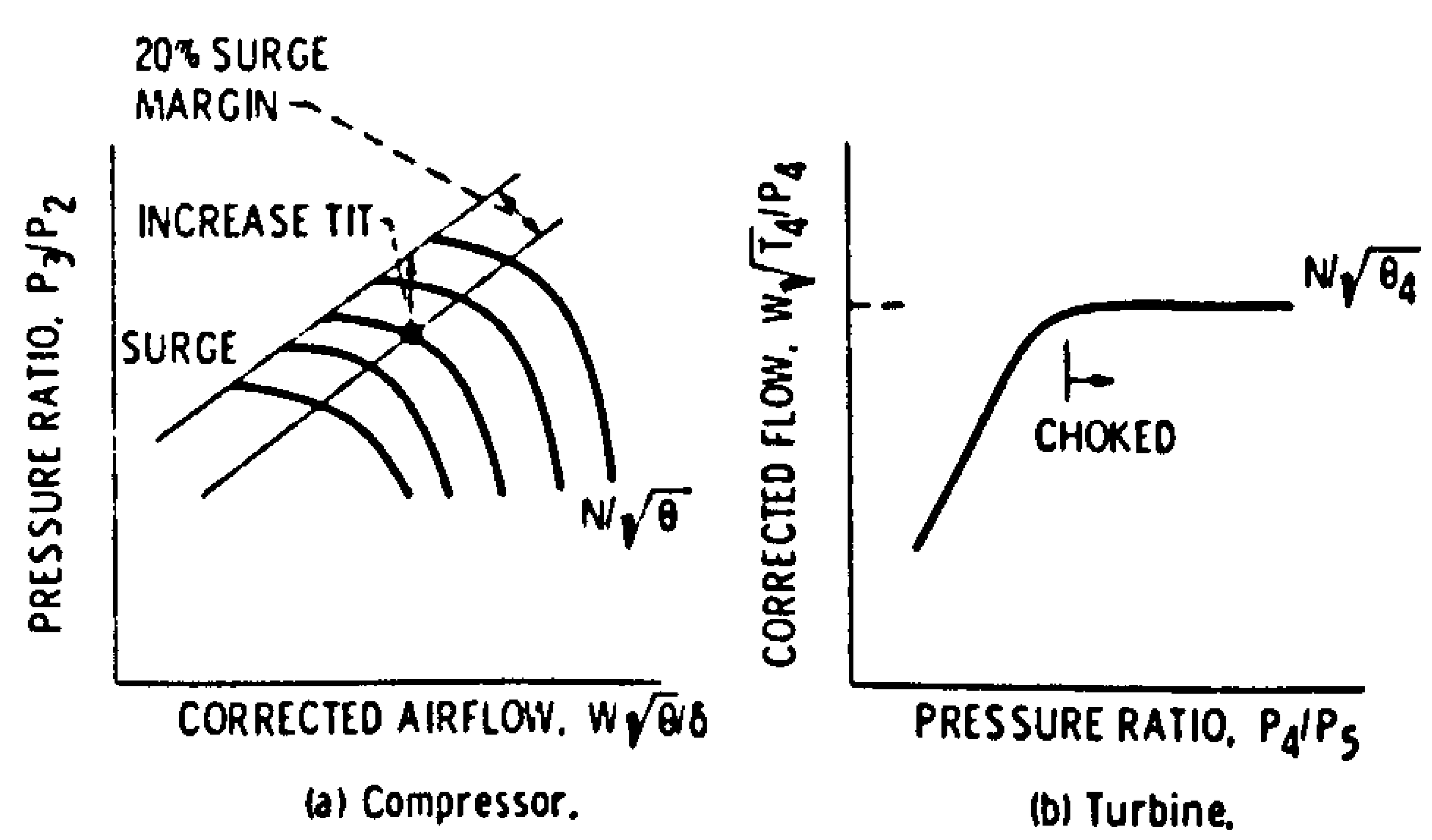



Fig. 5.1 Matching characteristics of a single spool turbojet [12]

such that both compressor mass flow and pressure ratio decrease. This can be taken as generally true for conventional gas turbines. An examination of the effect of the choice of the designed turbine area on engine performance will indicate that any attempt of designing a VCE should include the extensive use of the variable area turbine.

Typical characteristics of a fixed geometry compressor and turbine for a single spool turbojet are shown in Fig. 5.1. The turbine is choked for most of its operating range. In matching the two components, the compressor pressure ratio and mass flow change when turbine inlet temperature changes such that the turbine non-dimensional mass flow remains fixed if the turbine is operating in its choked region; that is,

$$\frac{W_4\sqrt{T_4}}{A_4P_4} = \text{Constant} \qquad 5.1$$

using the notation in Fig. 3.7. Therefore for a prescribed mass flow, W, a given fixed area turbine requires the compressor to operate at a high pressure ratio, that is, closer to surge, for a high TIT design, and towards choke for low TIT designs. The upper limit on TIT is therefore restricted by the amount of surge margin required, materials, and cooling techniques, whereas compressor efficiency and propelling nozzle area tend to restrict the lower bound on TIT. Therefore, a given compressor and turbine combination dictates the bounds on TIT excursions for an engine.

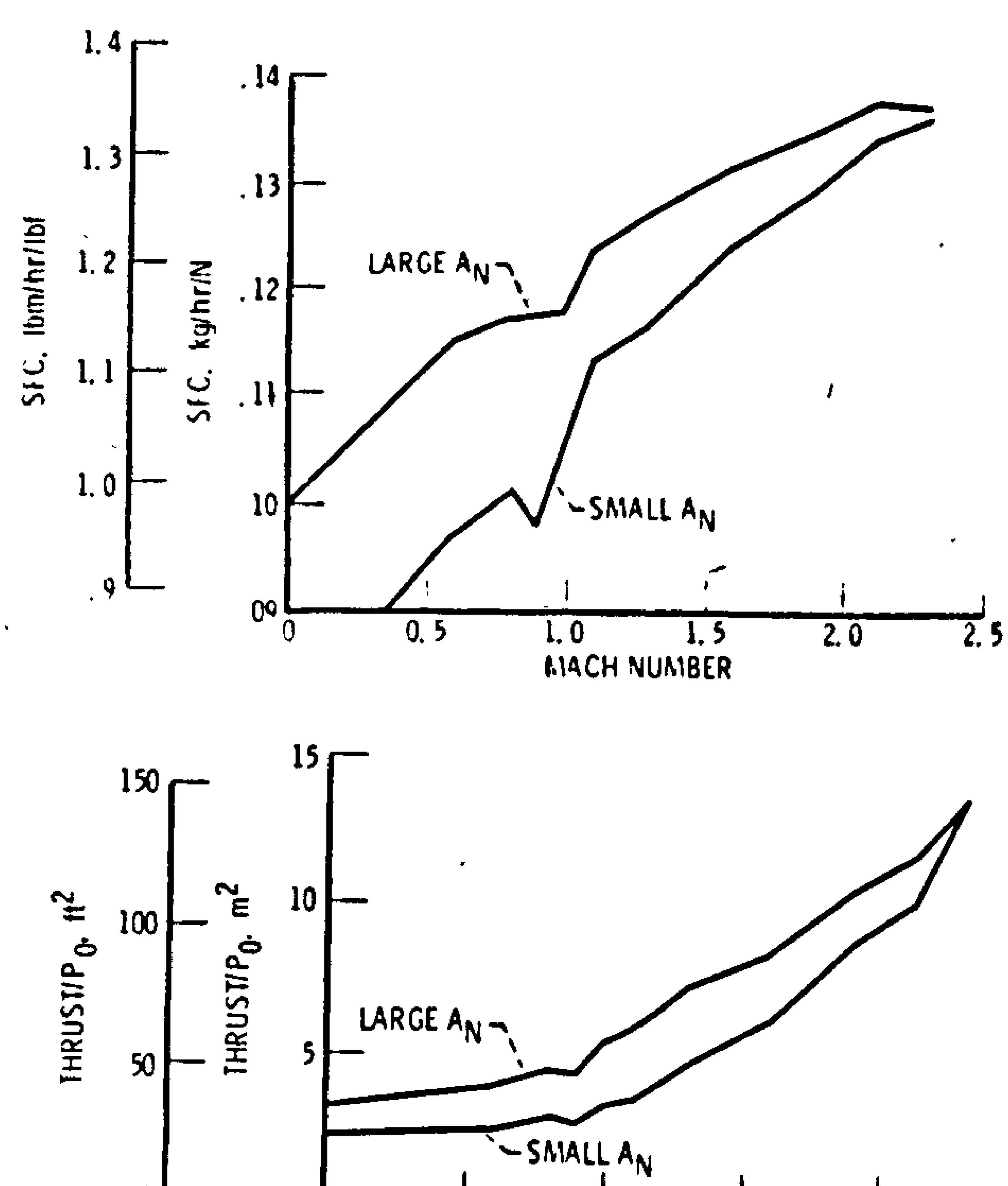



Fig 5.2 Effect of turbine area on acceleration performance of a turbojet [12]

Since the level of thrust produced by an engine is directly linked to the turbine inlet temperature, a larger turbine nozzle area is indicative of a high value of thrust while a small area gives lower thrust levels for a given compressor design point. The effect of the choice of the designed nozzle area on aircraft acceleration performance is illustrated in Fig. 5.2. The larger turbine area design permits the engine to be operated at the maximum TIT over

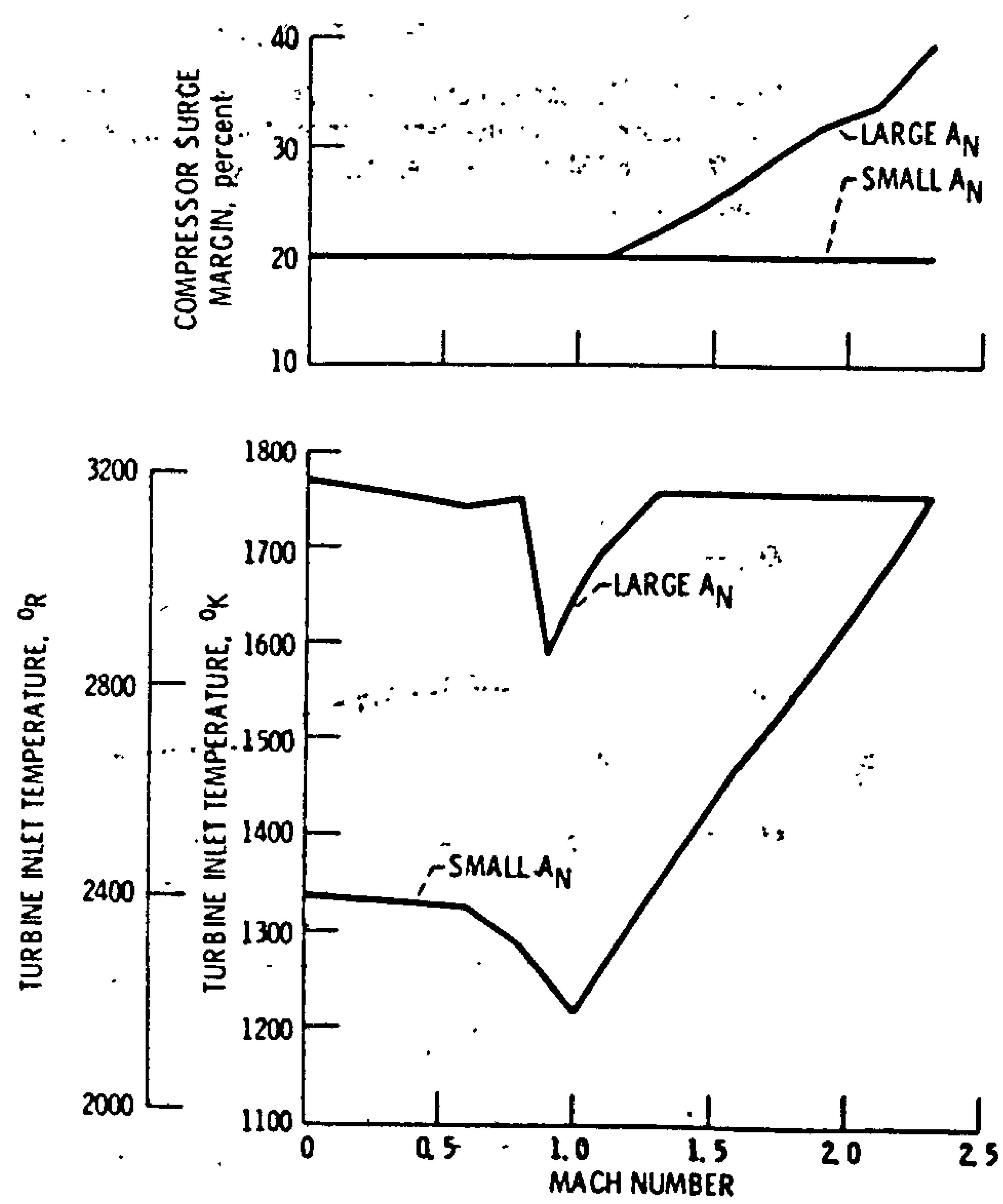



Fig. 5.3 Engine surge characteristics in relation to turbine size [12]

most of the Mach number range whereas the smaller turbine area design requires a lower TIT in order to prevent the compressor from surging, Fig. 5.3. A larger nozzle area is therefore desirable for high Mach flight, and referring to Fig. 5.2, transonic thrusts are about 60 percent greater although at the expense of a high specific fuel consumption.

The disadvantage of a high design nozzle area with its attendant high TIT is that the engine has to be severely throttled back to deliver the required cruise thrust at subsonic speeds. The decrease in airflow necessary to achieve this does not only reduce propulsive efficiency but also increase spillage and aft-end drags giving rise to poor installed performance. On the other hand, a smaller nozzle area requires a lesser reduction in TIT to deliver the cruise thrust and therefore incurs lower installation losses. Fig. 5.4 shows that a large saving in fuel flow can be realized at a typical subsonic cruise power setting for the smaller nozzle area design over a larger area. It is also clearly seen that the maximum dry thrust is substantially lower for the smaller turbine compared with that for the larger one. Therefore, if a VCE is to give

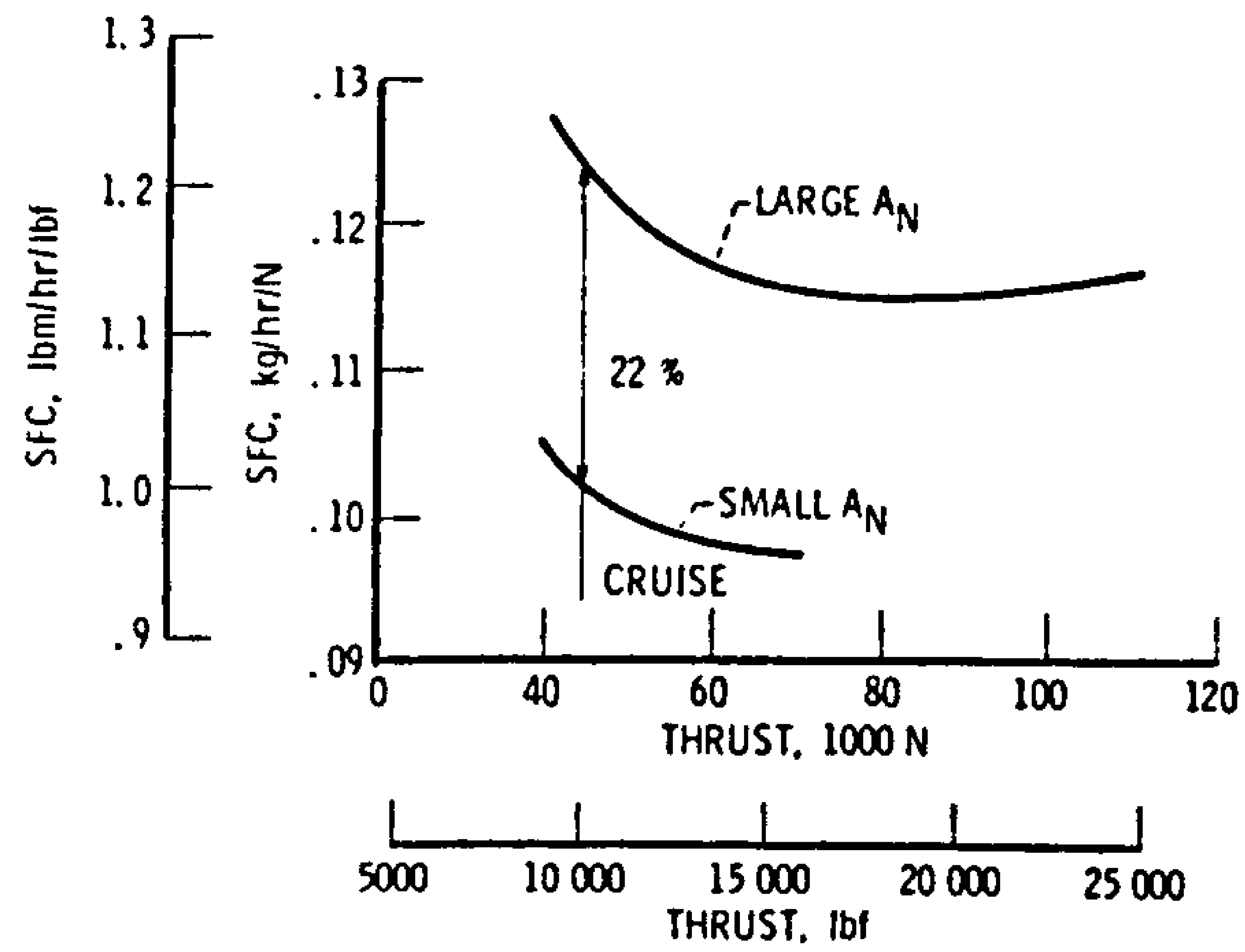



Fig. 5.4 Effect of turbine area on subsonic performance [12]

acceptable performance throughout the flight spectrum, then it should incorporate a variable area turbine to give the favourable high speed characteristics of a larger turbine area and the favourable low flight speed characteristics of a smaller turbine area.

## 5.2 Supersonic Propulsion

An aircraft with supersonic capability may spend a considerable amount of its flying time at subsonic speeds; for example, a supersonic transport aircraft will have to fly subsonically out of cities during its climb phase, and on the approach to landing there is the requirement for good hold endurance or diversion to alternate airports. Military aircraft may be required to cruise at subsonic speed on return from a combat mission. Hence, there is the requirement for good fuel burn in both legs of the mission.

The requirements for both efficient subsonic and supersonic propulsion can readily be found in literature on aircraft propulsion and will not be enumerated here save that high specific thrust is required to keep weight and drag down in supersonic flight whilst a high propulsive efficiency and hence, low sfc, is desirable for good subsonic performance. A turbojet engine could be designed to meet the requirements for supersonic flight efficiently but with poor subsonic performance whereas it is the other way round for the turbofan. Supersonic cruise aircraft designed in the West have all used the afterburning turbojet whereas the turbofan is favoured in the East. Present day mixed mission fighter aircraft use the mixed-flow turbofan and the thinking on how to meet future propulsion needs is along lines of variable cycle engines which will use some amount of bypass airflow in all segments of the flight envelope. The question now is what level of bypass ratio is suitable for adequate performance throughout the entire flight spectrum.

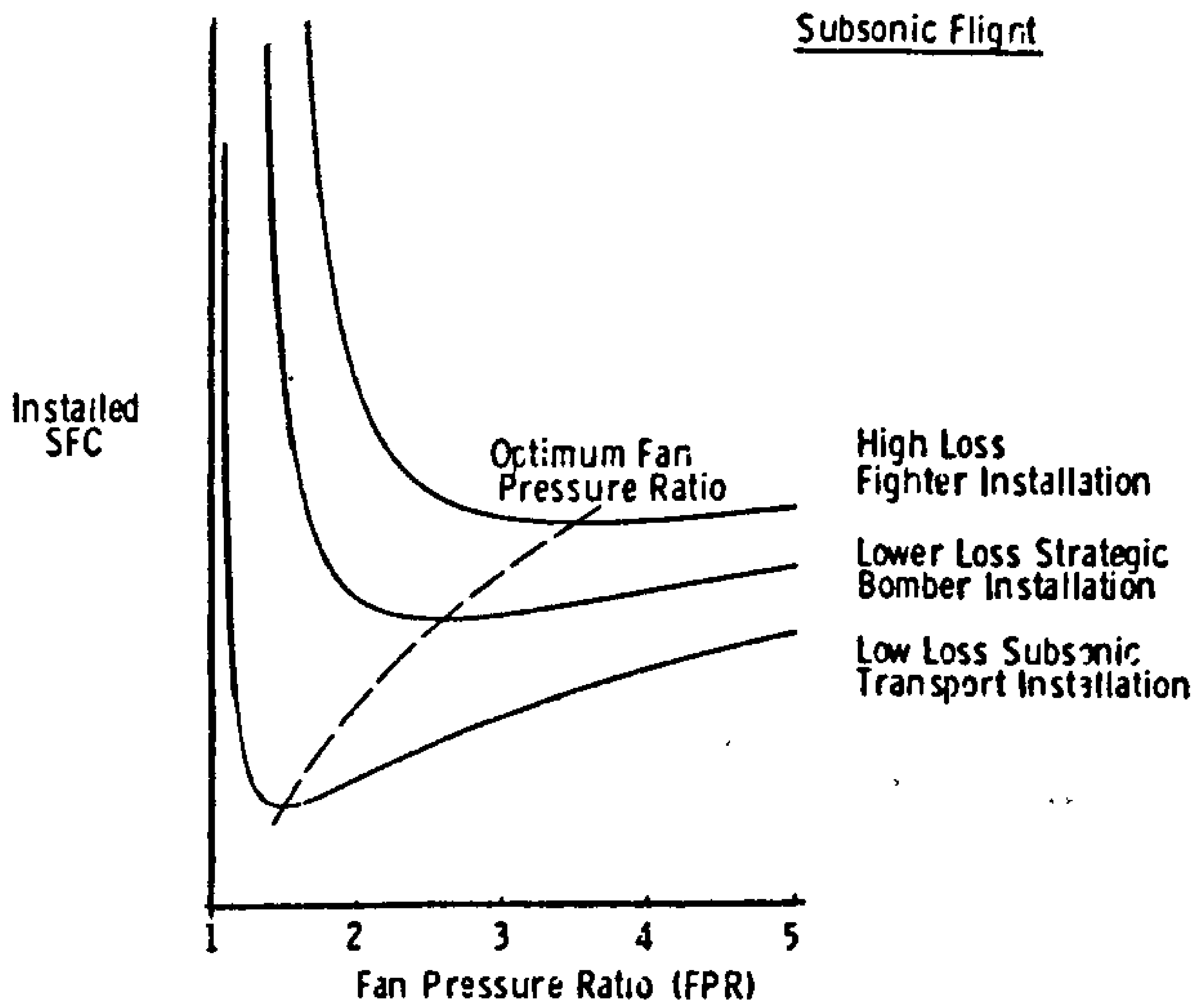



Fig. 5.5 Variation of sfc with fan pressure ratio for various installations [39]

One of the important cycle parameters of the turbofan is the bypass ratio to which the fan pressure ratio, FPR, is inextricably linked. The conflicting requirements for subsonic and supersonic cruise dictate a low bypass ratio and hence, a high fan pressure ratio, for supersonic flight whereas a high bypass ratio with its attendant low fan pressure ratio is needed for excellent subsonic cruise. These can be derived from analysis such as that given by Payzer [39] and as illustrated in Fig. 5.5, the type of mission will dictate the level of FPR or $\mu$ required to produce the necessary level of jet velocity needed to overcome propulsion losses. Some design considerations for both commercial and military supersonic engines are discussed next.

A commercial supersonic aircraft flies at supersonic speeds for most part of its mission, but the environmental problems with regards to noise and exhaust emissions impose stricter rules on its engine design as compared with those placed on its military counterpart. Remembering that noise is proportional to specific thrust, a low specific thrust and hence, high mass flow, is required for takeoff and climb in order to keep noise down. Also, engine weight has to be minimized to obtain good supersonic performance. The mass flow at takeoff will be determined by the level of noise and required thrust, and for a fixed mass flow, a higher bypass ratio will give a lower weight due to a reduction in core size, Fig. 5.6.

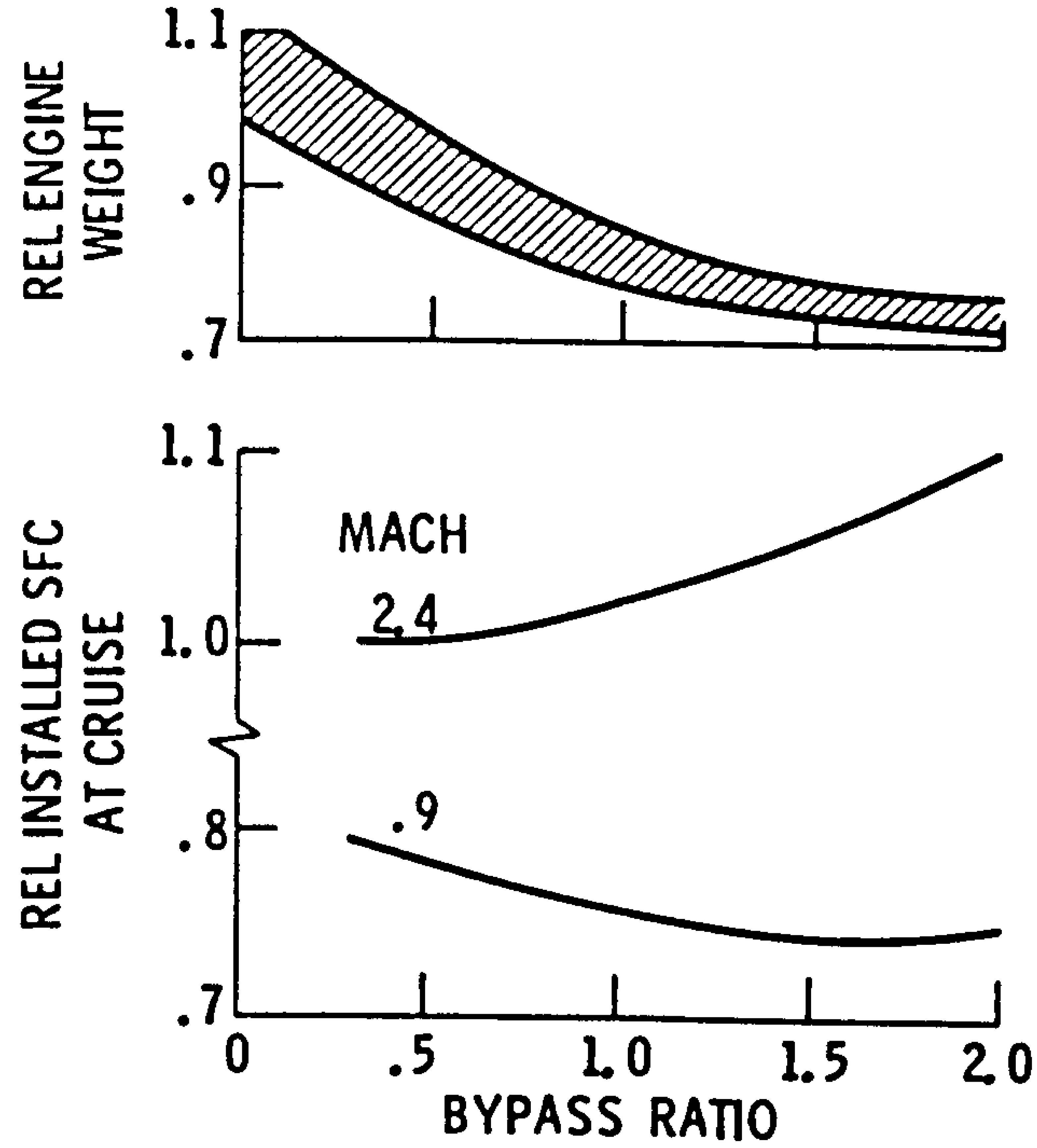



Fig. 5.6 Factors to consider in cycle selection for supersonic propulsion [59]

Figure 5.6 also shows the effect of bypass ratio on both supersonic and subsonic specific fuel consumption for a separated flow duct-burning turbofan in which the fan pressure ratio has been re-optimized at each bypass ratio. Maximum dry thrust was required at cruise for a bypass ratio of 0.5, therefore, at higher bypass ratios the duct burner had to be switched on to produce the same cruise thrust with a resultant higher level of sfc. At subsonic flight conditions, the higher the bypass ratio, the lesser is the throttling back required and therefore, the performance improves. Therefore, weight requirements for supersonic cruise cannot be met if takeoff noise is to be kept down to acceptable levels. This is not surprising as the bypass ratios required for subsonic and supersonic flights conflict one another as was mentioned earlier.

An obvious choice then is the variable cycle engine which has the ability to accept large swings in flow in transition from subsonic to supersonic flight and vice versa. The engine should be designed to accept large airflows at takeoff and subsonic flight with a greater amount bypassing the core. Fan pressure ratio will therefore be low. At supersonic speeds, the core should be high flowed to reduce $\mu$ with FPR increased to raise specific thrust. The range of FPR or $\mu$ is however limited by the requirements for good inlet-engine matching at supersonic cruise and minimal installation losses at subsonic flight [39]. These requirements also apply to military engines.

The performance demanded from supersonic military engines is more stringent than that expected of commercial engines. As was mentioned earlier, a significant portion of a fighter aircraft mission is flown at subsonic cruise, especially on return to base, where the thrust demands are quite low, and therefore any saving in fuel consumption is quite important. However, alternate mission requirements such as manoeuvrability, acceleration, etc, will drive the bypass ratio required for supersonic cruise only away from the optimum. A high bypass ratio, low FPR, is desirable for missions with little or no manoeuvre requirement if a significant portion of flying time is spent at subsonic speed, whereas a low $\mu$, high FPR, would be required for a similar mission carried out at supersonic speed.

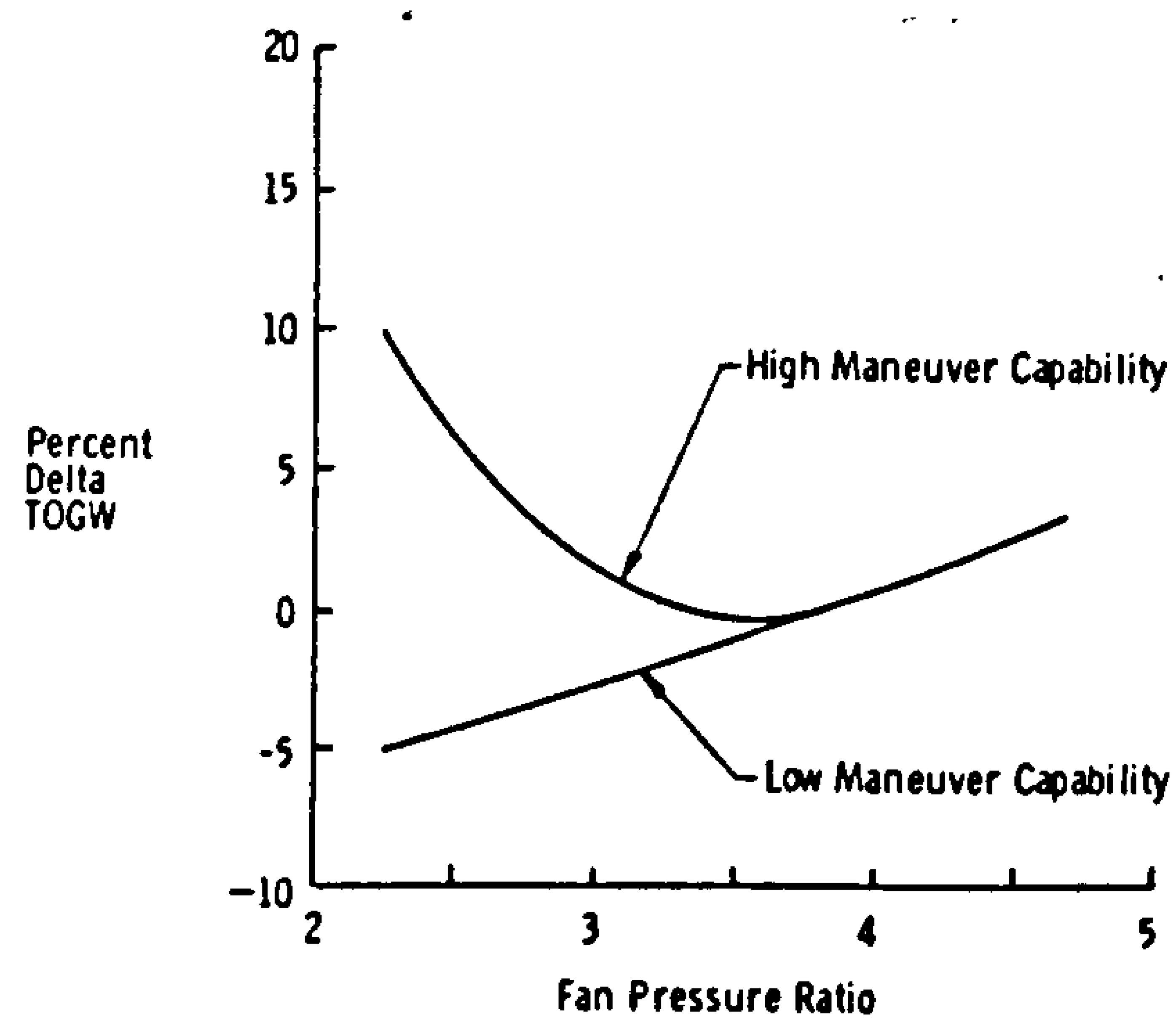



Fig. 5.7 Influence of mission requirements on engine weight [39]

When high manoeuvrability is required in either subsonic or supersonic leg, the cycle could be re-optimized to produce high specific thrust by lowering $\mu$ and thereby increasing FPR. Fig 5.7 shows that variable cycle could have a significant pay off in allowing the high fan pressure ratio cycle with good combat ability to also have good subsonic performance. The choice of cycle will however be decided by the superlative mission requirements.

## 5.3 Variable Cycle Engine Studies

Several variable cycle engine concepts for supersonic transport aircraft have been proposed in the past and the most promising ones are being developed further. They have all attempted to combine the advantages of both the turbofan and the turbojet while at the same time tried to address the problem of installation losses. A description of these engine concepts can be found in [19,39,59]. The three most promising concepts will be described here but before that is done, the principle of operation of a co-annular nozzle is first described as such a nozzle was found to play an important part in reducing noise generated by supersonic engines.

## The Co-Annular Nozzle

It was found that if two streams are arranged such that the stream on the outside is of a higher velocity than that on the inside, and if they exit from two coaxial nozzles with

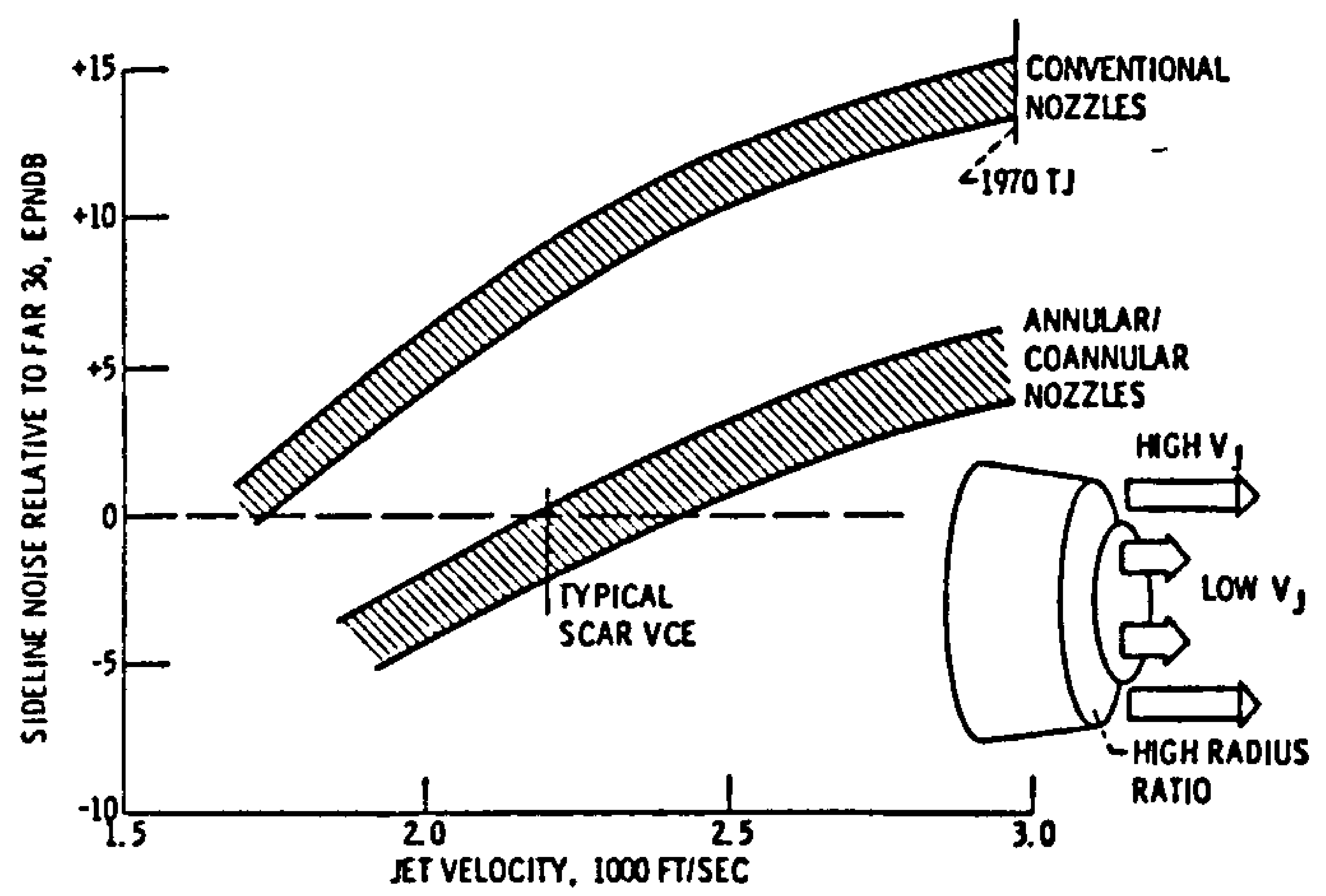



Fig. 5.8 Potential noise reduction of co-annular nozzles [59]

the outer nozzle having a high annular radius ratio, then the noise level emitted by this configuration is much lower than that emitted by two conical nozzles with identical mass flows and jet velocities. The more rapid mixing of the inverted flows is thought to be responsible for reducing noise.

Fig. 5.8 shows the impact that such a nozzle has on noise reduction. Such a nozzle is quite attractive for use with a VCE which may be capable of varying its bypass ratio to produce a low jet velocity at takeoff. Fig. 5.8 also shows that noise reduction well below that of the US Federal Aviation Regulation (FAR) 36 level can be obtained. A mixed flow bypass engine with a ventilated plug nozzle can also benefit from this effect if fan or inlet air can be made to exhaust from an annular slot in the afterbody.

## The Variable Stream Control Engine (VSCE)

The variable stream control engine is Pratt & Whitney's front runner concept to fulfil the propulsion requirements of a supersonic cruise transport aircraft. This engine, Fig. 5.9 is basically a two spool duct-burning turbofan with the ability to control its operating bypass ratio by making extensive use of rotor speed control, variable geometry in the fan, compressor, and both primary and secondary nozzles, in addition to its unique and complex throttle schedule. The throttle schedule is used to minimize installation losses by matching the inlet and engine at intermediate power and above at all Mach numbers.

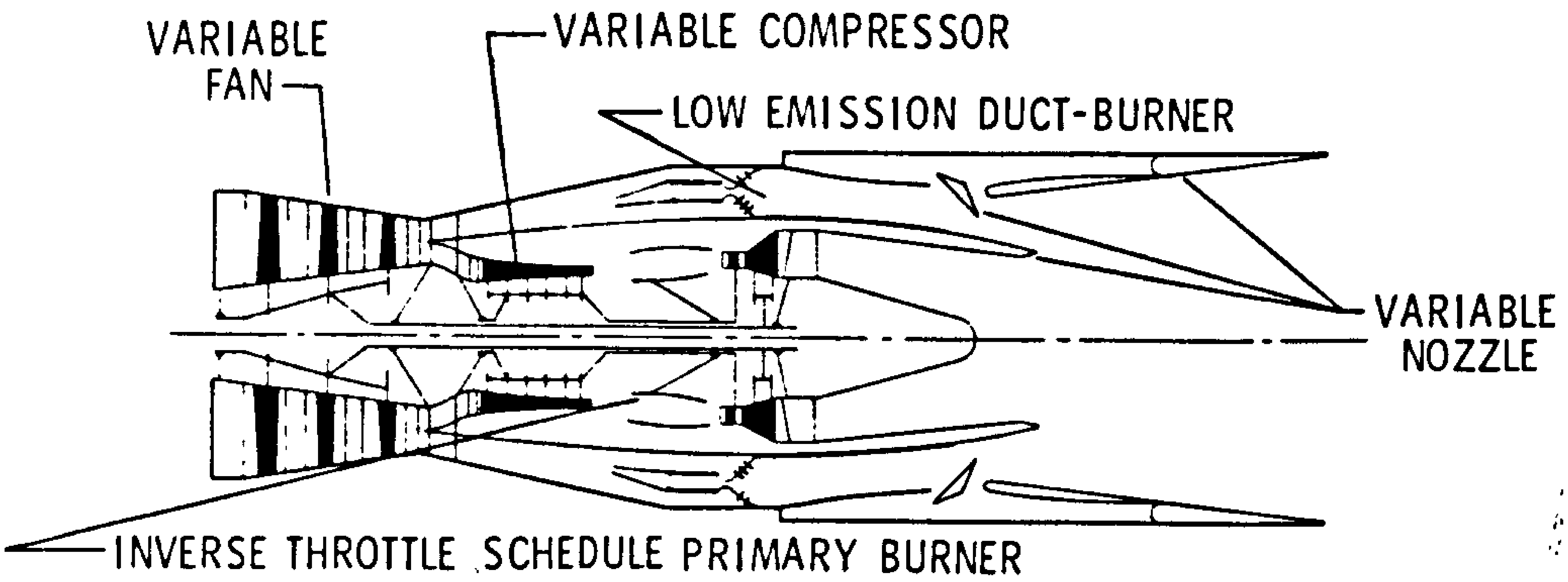



Fig. 5.9 Schematic of the Variable Stream Control Engine (VSCE)

At takeoff, acceleration, and supersonic cruise, high thrust is needed and this is obtained by lighting the duct burner. Since this configuration is well suited for a co-annular nozzle, the use of such a nozzle reduces noise, especially at takeoff, when the main burner temperature is well down on the maximum. At supersonic cruise, the core is speeded up by increasing main burner temperature and employing variable geometry to effect bypass ratio change.

For subsonic cruise, the duct burner is shut off and the engine is throttled back. The engine then operates as a separate flow conventional turbofan with moderate bypass ratio (about 1.5) and this provides good subsonic performance.

<u>The Rear-Valve Variable Cycle Engine</u>

This is another Pratt & Whitney concept and is basically a twin-spool ductburning engine with a mixer/crossover valve and an aft turbine downstream of a normal LP turbine. There are two distinct operating modes depending on the valve position.

In the takeoff, acceleration, and supersonic cruise modes, the valve is in the crossover position with the core stream being diverted around the second LP turbine into the outer nozzle annulus while the bypass stream from the lit duct burner goes through the aft turbine into the inner nozzle thereby forming two inverted turbojet cycles. Its supersonic performance is not that favourable as both cycles are not operating at optimum pressure ratio.

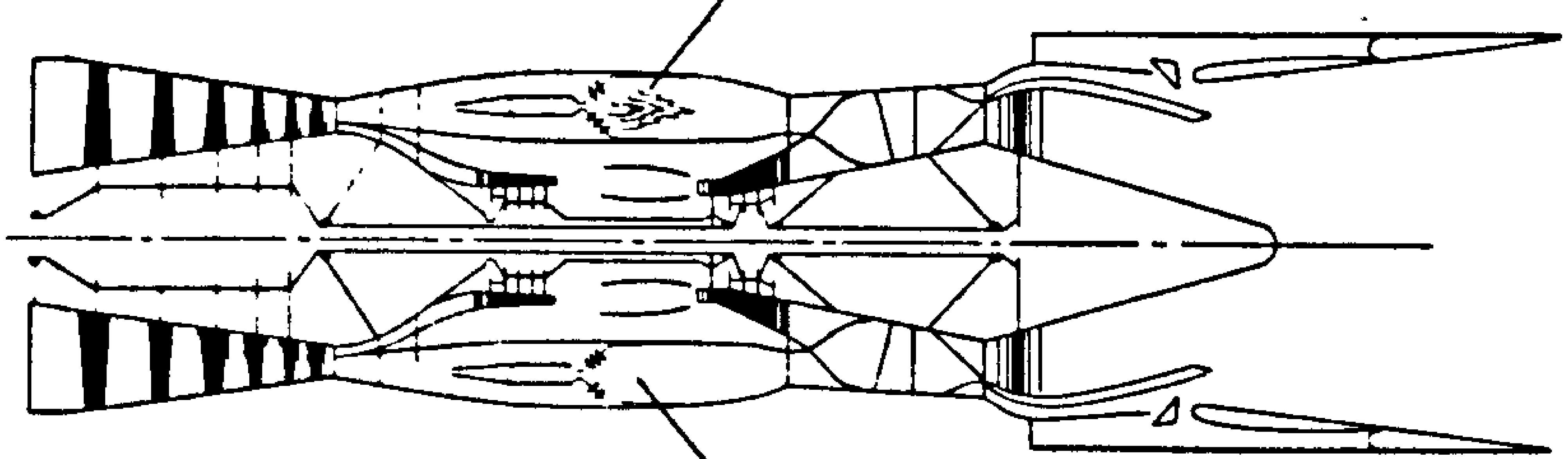



Fig. 5.10 Schematic of the Rear Valve Variable Cycle Engine (RVE)

In the subsonic mode of operation, the duct burner is not lit and since the aft turbine is designed to be operated choked, the bypass stream non-dimensional mass flow is supplemented by the core stream. As a result, the valve is in the "mix" position and all the burnt gases exit from the inner nozzle. The cycle is then that of a mixed flow turbofan with good subsonic performance. A major advantage of this cycle is that in the high thrust mode, the fact that the augmentor is upstream of a turbine stage implies that high augmentations can be obtained with a low sfc penalty and the resulting "flat" throttle curve provides the designer with the additional flexibility in terms of engine sizing [59]. A disadvantage is that the nozzle inner stream velocity is relatively high compared with that of the outer stream when in the twin turbojet mode and therefore, the nozzle does not make full use of the noise suppression benefits offered by the co-annular nozzle.

## The Double Bypass Variable Cycle Engine

The Double Bypass Engine (DBE) is a General Electric concept which evolved from the Modulating Bypass Cycle concept and has gone through several phase changes to get to its present configuration. A technology demonstrator has undergone tests with promising results.

This engine is a dual rotor variable low bypass ratio afterburning turbofan with variable geometry features which allow independent control of rotor speeds to provide higher airflow at subsonic part power, acceleration, and supersonic operating conditions than are possible with conventional mixed flow turbofans. The unconventional features include a variable geometry fan split into a forward block driven by a variable area LP turbine, and a rear block driven by the core. There are two possible bypass streams, one immediately downstream of each fan block which are mixed in a forward variable area bypass injector (VABI). There is also a rear VABI which mixes part of the bypass flow with the core stream. A co-annular variable area nozzle is used for noise reduction. A schematic of this engine is shown in Fig. 5.11.

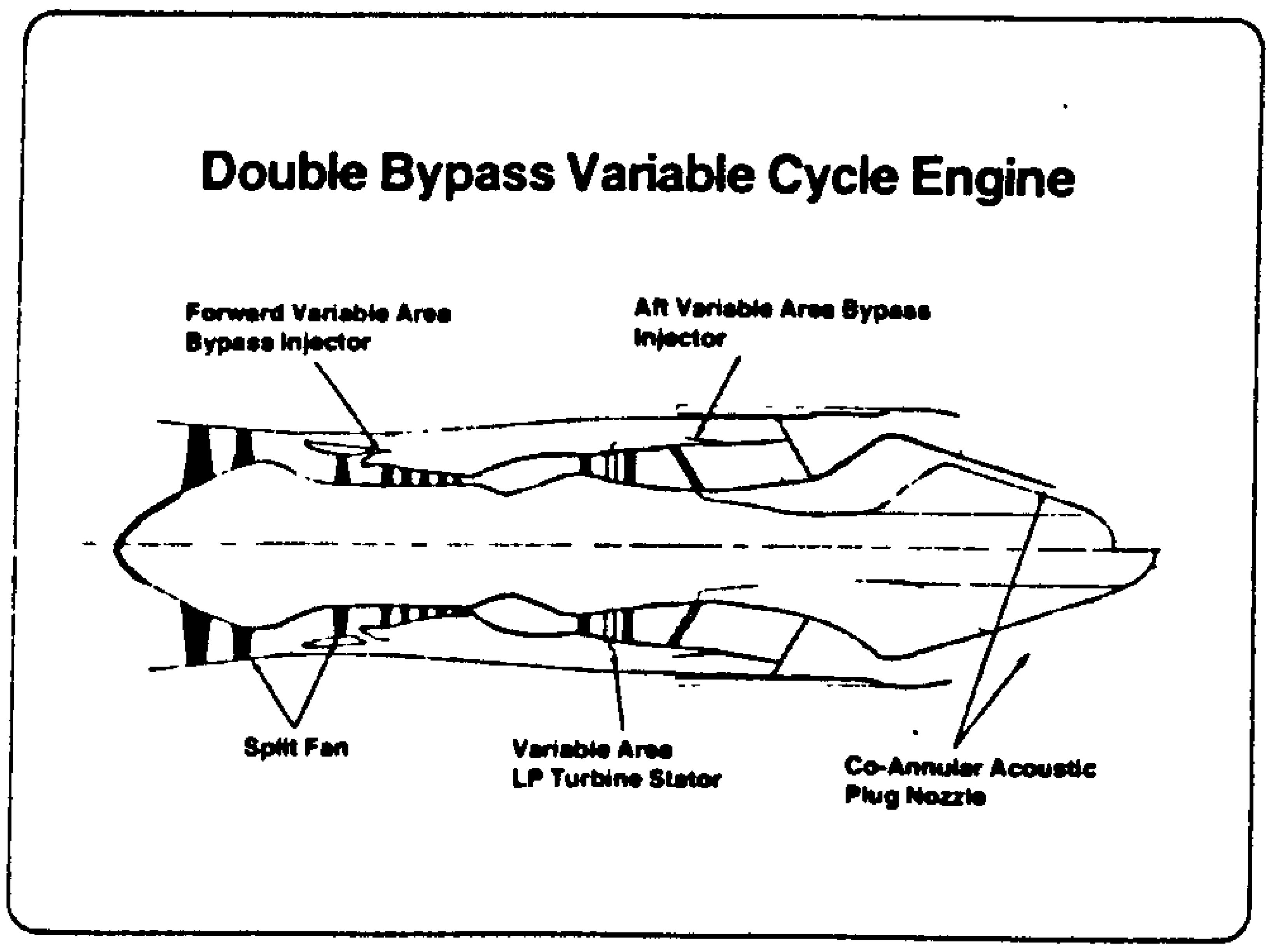



Fig. 5.11 Schematic of the Double Bypass Engine (DBE)

At takeoff, the front fan block is set at its fully open position and the rear block is set to give the desired thrust/noise relationship. As in all subsonic operation, part of the bypass airflow mixes with the core stream and is diverted into the outer nozzle while the rest exits from the inner nozzle thereby making use of the co-annular nozzle's noise reduction capability. However, in subsonic operation, the front fan block is set to match the engine with the intake right down to the cruise thrust setting thereby eliminating spillage drag and improving aft end performance. The rear fan block is low-flowed to provide the required thrust thereby driving the bypass ratio up and reducing sfc.

The engine operates in the single bypass mode during acceleration and supersonic cruise by closing the mode selector valve which shuts off the forward bypass stream, and again the front block is set to meet the inlet flow supply. The rear fan block is fully open to provide a proper flow match with the front block. Scheduled compressor stators would be desired to maintain performance at the desired bypass ratio. The LP turbine nozzle is fully opened to load the HP turbine while the rear VABI and nozzle are set to maintain optimum bypass ratio and back pressure to produce high specific thrust. Supersonic cruise is obtained at dry power setting whilst low augmentation may be used during acceleration.

# CHAPTER 6

# VATEMP PROGRAM DESCRIPTION

Computer programs have been used for many years to predict the performance of gas turbine engines and they have proved to be invaluable tools to gas turbine engineers in all phases of the development of a gas turbine engine.

During the conceptual studies phase, computer programs give engineers the opportunity to study many engine cycle concepts within a short period of time and to choose those that are most promising to meet the given requirements. Those candidates that survive the initial screening process are further evaluated to define and assess interactions among requirements, performance, cost, and effectiveness from which one or two preferred concepts are selected for preliminary design and development.

In the design and development phases, the use of computers make it possible to investigate within a short time period the effect of a change in one or more variables on the performance of the engine at the design point, or on the steady state or transient performance at off-design. Development of gas turbine controls also relies heavily on computer simulation techniques. Over the last ten to fifteen years, computers have found increasing roles in engine testing and more recently in engine health monitoring and deteriorated performance modelling and prediction.

Many computer programs have been developed over the last three decades or so to aid engineers in their quest for efficient, reliable, and compact gas turbine units. In general, these programs which may be general- or special-purpose built have the capability to predict either the steady state [11,21,24] or transient [50] performance of a given engine configuration, or may even simulate engine control systems [36] or analyse rig test performance [37]. References 1, 9, and 24 either describe or refer to various programs that have been developed for gas turbine performance simulation.

In order to keep in line with the development of advanced and complex gas turbine engines, more and more advanced and complex computer programs are being developed and most recently Douglas [9] developed a single program which can handle most facets of gas turbine performance simulation while Singh-Grewal [53] has considered the effect of in-service performance deterioration in his program.

While all the above mentioned works have looked into certain aspects of performance simulation, one thing they all have in common is that they can be used only for engines with fixed geometry or simple variable geometry. Future gas turbines, both aeronautical and non-aeronautical, will employ varying degrees of variable geometry in search of cycles that will give performance levels that are much higher than what present day cycles can give. Also, more consideration is being given to the problem of aircraft-engine integration and this is having a great impact on the design of future aircraft propulsion systems. Therefore, propulsion engineers will in the future have to consider installation effects quite early (if not in the initial screening phase) in the development of the aircraft gas turbine in order to tailor the cycle to airframe characteristics such that integration may be optimized. A computer program has been developed at Cranfield, herein described, which has the capability to simulate the steady state performance of gas turbine engines with variable geometry in one or more gas path components.

## 6.1 The Computer Program

VATEMP, Variable Area Turbine Engine Matching Program, is a component matching computer program developed to analyse both the design point and off-design performance of any gas turbine engine with variable or fixed geometry components, and in the case of aircraft engines, it could analyse both the uninstalled and installed performances. It is an extended version of the TURBOMATCH program developed at Cranfield with added graphics capability. The mechanics of operation of Turbomatch are described in [24,34] and will therefore not be explained here save only what is necessary to understand the operation of VATEMP.

Because of the steady flow nature of the gas turbine engine, each component can handle one thermodynamic process at a time and therefore it is possible to simulate the thermodynamic processes going on in the various components in separate sub-programs called "bricks" which may be called upon at any time to calculate the conditions of the fluid at exit from a component, knowing its conditions at entry in addition to component characteristics. As a result, an engine can be considered as being constructed in modular form and by assembling the various bricks in a given fashion in accordance with the turbocode [35], the performance of arbitrary gas turbine engines can be simulated.

The component matching procedure used is not too different from that described in [7] for predicting the off-design performance of gas turbines. The "guesses" and "checks" that are made at various stages as suggested in [7] are replaced by "variables" and "errors"; however, in this case, the errors are used to change all the variables simultaneously in a manner that realistically describe the actual engine behaviour. The balancing technique described in Appendix A is used to reduce the errors to zero which occur only when the engine is in equilibrium.

### 6.1.1 Program Potential

When an engine is running in equilibrium, compatibility of operation must be maintained between adjacent components and also between components coupled to the same shaft. These conditions give rise to four equations of compatibility, viz;

(a) speed,
(b) mass flow,
(c) work, and
(d) pressure (static)

which must be satisfied at all power settings. As a result, it is possible that at certain operating conditions defined by ambient or flight conditions and power setting, loss of engine performance as dictated by component matching may be unacceptable. The use of variable geometry in one or more components could under these circumstances improve the performance of the engine. Vatemp has the capability to vary the geometry of the following gas path components;-

1. Intake,
2. Compressor,
3. Turbine,
4. Mixer,
5. Bypass splitter, and
6. Nozzle.

The various geometries can either be scheduled (with the exception of the intake) or allowed to float (bypass splitter excepted). In the former case, the geometry could be given as a function of one or two parameters while in the latter, the geometry is free to take up a unique position which will satisfy the necessary matching constraints.

The program is structured to simulate the performance of conventional engines with fixed geometry components and variable geometry engines. In the variable geometry mode, it may not be possible to continue operating one or more components as a variable geometry component, especially so when the area is allowed to float, due to the fact that this component or another has reached some limiting condition or constraint. The various possible limiting conditions will be mentioned in the relevant sections when the mechanics of operation of the possible variable geometry components are described later on.

When a variable geometry component reaches a critical limiting area, one of two possible modes of area operation may occur at power setting at and beyond the critical. Either the area is held fixed at the value obtained at the critical power setting or is scheduled as a function of one or two parameters. It is obvious that for any ambient or flight operating conditions there may be two possible critical power settings for each component with variable geometry capability at which variable geometry operation will be aborted since the area of the component can physically attain a maximum and a minimum value. Therefore, a reference is needed which will determine whether the limiting area is on the "low" or "high" side.

Whenever a constraint is violated, for example, a perturbed area taking a value lower than the minimum, switches are activated which will abort operation of the relevant components in the variable geometry mode. The parameter that is used to control engine power is thereafter perturbed in a logical fashion until a value is obtained at which the component in question will operate in the variable geometry mode at the limiting value of area setting. This power setting is taken as a critical power condition and is stored internally so that subsequent power settings at the same altitude, Mach number, and ISA deviation will be compared with it to determine if the components affected will operate with fixed or variable geometry at the desired power setting. Once any of the three parameters mentioned above changes, the stored critical power is lost and the engine will revert to the variable area mode.

If the trial point obtained for the perturbed variable happens to temporarily fall outside the bounds of variable geometry operation, in other words, a false alarm triggered off the switches, the critical power obtained will give an area which lies within the range of variable geometry operation thereby "telling" the program that it was a false alarm. The perturbed variable is then adjusted with the hope that another false alarm will not be given.

A unique feature of Vatemp is its ability to change the number and/or type of variable in a given run. This is essential as the number or type of variable may change when operation from variable to fixed geometry mode is effected, or vice versa. In setting up the input data, the rule is that the engine is configured to operate in the fixed geometry mode in accordance with the rules set out in [34]. In this case, the number of variables must equal the number of errors. Vatemp incorporates one more possible variable which is static pressure, used only with a convergent nozzle when this component is part of a variable area mixer. For variable geometry operation, some of the prescribed variables as set out for fixed geometry operation may be either removed or altered such that only those necessary to effect the desired mode of operation are left. The number of errors may not be equal to the number of variables after this exercise is completed. The program internally alters the errors such that only the relevant errors are activated. This is simple to achieve as a particular error can be thought of as being associated to a particular variable for a given configuration. When fixed geometry operation is effected, the variables and errors may change internally so that the configuration is that of the desired fixed geometry configuration.

The modular structure of the original program is maintained but there may be interaction between components that are not connected by being either on the same shaft or adjacent; this especially happens when a critical power point is being sort and will surely increase execution time which could have been minimized by increasing storage capacity.

### 6.1.2 Brick Structure

It was the intention of the author to maintain the simplicity of the original program so that the procedure for setting up the input data file will not change much. This was accomplished to a certain extent and with difficulty as it would have been easier in certain cases if a different approach was adopted. Many new bricks have been incorporated which perform the calculations for the various variable geometry components. It was decided not to add any new brick names to the codewords but that these bricks will be accessed by switches given as brick data or by the fact that a particular brick or station vector data is given as a variable or it takes a value which has a significant meaning. The most important changes will now be described.

<u>Intake</u>

The elements for the brick data are shown in Appendix B1; the first three do not need an explanation. The intake pressure recovery could either be given explicitly as a positive value not greater than unity or take negative integer values in the range -6 to -1 signifying that pressure recovery is a function of Mach Number. A value of -1 indicates that the USAF standard values are required, otherwise, pressure recovery is obtained from maps representative of typical intake types covering a wide Mach number range.

There are several built in maps which could be used to evaluate the spillage drag of several intake types when installed performance is being evaluated. The map number, in the range 1-4, of the particular drag system used is given as brick data item 5. The intake characteristic is specified by brick data item 6 taking a positive value. The characteristic depicts airflow supply as a function of Mach number. A typical characteristic is shown in Fig. 6.1. Either the value of the map number used or an absolute value of corrected non-dimensional airflow should be given. In the case of the former, the absolute value is determined by the flight Mach number. One such map is supplied internally, but there is provision for the user to supply his own maps up to a maximum of five. If a map is specified, a scale factor is calculated and stored internally for future use.

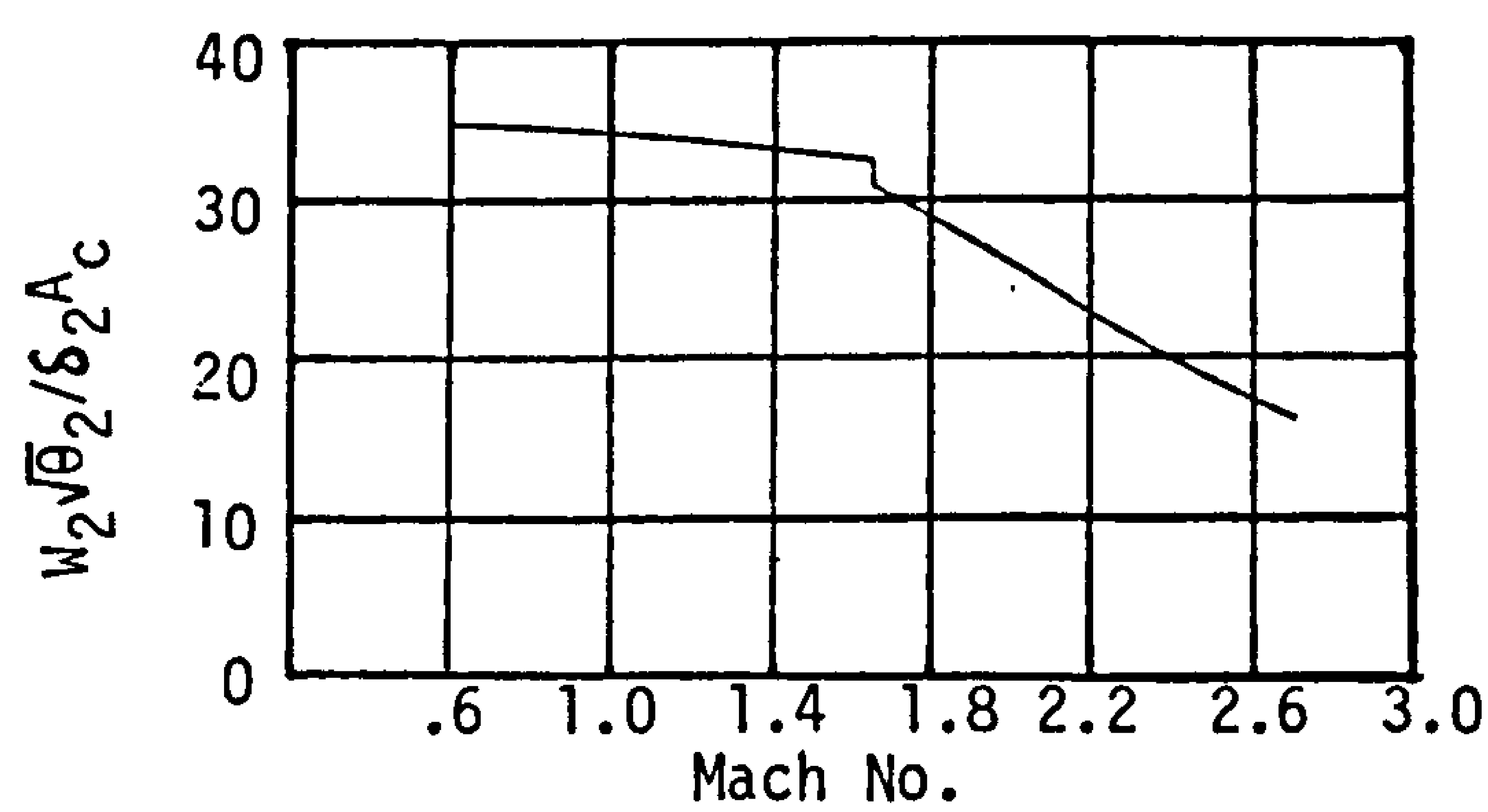



Fig. 6.1 Inlet maximum flow capacity [73]

A simple form of variable geometry can be simulated if it is assumed that the inlet throat area is always set such that only the airflow demanded by the engine is captured. The inlet capture area is calculated if brick data 6 is positive or if the flight conditions for which the inlet is sized are given when installed performance is being evaluated. This will be discussed further in the section on installed performance.

Compressor

Appendix B2 shows the various brick data needed for the compressor module. Not all of these are needed to effect brick calculations. Maps are supplied for both fixed and variable geometry compressors, one such map was shown in Fig. 3.1 for a given fixed geometry and these are stacked for suitable interpolation when the geometry varies. The user has the option to supply his own maps in a specified format. Brick data 7 takes a value other than -100 at the design point if one of the variable geometry maps is used.

The maximum rotational speed may not necessarily be equal to the design speed and this is taken care of in Brick Data item 8. Item 10 is a switch which will indicate if the compressor is operated at constant pressure ratio, and if this is so, a pressure ratio error will be generated. For both constant pressure ratio and constant turbine inlet temperature operation the surge margin may be approached quite closely at low power settings and therefore, it becomes necessary to avoid this. A limit is placed on the surge margin, item 9, which when reached will cause the compressor to operate either conventionally or at the limiting surge margin or with IGVs scheduled, item 11.

The compressor could be allowed to operate at constant speed while variable geometry is used to control airflow into the engine. This is effected either by specifying a value for IGV angle, brick data 7, or by letting this brick data be a variable. When power setting is such that vane travel reaches the maximum or minimum limit, the vanes are held fixed in this position and speed allowed to vary to attain equilibrium. It may be desirable to let such a constraint cause the next compressor downstream to follow suit if it is in the variable geometry mode as well. The reverse is not true; a constraint on a compressor will not cause the adjacent compressor at the lower pressure end to stop operating in its current mode. Brick data 12, which is a switch, allows compressor operations to be connected.

Turbine

The key to improved performance of a variable geometry engine is the variable area turbine as turbine flexibility makes it possible to operate at points on the compressor characteristics that would otherwise not be explored due to the matching constraints imposed by a fixed geometry turbine. (A variable area propelling nozzle can be used to shift the working line on a LP compression system).

There are 15 possible brick data items that are required for turbine calculations as shown in Appendix B3. The types of map used to describe turbine performance are shown in Fig. 6.2 for the fixed area turbine and in Fig. 6.3 for the variable area turbine. The variable geometry maps can also be used for fixed geometry designs if a particular ratio of design/minimum area is specified.

Because of the different forms of presentation of the two types of map, items 2 and 3 of the brick data menu are the design values of either TFF and CN or $\Delta H/T$ and $\Delta H/N^2$ for the fixed or variable geometry maps, respectively. If the variable geometry maps are used then the switch in item 10 is set to a positive value, otherwise, it takes a zero or negative value. Items 11-15 are specified accordingly as is necessary. The LP turbine exit Mach number is necessary to define the area at exit which is used either to calculate the exit Mach number at off-design to check if this turbine has reached limit loading, or to define the "hot" area at design for a mixer, whether variable or fixed.

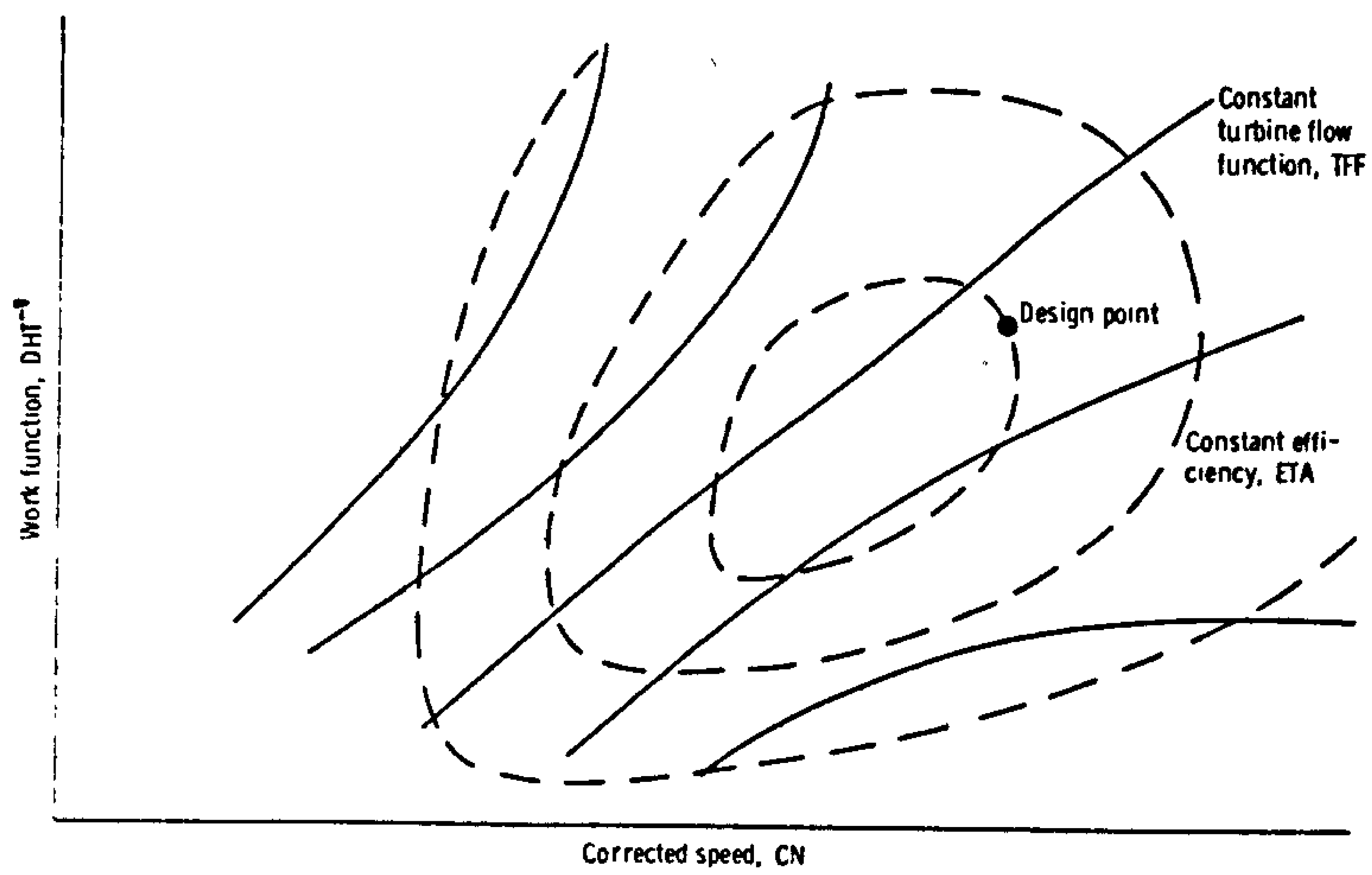



Fig. 6.2 Fixed geometry turbine performance representation

The phenomenon of limit loading is explained in Chapter 9. To preclude this effect which may occur at low power setting, brick item 13 specifies the limiting exit Mach number which will cause the turbine to stop operating in the variable area mode.

It may be desired to control the operation of an engine such that a rotor speed is held constant while power is modulated. Depending on the type of control employed, variable geometry may be needed in the relevant turbine. When a constraint is violated which will cause the turbine to operate with fixed area, it is possible to continue operating the engine with the rotor speed still held constant. This can be accomplished by using variable geometry in the relevant compressor to control the flow. If item 14 in the brick data menu is positive then the rotor speed will be held fixed as will the turbine area, otherwise, it will vary.

As with a variable geometry compressor, a variable area turbine reverting to fixed area mode of operation may cause an adjacent turbine to follow suit, but unlike the compressor, the turbine affected is the one immediately upstream. If the switch in item 15 is set to a positive value, then the mode of operation of the upstream turbine is affected by this turbine. Of course, the upstream turbine should have variable geometry.

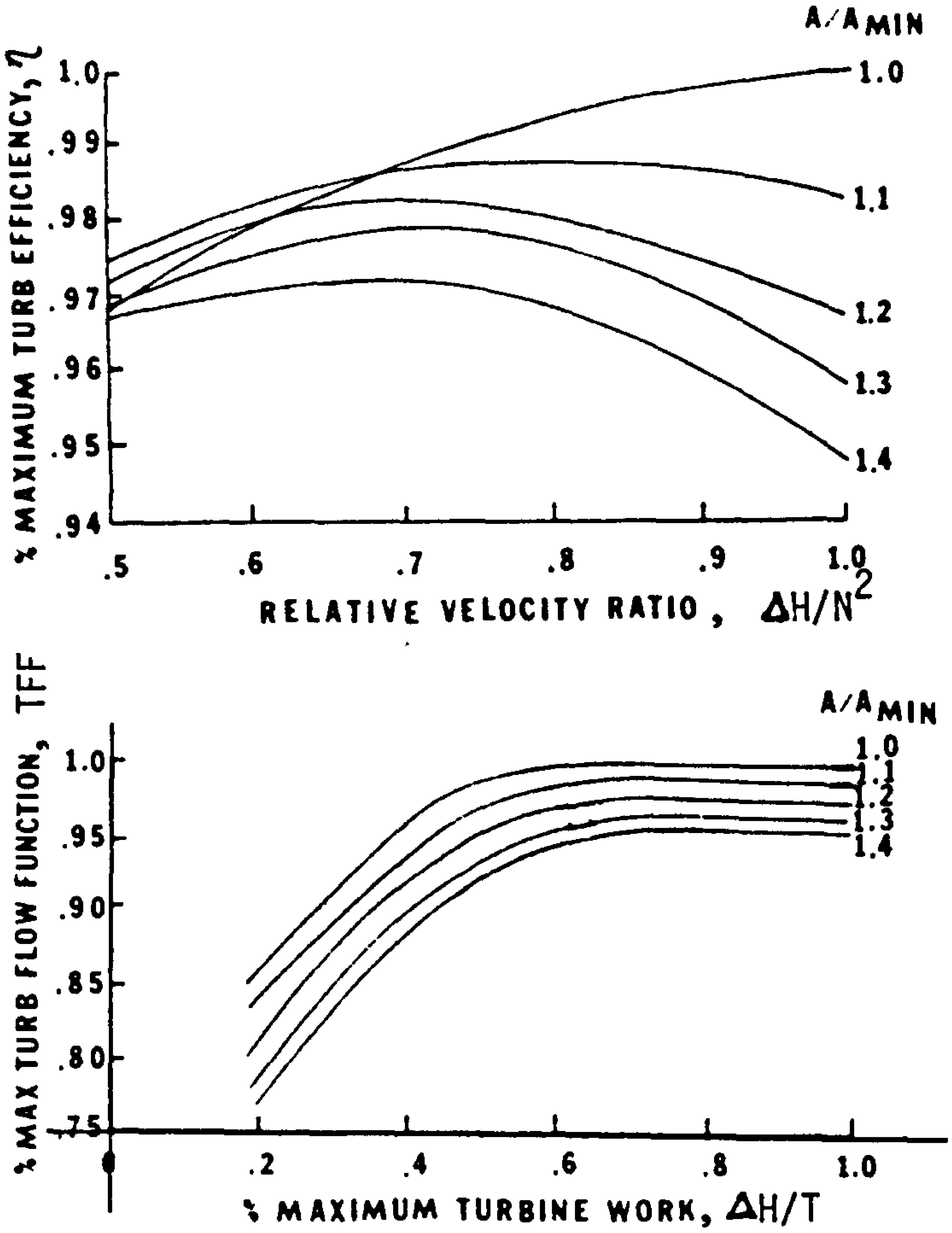



Fig. 6.3 Variable geometry performance maps [27]

When two or more compressors are connected to the same shaft, the compressor number given as brick data 6 is that of the compressor at the highest pressure end but the speed of the shaft is given in brick data item 2 for the compressor at the lowest pressure end.

There are three causes for a variable area turbine returning to the fixed area mode of operation which are directly related to the operation of the turbine and not the engine or some other component. These are,

1. Area limit,
2. Limit loading, and
3. Operating point outside map.

Case 3 is an escape route and does not have any physical meaning except that the turbine cannot supply the power to pass the required mass flow at the given speed. There are of course other causes for a VAT aborting its variable mode of operation such as surge margin limit being reached on a compressor during constant pressure ratio or TIT operation. When this is necessary, the critical power setting as defined by the control parameter, PCN, TIT, etc., is found and the necessary critical areas of the variable geometry affected by this turbine are obtained.

In the variable mode, the turbine generates one error caused by the mismatch of turbine flow function. In the fixed mode, an additional error, turbine work, is also generated as this parameter now becomes a variable which was not possible when the area varied. It is also possible to schedule turbine area with at most two other parameters.

Mixer

The "MIXFUL" brick performs the calculations for the constant area mixing of two flows taking into account total pressure change resulting from momentum balance. Since mixing occurs at equal static pressure, a matching error is generated which reduces to zero at equilibrium.

Variable area is simulated by inserting a variable area convergent nozzle in the bypass stream which is allowed to expand to a specified static pressure which is either the static pressure of the main stream at the last "loop" or the value specified by the perturbed variable which will of course be static pressure. Since the calculations for the bypass stream are done before those of the main stream, the static pressure of the main stream is most likely to be different from the previous value to which the "floating area" nozzle in the bypass stream was allowed to expand. Therefore, an error will be generated in either case. It is assumed that mixing will always take place at constant final area.

A limit may be placed on the Mach number in the bypass stream which is to be given as brick data item 4 in Appendix B4. When this limit is reached, the mixer areas will be fixed and will cause the variable area turbine to which this mixer is associated to operate with fixed area. Depending on engine configuration, the variable associated with this mixer, that is, static pressure, may be temporarily deleted in order to keep the number of errors and variables equal. An error is introduced in the nozzle, that of total pressure mismatch, once the area becomes fixed.

A special case of the mixer occurs when the bypass duct is closed during certain engine operating conditions for certain types of engine. This being the case, the conditions of the gas at exit from the mixer will be equal to those of the mainstream. It is clearly envisaged that for such a case a change in the number of variables and errors is needed to keep the program working. This is also the case when the bypass duct is re-opened.

Since a value of the static pressure to which the nozzle is to expand has to be given at the design point, it is necessary that at the design point the nozzle representing the variable area mixer be removed. The fixed area mixer would then give a static pressure which should be given as a design point value for operation of the convergent nozzle at off-design.

Premas

Future variable cycle engines will have the capability to vary bypass ratio and in certain cases this may be reduced to zero to operate as a turbojet. The bypass splitter can be seen as operating like a valve varying area from fully closed to fully open positions. This process can be simulated by specifying the bypass duct inlet area.

At the design point, the Mach number at inlet to the duct is specified and this will determine the inlet area. Brick data item 6 in Appendix B5 makes provision for this. Strictly speaking, this component, the splitter, cannot be thought of as having variable geometry. It can be considered more as a two position valve which either allows flow or not. When the valve is in the open position, the Mach number at the throat may vary considerably with area change if bypass nozzle area remains fixed. When a zero value is specified, the bypass ratio is no longer a variable and therefore an error will also disappear which will again be introduced when a positive value of area is later on specified to once again activate the bypass stream.

The intake calculations assume that the airflow demanded by the engine is that swallowed by the intake and therefore when the engine is balanced, the intake airflow is equated to that of the engine. Intakes may have blow-in or dump doors which when open will surely cause the intake and engine airflows to differ. Item 5 in this brick data menu is a switch which will indicate if this brick represents such a door in the intake so that the actual intake flow will be evaluated having known the engine airflow.

Nozzle

A variable area propelling nozzle is widely used nowadays especially in military installations and therefore would not be expected to pose serious operating difficulties when used in future variable geometry engines. Since the components downstream of the combustor split the energy in the burnt gases, a variable area propelling nozzle will have to be used in conjunction with a VAT. Appendices B6 and B7 describe the brick data needed to effect operation of convergent and con-divergent nozzles, respectively. If variable geometry is to be used in the nozzle then the exit area may as well be positioned to give full expansion. This philosophy is used in all variable area calculations.

If brick item 1 in Appendix B6 is given a positive value, then the area floats so that full expansion to the relevant static pressure occurs, otherwise, the exit area is fixed with the likelihood of operating either under- or over-expanded. Brick data 2 is used more like a switch indicating that the nozzle is part of a variable area mixer. Since the operation of a variable area mixer can be related to that of a variable area turbine, the relevant turbine number is input. Item 1 in Appendix B7 is similarly used as that for the convergent nozzle. However, since two areas are involved, there is provision for allowing one area to float while the other is held fixed. If any of the areas for either nozzle is held fixed, it can be changed during a run by either specifying or scheduling it.

Duct/After-Burner

Due to the change in density of a fluid when heated, an area downstream of a duct with fuel burning capability has to change if the engine operating point should not change when the duct is lit. Usually, a nozzle with variable area is situated immediately downstream to cater for this. The switch in item 1 of Appendix B8 can take one of four possible values indicating the present state of the duct.

The burner can be considered as being attached to or separated from the engine, when lit, by the desired choice of the switch. If the burner is separated from the engine, then the engine does not "know" whether or not the burner is on and therefore, its operating point will not change when the burner comes on. When the switch takes a value of 2, the engine is balanced on the assumption that the nozzle downstream is operating with fixed area with the burner off. When equilibrium is obtained, the burner is then switched on and the nozzle areas allowed to float to give full expansion. If the burner is attached to the engine, the operating point of the engine may change when the burner is lit depending on the value of nozzle throat area specified.

Item 5 of the appendix specifies a Mach number at design which is used to evaluate the design exit area. This is only needed for ducts immediately preceding a nozzle as the program assumes that the nozzle exit area will not exceed the inlet area in value. If the nozzle tries to operate with an exit area greater than that at inlet, the nozzle area is then set to operate at this value thereby incurring under-expansion losses.

### 6.1.3 Installed Analysis

The installed performance of an aircraft engine is obtained by evaluating the terms on the right hand side of Eq. 4.1. Since the throttle dependent trim drag is normally small, it is neglected here. Also, the thrust loss due to power and bleed off-take is neglected since this is installation dependent and may be quite difficult to quantify. Therefore, Eq. 4.1 reduces to

$$F_{INST} = F_G - D_R - \Delta F_{\eta R} - F_G\left(1-\frac{C_{FGR}}{C_{FGT}}\right) - \Delta D_{INL} - \Delta D_{AFT-END} \qquad (6.1)$$

Since most calculations do involve evaluating the uninstalled thrust only, the performance engineer would normally include the thrust loss due to pressure recovery and nozzle internal performance as he would have chosen both the intake and nozzle and therefore would like to take into account the performances of these components. If the switch to indicate that installed performance is required is not activated, then the program takes the first four terms on the right hand side of Eq. 6.1 as that for installed thrust with $C_{FGT}$ set equal to 1, otherwise, the true expression for uninstalled thrust is used and this is equal to the first two terms on the right hand side.

If the user wants to evaluate installed performance, a value for the ratio of maximum nozzle exit area to maximum fuselage cross-sectional area, $A_{9max}/A_{10}$, see Fig. 4.7, must be given. This actually activates the installed performance switch. A rigorous and detailed evaluation of installed performance will involve writing a separate program which could even be as large as Vatemp; this aspect of aircraft development is usually carried out by airframe manufacturers. A simple analysis using the procedure outlined in Chapter 4 is used here to obtain a first hand knowledge of installation effects.

The data for inlet and aft-end drags do not normally contain only the throttle-dependent portions of aircraft drag, therefore, the reference conditions have to be set up and the drag thus obtained should be included in the aircraft drag polar. Since the expressions for inlet and aft-end drags do involve the inlet capture and maximum fuselage areas respectively, these two areas must be evaluated. It is assumed that the maximum cross-sectional area to which the nozzle exit area will expand equals that at jet pipe exit. Therefore, by specifying a design Mach number at the duct exit as explained in the section on duct/after-burner operation, the maximum nozzle exit area is obtained. Therefore, $A_{10}$ can be obtained by specifying a representative value for $A_{9max}/A_{10}$. Immediately after the design point calculations are performed, the flight conditions and power setting at which the intake is sized are input. This will define the intake capture area used to evaluate intake drag.

As explained in Chapter 3, a correction has to be performed on nozzle thrust coefficient to take into account changes in this parameter when the test nozzle differs from the actual. Since the program has only one nozzle internal performance map which is represented in the same variables as those of Fig. 4.6, it is not possible to simulate engine performance with different test and actual nozzles. Therefore, it is assumed that the test nozzle is an ideal one which will give a thrust equal to the ideal as obtained from fluid measurements at nozzle inlet. Therefore, to obtain the uninstalled thrust, the engine is balanced with both inlet pressure recovery and nozzle velocity coefficient set to unity. In other words, both the inlet and nozzle are removed. The thrust loss due to nozzle internal performance is obtained by re-introducing the nozzle and subtracting the thrust so obtained from that with no nozzle.

To calculate the remaining terms in the expression for installed thrust, another pass is made to balance the engine with both inlet and nozzle in place. The inlet drag coefficient is obtained from a map which gives the additive drag coefficient as a function of Mach number and mass flow ratio. Data are not available to enable calculation of the entire inlet drag. Therefore, taking additive drag as being representative of inlet external performance will introduce serious errors into the results. However, for trend studies, meaningful conclusions can be drawn concerning inlet performance since a greater portion of inlet drag is accounted for by additive drag.

The reference inlet drag which is to be included in the aircraft drag polar is defined here as the inlet drag which occurs at maximum power conditions at any given flight conditions. In order to determine this drag, the program is first run with the data for maximum power at the specified flight conditions. Thereafter, the inlet drag is obtained by taking the difference between the present value and that for maximum power.

The exhaust system reference drag is defined as the drag occurring when the nozzle flaps are fully opened to give a cylindrical contour with the fuselage at the "customer connection" point, that is, $A_9 = A_{11}$, Fig. 4.7, and with fully expanded flow. Since the nozzle aft-end drag coefficient is a function of $A_9/A_{10}$ and $P_{s9}/P_a$, the reference drag coefficient is easily obtained from the value of area ratio which turns on the installed performance switch and a static pressure ratio of unity.

The data available within the program for evaluating aft-end performance is for a Mach number of 0.9. If an attempt is made to evaluate a drag coefficient at any other value of Mach number, a zero value will be returned. The drag coefficient at any power setting is obtained after taking into consideration the reference value.

On subtracting the various thrust and drag terms from the value of installed thrust obtained on the second pass, the thrust loss due to pressure recovery is obtained. The data print out contains the component performance for the installed engine with a break down of both the uninstalled and installed performances.

### 6.1.4 Special Features

Some special features have been incorporated in Vatemp which gives it some degree of flexibility. Some of these are described below.

Change of Variables

As was mentioned earlier, the fact that an engine's mode of control may change as a result of some constraint being violated may cause some of the variables associated with certain components to change. This is done automatically within the program and requires no assistance from the user. However, there may be an instance when the user desires to change the mode of control. For example, power may be modulated by controlling fan speed, say. The user may wish to fix fan speed at some time in the run and modulate power by specifying IGV angle; or, he may want to specify TIT while allowing fan speed to vary. This can be accomplished by inputting the change as an off-design point sandwiched by two minus fours (-4s). The procedure is described in Appendix C1.

The Scheduling Function

It may be desired to schedule a particular parameter rather than give a value for it at an appropriate time during a run. A Codeword Descriptor starting with the letter "A" makes this possible. The procedure is described in Appendix C2.

The proposed parameter can be scheduled as a function of one or two parameters. Each parameter in the functional relationship can either be a single brick data or station vector item, or it can be a combination of these, within limits. The relationship is given graphically and the data are input according to the specified format, Appendix C3. When assembling the input data, the data for all scheduling functions are given immediately after the "-1" which indicates the end of inputting all optional turbine maps. The data are also terminated with a "-1".

### 6.1.5 Graphical Output

A routine is included in Vatemp which gives it graphics capability. The graphics session occurs after all off-design points have been evaluated. There is no facility for plotting design point performance variables. Any two of twenty possible variables can be plotted and the plotting codes can be read either interactively or from the input file. The data to be plotted are read from the relevant data files in accordance with the codes. The graphs are fully labelled and titled.

A maximum of six different graphs can be plotted on a page with up to six plots per graph. The page can be orientated with the abscissa axis lying along the longer or shorter side of the sheet to give a landscape or portrait plot, respectively. Appendix D gives the menu of the variables that can be plotted and it also describes the mechanics of operation of the routine.

# CHAPTER 7

## SHAFT POWER CYCLE STUDIES

The shaft power cycle gas turbine is used in a wide variety of applications to meet the varying and diverse power requirements of many mechanical systems. The type of application may range from providing torque to drive a ship's propeller to providing power for traction for vehicular systems. These turbines may also be required to generate power for a mechanical drive system such as a pump.

In some applications, the turbine engine is required to operate at or quite close to its design point for extended periods of time whereas with the vast majority, a wide range of power excursion is desirable with a considerable amount of running time being spent at intermediate and low power settings. Since the gas turbine is characterized by operating efficiently over a narrow range, usually at conditions quite close to the designed, some operational problems may be encountered when the engine is required to operate at low power for extended periods of time. Two such applications are marine and vehicular propulsion. The poor part load performance of the gas turbine has rendered it an unsuitable candidate for applications where performance at low power is of great importance.

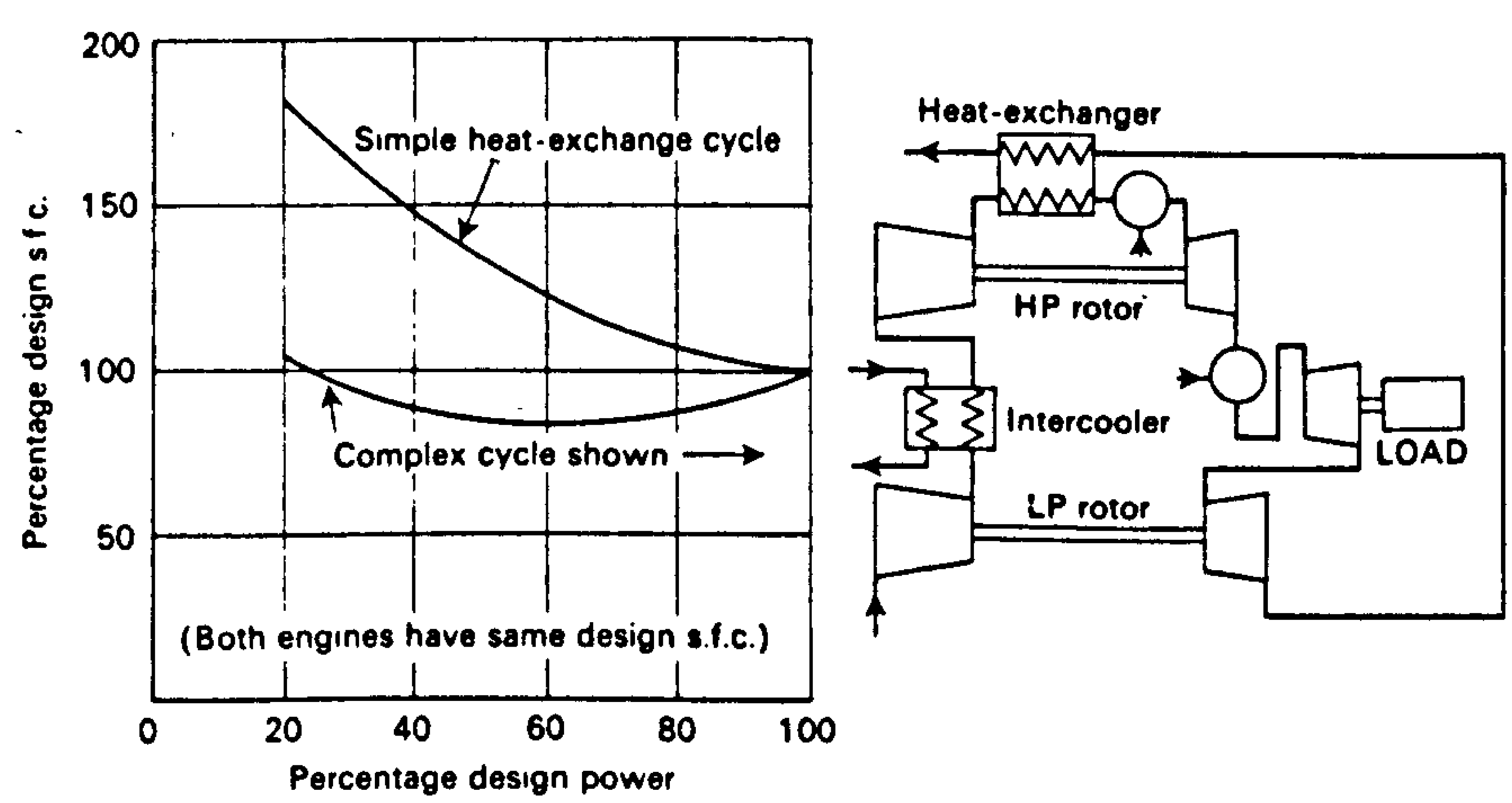



Fig. 7.1 sfc curves for simple and complex cycles [7]

Early attempts to solve the poor part load performance looked at compound cycles involving heat exchange, reheat, and intercooling. At a first glance, the thermodynamic merits of such cycles, Fig. 7.1, justified the use of these components; however, the enormous mechanical complexity involved rendered these concepts unsuccessful.

A greater number of shaft power gas turbines in use today are either of the high pressure ratio simple cycle type or the low or moderate pressure ratio type incorporating heat exchange. As was shown in Fig. 3.6, the thermal efficiency of the simple cycle type is dependent on both the overall pressure ratio and the ratio of turbine inlet temperature to compressor face temperature while the latter alone affects the thermal efficiency of the regenerative cycle at very high turbine inlet temperatures. For either cycle with fixed geometry components, the turbine inlet temperature must be reduced in order to reduce power, and it turns out that the rapid drop of turbine temperature with reduction in power is mainly responsible for the poor part load performance of the shaft power gas turbine.

Any means which will cause the turbine inlet temperature or compressor pressure ratio to be held at a level higher than that given by conventional engines will surely improve cycle performance. One such means is the variable area turbine. By allowing the appropriate turbine to vary its area, the turbine inlet temperature or a compressor pressure ratio can be maintained at the design or a higher value throughout the entire power range or most of it. An investigation of the use of variable geometry in both the turbine and compressor components is presented in this chapter. Both the constant pressure ratio and constant turbine inlet temperature cycles will be examined for both the simple and regenerative cycles.

## 7.1 The Simple Cycle

The cycle chosen for study is that of a single spool free turbine engine. The engine selected for simulation is one that has been in service for quite a number of years and an interest was taken in it as it utilizes variable geometry in the compressor. The design point performance is given in Table 7.1. Representative pressure losses and component efficiencies were judiciously chosen to give the specified performance for the given pressure ratio, mass flow, and fuel flow. The turbine inlet temperature is that which resulted from the specified fuel flow.

Design Point Performance (ISA SLS)

| | | |
|---|---|---|
| Pressure Ratio | | 14.8 |
| TIT | (K) | 1388 |
| Airflow | (kg/s) | 13.3 |
| Power | (MW) | 3.266 |
| sfc | ($\mu$g/J) | 79.4 |
| $\eta_{th}$ | (%) | 29.2 |

Table 7.1 Turboshaft Performance

## Choice of Design Point Surge Margin

It can be perceived that for the constant pressure ratio cycle as speed is reduced to decrease power, the surge margin will be eroded and therefore remedial action will have to be taken to avoid the operating line running into the surge line at low powers or during acceleration from low speeds. The problem then arises of choosing a design point that will permit unrestricted operation while maintaining acceptable efficiency levels.

As is shown in the figure below, design point A will cause the engine to approach surge at intermediate powers whereas point B may give unrestricted operation but as is clearly seen, point B operates at a lower efficiency though there is the tendency for efficiency to improve as power is

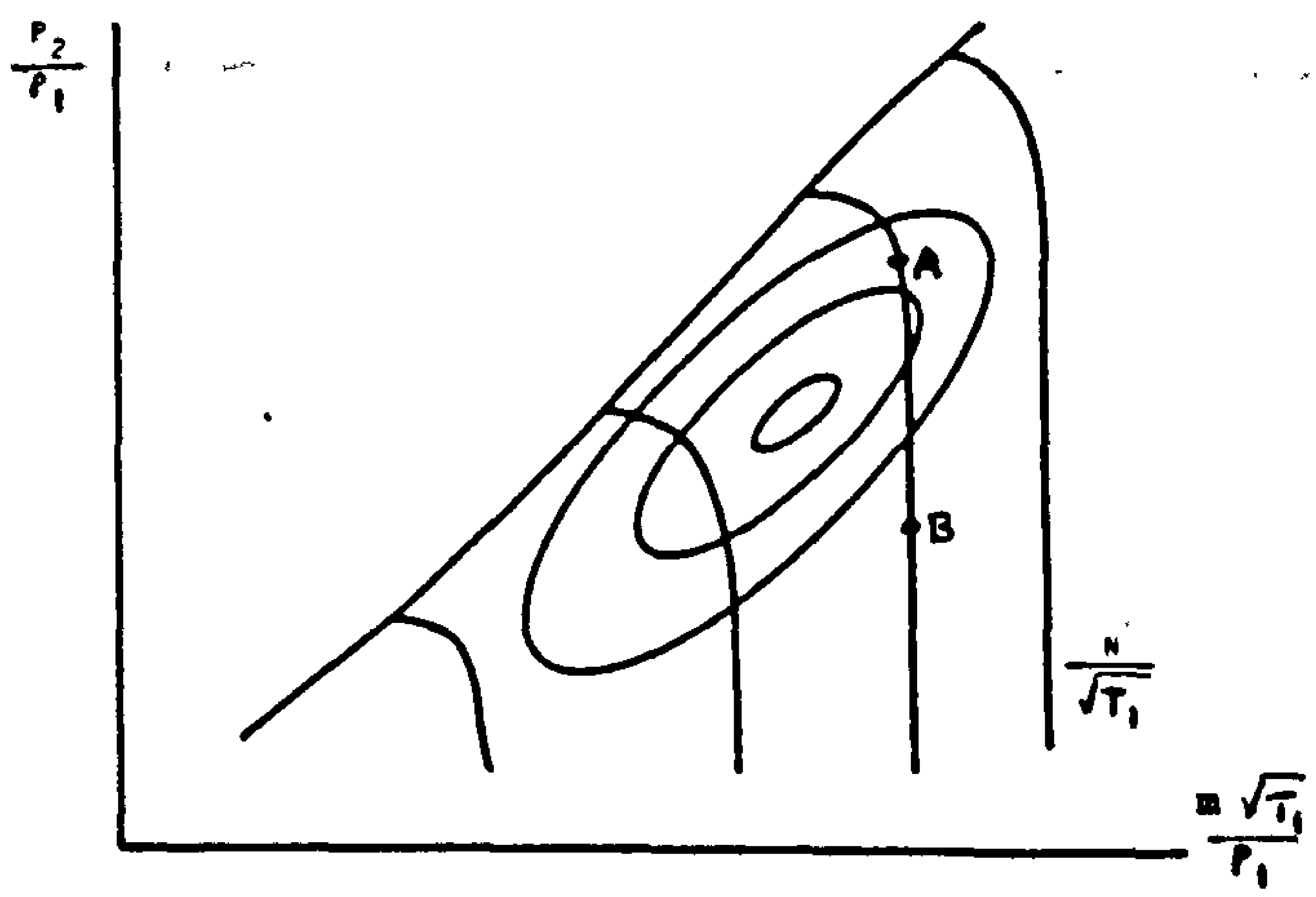



Fig. 7.2 Effect of design point surge margin on off-design operation

reduced. This problem arose owing to the nature of the type of compressor characteristics used. The available compressor maps have contours of highest efficiency at intermediate speeds and are therefore more suitable for aircraft application. A compressor designed to have peak efficiency at point B would be quite suitable for constant pressure ratio application. (Note: The above map will have

## Surge Margin Effect On Turboshaft Performance

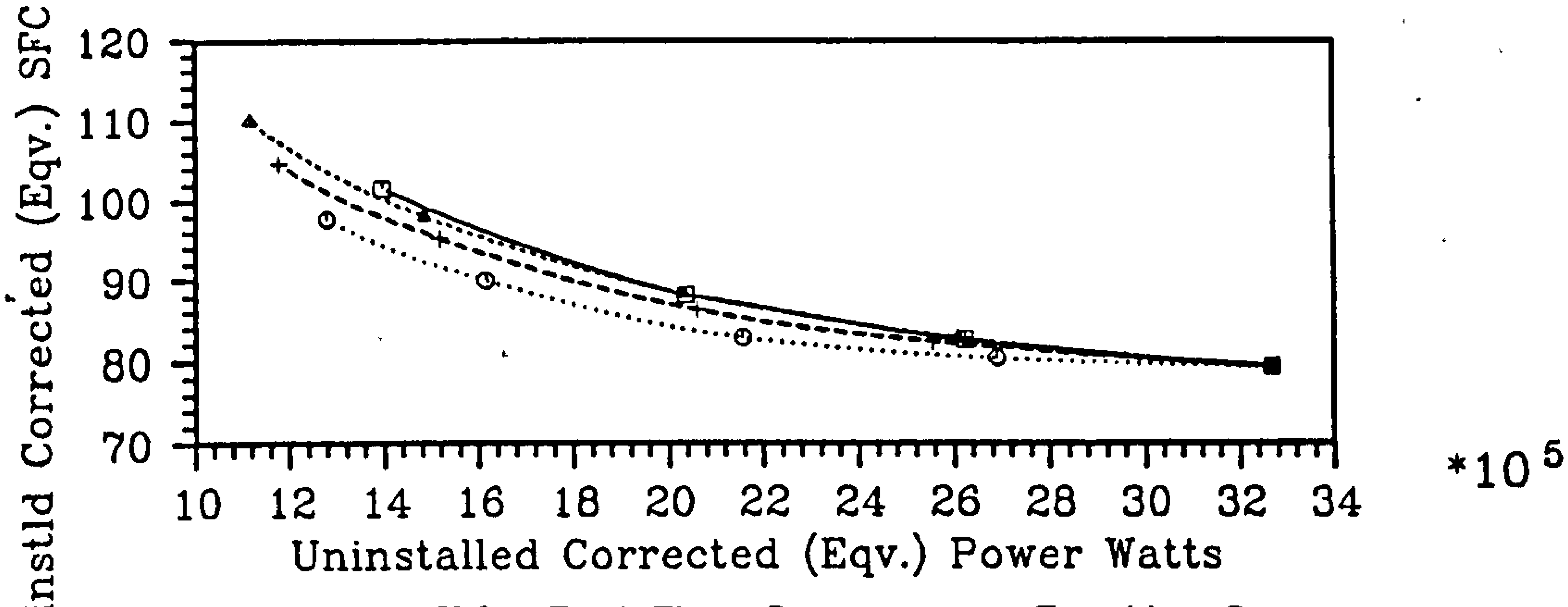



Fig. 7.3a Fuel Flow Requirement For Unit Power

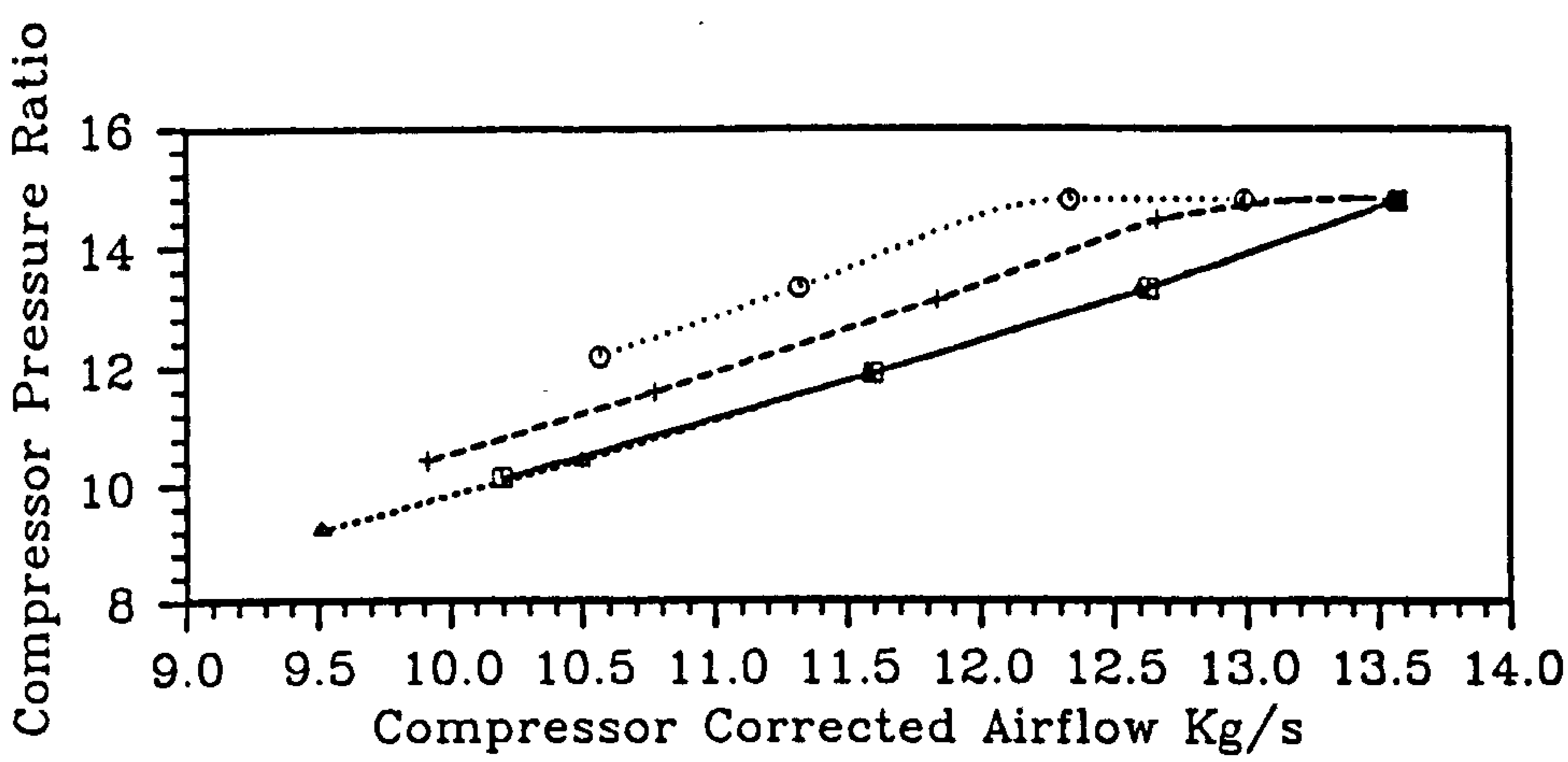



Fig. 7.3b Compressor Operating Line

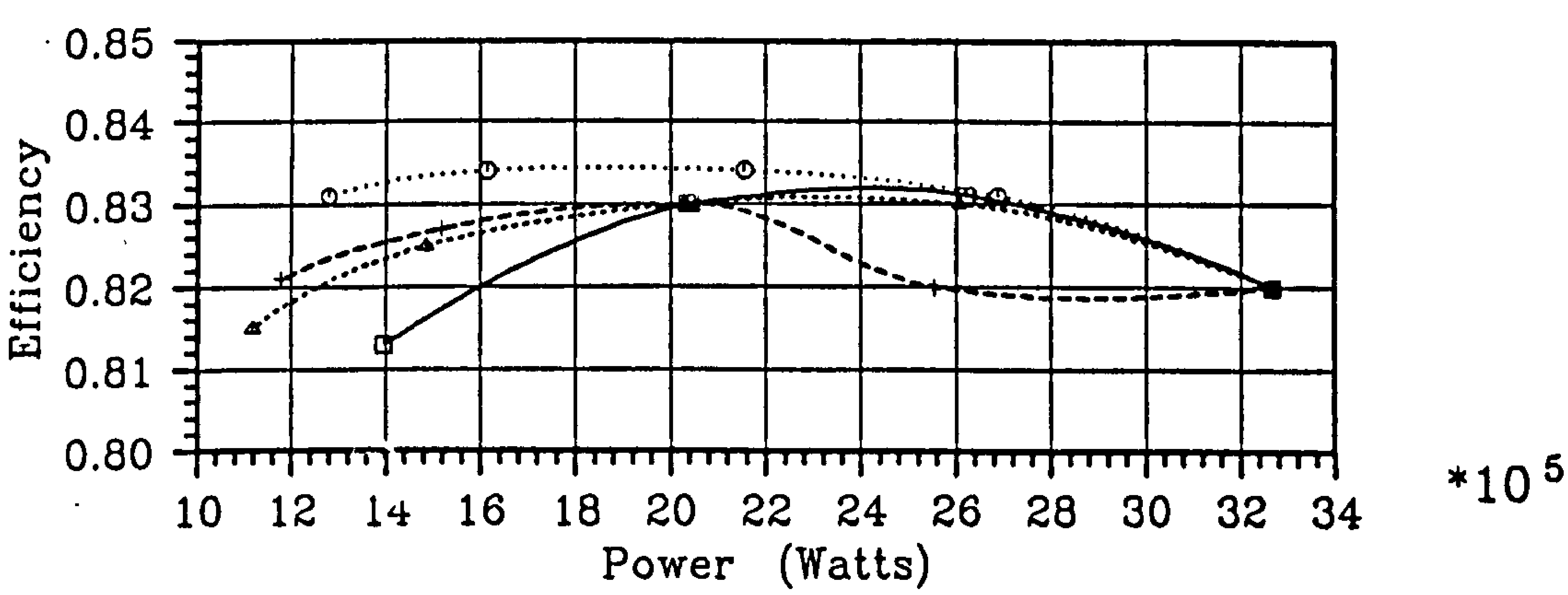



Fig. 7.3c Compressor Efficiency Variation

Z = .74 Conventional Control

Z = .74 Constant Pressure Ratio/Conventional

Z = .85 Conventional Control

Z = .85 Constant Pressure Ratio/Conventional to be scaled such that points A and B give the same design point parameters).

Fig. 7.3 shows how the performance of a constant pressure ratio engine compares with that of a conventional engine for two different design point surge margins. Both engines have the same design point performance. The first two plots are for the engines having a Z parameter value of .74 whereas the other two have a Z value of .85. Variable geometry is used in the HP turbine to hold pressure ratio fixed. A minimum value of surge margin equal to 10 percent was maintained.

The sfc's are compared in Fig. 7.3a in which it is seen that the position of the design point on the compressor map slightly affects the fuel consumption of the conventional engine whilst a lower surge margin at design point causes the surge margin limit to be reached prematurely and therefore affects the overall performance of the constant pressure ratio engine. In this case, the constant pressure ratio cycle offers no significant performance improvement of over that of the conventional cycle whereas at about half load, the cycle with about 26 percent surge margin at the design point gives a saving of about 6.7 percent in fuel consumption over the conventional. Fig. 7.3b shows that at this loading, the constant pressure ratio cycles are operating conventionally as surge margin limit had been reached. Fig. 7.3c shows the variation of compressor efficiency.

<u>Engine Control at Surge Margin limit</u>

When the running line approaches the surge line dangerously close, action is to be taken to avoid surge and possible catastrophe. The simplest remedy is to abort constant pressure ratio operation and revert to conventional control. It is possible that this may not provide a solution as the operating line in the fixed geometry mode may as well run into surge. Two other possible modes of control were considered for operation at power levels below that obtained at the surge margin limit. The first uses variable geometry in the compressor while the other considers variable geometry in the relevant driving turbine to hold surge margin fixed at the limiting value.

The magnitude of performance improvement of the constant pressure ratio cycle over that of a conventional cycle in the power range where constant pressure ratio is no longer possible depends greatly on the amount of power modulation obtained at constant pressure ratio and also on the type of control employed when surge margin limit is reached. The latter will be discussed in detail in the next section while the former has already been highlighted when discussing Fig. 7.3 in the above section. As was explained earlier, the choice of a higher Z parameter which is equivalent to a lower surge margin at the design point causes the engine to reach surge margin limit over a narrower power range resulting in a small or insignificant improvement in performance, Fig 7.3a. The importance of the driving turbine's performance in this power range will be discussed later.

### 7.1.1 The Constant Pressure Ratio Cycle

The performance of the constant pressure ratio cycle will be compared with that of a conventional cycle, and also, three different modes of control will be considered for the constant pressure ratio cycle when the limiting surge margin is reached. These are,

1. Conventional,
2. Constant Surge Margin, and
3. Scheduled IGVs.

The surge margin at design point was set at 20 percent and a surge margin limit of 12 percent was specified for the compressor. This value ensured that this constraint and not the minimum area constraint caused the engine to abort constant pressure ratio operation.

Fig. 7.4a shows the various compressor running lines. The compressor characteristics were not plotted as one of the engines used variable IGVs. Plot one is that for the conventional engine with all geometries fixed and the remaining three are for the constant pressure ratio case in the order as listed above. The sfc curves are compared in Fig. 7.4b where it is seen that the constant pressure ratio cycle gives better performance compared with the conventional. At a power setting of about 2 MW which is about 60 percent of the maximum, a saving of about 4 percent is obtained in fuel consumption while at about 50 percent power, this value varies from 1.7-4.0 percent depending on the type of control employed at power levels lower than the critical.

As TIT decreases to reduce power, the HP turbine area decreases to maintain the compressor pressure ratio fixed while the turbine operates choked. This is shown in Fig. 7.5a where the area progressively decreases up to the point where the surge margin limit is reached. If the surge margin is held fixed at low power levels, the turbine area opens up to provide at a lower pressure ratio, the power to drive the compressor along a line of constant surge margin. Since the pressure ratio across the turbines is also held fixed when the compressor pressure ratio is fixed, the pressure ratio across the HP turbine and therefore, turbine work function increases as the LP turbine extracts less power, Fig. 7.5b. The LP turbine pressure ratio variation

## Turboshaft Performance With Different Controls

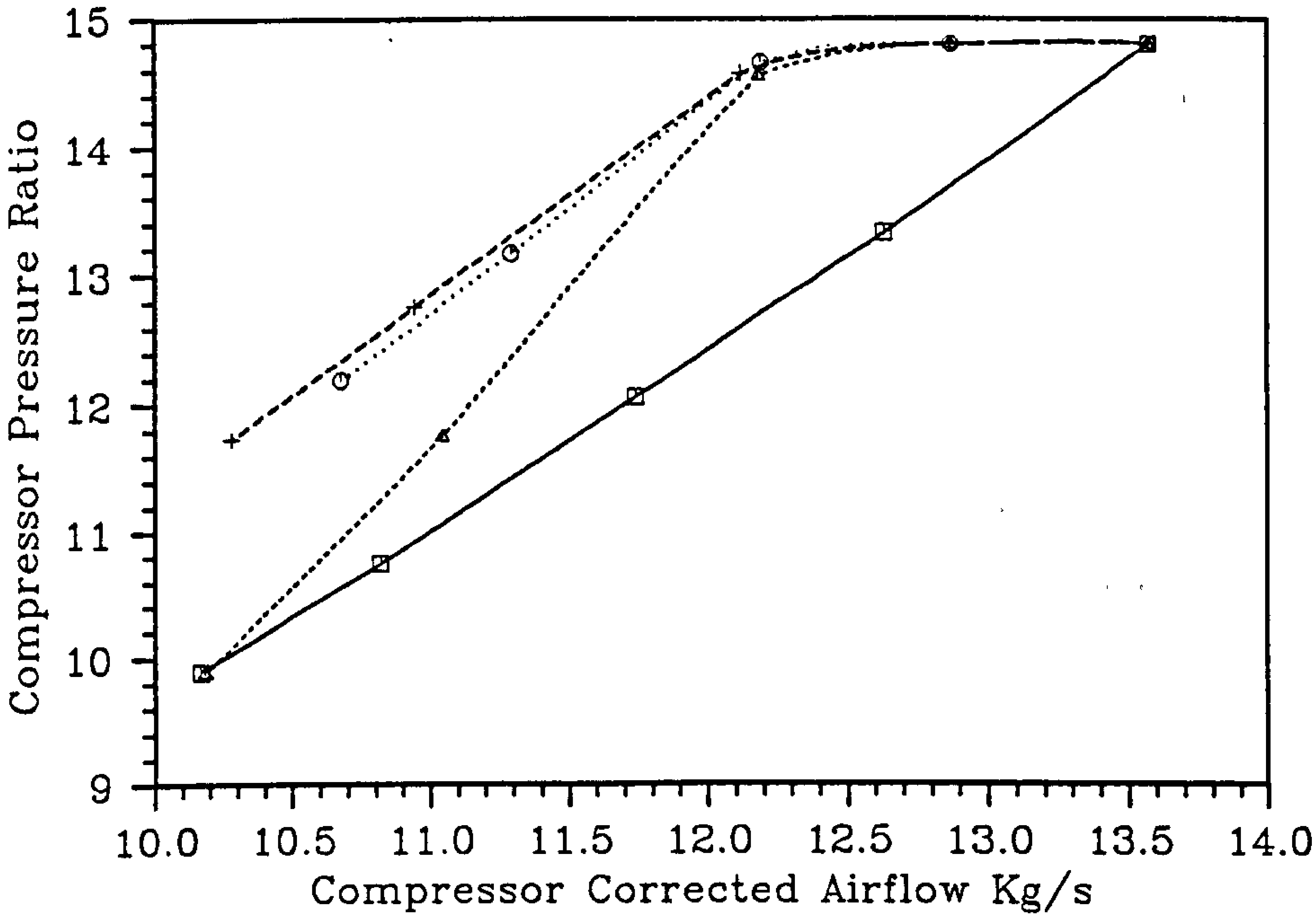



Fig. 7.4a Compressor Operating Line

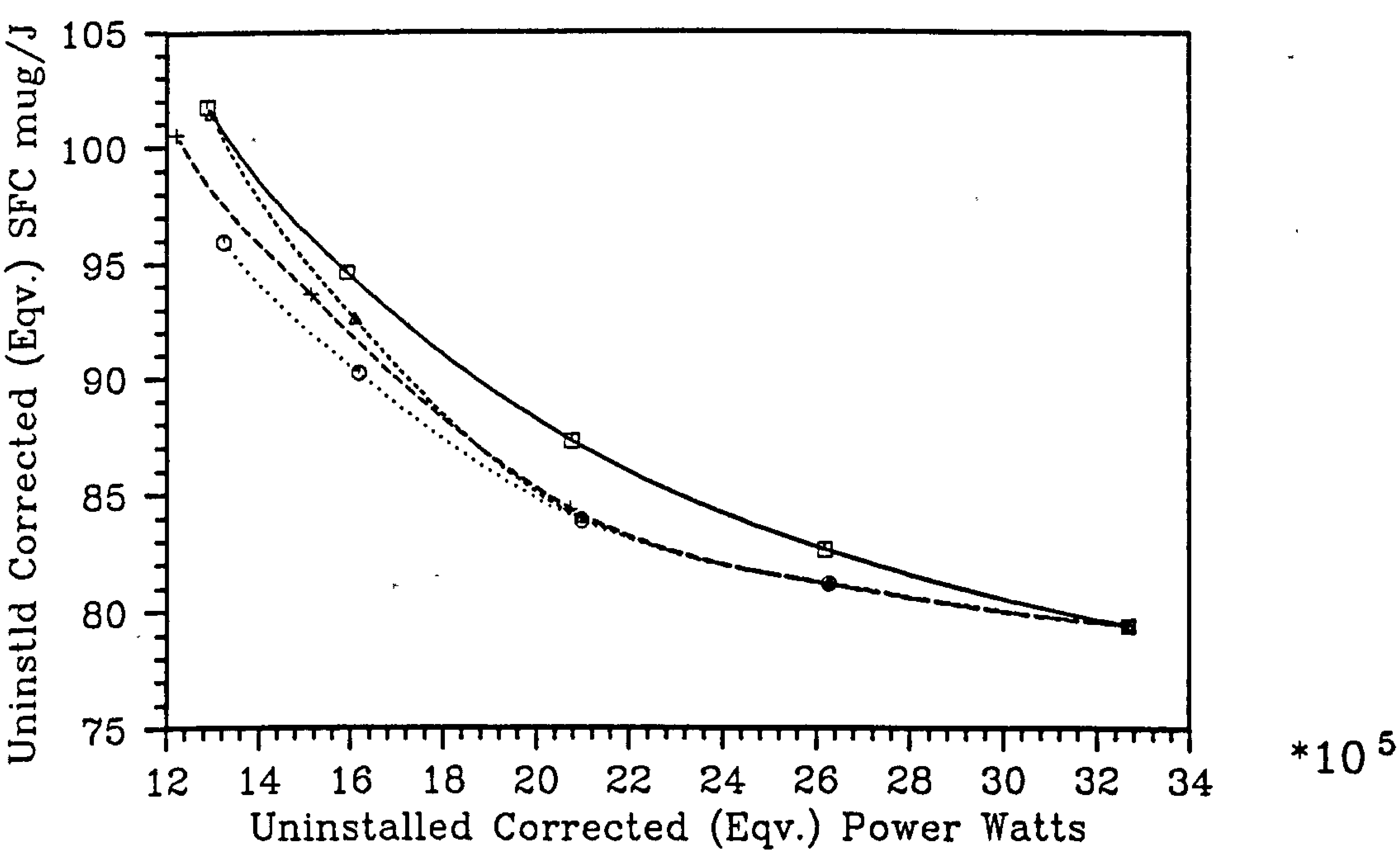



Fig. 7.4b Fuel Flow Requirement For Unit Power

Alt (m) 0.0
Mn 0.0

□—□ Fixed Comp. Vanes/ Fixed Turbine Area
○····○ Fixed Comp. Vanes/ Var. HP Turbine Area
▲····▲ Fixed Comp. Vanes/ Var. HP Turbine Area
+---+ Var. LP Comp. Vanes/ Var. HP Turbine Area

# Turboshaft Performance With Different Controls

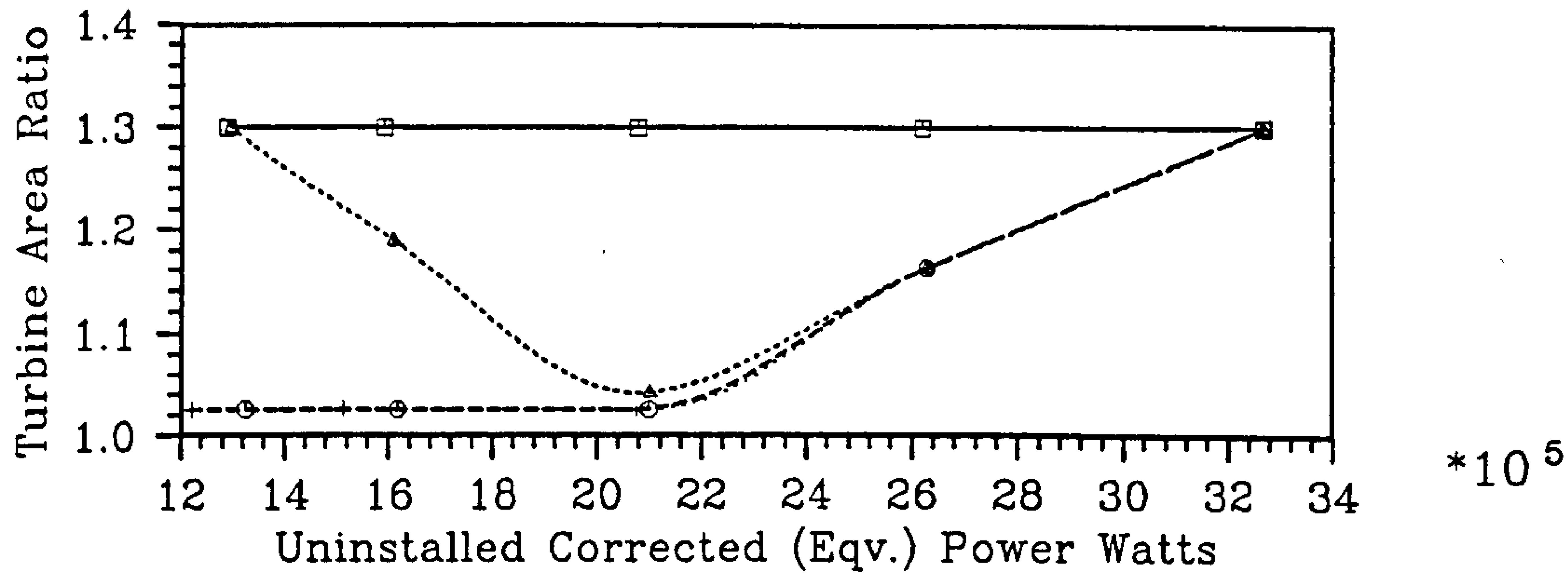



Fig. 7.5a HP Turbine Area Schedule

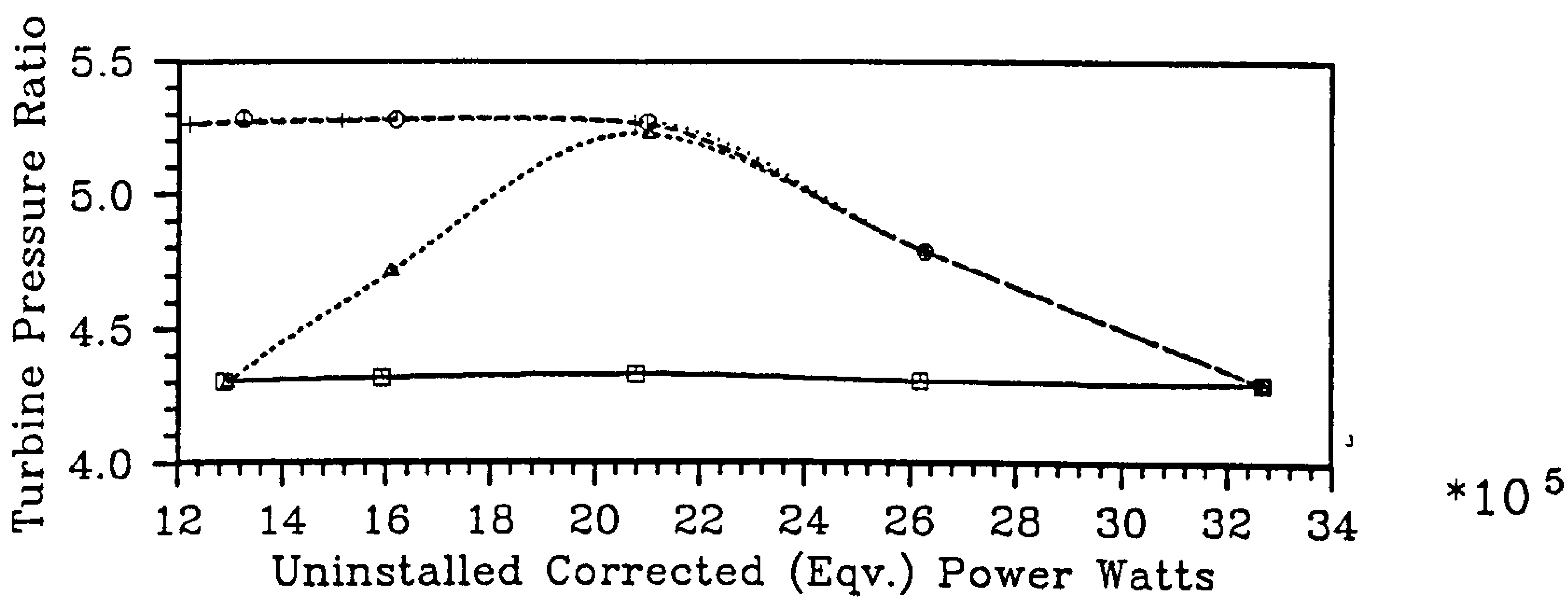



Fig. 7.5b HP Turbine Pr. Ratio

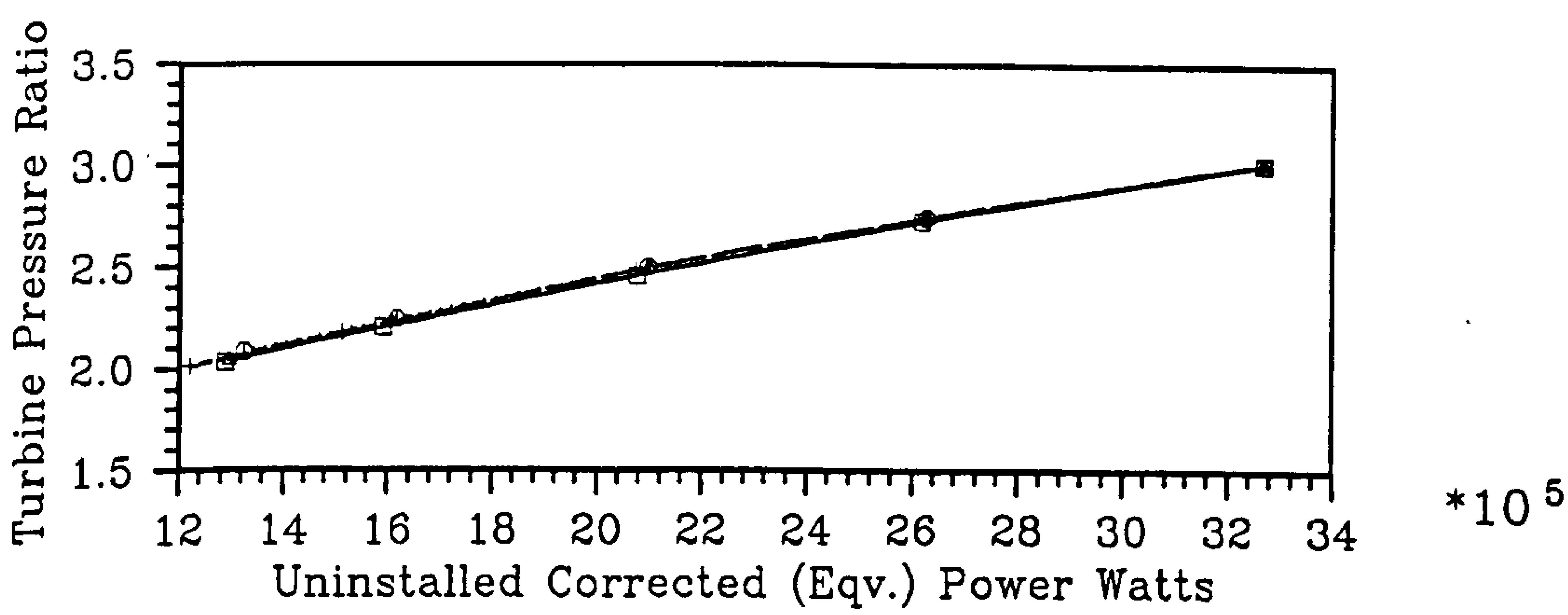



Fig. 7.5c LP Turbine Pr. Ratio

Alt (m) 0.0
Mn 0.0

⊞—⊡ Fixed Comp. Vanes/ Fixed Turbine Area
○····○ Fixed Comp. Vanes/ Var. HP Turbine Area
▲····▲ Fixed Comp. Vanes/ Var. HP Turbine Area
+---+ Var. LP Comp. Vanes/ Var. HP Turbine Area

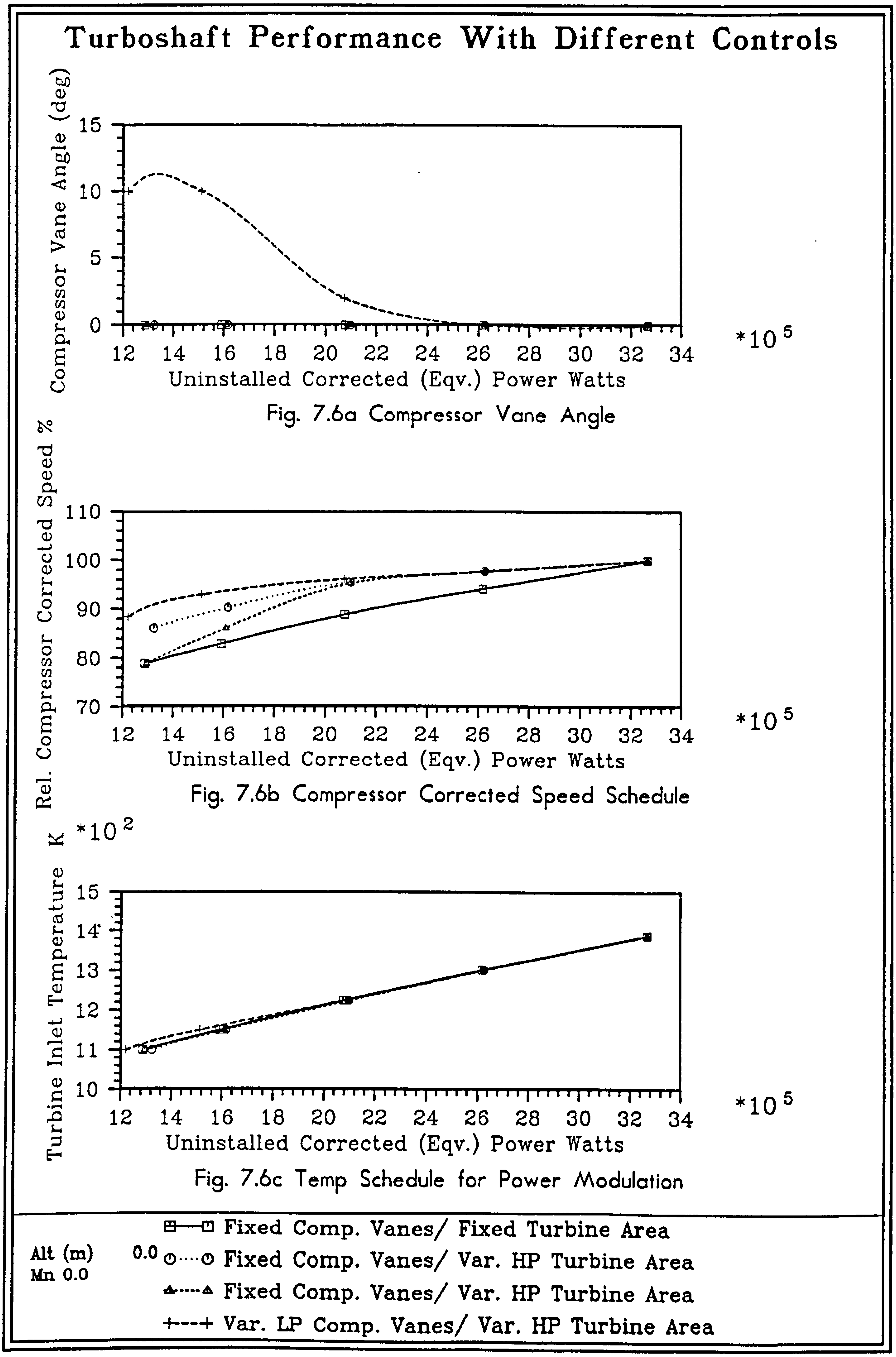
is shown in Fig. 7.5c and not surprisingly, the pressure ratio progressively reduces with power.

It is clearly seen in Fig. 7.4a that if the engine is allowed to operate in a conventional fashion once the surge margin limit has been reached, the components match such that the margin is further eroded. Therefore, another mode of control would be more desirable in this case. Surge margin can be held constant as shown by the third curve in the same figure or the IGVs can be scheduled so as to maintain adequate surge margin. The IGV schedule shown in Fig. 7.6a was used. It was found out that this did not give any improvement in surge margin over that of conventional control. Adequate surge margin could however be maintained but this would require operating at undesirably low speeds. An investigation was carried out to find out if an improvement in performance can be obtained in terms of fuel saving and surge margin maintenance by allowing rotor speed to remain fixed at the value obtained at the critical power, that is, the power at which surge margin limit is reached, while allowing the IGVs to vary to give the required matching. It was found out that surge margin could not be kept within limits and the fuel consumption was higher due to poor compressor performance.

Fig. 7.6b shows how the speeds compare. The constant pressure ratio engine gives a better response and the response at power levels below the critical power varies with the type of control deployed. As can be seen in Fig. 7.6c, the HP turbine runs at the same temperature in all modes of control at any given power setting.

Going back to Fig. 7.4b, one would attempt to conclude that there is no point in using any other control at power levels below the critical in an attempt to get improved performance as the conventional control of allowing speed, pressure ratio, and mass flow to fall with decrease in fuel flow gives the best cycle performance in this power range. However, as is seen from the plot of the compressor operating lines in Fig. 7.4a, this would entail violating the surge margin constraint. Operation with constant surge margin gives sfc values progressively approaching those for the conventional fixed geometry engine. This is to be expected as the operating line steadily approaches that of the fixed geometry engine as power is reduced, with an attendant increase in compressor efficiency but with a decrease in HP turbine efficiency, Fig. 7.7. The performance of the LP turbine is not at all influenced by that of the HP turbine.

The improvement in compressor efficiency is outweighed by the detrimental effects of the decrease in turbine efficiency as turbine area increases to hold surge margin


Turboshaft Performance With Different Controls
Compressor Vane Angle (deg)
15
10
5
0
12 14 16 18 20 22 24 26 28 30 32 34
*10 5
Uninstalled Corrected (Eqv.) Power Watts
Fig. 7.6a Compressor Vane Angle
Rel. Compressor Corrected Speed %
110
100
90
80
70
12 14 16 18 20 22 24 26 28 30 32 34
*10 5
Uninstalled Corrected (Eqv.) Power Watts
Fig. 7.6b Compressor Corrected Speed Schedule
Turbine Inlet Temperature K *10 2
15
14
13
12
11
10
12 14 16 18 20 22 24 26 28 30 32 34
*10 5
Uninstalled Corrected (Eqv.) Power Watts
Fig. 7.6c Temp Schedule for Power Modulation
Alt (m) 0.0
Mn 0.0
Fixed Comp. Vanes/ Fixed Turbine Area
Fixed Comp. Vanes/ Var. HP Turbine Area
Fixed Comp. Vanes/ Var. HP Turbine Area
Var. LP Comp. Vanes/ Var. HP Turbine Area

# Turboshaft Performance With Different Controls

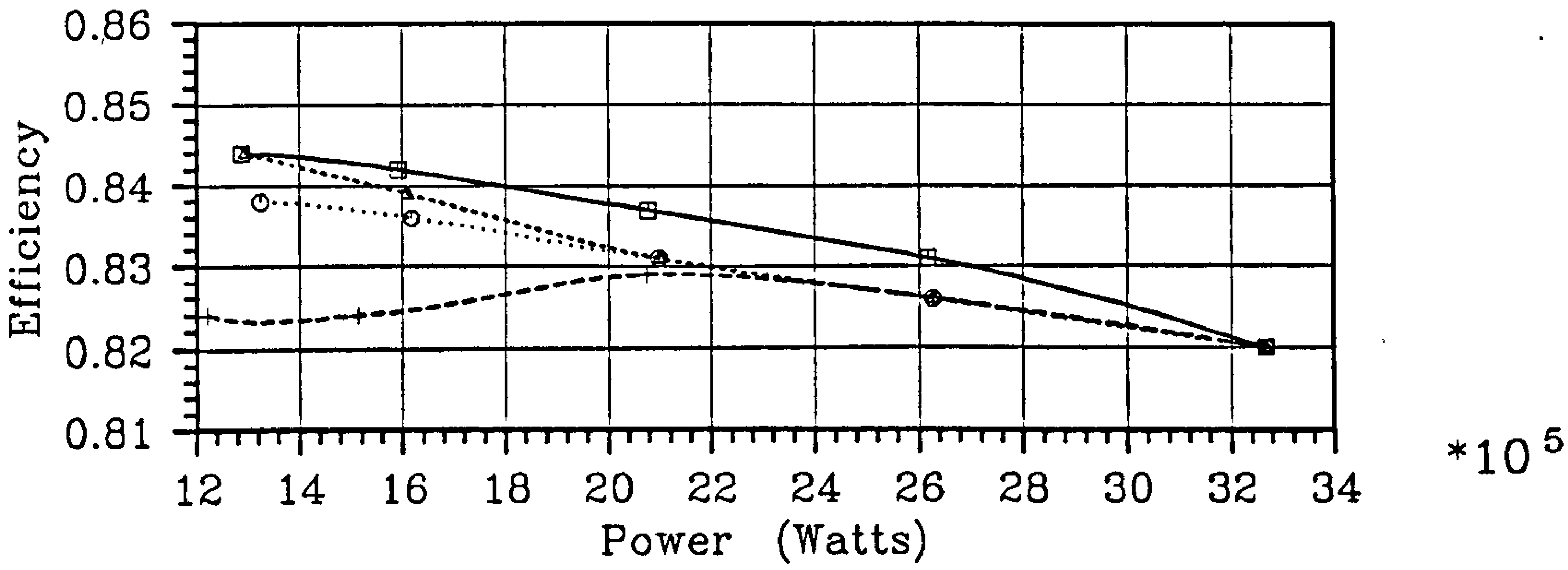



Fig. 7.7a Compressor Efficiency Variation

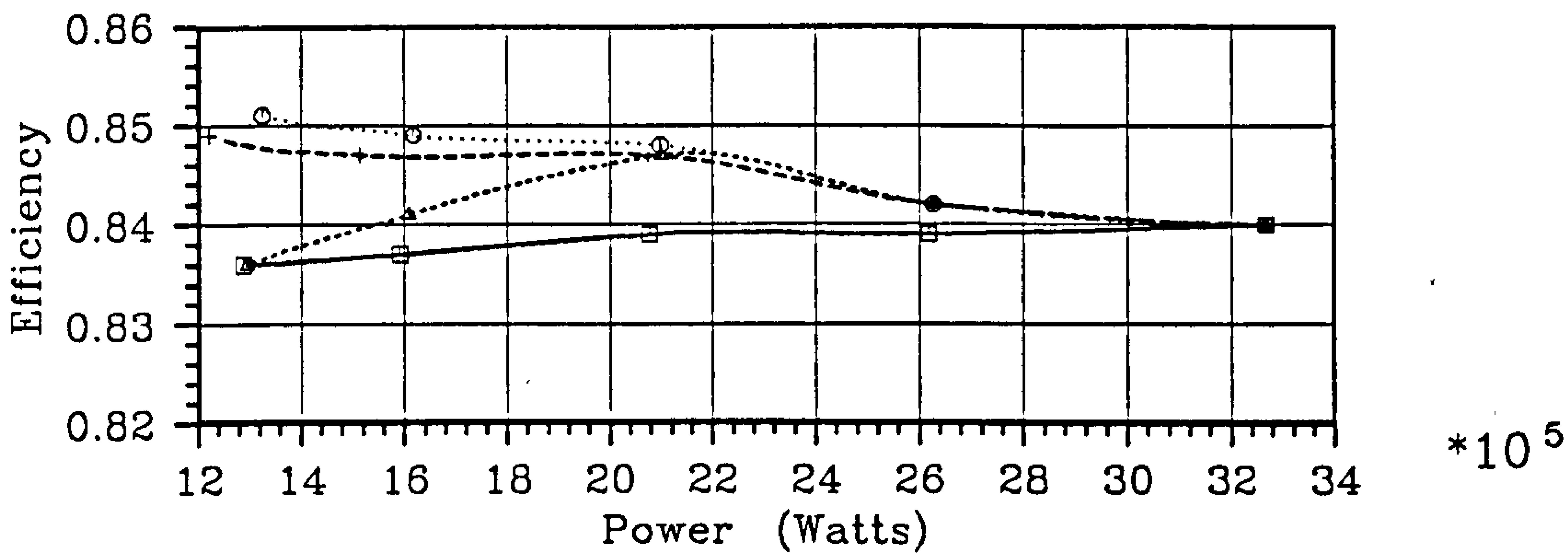



Fig. 7.7b HP Turbine Efficiency Variation

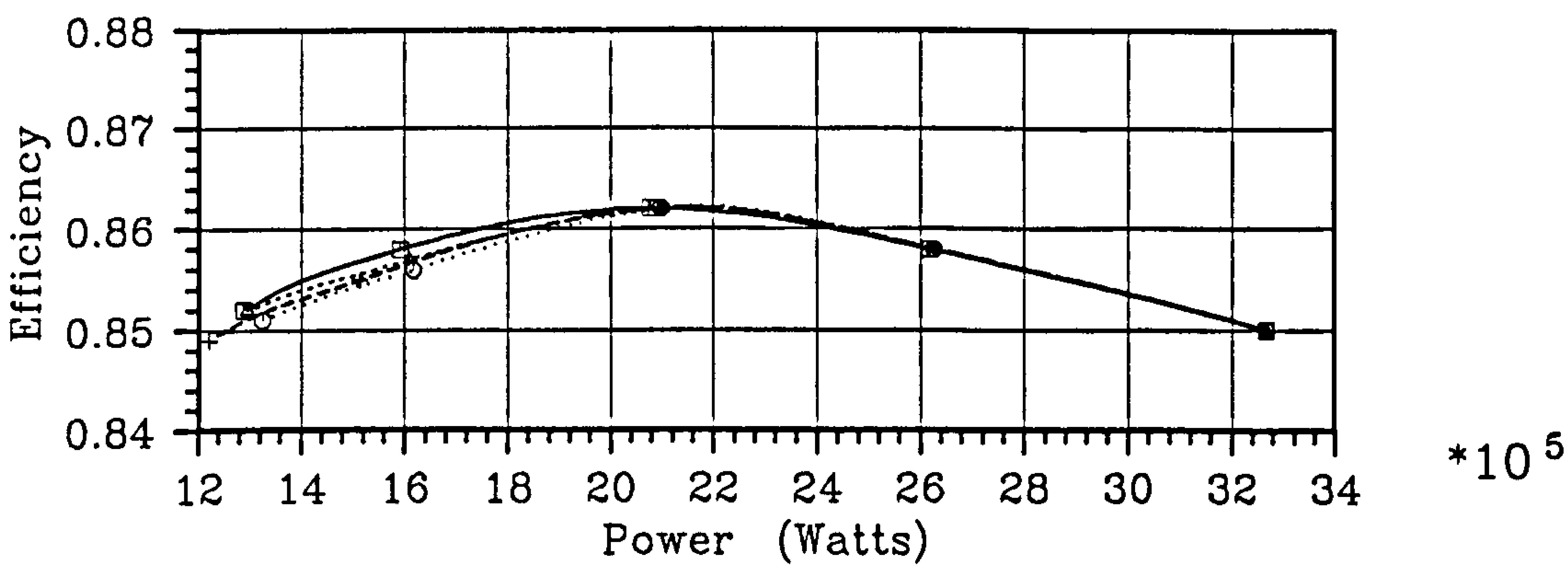



Fig. 7.7c LP Turbine Efficiency Variation

Conventional Control

Constant Pressure Ratio/Conventional

Constant Pressure Ratio/Constant Surge Margin

Constant Pressure Ratio/Scheduled IGVs

constant. If turbine efficiency could be improved at low powers, then the constant surge margin operation would be the best mode of control to be deployed at low and intermediate powers.

## The Influence of Turbine Performance

It is expected that a variable area turbine would lose a few points in efficiency as its area changes from that at highest efficiency. Also, since a given drop in turbine efficiency affects cycle performance more than an equal drop in compressor efficiency, it was decided to investigate the performance of the constant pressure ratio cycle when equipped with a variable area turbine having characteristics very dissimilar to those used in the above investigation.

Fig. 7.8a shows the efficiency function of the turbine used in the above analyses while that of a different turbine is shown in Fig. 7.8b. These will be referred to as Maps 1 and 2 respectively. Map 1 has a characteristic of having increased turbine efficiency with decreased area while Map 2 has highest efficiency at intermediate values of area.

As turbine area decreases with reduction in power, Map 1 progressively gives an increase in turbine efficiency up to the critical power point and thereafter remains almost constant except in the constant surge margin control mode where the turbine area is opening up with an accompanying decrease in efficiency as power is further reduced, Fig. 7.7b. The efficiency given by Map 2 rises to a maximum at intermediate power and falls slightly to a level where it is held constant when constant pressure ratio could no longer be maintained except in the constant surge margin operation mode where it falls further and then rises again in the low power spectrum, Fig. 7.9b.

Because of the higher efficiencies given by Map 1 at reduced power, the saving in fuel consumption over that of the fixed geometry engine is bigger than that given by Map 2. The sfc curves are shown in Fig. 7.9a. At half load, the saving in fuel consumption ranges from about 1.4 to 2.7 percent. Because of the improvement in turbine efficiency, the constant surge margin control mode gives a performance similar to the conventional control mode and when compared with Fig. 7.4b, it is seen that the performance of the variable area turbine plays a significant part in determining the performance gains obtained with a variable geometry turbine.

## Closure

The magnitude of performance obtained by controlling a simple cycle single shaft free turbine turboshaft such that

# Effect Of Turbine Performance

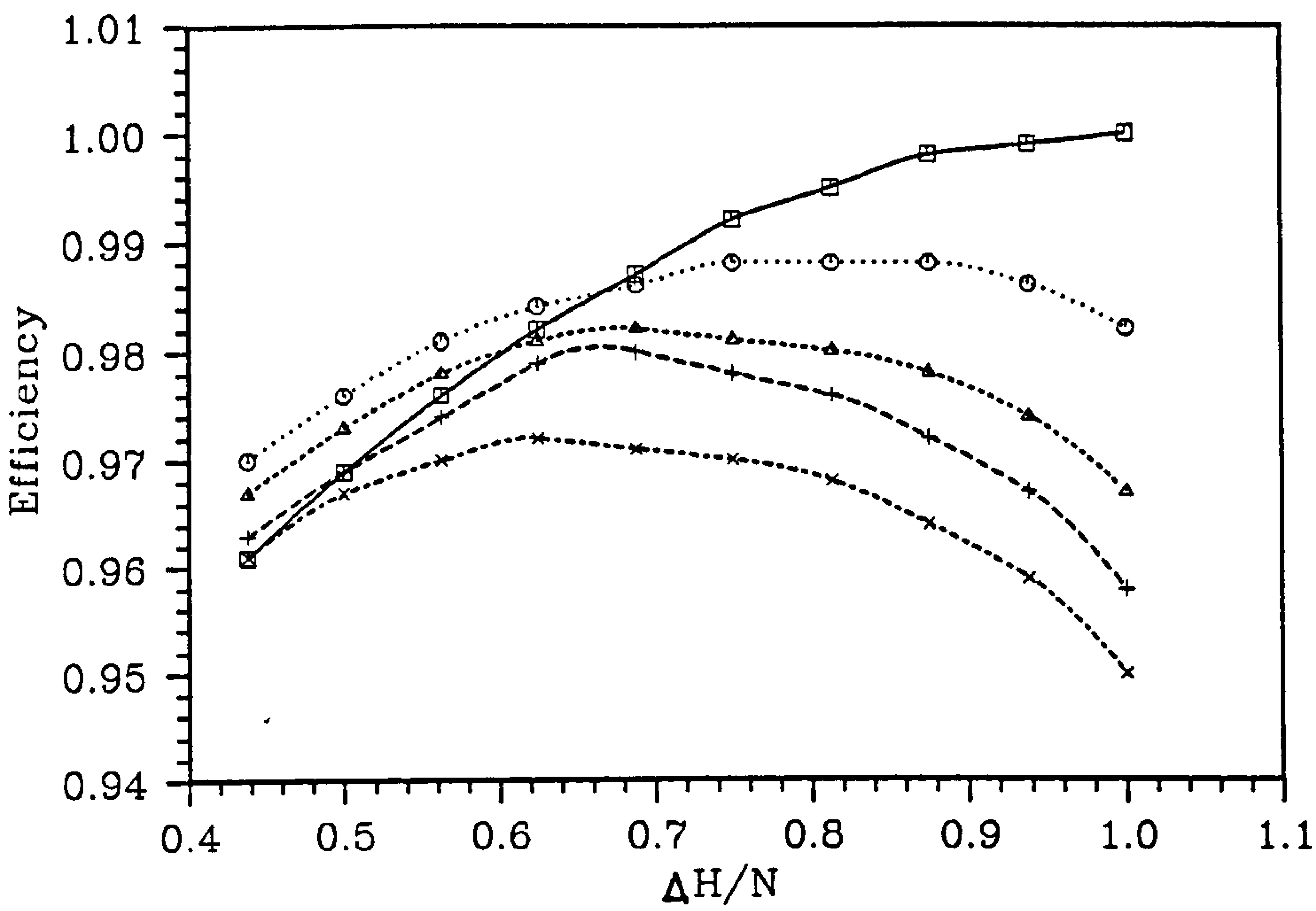



Fig. 7.8a Turbine Performance (Map 1)

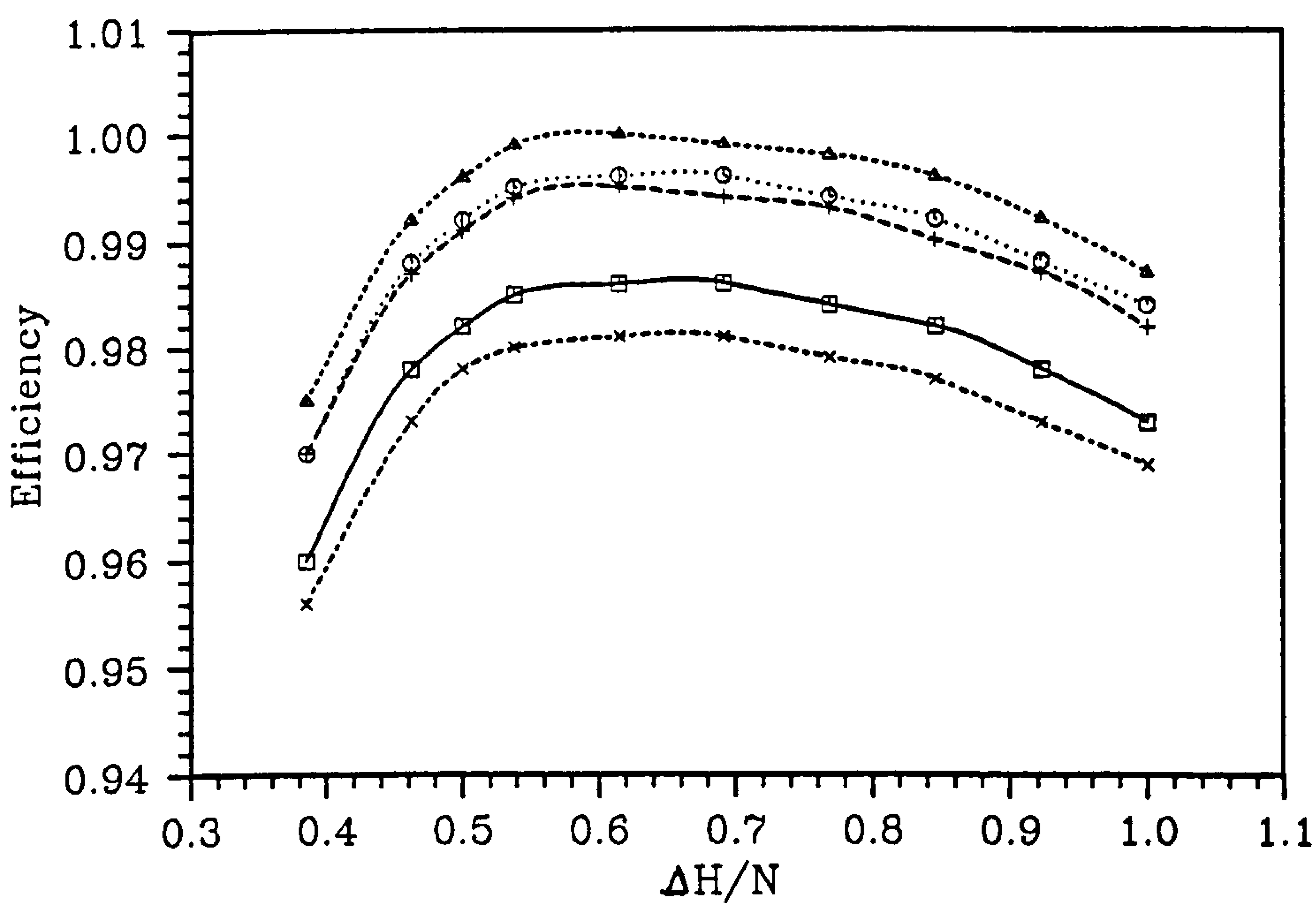



Fig. 7.8b Turbine Performance (Map 2)

1.0
1.1
1.2 A/Amin
1.3
1.4

# Effect Of Turbine Performance

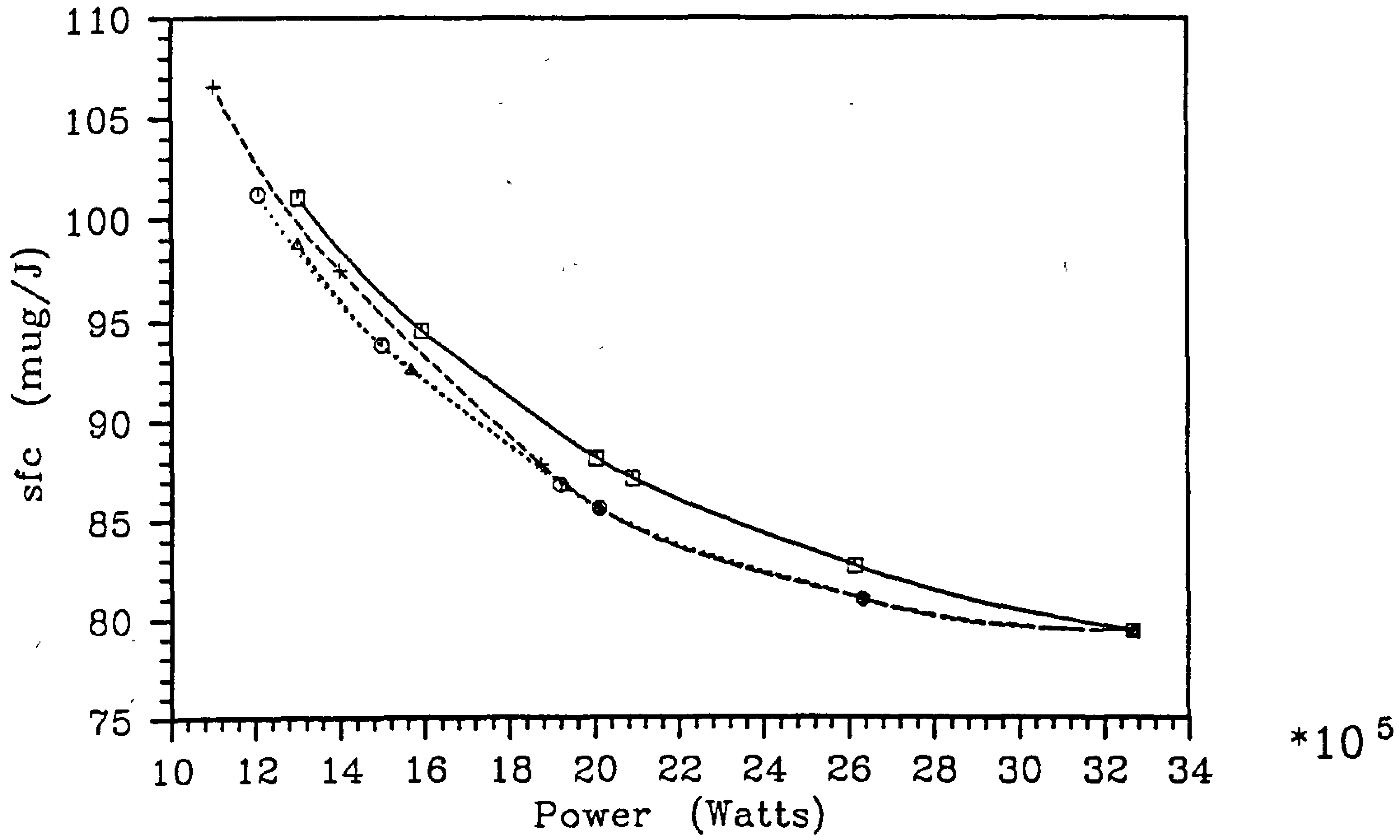



Fig. 7.9a Fuel Flow Requirement For Unit Power

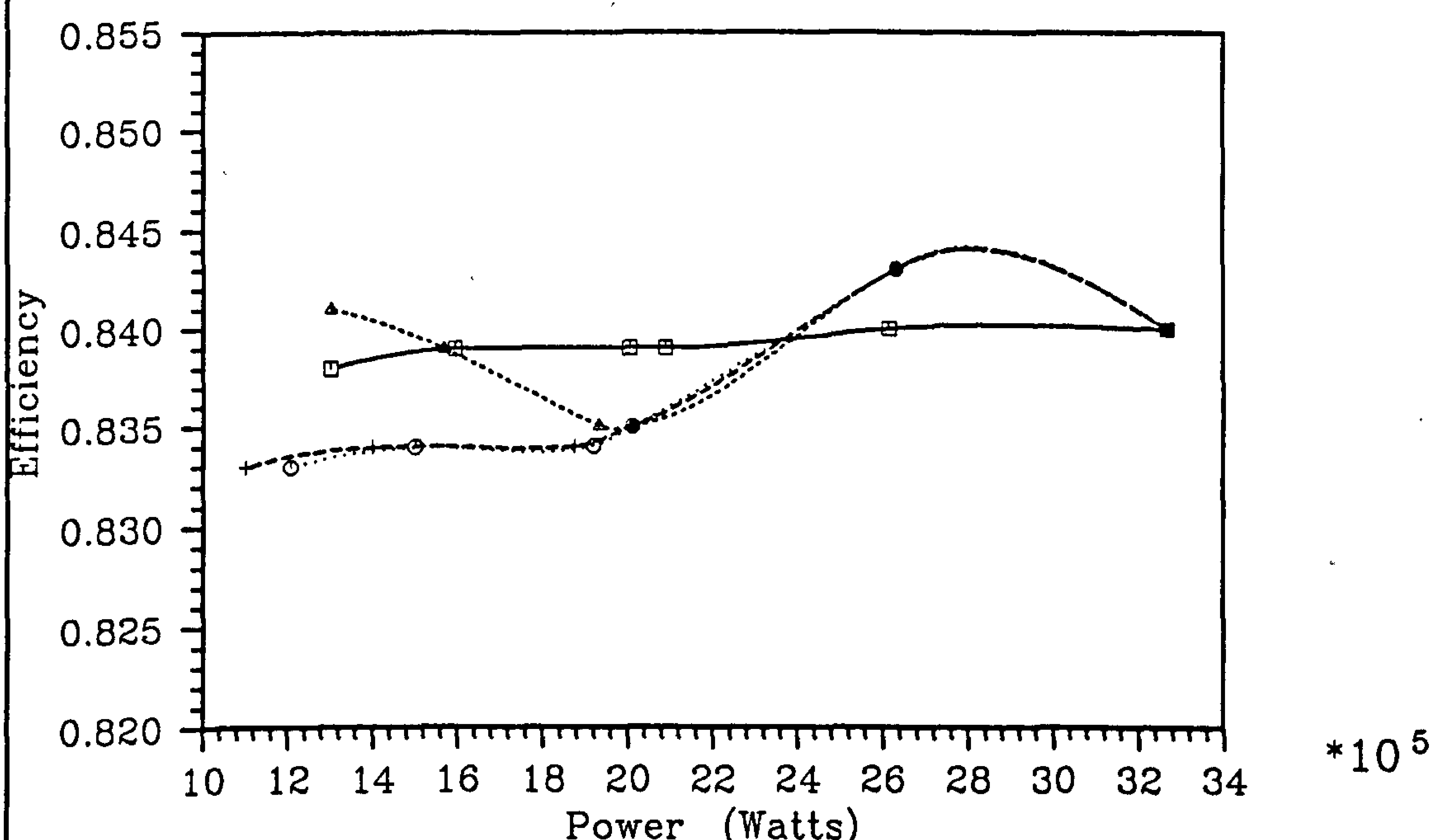



Fig. 7.9b HP Turbine Efficiency Variation

□—□ Fixed Component Geometry
○····○ Constant Pressure Ratio/Conventional
△-----△ Constant Pressure Ratio/Constant Surge Margin
+---+ Constant Pressure Ratio/Scheduled IGVs

the compressor pressure ratio is held constant depends on the performance of the driving turbines and the amount of surge margin maintained at maximum power. A variable geometry compressor does not offer any appreciable advantage when used to extend the range over which constant pressure ratio can be maintained. In certain cases, it may be desirable to operate the engine at intermediate and low power settings such that surge margin is held constant to give improved performance and safety.

### 7.1.2 The Constant Turbine Inlet Temperature Cycle

Variable geometry is used in the LP turbine to effect this mode of operation. As with the constant pressure ratio cycle, surge margin is constantly eroded as power is reduced and at low power operation, it may become necessary to change the mode of operation to maintain adequate surge margin.

The manner of investigation was carried out along similar lines as those of the constant pressure ratio cycle and the overall performance of the two cycles are completely dissimilar in all respects. These two cycles will be compared and an attempt will be made to explain the reasons for the difference in the performance characteristics in a later section.

The compressor operating lines are shown in Fig. 7.10a while the sfc curves are given in Fig. 7.10b. At 50 percent load, the saving in fuel consumption ranges from .7 to 2.1 percent. The TIT schedule, LP turbine area and speed variations are shown in Fig. 7.11 while the LP and HP turbine pressure ratio variations and compressor IGV schedule are shown in Fig. 7.12.

## 7.2 The Regenerative Cycle

A single spool free turbine turboshaft cycle incorporating heat exchange and with similar controls was also investigated to find out how its performance compares with that of the simple cycle. The component efficiencies and pressure losses chosen are akin to those of the simple cycle and a pressure ratio was selected to optimize specific power at the same TIT as the simple cycle. The mass flow was chosen to give the same power output as the simple cycle. The performance at the design point is given in Table 7.2.

# Performance Of An Advanced Turboshaft Engine

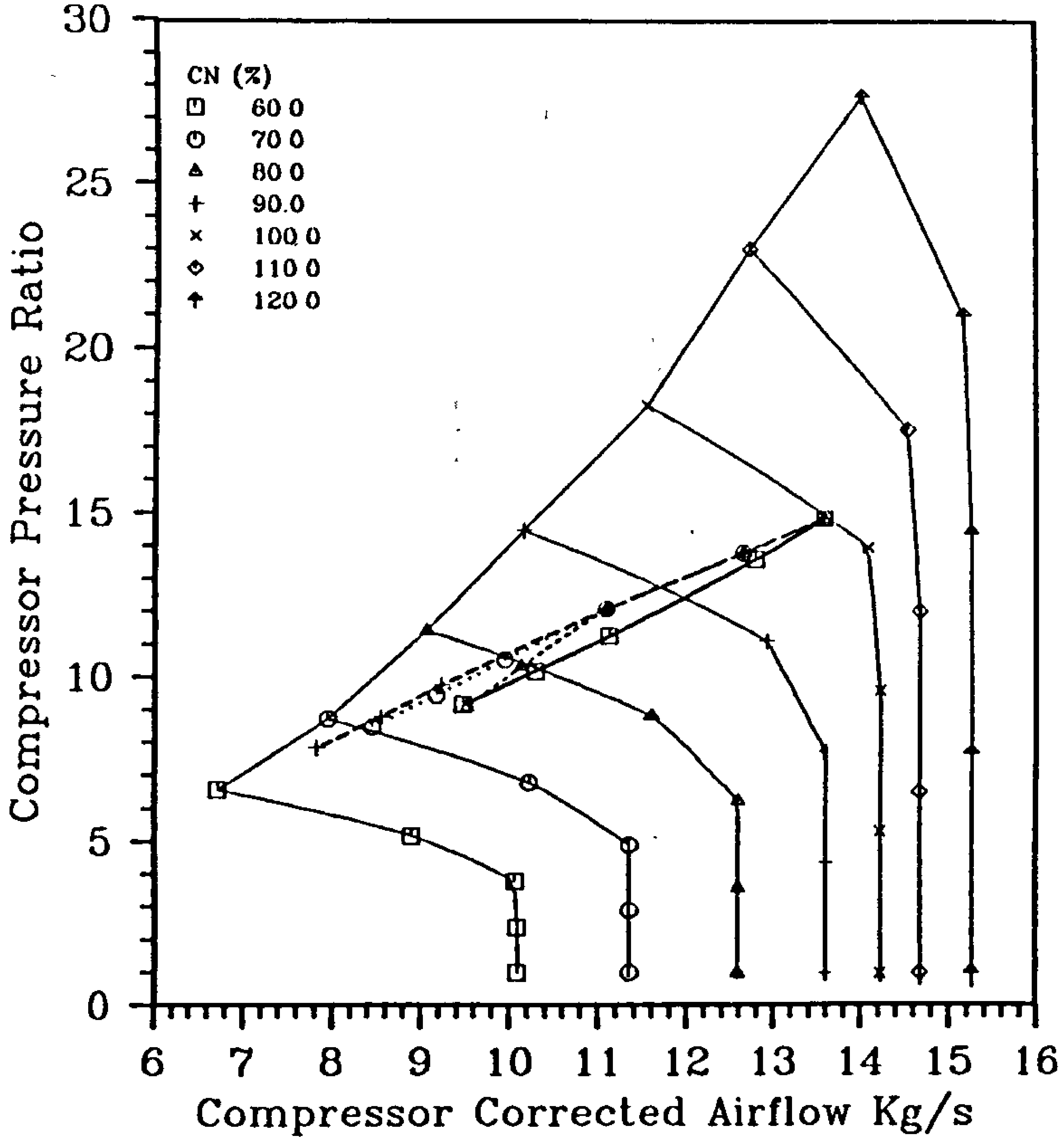



Fig. 7.10a Compressor Operating Characteristic

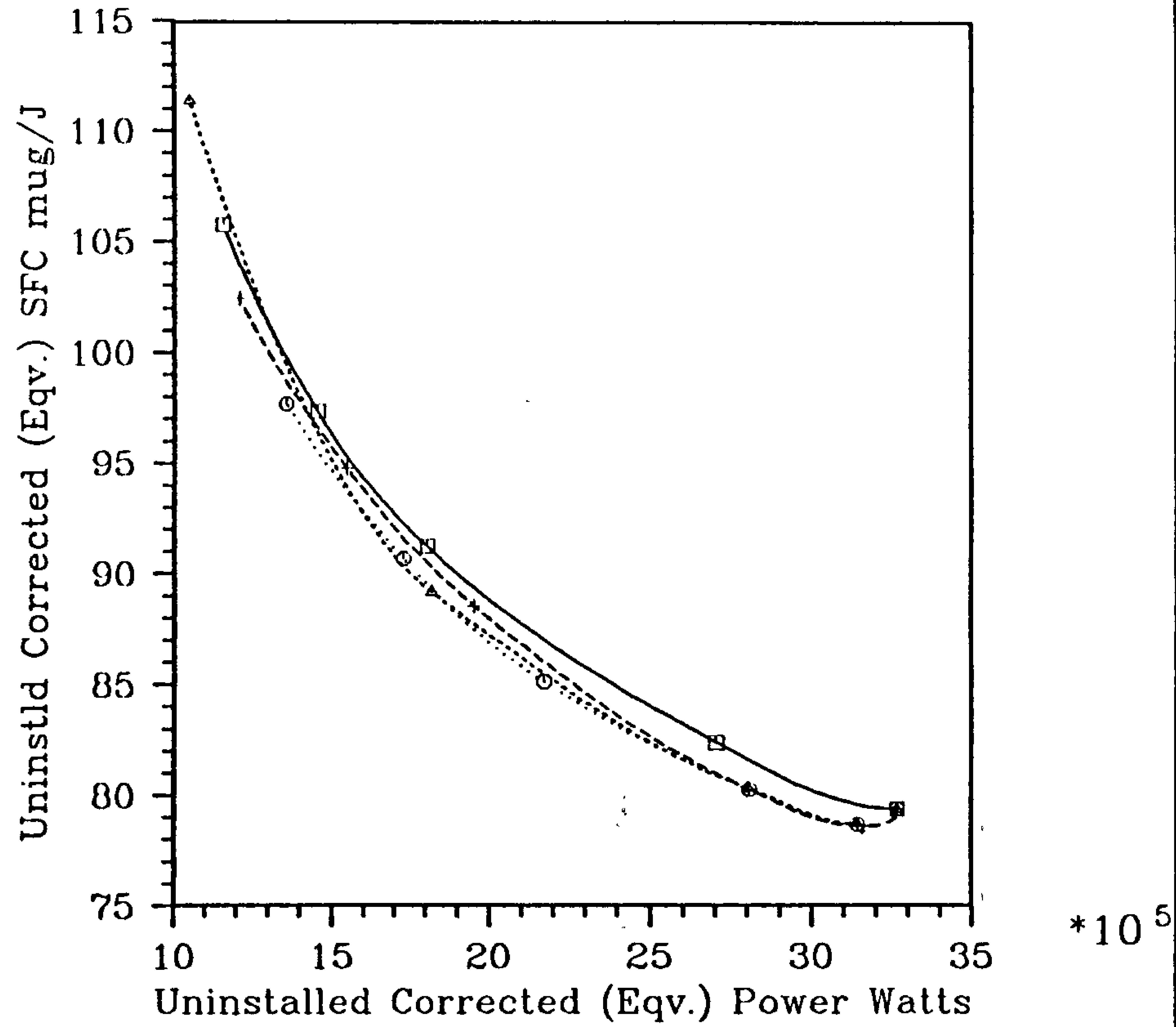



Fig. 7.10b Fuel Flow Requirement For Unit Power

Alt (m) 0.0
Mn 0.0

- Fixed Comp. Vanes/ Fixed Turbine Area
- Fixed Comp. Vanes/ Var. LP Turbine Area
- Fixed Comp. Vanes/ Var. LP Turbine Area
- Var. LP Comp. Vanes/ Var. LP Turbine Area

## Performance Of An Advanced Turboshaft Engine

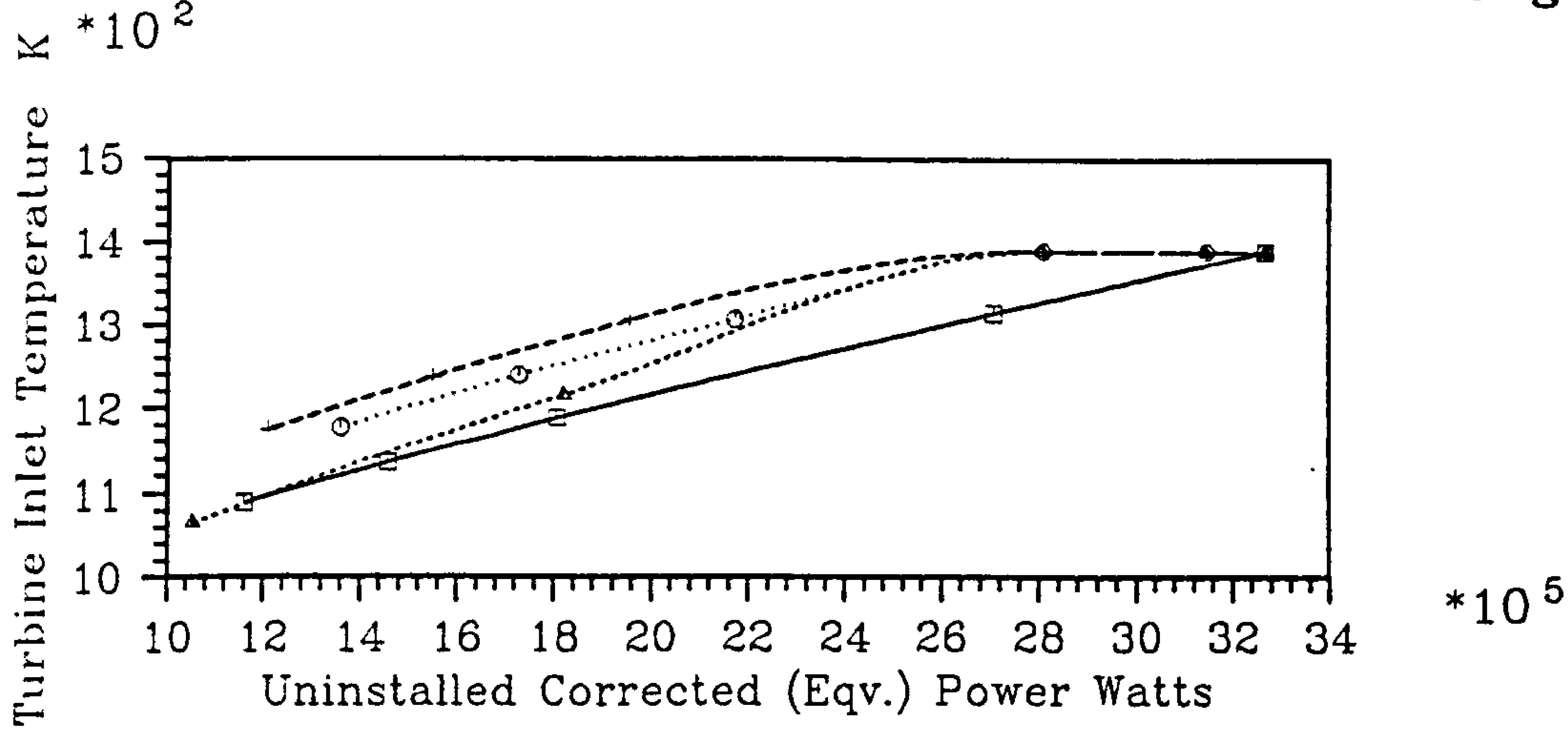



Fig. 7.11a Temp Schedule for Power Modulation

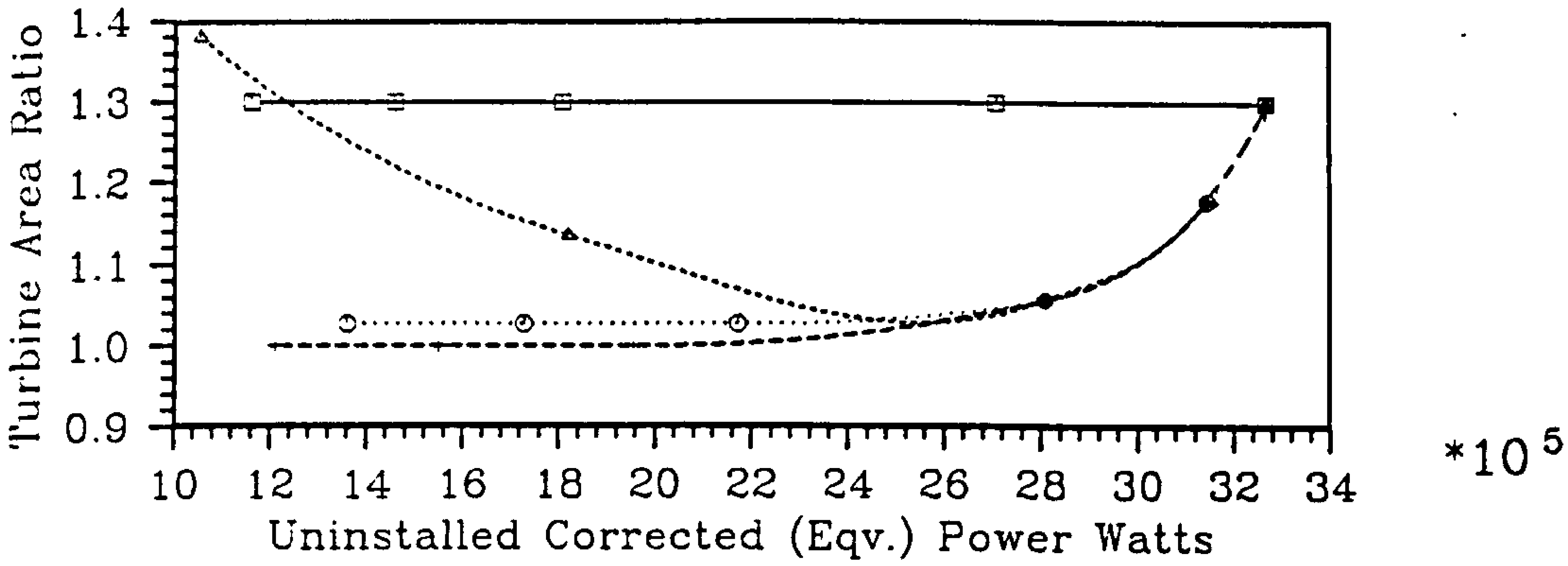



Fig. 7.11b LP Turbine Area Schedule

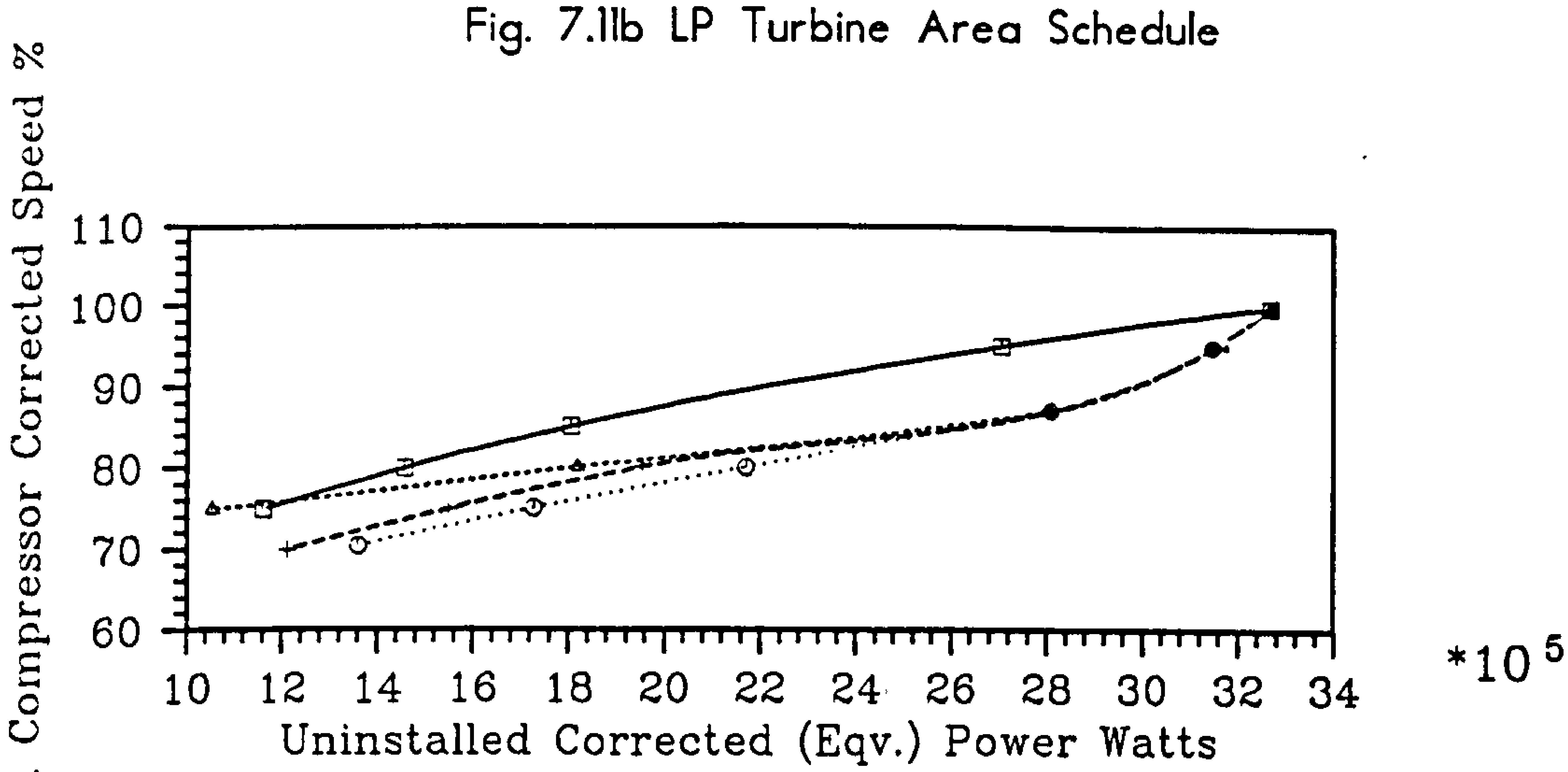



Fig. 7.11c Compressor Corrected Speed Schedule

Alt (m) 0.0
Mn 0.0

- ⊟—⊟ Fixed Comp. Vanes/ Fixed Turbine Area
- ⊙····⊙ Fixed Comp. Vanes/ Var. LP Turbine Area
- ▲----▲ Fixed Comp. Vanes/ Var. LP Turbine Area
- +---+ Var. LP Comp. Vanes/ Var. LP Turbine Area

# Performance Of An Advanced Turboshaft Engine

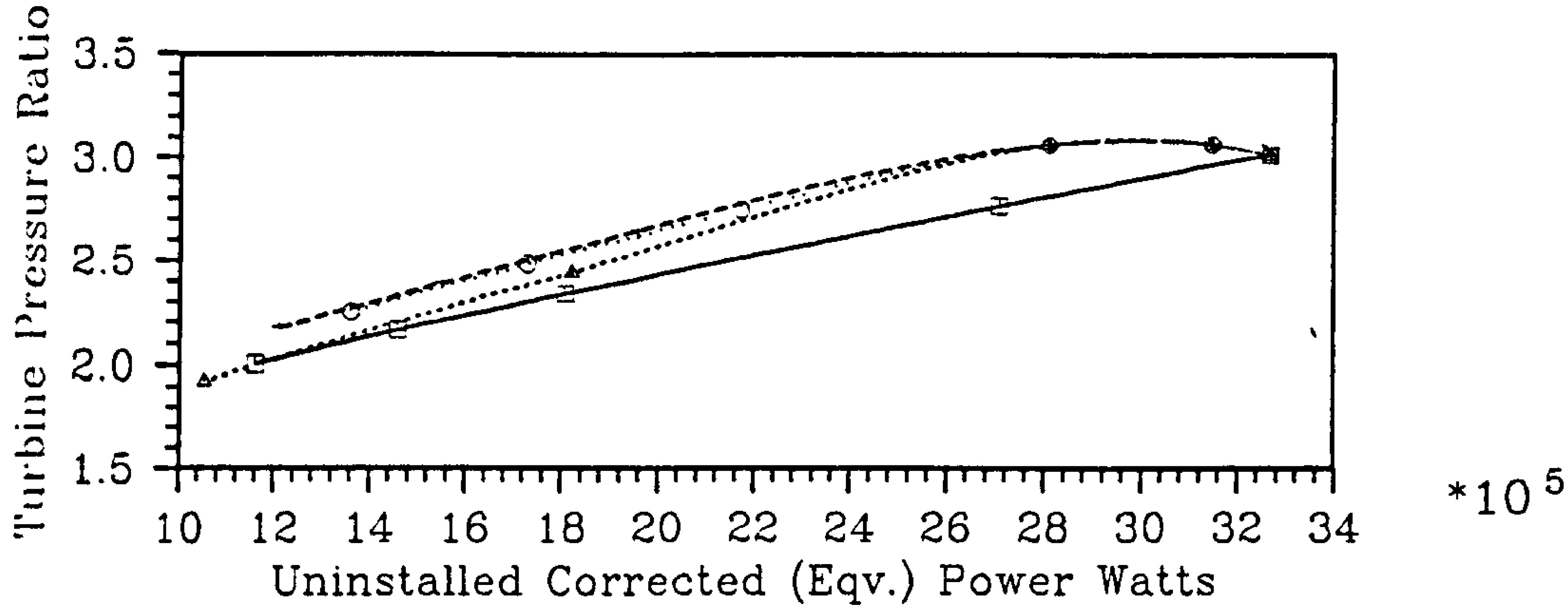



Fig. 7.12a LP Turbine Pr. Ratio

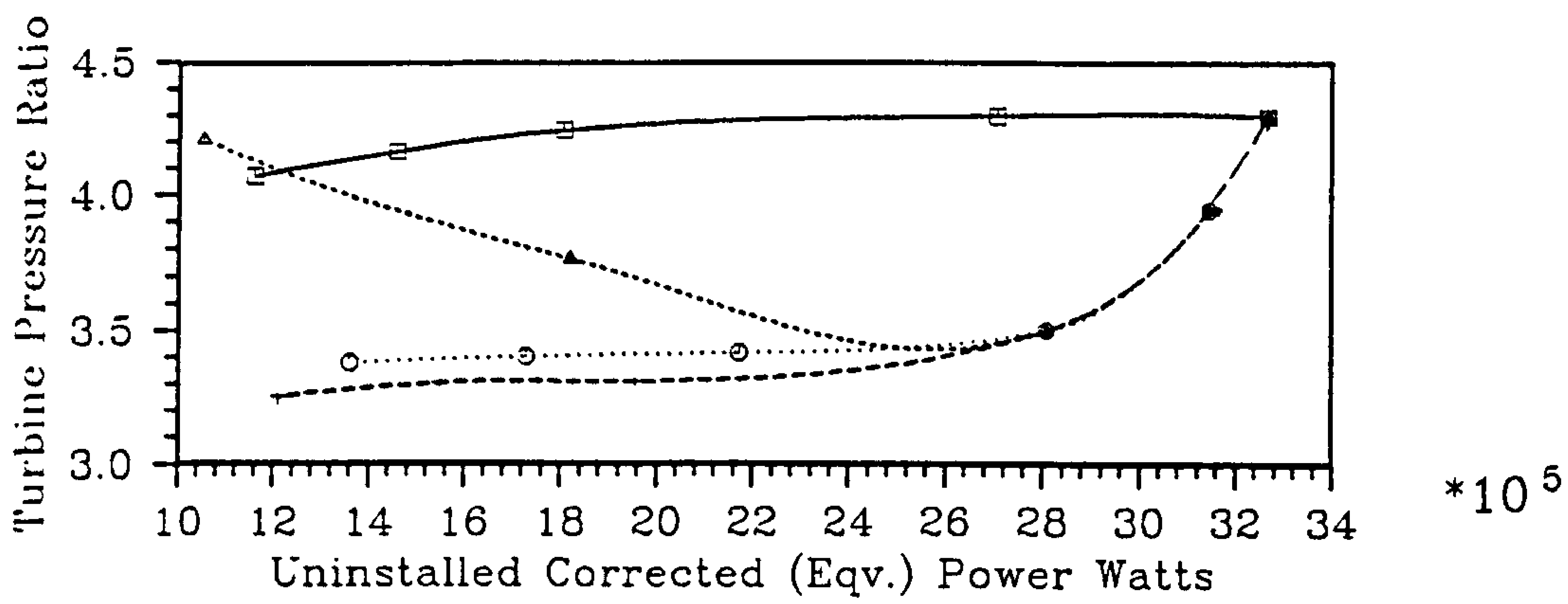



Fig. 7.12b HP Turbine Pr. Ratio

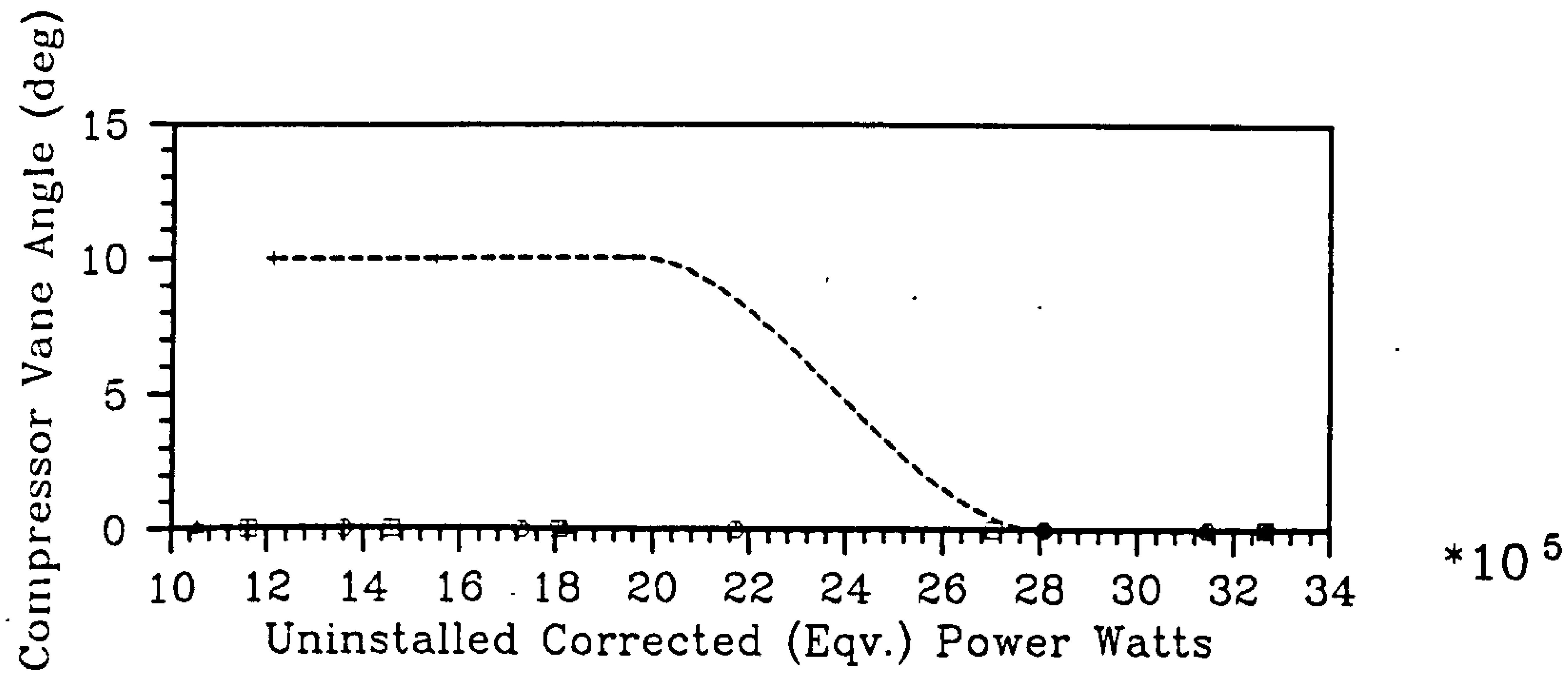



Fig. 7.12c Compressor Vane Angle

Alt (m) 0.0
Mn 0.0

- ⊟—◻ Fixed Comp. Vanes/ Fixed Turbine Area
- o····o Fixed Comp. Vanes/ Var. LP Turbine Area
- ▲····△ Fixed Comp. Vanes/ Var. LP Turbine Area
- +---+ Var. LP Comp. Vanes/ Var. LP Turbine Area

Design Point Parameters (ISA SLS)

| | | |
|---|---|---|
| Pressure Ratio | | 9.0 |
| TIT | (K) | 1388 |
| Airflow | (kg/s) | 14 |
| Power | (MW) | 3.266 |
| sfc | ($\mu$g/J) | 63.7 |
| $\eta_{th}$ | (%) | 36.4 |
| Heat Ex. Effect. | | .85 |

Table 7.2 Heat Exchange Cycle Performance

### 7.2.1 Performance of Study Cycles

The trends in performance of both the constant pressure ratio and constant turbine inlet temperature cycles are similar to those observed for the simple cycle case with the exception that there is a bigger saving in fuel consumption for the constant TIT cycle over that of the conventional cycle while the constant pressure ratio cycle gives a poorer performance compared with that of the conventional cycle.

Since the performance trends for these cycles are qualitatively similar to those of the simple cycle which have already been shown, the performance plots of these cycles are not shown as it would amount to a mere duplication of plots. However, some of these plots are given in the following section which compares the performances of the constant pressure ratio and constant TIT cycles with the conventional fixed geometry cycle for both the simple and regenerative types.

## 7.3 Performance Comparison of the Various Cycles

As was mentioned earlier, the constant pressure ratio cycle has a performance characteristic dissimilar to that of the constant TIT cycle, and the magnitude of the performance improvement of any of the two cycles over that of the fixed geometry engine cycle differs considerably in going from the simple to the regenerative type.

The following figures compare the performances of both the simple and regenerative types for the three different cycles established earlier, viz, conventional, constant pressure ratio, and constant TIT. For both the constant pressure ratio and constant TIT cycles, the engines are controlled in a conventional manner at power levels below the critical power at which surge margin limit is reached, that is, pressure ratio, mass flow and speed are all allowed to drop as TIT is decreased to reduce power. The plots for the simple cycle case are shown in the "a" figures while those of the regenerative cycle appear in the "b" figures. As a reminder, variable geometry is used in the gas generator turbine to effect constant pressure ratio operation whereas a variable area power turbine makes possible operation at constant TIT. All comparisons are made with reference to the conventional cycle.

The compressor operating characteristics are shown in Fig. 7.13, and as can be clearly seen, at any given speed both the constant TIT and constant pressure ratio cycles operate at a higher pressure ratio and hence, lower surge margin, when compared with the conventional cycle. The TIT schedules are shown in Fig. 7.14. For the simple type, both the constant pressure ratio and constant TIT cycles give a slightly better cycle performance than does the conventional cycle, Fig. 7.15a. By operating at constant pressure ratio, there is a reduction in fuel flow as the compressor delivery temperature is maintained almost constant as power is modulated, whereas operation at constant turbine inlet temperature increases the efficiency of heat addition and therefore, both cycles give rise to a saving in fuel consumption. The constant turbine temperature mode of operation burns a higher mass of fuel to raise the temperature of the exhaust gases which are wasted; this causes the constant pressure ratio operation to be a little bit more efficient.

With the regenerative cycle, as power is reduced, the gases at the hot side of the heat exchanger in the constant pressure ratio cycle continuously exchange heat at reduced inlet temperature with the air at the cold side whose temperature is almost held constant. Therefore, the amount by which the compressor delivery temperature is raised is progressively lower than that of the conventional cycle. The constant TIT mode of operation uses the higher exhaust gas temperature to heat the compressor delivery air before entering the combustor, with a resultant decrease in fuel flow. This effect becomes bigger as power is reduced. At about 50 percent power, there is a 10 percent saving in fuel consumption for the constant TIT mode over the conventional, whereas the constant pressure ratio mode of control consumes about 10 percent more fuel compared with the conventional cycle. At this power setting, it was no longer possible to hold either the constant pressure ratio or constant TIT mode of control as the former had its HP turbine operating at minimum area while the latter had reached surge margin limit at a higher power setting. Both are operating at a surge margin lower than the specified minimum.

As TIT is reduced to decrease power at constant compressor pressure ratio, the HP turbine area decreases to maintain the choking mass flow, Fig. 7.16. The HP turbine does not need variable geometry in the constant TIT mode. Since the

# Turboshaft Performance Comparison With Different Controls

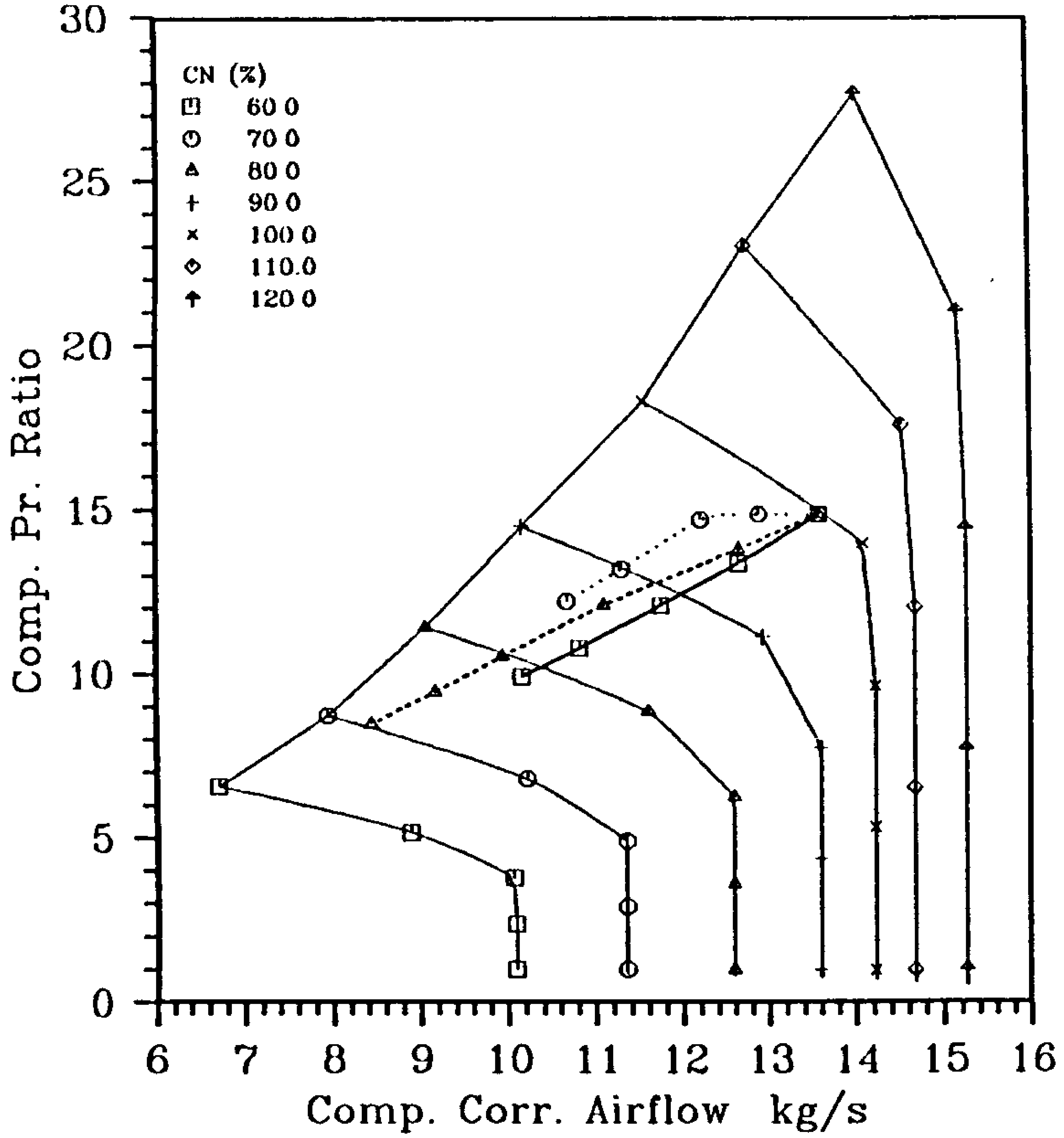



Fig. 7.13a Compressor Operating Characteristic

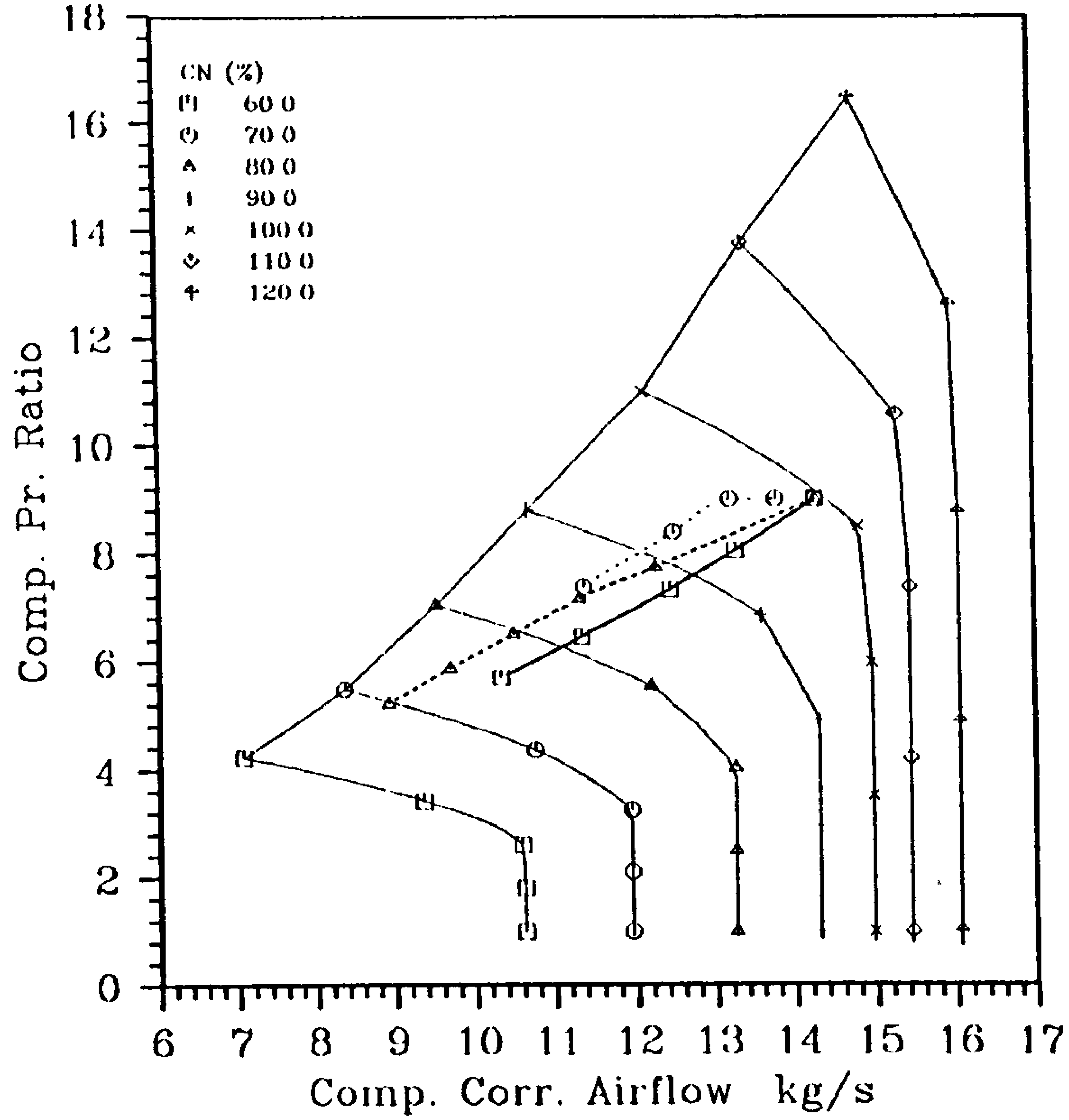



Fig. 7.13b Compressor Operating Characteristic

Alt (m) 0.0
Mn 0.0

- Fixed Comp. Vanes/ Fixed Turbine Area
- Fixed Comp. Vanes/ Var. HP Turbine Area
- Fixed Comp. Vanes/ Var. LP Turbine Area

# Turboshaft Performance Comparison With Different Controls

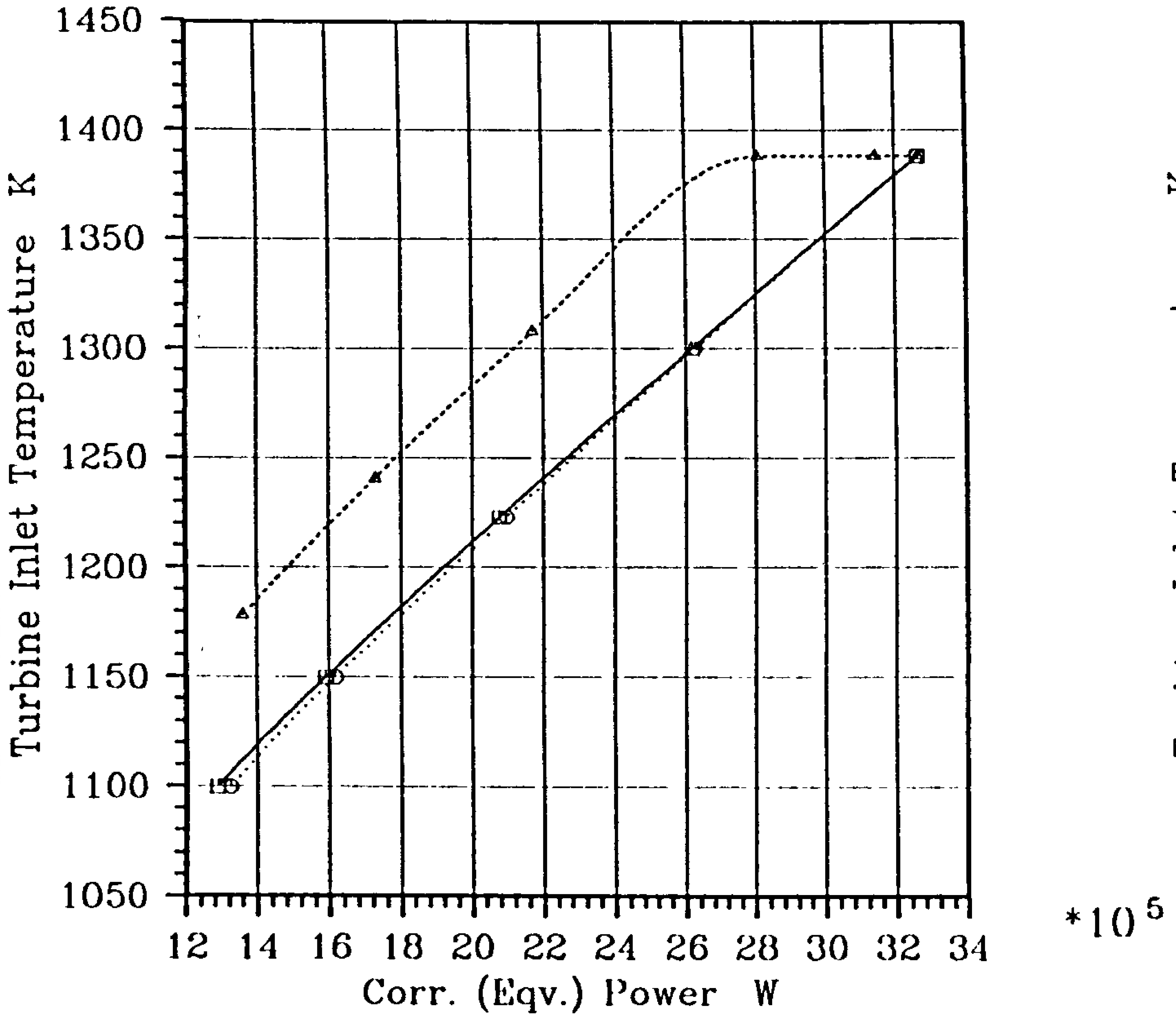



Fig. 7.14a Temp Schedule for Power Modulation

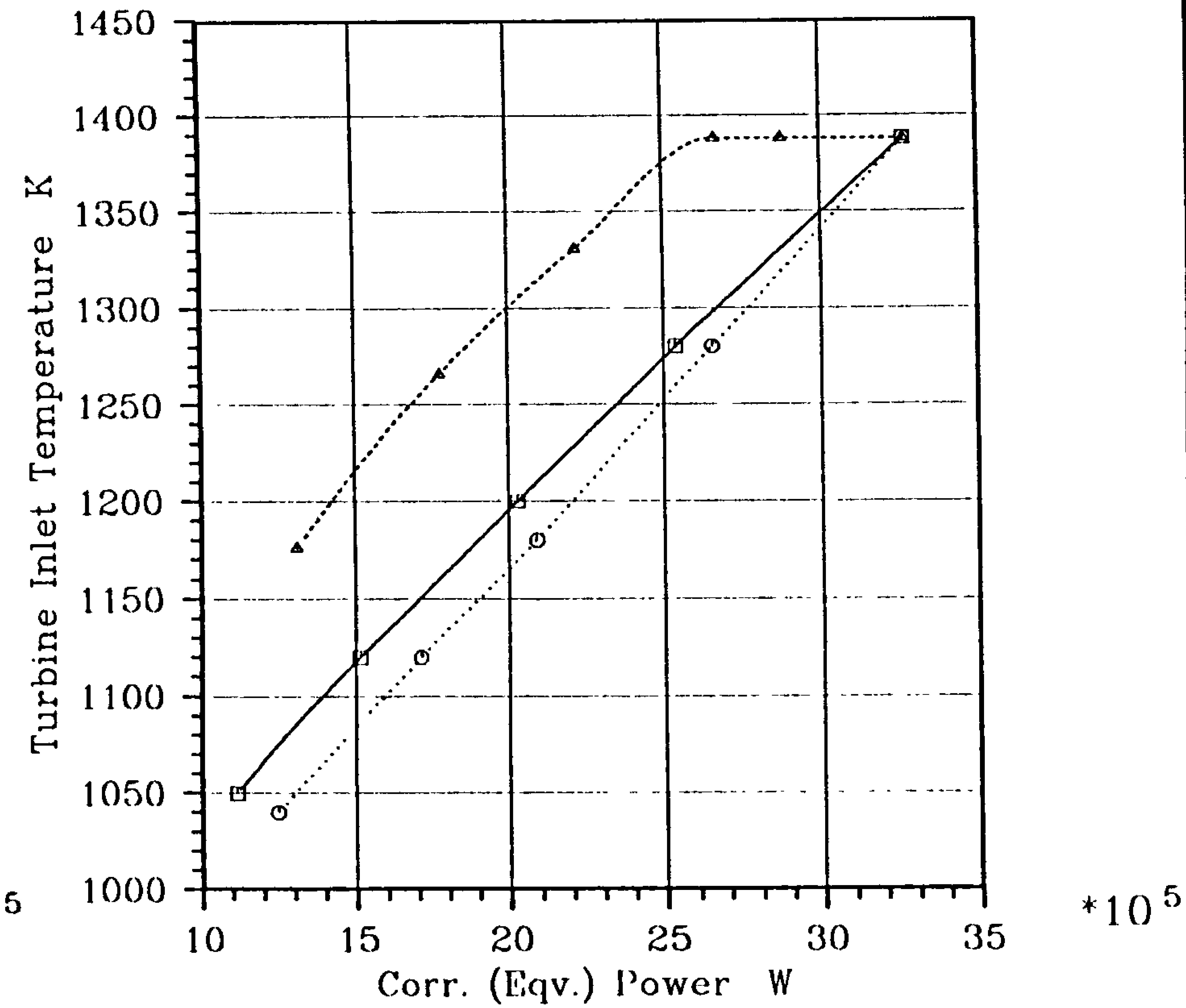



Fig. 7.14b Temp Schedule for Power Modulation

Alt (m) 0.0
Mn 0.0

Fixed Comp. Vanes/ Fixed Turbine Area
Fixed Comp. Vanes/ Var. HP Turbine Area
Fixed Comp. Vanes/ Var. LP Turbine Area

# Turboshaft Performance Comparison With Different Controls

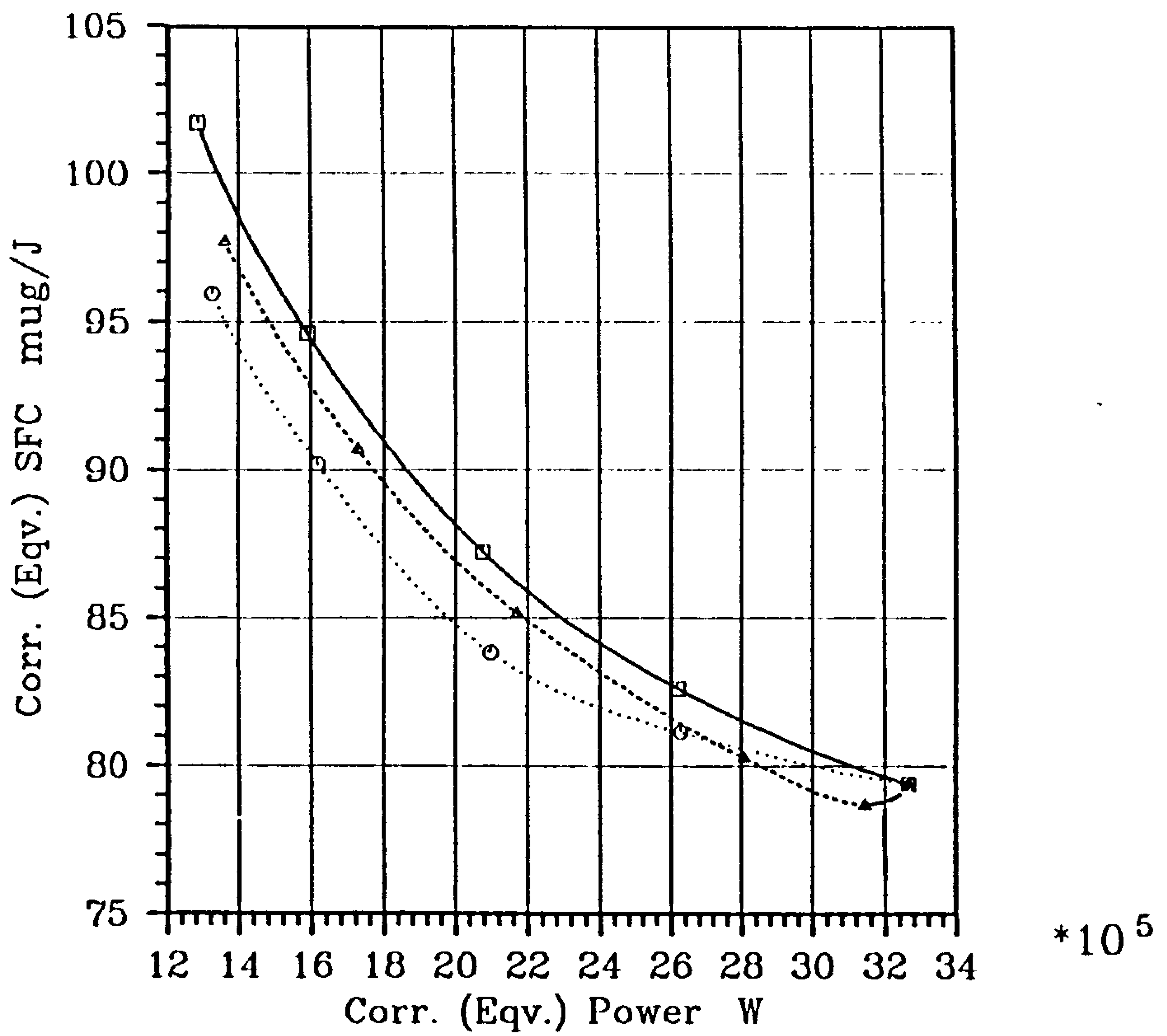



Fig. 7.15a Fuel Flow Requirement For Unit Power

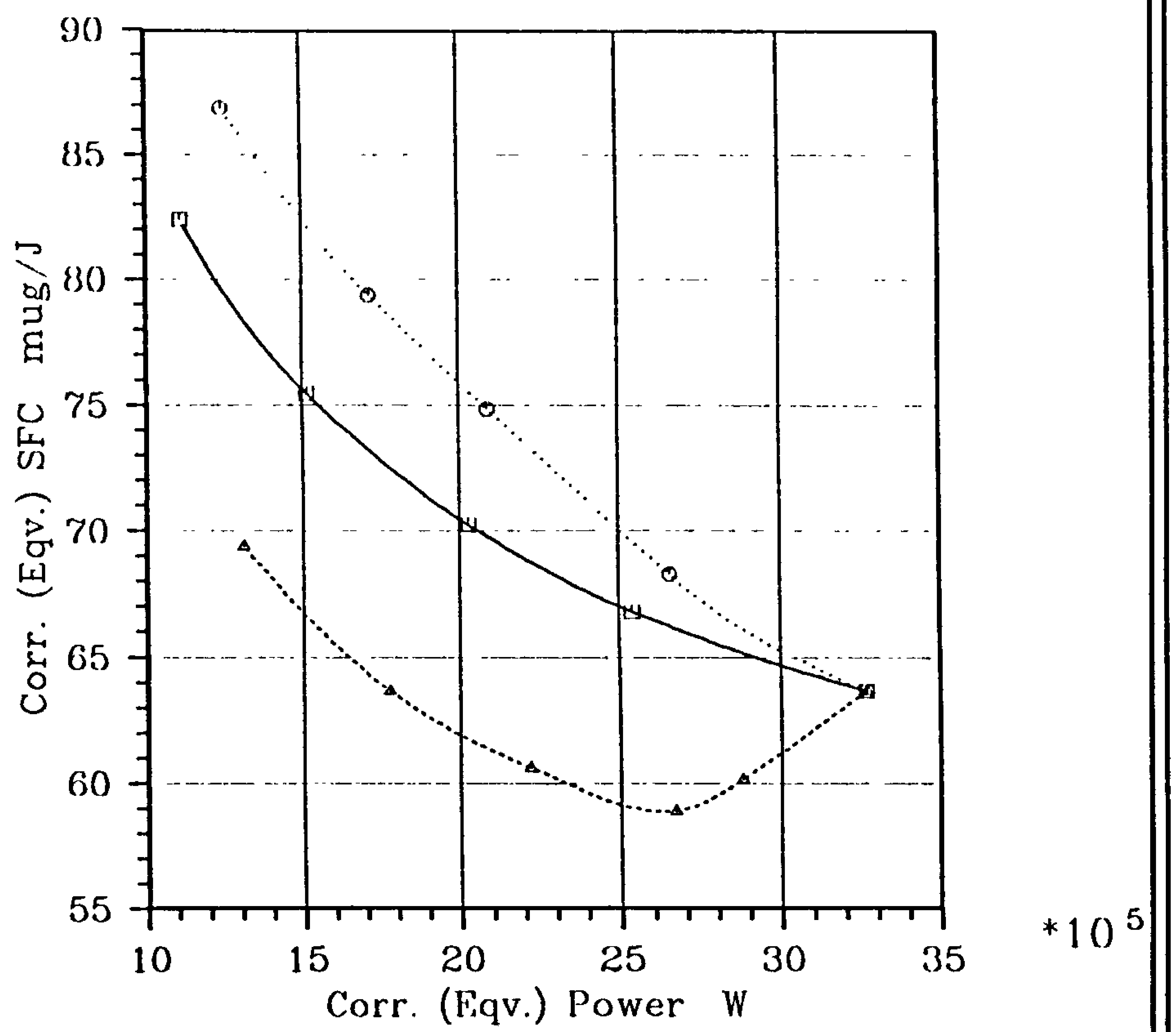



Fig. 7.15b Fuel Flow Requirement For Unit Power

Alt (m) 0.0
Mn 0.0

- Fixed Comp. Vanes/ Fixed Turbine Area
- Fixed Comp. Vanes/ Var. HP Turbine Area
- Fixed Comp. Vanes/ Var. LP Turbine Area

## Turboshaft Performance Comparison With Different Controls

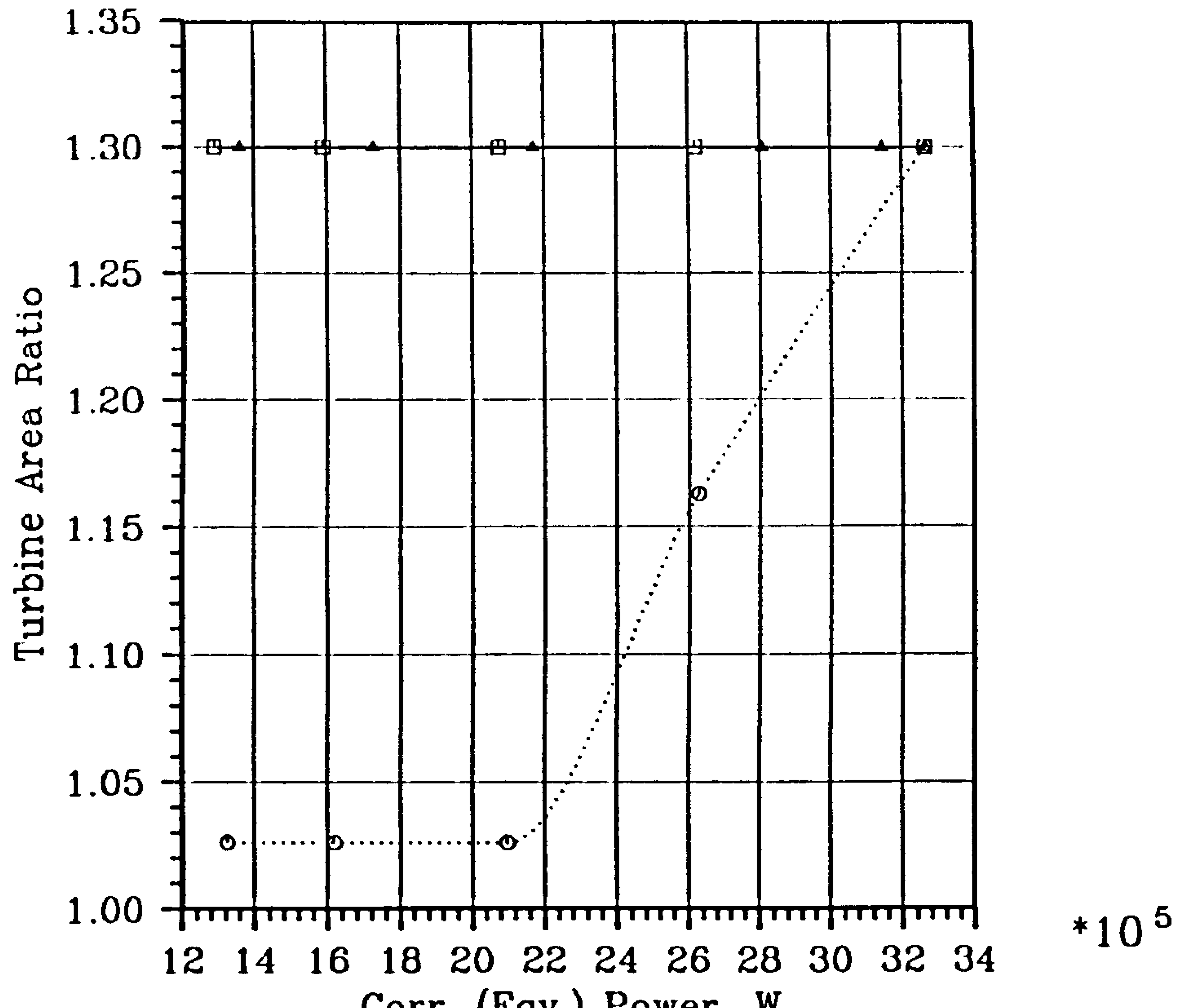



Fig. 7.16a HP Turbine Area Schedule

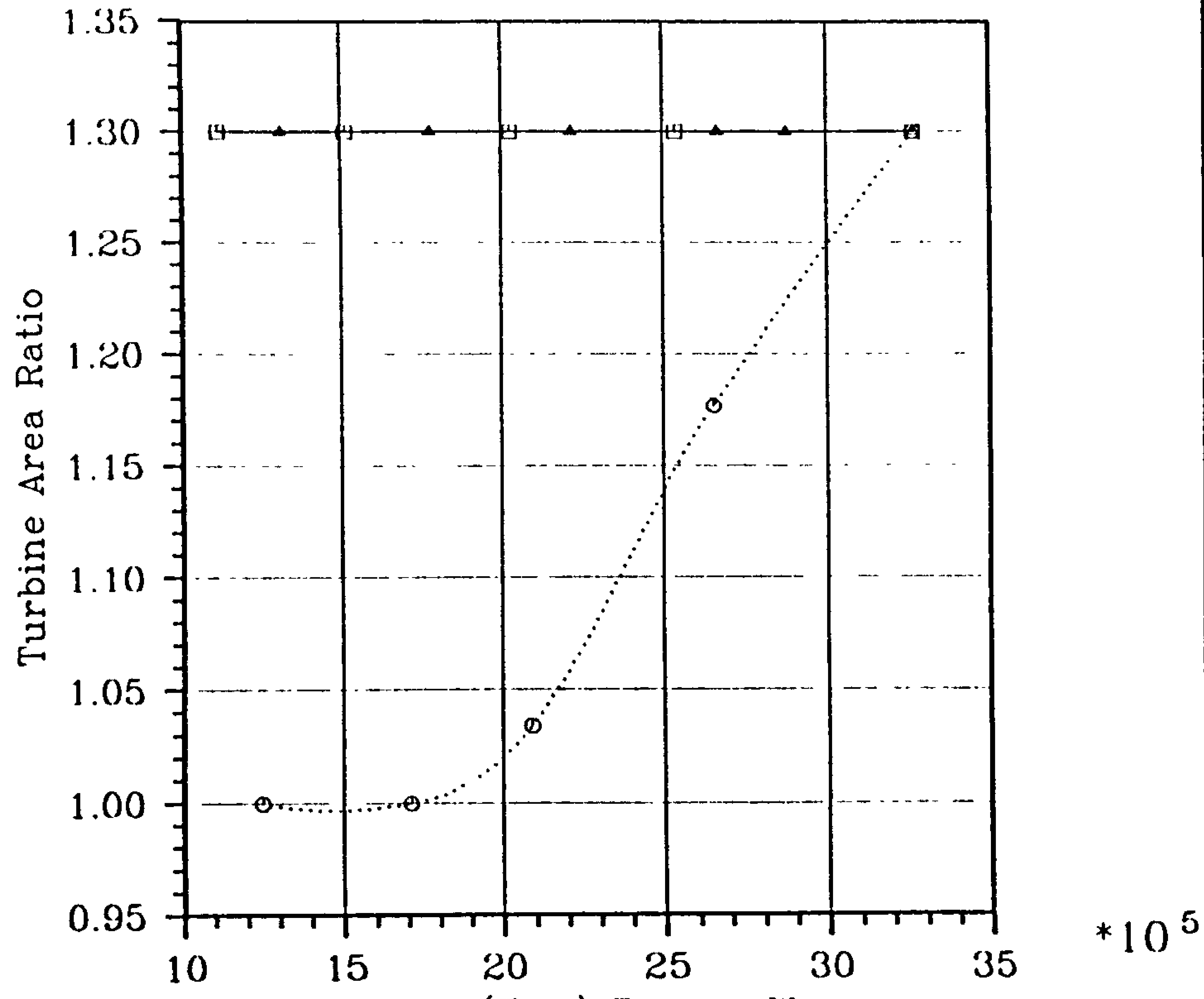



Fig. 7.16b HP Turbine Area Schedule

Alt (m) 0.0
Mn 0.0

- Fixed Comp. Vanes/ Fixed Turbine Area
- Fixed Comp. Vanes/ Var. HP Turbine Area
- Fixed Comp. Vanes/ Var. LP Turbine Area

## Turboshaft Performance Comparison With Different Controls

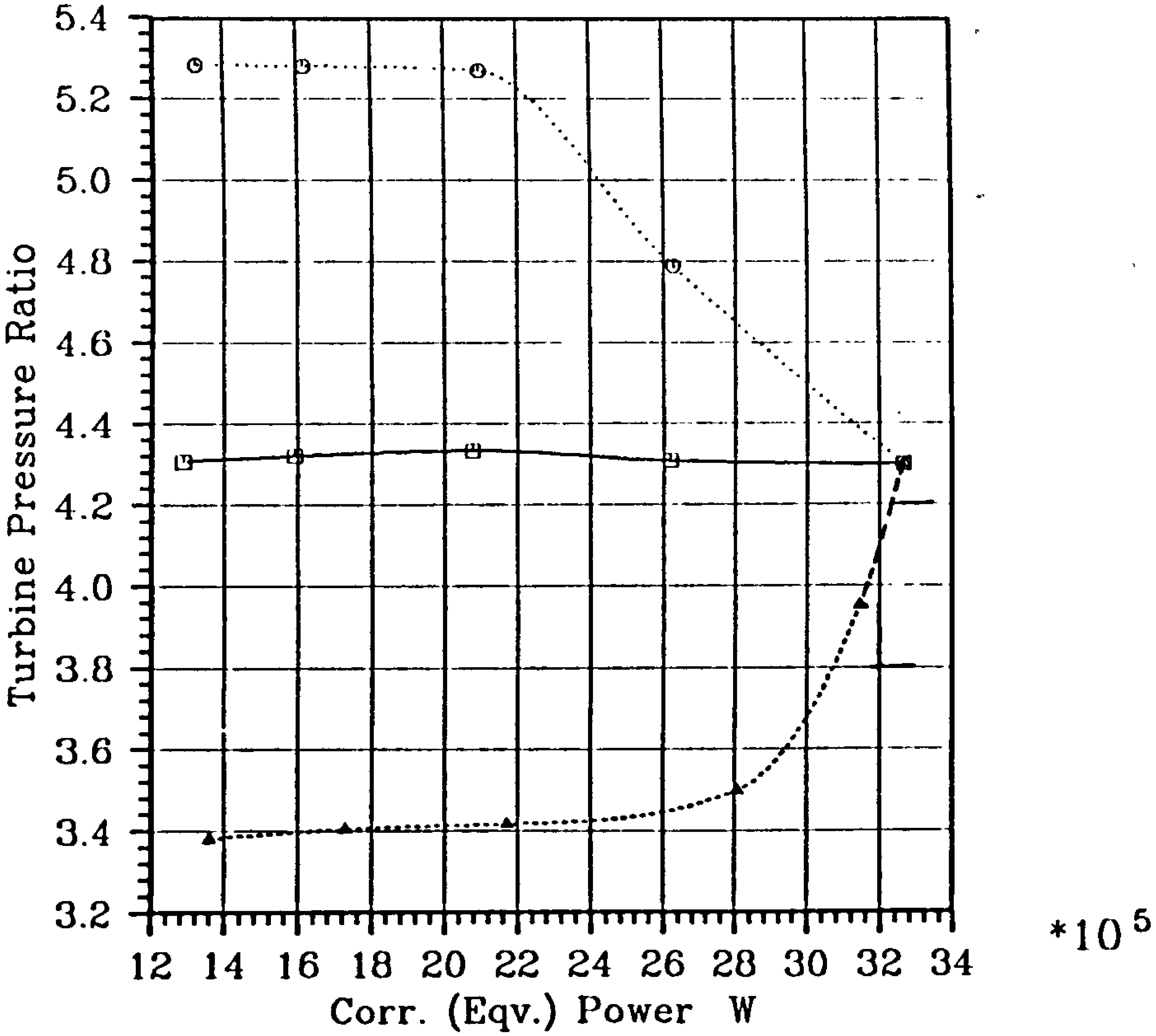



Fig. 7.17a HP Turbine Pr. Ratio

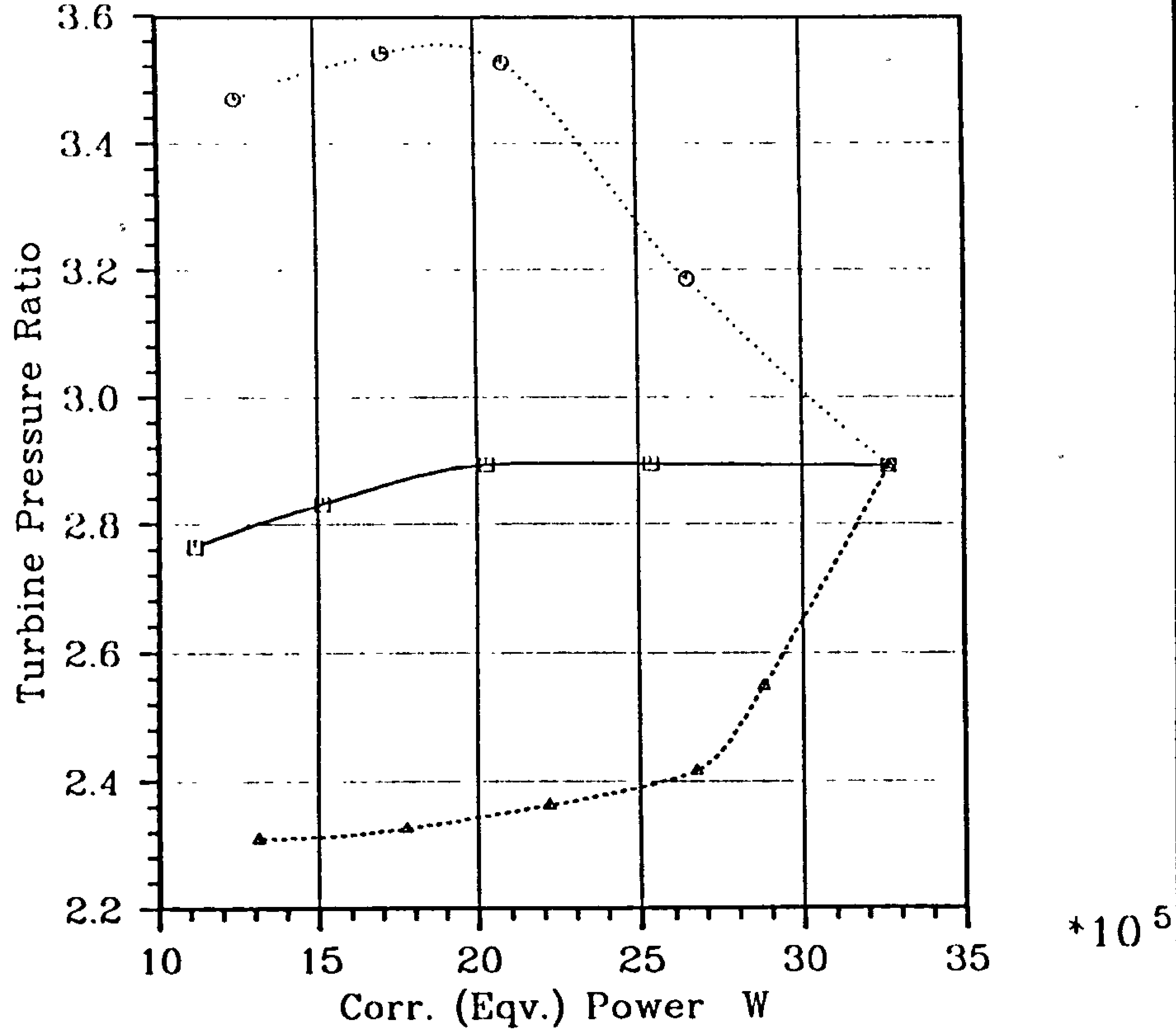



Fig. 7.17b HP Turbine Pr. Ratio

Alt (m) 0.0
Mn 0.0

- Fixed Comp. Vanes/ Fixed Turbine Area
- Fixed Comp. Vanes/ Var. HP Turbine Area
- Fixed Comp. Vanes/ Var. LP Turbine Area

overall pressure ratio across the turbines hardly changes when compressor pressure ratio is held fixed, the pressure ratio across the HP turbine increases with a reduction in power since LP turbine pressure ratio decreases, Fig. 7.17. By contrast, the HP turbine in the constant TIT mode of operation does less work at the same inlet temperature as power is reduced, therefore, its pressure ratio decreases with reduction in power.

The power turbine in the constant pressure ratio cycle does not utilize variable geometry, but with constant TIT operation, since the temperature at inlet to the power turbine increases as power is reduced, a reduction in HP turbine pressure ratio from point A to B in Fig. 7.18 causes the turbines to match such that the power turbine

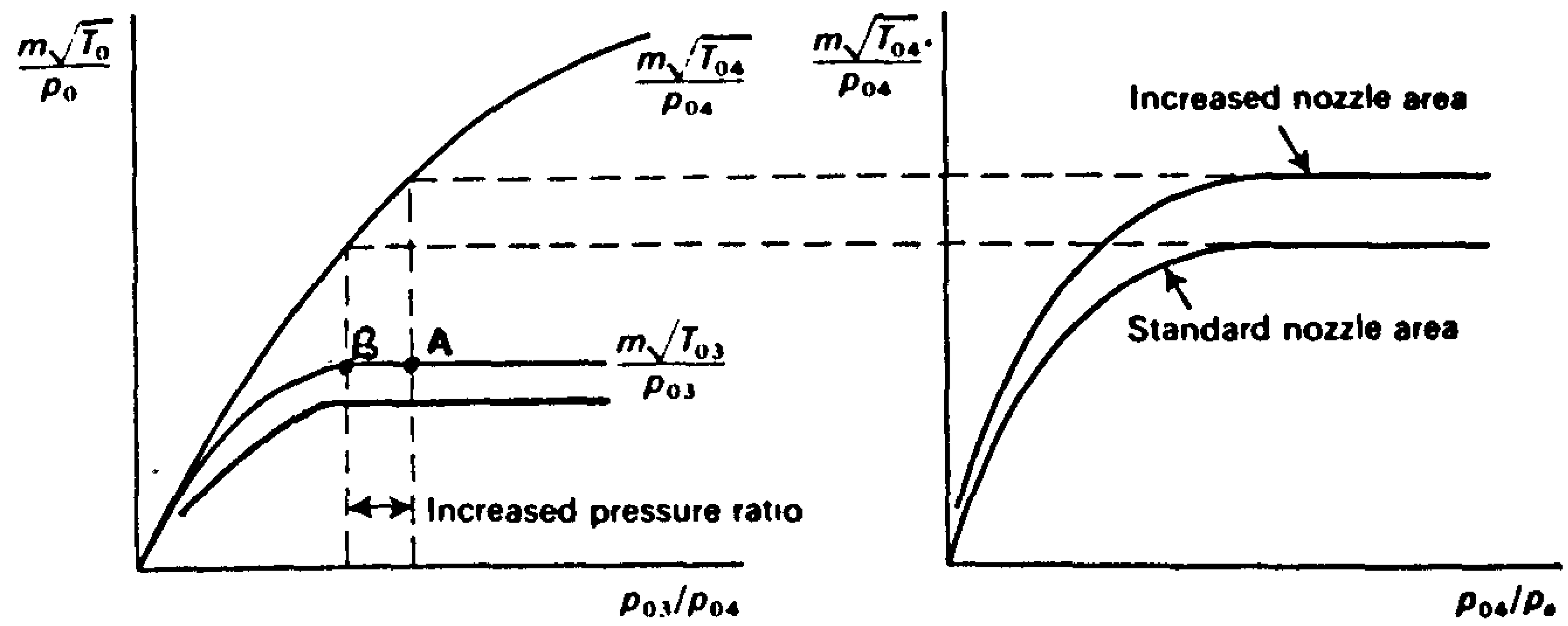



Fig. 7.18 Matching of two turbines in series [7]

area decreases as shown in Fig. 7.19. For the constant pressure ratio operation, since the variable area HP turbine does not affect the matching characteristics of the power turbine, the pressure ratio across the power turbine is the same as that required by the conventional cycle at the same power output, Fig 7.20. Since the temperature at inlet to the power turbine increases with reduction in power in the case where TIT is held constant, and with the fact that the power turbine area decreases, W/P must decrease to maintain choked turbine conditions. It is not clear why the pressure at inlet to the power turbine increases slightly for the simple cycle thereby causing the power turbine pressure ratio to increase as power output decreases. One would expect that both W and P will decrease but with W decreasing much faster to maintain turbine non-dimensional mass flow.

Fig. 7.21 shows that the gas generator operates at a higher speed when constant pressure ratio is maintained. Figs 7.13 and 7.14 explain this. As shown in Fig. 14 the constant pressure ratio control causes the cycle to operate at the same TIT as the conventional cycle at all power settings. Therefore, if both cycles are to operate at the

## Turboshaft Performance Comparison With Different Controls

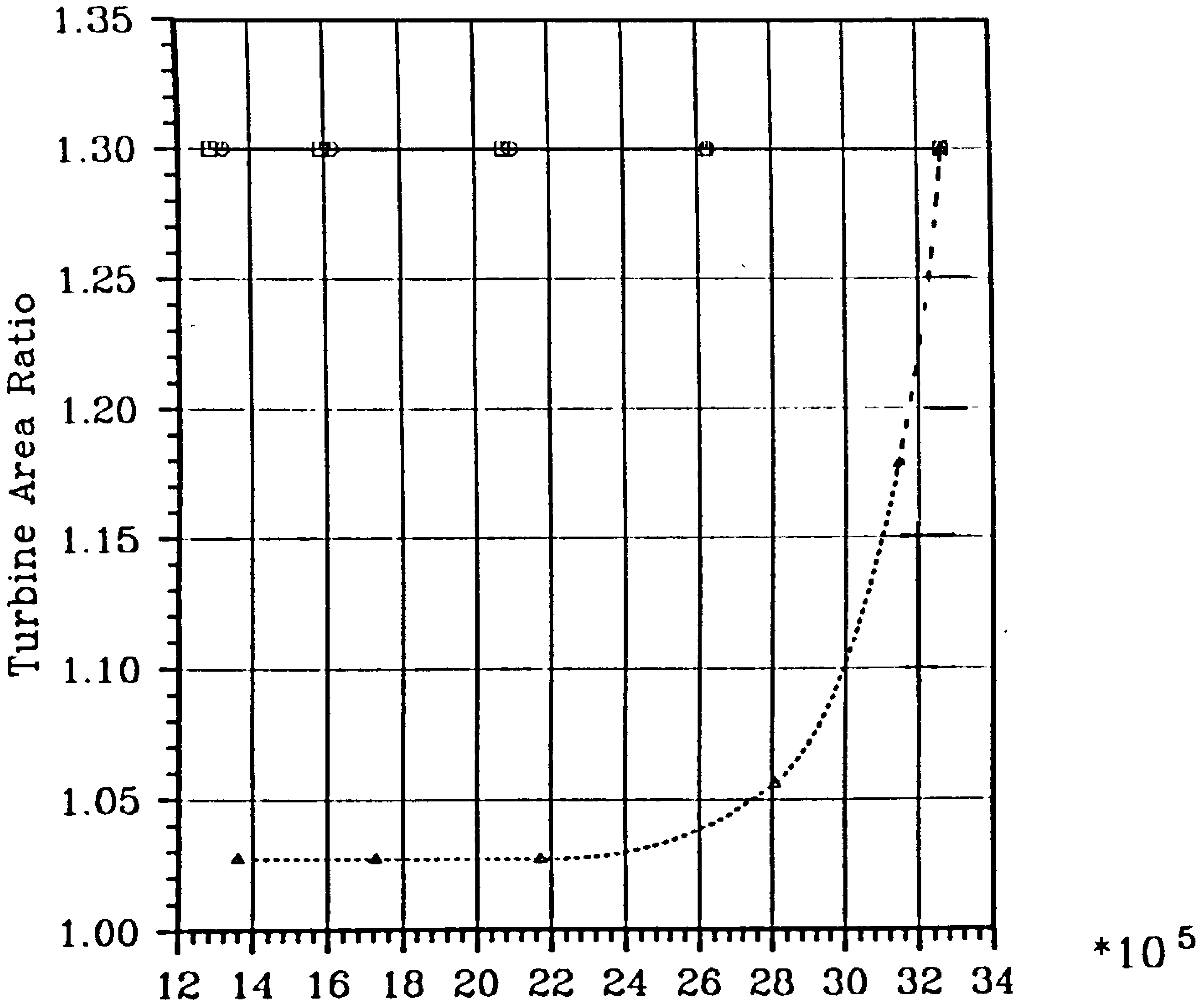



Fig. 7.19a LP Turbine Area Schedule

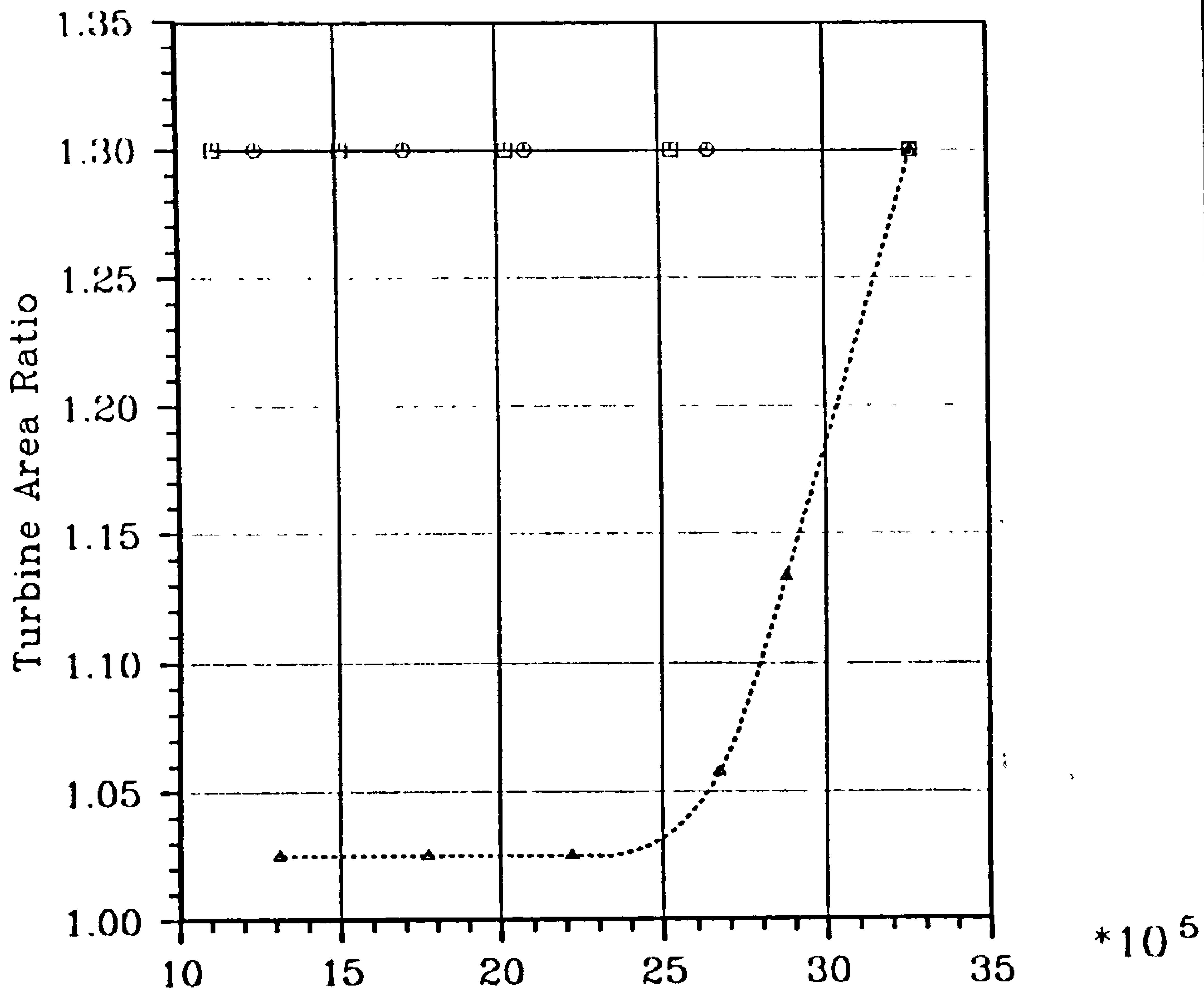



Fig. 7.19b LP Turbine Area Schedule

Alt (m) 0.0
Mn 0.0

- Fixed Comp. Vanes/ Fixed Turbine Area
- Fixed Comp. Vanes/ Var. HP Turbine Area
- Fixed Comp. Vanes/ Var. LP Turbine Area

# Turboshaft Performance Comparison With Different Controls

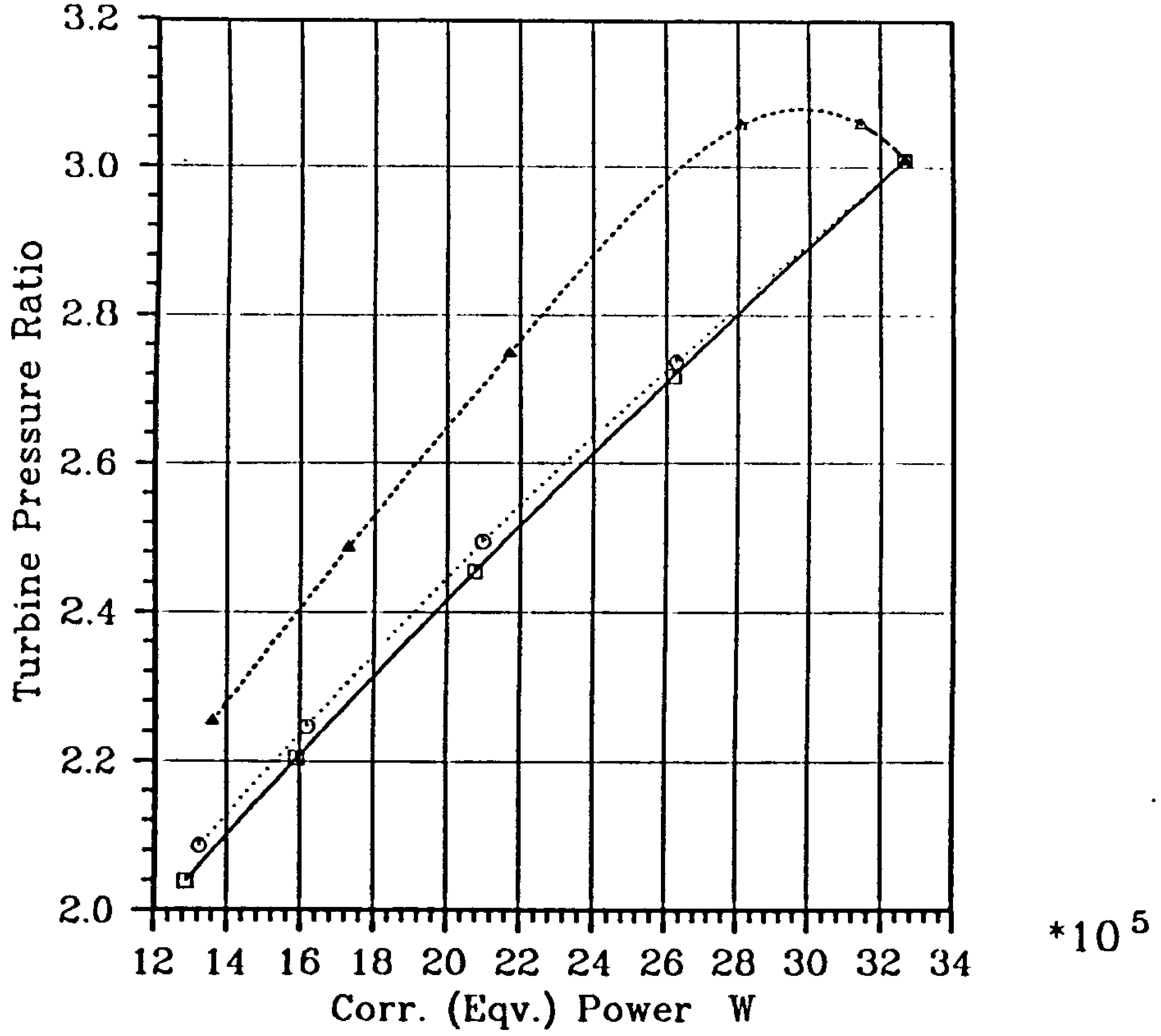



Fig. 7.20a LP Turbine Pr. Ratio

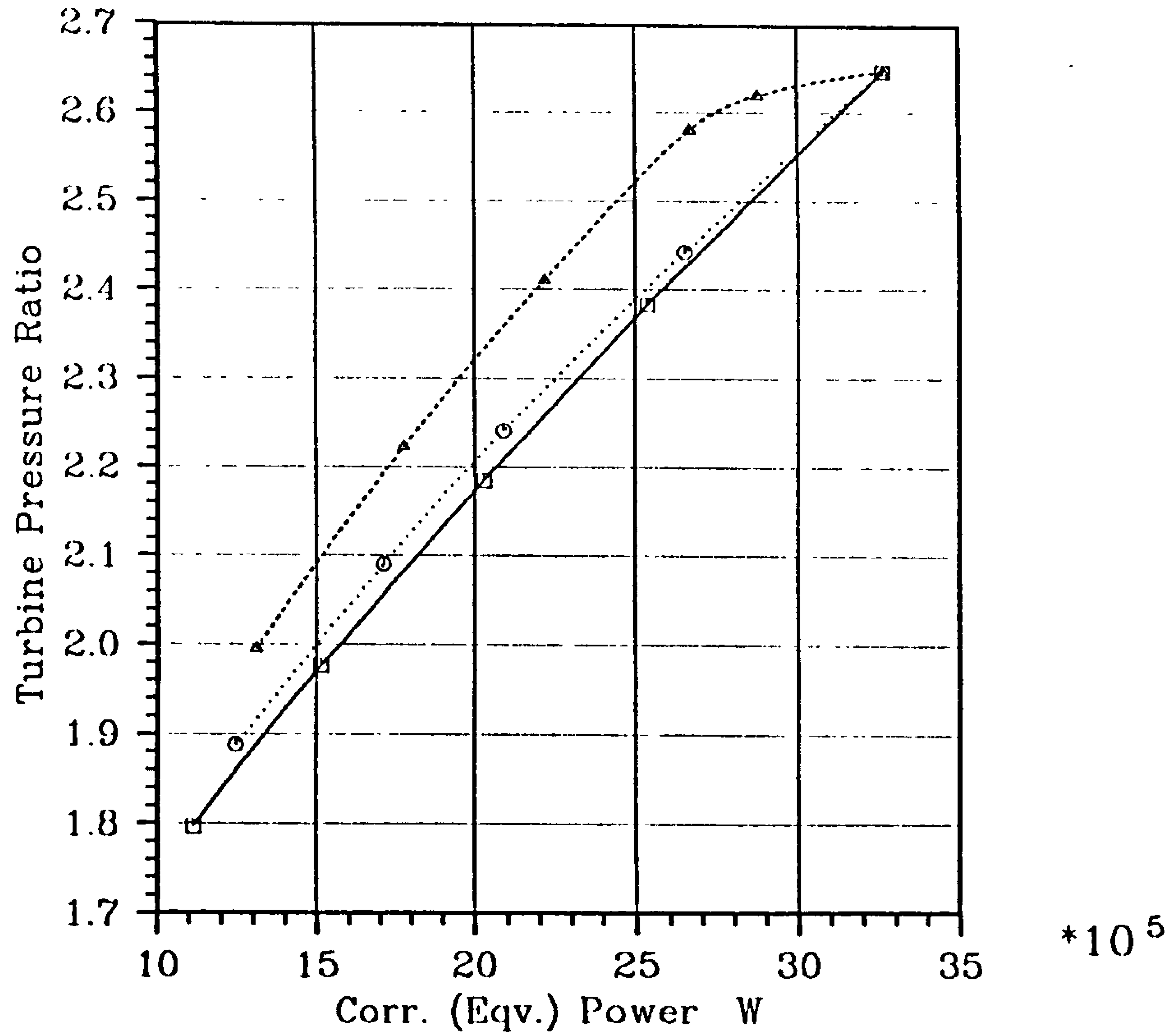



Fig. 7.20b LP Turbine Pr. Ratio

Alt (m) 0.0
Mn 0.0

- Fixed Comp. Vanes/ Fixed Turbine Area
- Fixed Comp. Vanes/ Var. HP Turbine Area
- Fixed Comp. Vanes/ Var. LP Turbine Area

# Turboshaft Performance Comparison With Different Controls

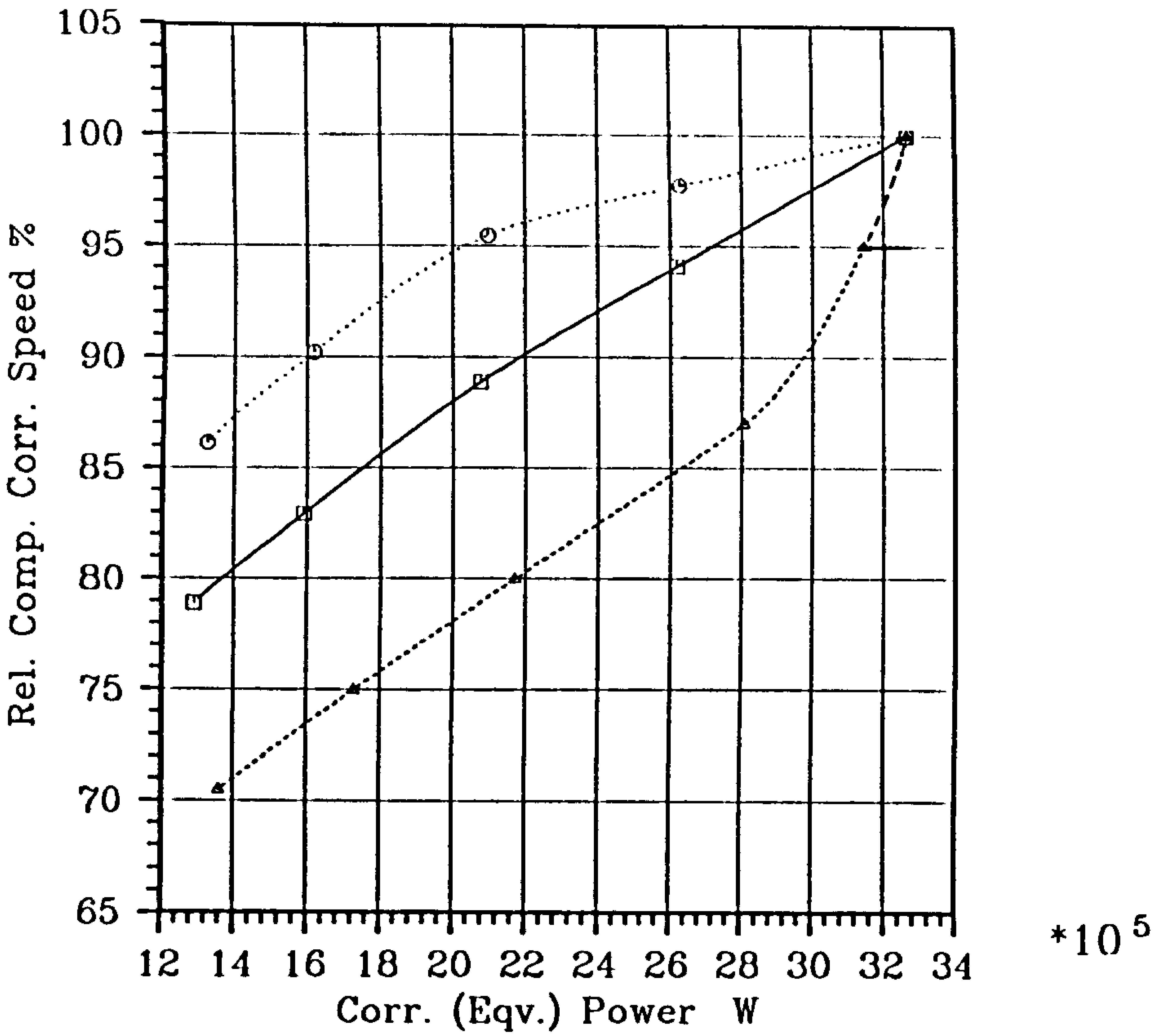



Fig. 7.21a Compressor Corrected Speed Schedule

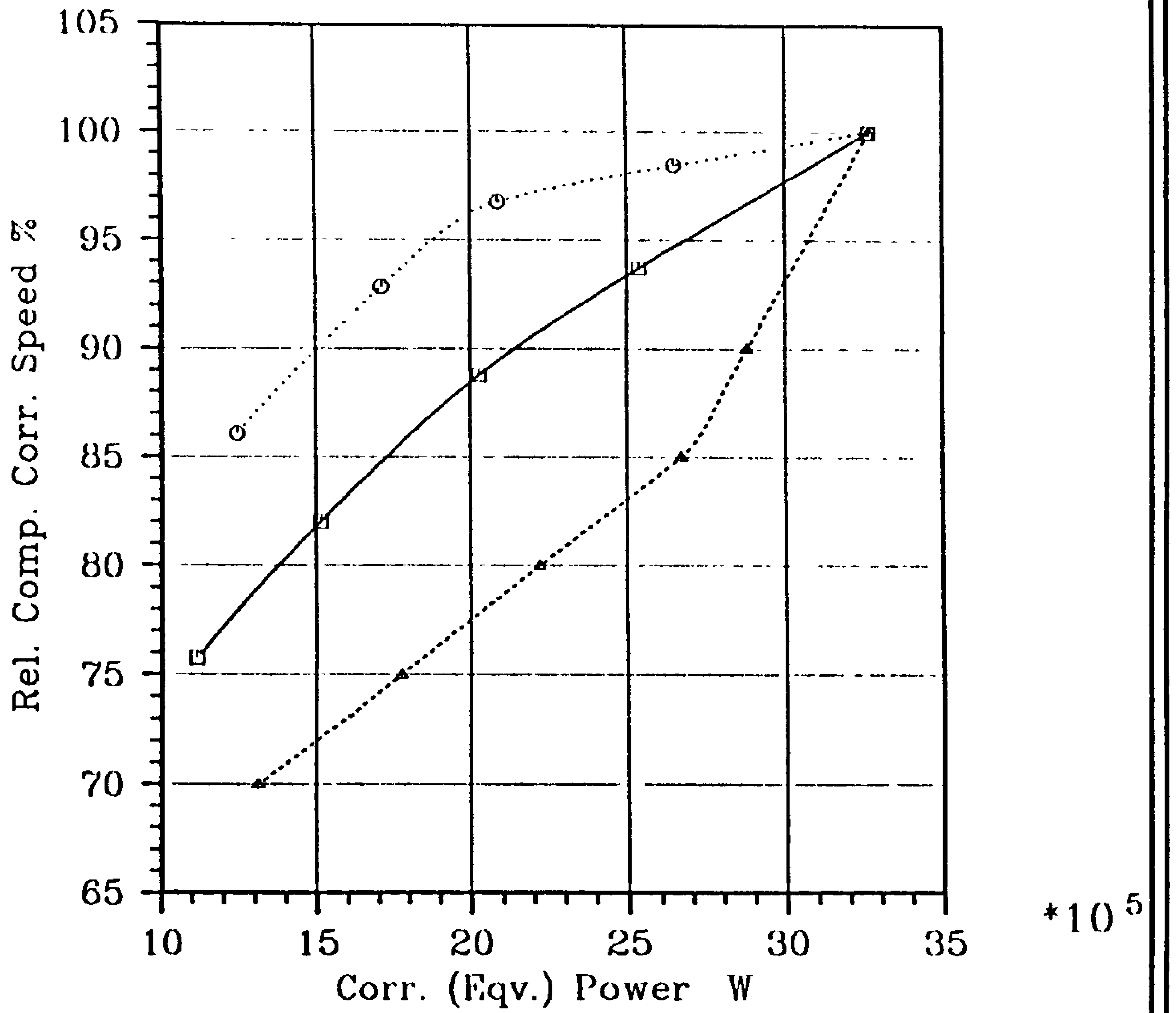



Fig. 7.21b Compressor Corrected Speed Schedule

Alt (m) 0.0
Mn 0.0

- Fixed Comp. Vanes/ Fixed Turbine Area
- Fixed Comp. Vanes/ Var. HP Turbine Area
- Fixed Comp. Vanes/ Var. LP Turbine Area

same temperature ratio, then the constant pressure ratio cycle has to operate at a higher speed as can be deduced from Fig. 7.13. With the constant TIT operation, the lower mass flow at which the cycle operates compared with the conventional cycle causes the gas generator rotor to operate at a lower speed at the same power output. Therefore, the constant pressure ratio mode of control gives greater acceleration rate while the constant TIT mode of operation gives a sluggish response.

As can be deduced from Fig. 7.15, if a variable area turbine is to be used to improve the part load performance of a free turbine turboshaft engine, then a significant performance gain can be obtained if a variable area power turbine is used to hold turbine inlet temperature at the maximum value in a regenerative cycle.

## 7.4 Comparison with Published Results

The authors of Ref. 29 carried out some investigations on various engine types in an attempt to establish the potential performance benefits that can be gained with the use of variable area turbines. The results obtained in this investigation are similar in trend to those given in Ref. 29 for the free turbine engine.

For the simple cycle case, a fuel saving of 5.7% over the fixed geometry engine was quoted at 50 percent power for the constant pressure ratio mode of control. The results for the constant TIT mode of operation were not included as this mode of control gave a poorer performance compared with the conventional fixed geometry engine. The cycle chosen had a pressure ratio of 10 and a TIT of 1644 K. A regenerative cycle of the same pressure ratio and TIT had a fuel consumption 9.5 percent lower than that of the conventional engine when controlled at constant TIT. The constant pressure ratio cycle gave a poorer performance than that of the conventional cycle.

Another cycle with a pressure ratio of 16 and TIT of 1920 K was also investigated. The improvement in sfc was a bit lower than the figures quoted above.

## 7.5 The Rapid Response Turboshaft

In general, gas turbines operate such that a change in gas generator rotor speed is needed to change power. The usual exception is normally found in applications for electrical power generation where a single shaft turboshaft is utilized. Due to rotor inertia, there is a time lag between fuel flow change and speed change, therefore, time is required to change output.

Variable geometry in compressors has been proposed as one means of reducing the response time of gas turbine engines and results of such investigation have been reported in many literature [38,51]. These studies considered the case wherein rotor speed is changed with variable IGV/stator vanes being used to shorten response time. However, there are certain cases where instantaneous power change may be desired such as in emergency or terrain contour flying and therefore power change with little or no rotor speed change may be beneficial. This can be accomplished by using compressor variable geometry and it may be possible to cover the entire power range if the vanes can be rotated through a wide angle change, within limits.

### 7.5.1 Factors Influencing Engine Response

Engine response time for a single spool turboshaft is given by

$$\Delta t = I_p \int_{\omega_i}^{\omega_f} \frac{\omega d\omega}{P} \qquad (7.1)$$

where,

| | | |
|---|---|---|
| $\Delta t$ | = | response time |
| $I_p$ | = | polar moment of inertia of rotor |
| $\omega$ | = | rotor speed |
| $P$ | = | distribution of available power during transient |
| $\omega_i, \omega_f$ | = | initial and final speeds, respectively |

The speed range, $d\omega$, can be reduced by employing variable geometry in the compressor as was explained in Chapter 3. A restriction on the amount of variable geometry that can be used is dictated by the losses occurring in the compressor at large angles of rotation. Loss of compressor efficiency increases rotor speed at both the low and high speed ends at a given pressure ratio and so to maintain power at the given pressure ratio, turbine inlet temperature has to increase; therefore, excessive loss of compressor efficiency should be avoided whenever variable geometry is used in compressors.

The polar moment of inertia of the shaft is influenced by compressor design parameters and will be fixed for the rotor. However, the compressor design parameters required to give a relatively low polar moment of inertia have been shown to be in contrast to what is required to maximize the effects of inlet guide vanes in order to reduce the speed range required. Power should be defined over the specified speed range with any limiting temperatures taken into account. Hence, if power can be changed while keeping shaft speed fixed, engine response time can be greatly reduced.

### 7.5.2 Engine Performance Analysis

The same engine model as that described in the previous sections on the constant pressure ratio and constant TIT simple cycles was simulated with variable IGVs in the compressor to find out what sort of performance level can be obtained when IGVs are used to control engine flow.

Choice of Design Point

The compressor characteristics used have a VIGV angle setting ranging from -25 to 42.5 degrees preswirl, amounting to a maximum change of 67.5 degrees. In actual fact, the characteristics are those of a VIGV fan designed at an IGV angle of 0 degree with 11 percent surge margin at a corrected speed of 100 percent. When the engine model was set up such that the design point of the compressor corresponded to that of the actual characteristics at the point of maximum efficiency, compressor surge occurred at low power; therefore, the surge margin at the design point had to be made larger. This posed some operational problems as such a point lies on the map in the region of low efficiency with the possibility that very high efficiencies may be obtained at medium and low power settings. A compromise point was chosen which gave a surge margin of 30 percent at the design speed and IGV angle of the actual characteristics.

Preliminary Analysis of the Effects of Variable Geometry

As was mentioned above, it was intended to use IGVs only to modulate power while keeping rotational speed fixed. A preliminary investigation into the feasibility of this showed that it would not be possible to obtain very low powers as the angle through which the vanes could be turned (0-42.5 deg.) was not sufficient enough and also, low compressor efficiencies encountered as the vanes were closed resulted in very high sfc's. As was mentioned in the above paragraph, it was undesirable in this case to use lower (in magnitude) IGV settings at design to avoid excessively high efficiencies at medium and low power settings. It should be borne in mind that appropriate characteristics would give a larger range of power without encountering high compressor efficiencies while maintaining adequate surge margin at low speeds.

As it was not possible to obtain very low powers using IGVs alone, it was decided to investigate the possibility of using variable geometry in one or more turbines in order to decrease the rate of change of IGV angle with power. The results of the preliminary investigation are shown in Figs 7.22 and 7.23. In these figures, the conventional engine wherein gas generator speed is changed to control engine airflow is compared with the constant speed VIGV control, "clean" and with one or more turbine areas variable.

The conventional control required a speed change of about 30 percent to reduce power down to about 25 percent, Fig. 7.22a, whereas the fully closed VIGVs reduced power to about 46 percent while maintaining constant speed, Figs 7.22b,a. Because of the very high sfc's generated by the VIGV control at medium power and also because of the low power range, Fig. 7.22c, it was decided to use variable geometry in one or more turbines to find out if any performance gains can be obtained. Both a gradual and step change in turbine areas were considered; the former was preferred. Variable geometry in either turbine reduced the rate of change of IGV angle with power equally well as shown in Fig. 7.22b. Some improvement in sfc was obtained, Fig. 7.22c. Deployment of variable geometry in both turbines gave an even better performance.

If we now turn our attention to Fig. 7.23d, it is clearly seen that variable geometry in the HP turbine is not desirable as there is the danger of running into surge at very low power settings, whereas variable geometry in the LP turbine gives a healthier running line. This is not surprising as a similar analysis to that given in [7] should show this.

The TIT variations are shown in Fig. 7.22d. TIT rises very steeply as the IGVs are further closed to lower power. This point was mentioned in an earlier section and the reason for this rise in temperature can be found in Fig. 7.23c where it is seen that there ia a sharp drop in compressor efficiency when the vanes are closed beyond 25 degrees. As a result of this loss in efficiency, the sfc rises very steeply with reduction in power as shown in Fig. 7.22c. The variation of HP turbine area with power is shown in Fig. 7.23a while that of the LP turbine is shown in Fig. 7.23b. Because of the rapid deterioration of surge margin when variable geometry is used in the HP turbine, variable geometry in this component was not further investigated even though it gives a good fuel consumption characteristic.

## Results

The investigation carried out on using variable geometry in the compressor revealed that it is unlikely that power modulation over the full power range can be accomplished at constant speed, not even with variable geometry in the power turbine as well. Therefore, the speed was allowed to drop, starting at about half power, to give improved performance over that when only IGVs were used to control engine power. At power levels below that obtained when the IGV angles were set at 30 degrees at the maximum speed, the

# Variable Geometry Effects On Rapid Response Turboshaft

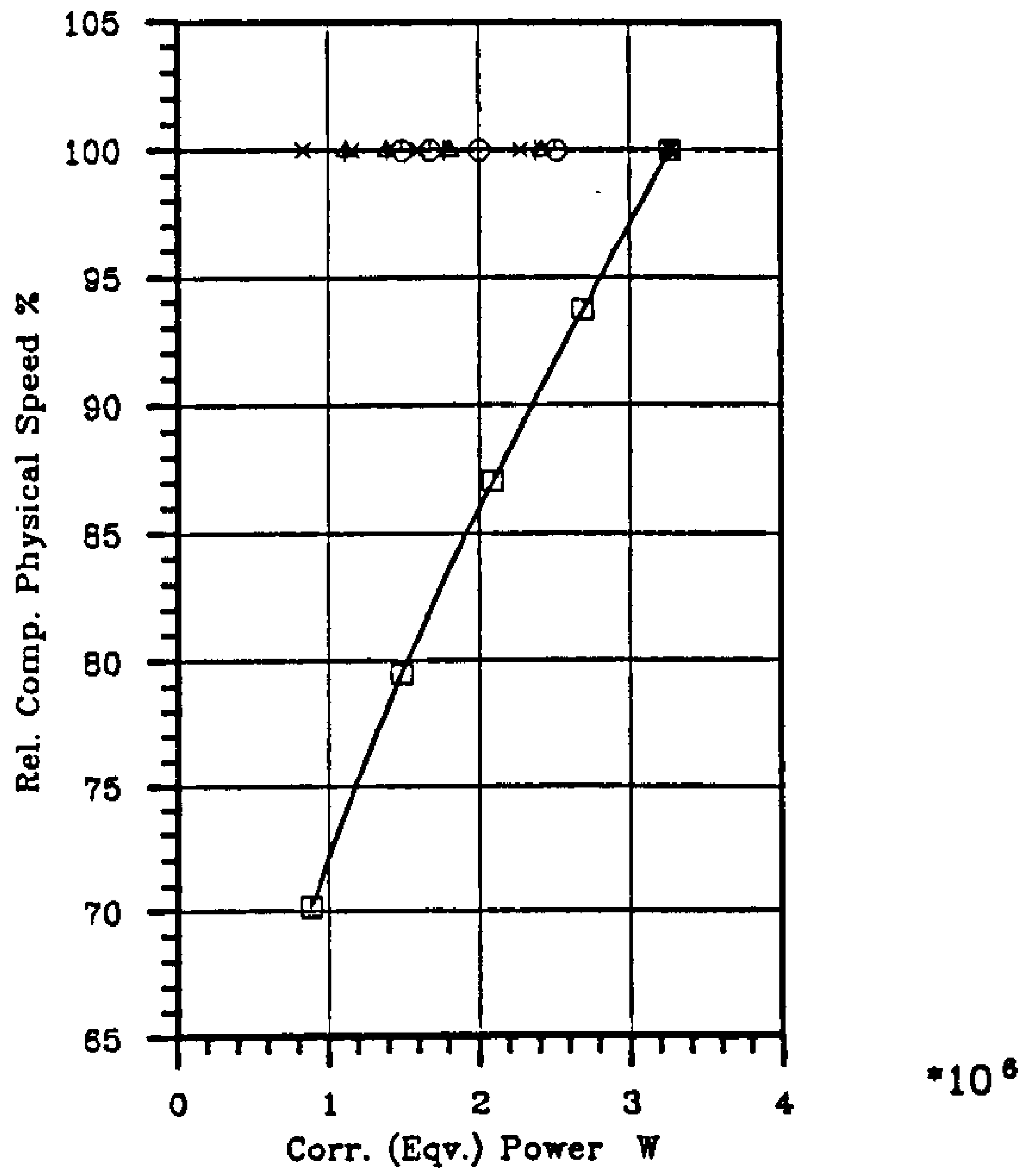



Fig. 7.22a Compressor Physical Speed Schedule

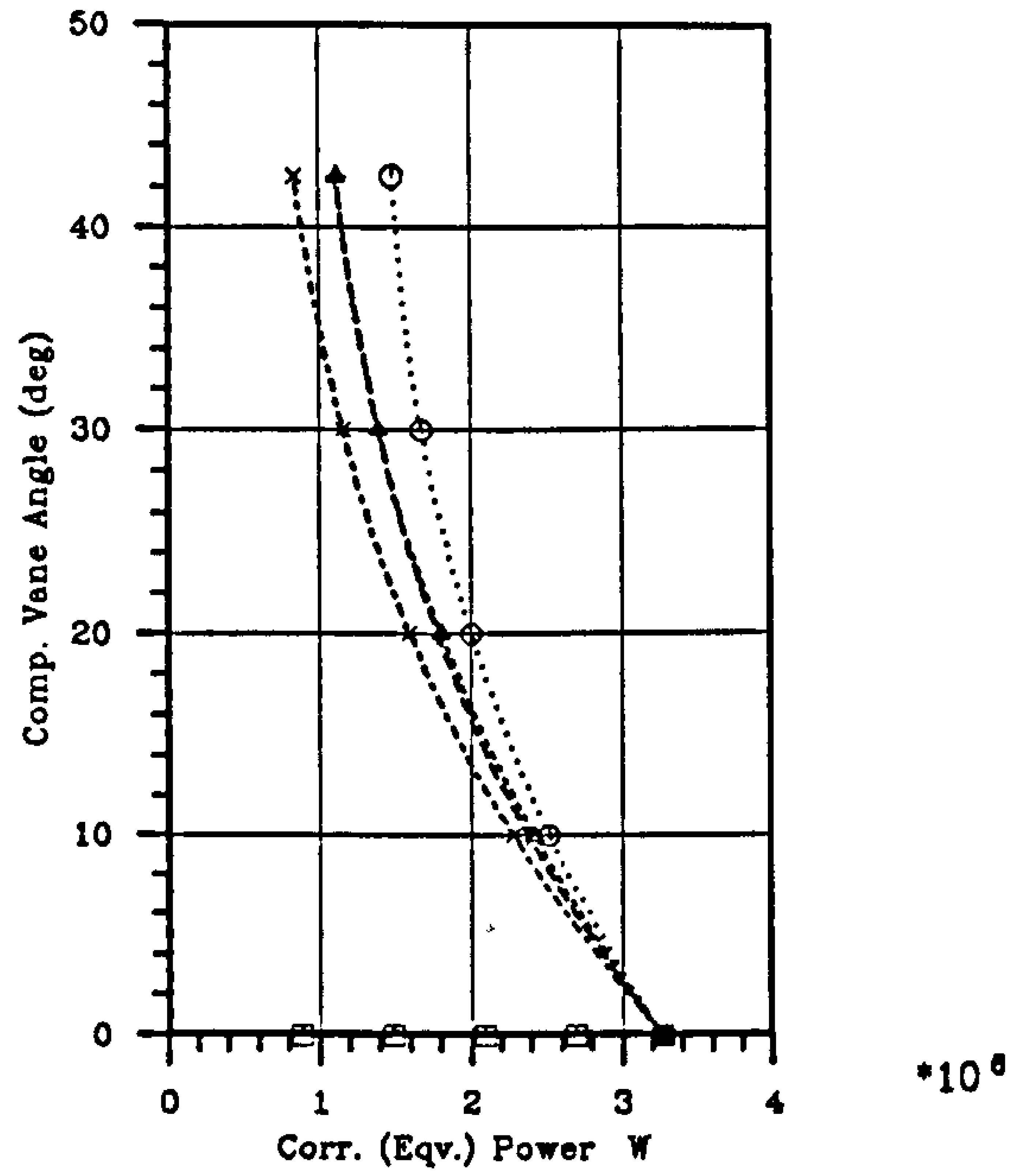



Fig. 7.22b Compressor Vane Angle

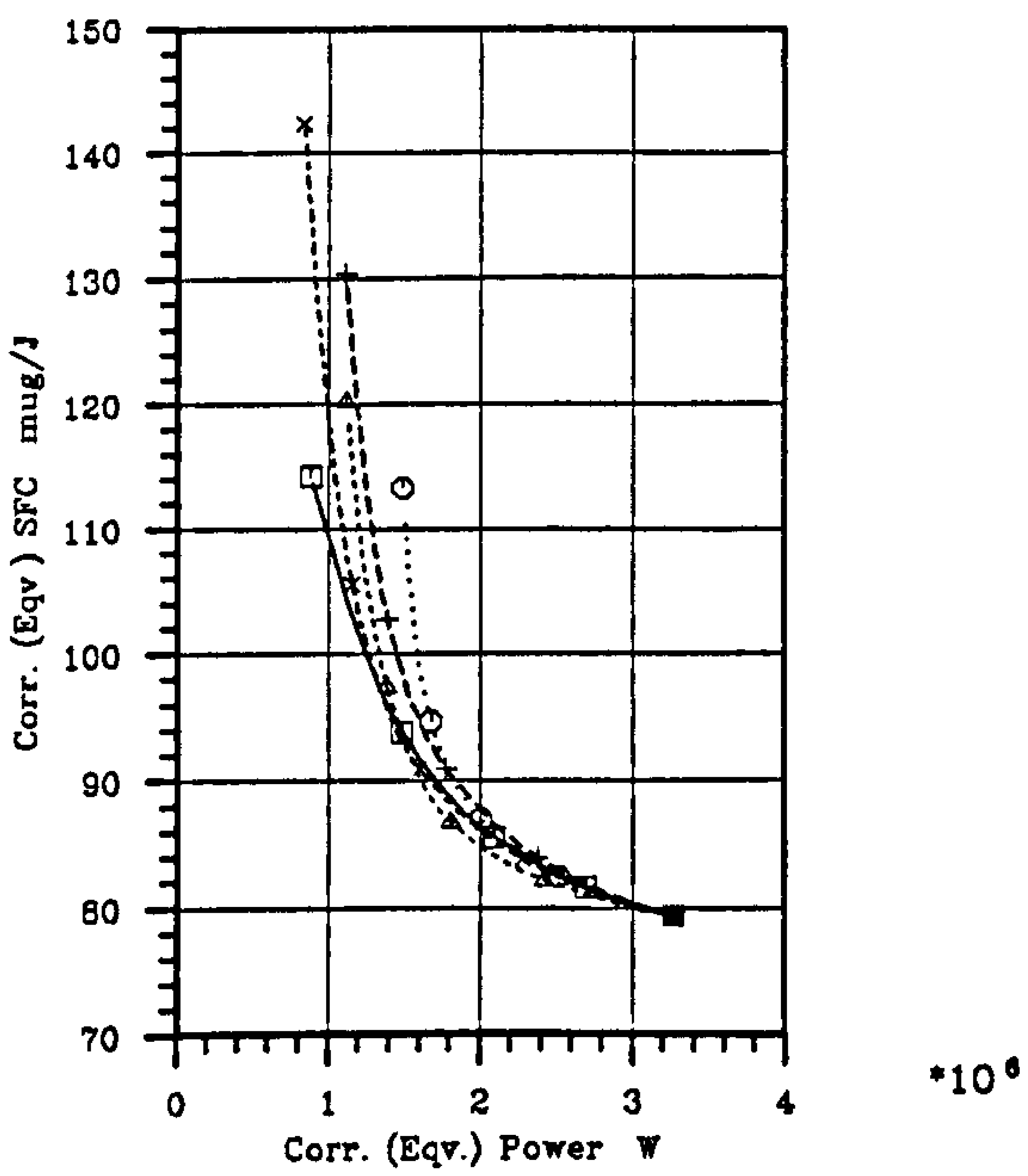



Fig. 7.22c Fuel Flow Requirement For Unit Power

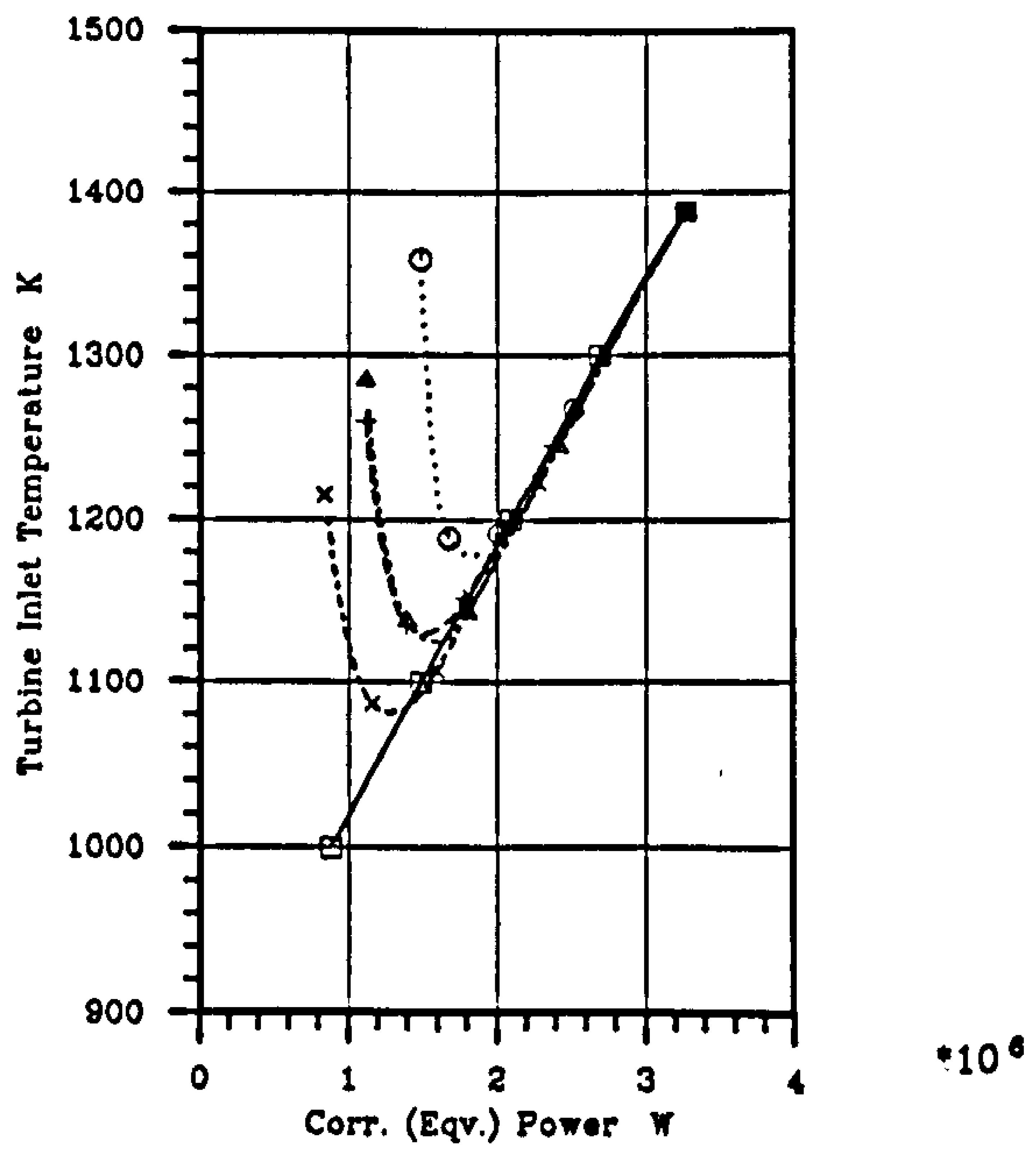



Fig. 7.22d Temp Schedule for Power Modulation

Alt (m) 0.0
Mn 0.0

- □—□ Fixed Comp. Vanes/ Fixed Turbine Area
- ⊙····⊙ Var. LP Comp. Vanes/ Fixed Turbine Area
- △----△ Var. LP Comp. Vanes/ Var. HP Turbine Area
- +---+ Var. LP Comp. Vanes/ Var. LP Turbine Area
- ×----× Var. LP Comp. Vanes/ Var. LP,HP Turbine Area

# Variable Geometry Effects On Rapid Response Turboshaft

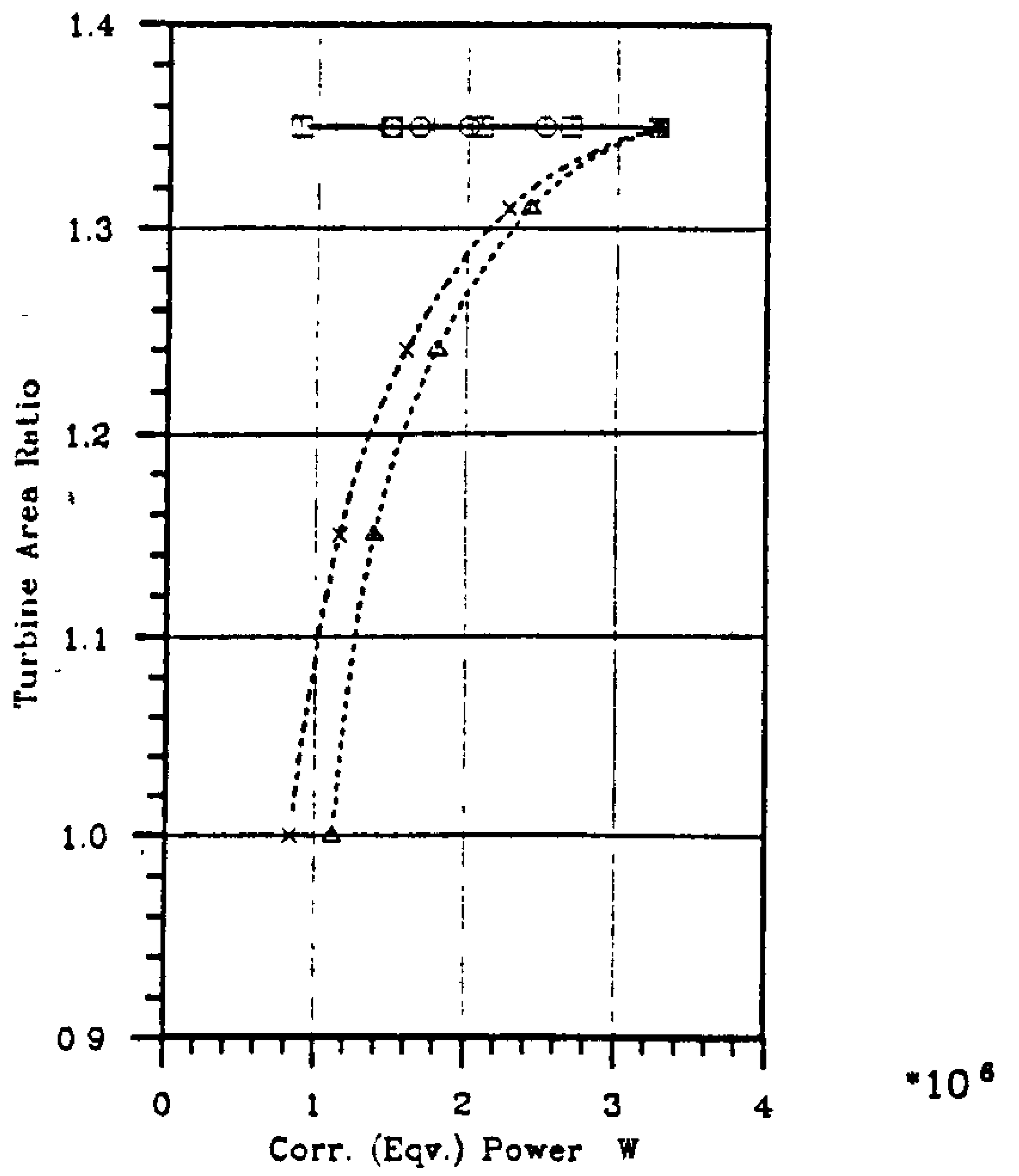



Fig. 7.23a HP Turbine Area Schedule

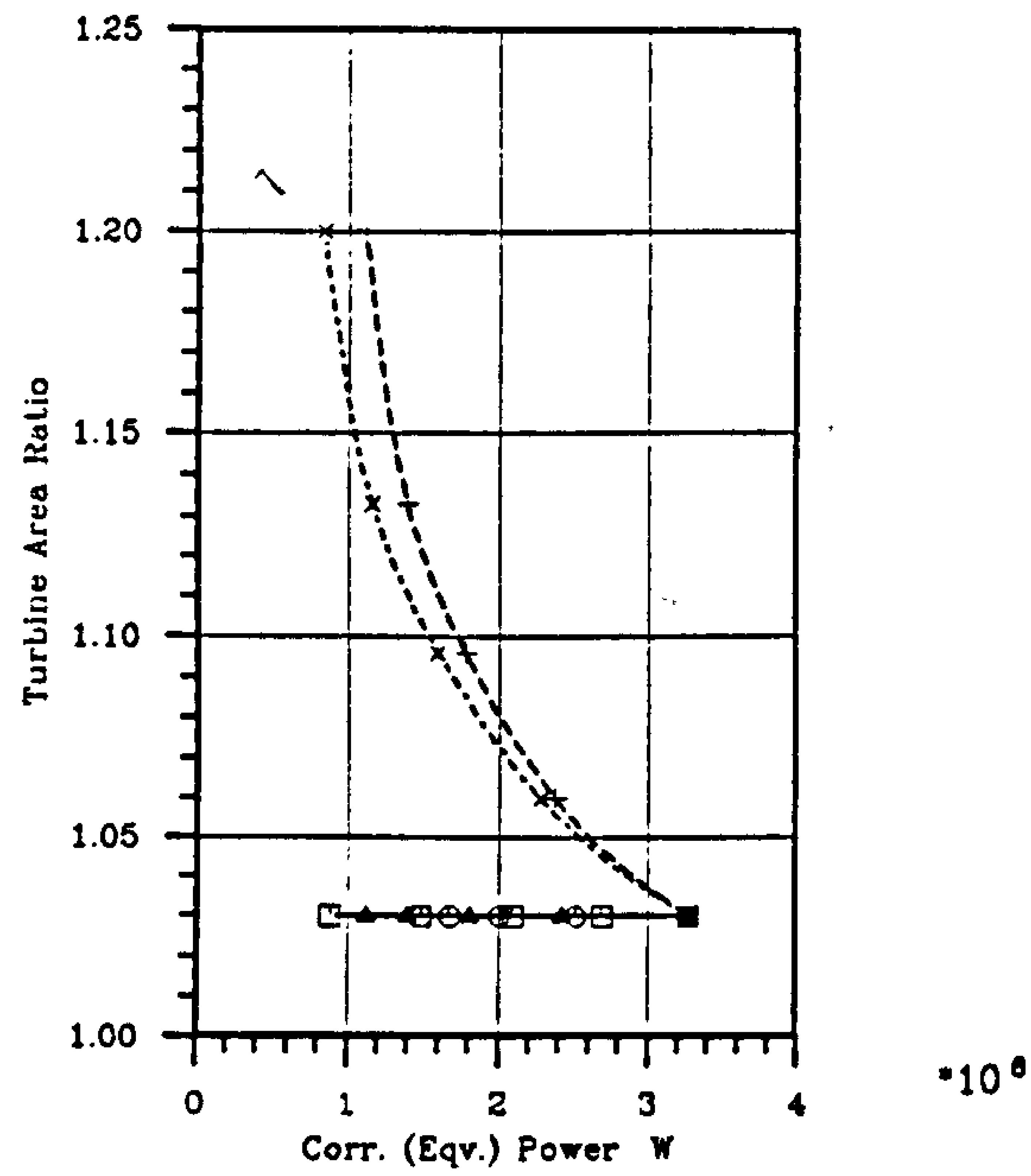



Fig. 7.23b LP Turbine Area Schedule

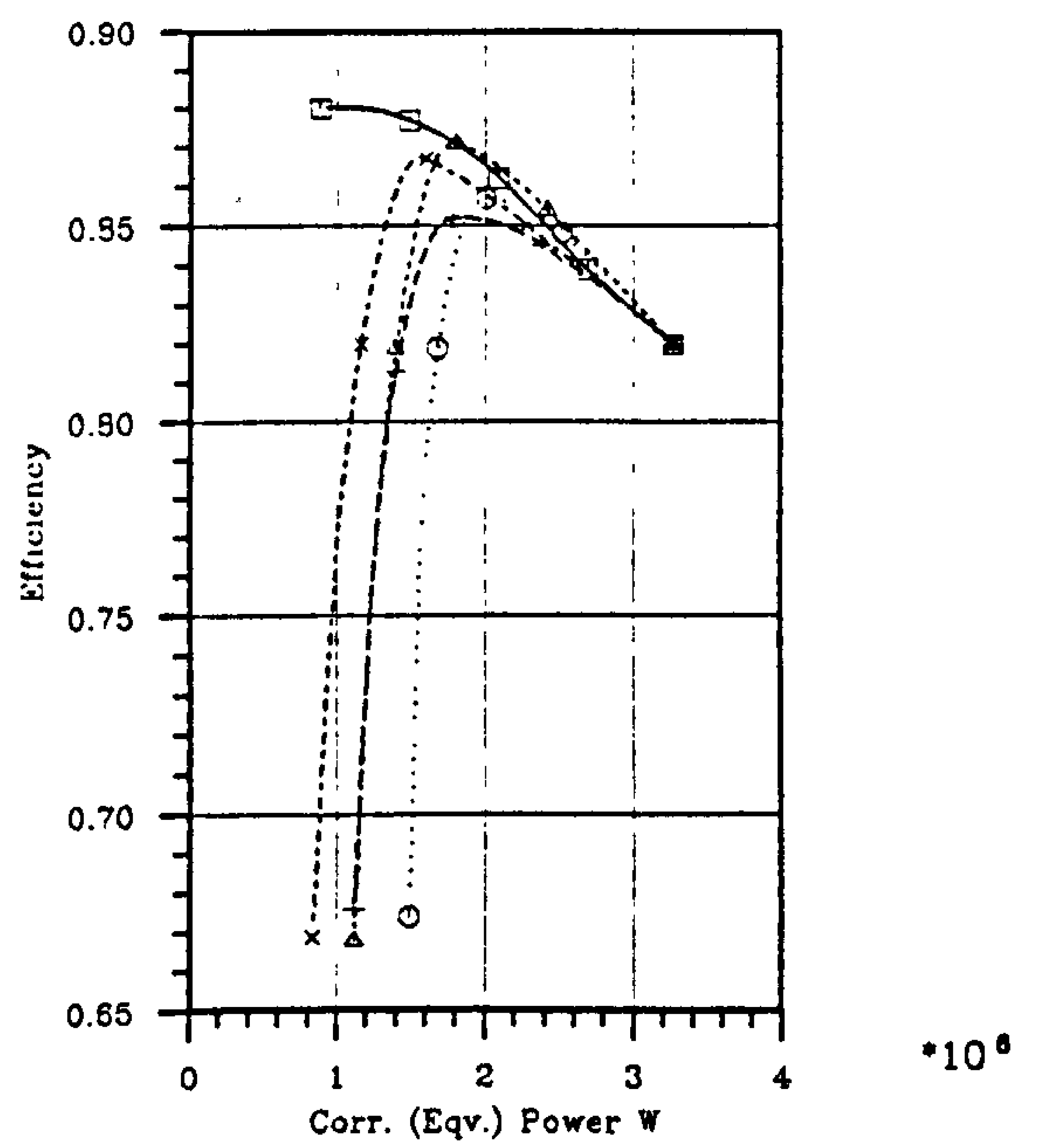



Fig. 7.23c Compressor Efficiency

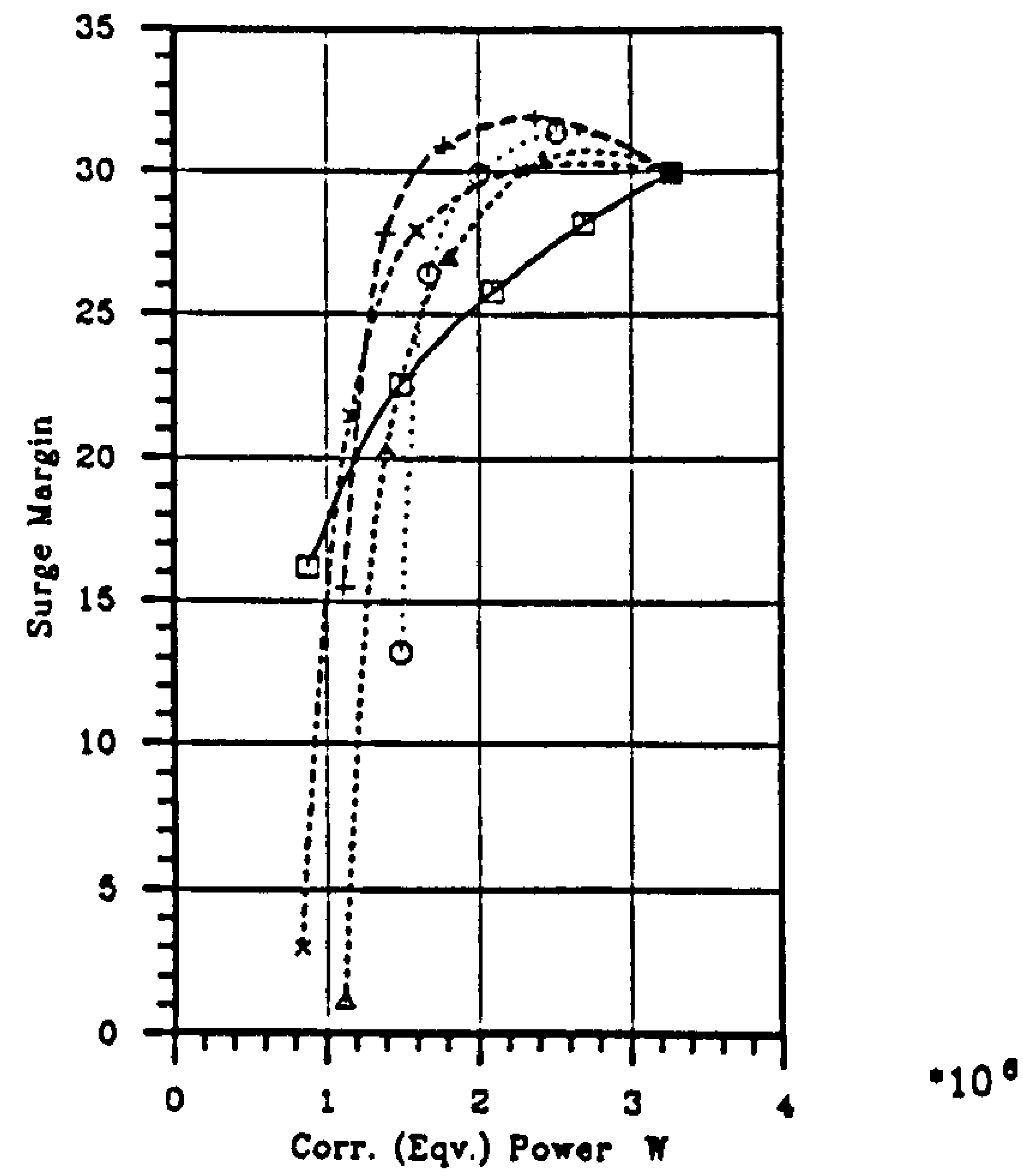



Fig. 7.23d Operating Surge margin

Alt (m) 0.0
Mn 0.0

- ⊞——□ Fixed Comp. Vanes/ Fixed Turbine Area
- ⊙······⊙ Var. LP Comp. Vanes/ Fixed Turbine Area
- ▲······△ Var. LP Comp. Vanes/ Var. HP Turbine Area
- +- - -+ Var. LP Comp. Vanes/ Var. LP Turbine Area
- ×- - - -× Var. LP Comp. Vanes/ Var. LP,HP Turbine Area

IGVs were held fixed at this value and speed decreased to effect a decrease in power.

The performance of the conventional engine is compared in the following figures with those of two other engines, one of which uses IGVs alone to modulate power while the other allows for some speed change. The variations of compressor corrected speed, IGV angle, and TIT with power are shown in Fig. 7.24. By allowing speed to drop while holding IGVs fixed, operation of the compressor at reasonably high efficiencies is possible thereby permitting the gas generator turbine to operate at moderate inlet temperatures. The sfc curves are shown in Fig. 7.25a where it is seen that there is a great improvement at low powers when speed is allowed to drop. The compressor operating lines are shown in Fig. 7.25b and although a plot of the surge margins is not given, a close examination of this figure and Fig. 7.23d will indicate that healthy running lines are obtained.

As altitude increases, the physical rotational speed would have to be dropped to maintain fixed corrected speed. The IGVs are opened further at the same turbine inlet temperature but with decreased output. As flight speed increases, a higher power is delivered at reduced guide vane angle with increased rotational speed to maintain corrected speed.

It is beyond the scope of this work to show that engine response will be improved by the use of variable geometry in the compressor. However, work carried out at Cranfield has supported this concept and it may be necessary to augment variable guide vane control to cover the entire power range.

# Performance Of A Rapid Response Turboshaft

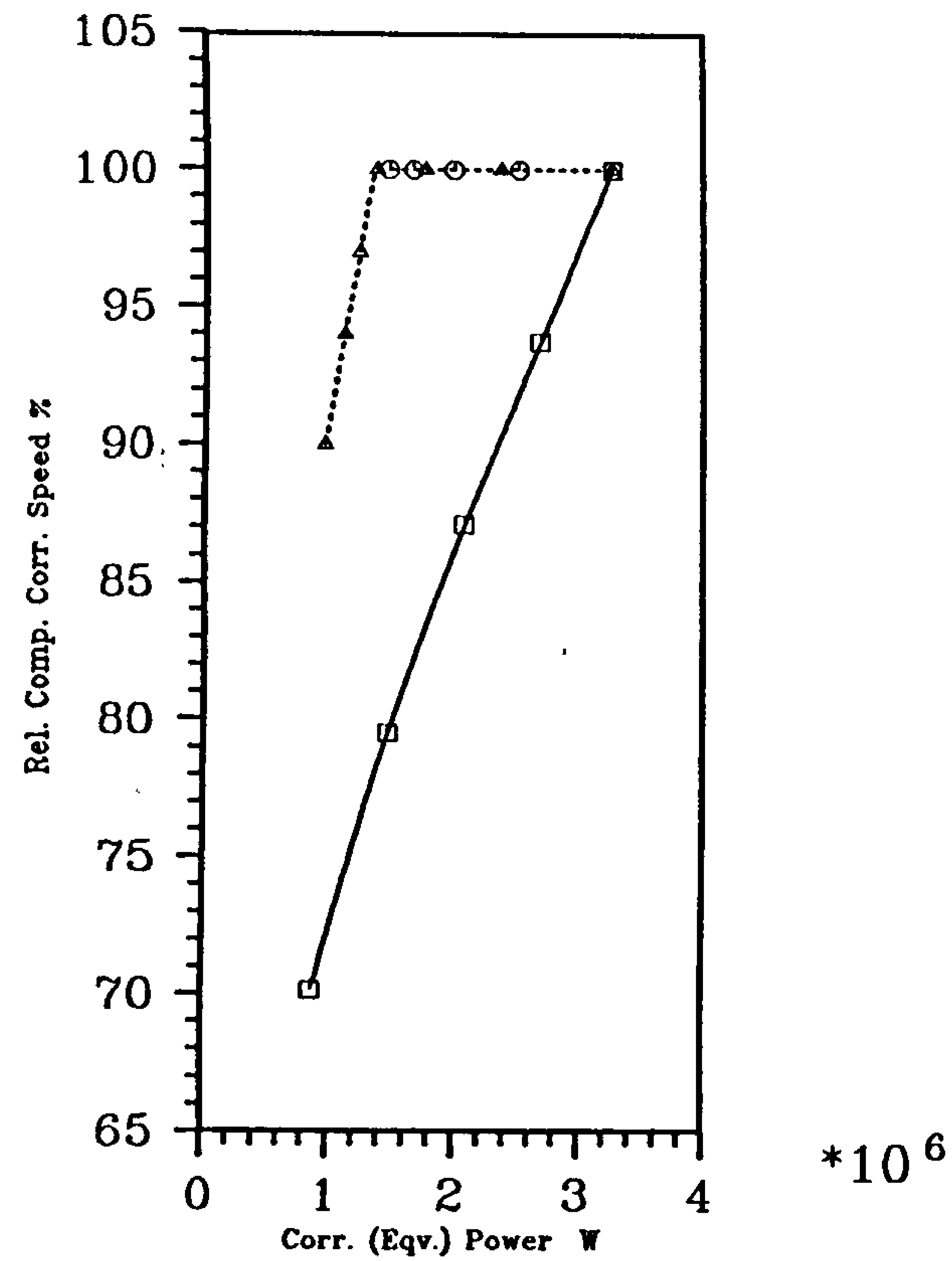



Fig. 7.24a Compressor Corrected Speed Schedule

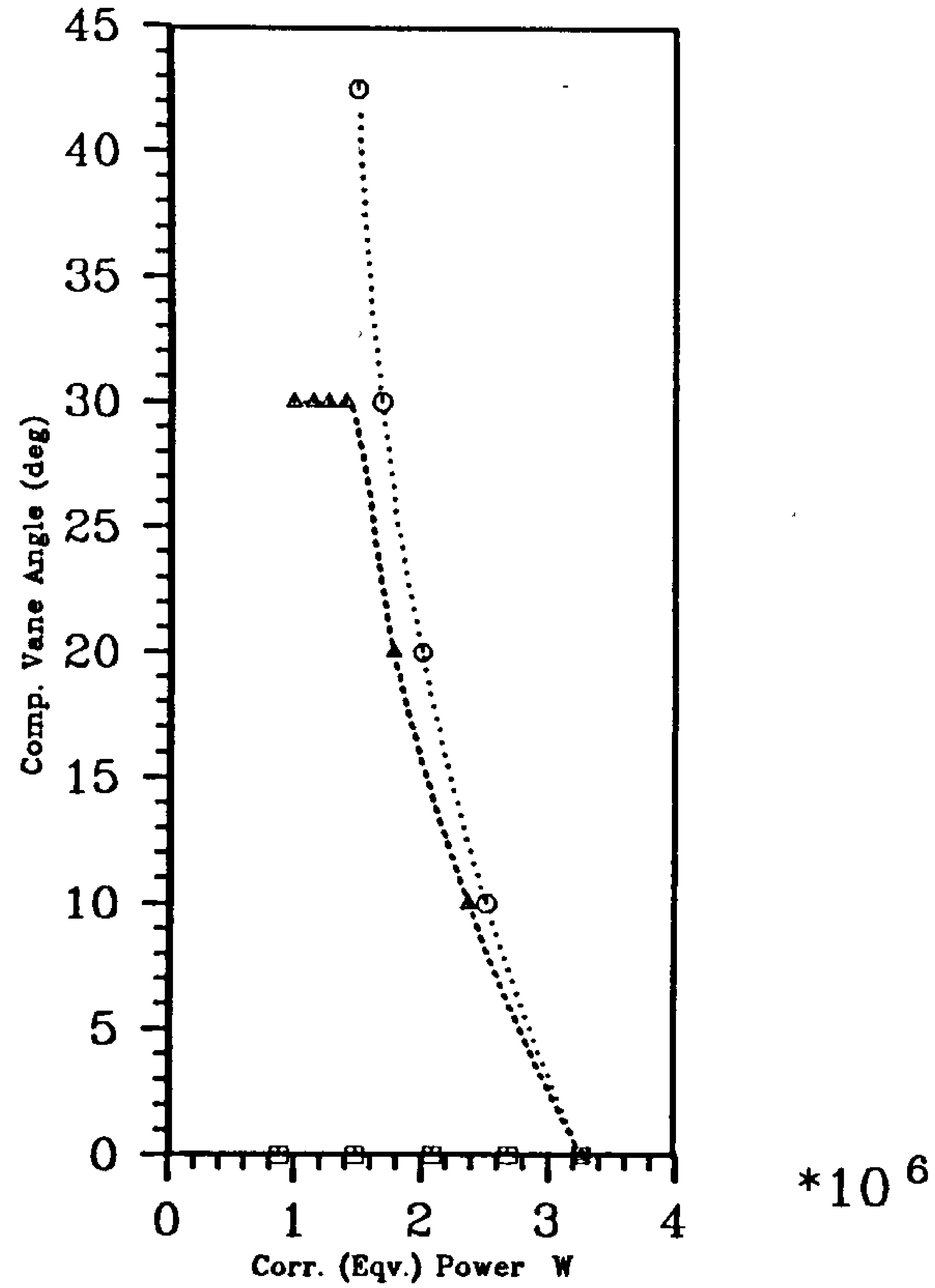



Fig. 7.24b Compressor Vane Angle

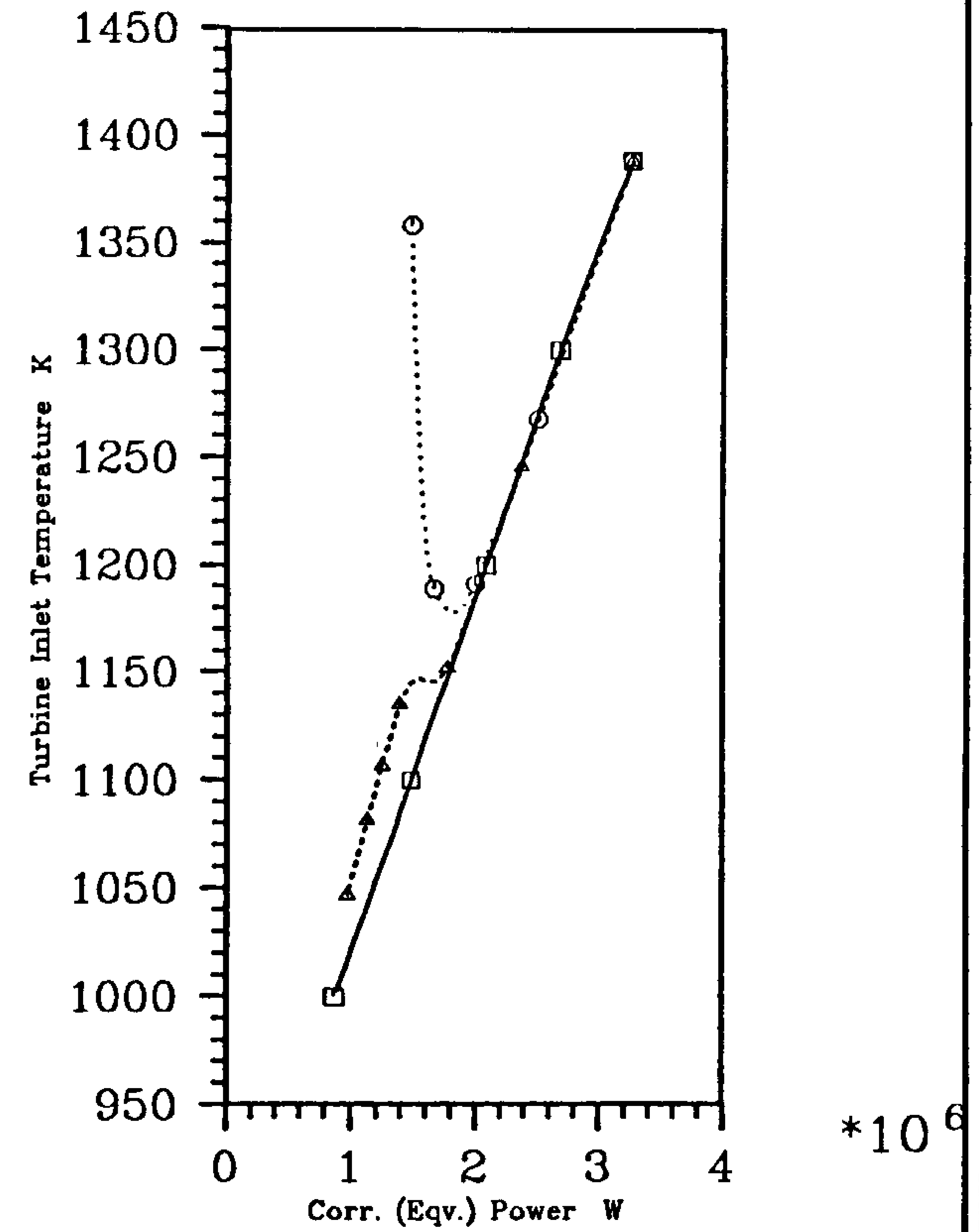



Fig. 7.24c Temp Schedule for Power Modulation

Alt (m) 0.0
Mn 0.0

- Fixed Comp. Vanes/ Fixed Turbine Area
- Var. Comp. Vanes/ Fixed Turbine Area
- Var. Comp. Vanes/ Var. LP Turbine Area

# Performance Of A Rapid Response Turboshaft

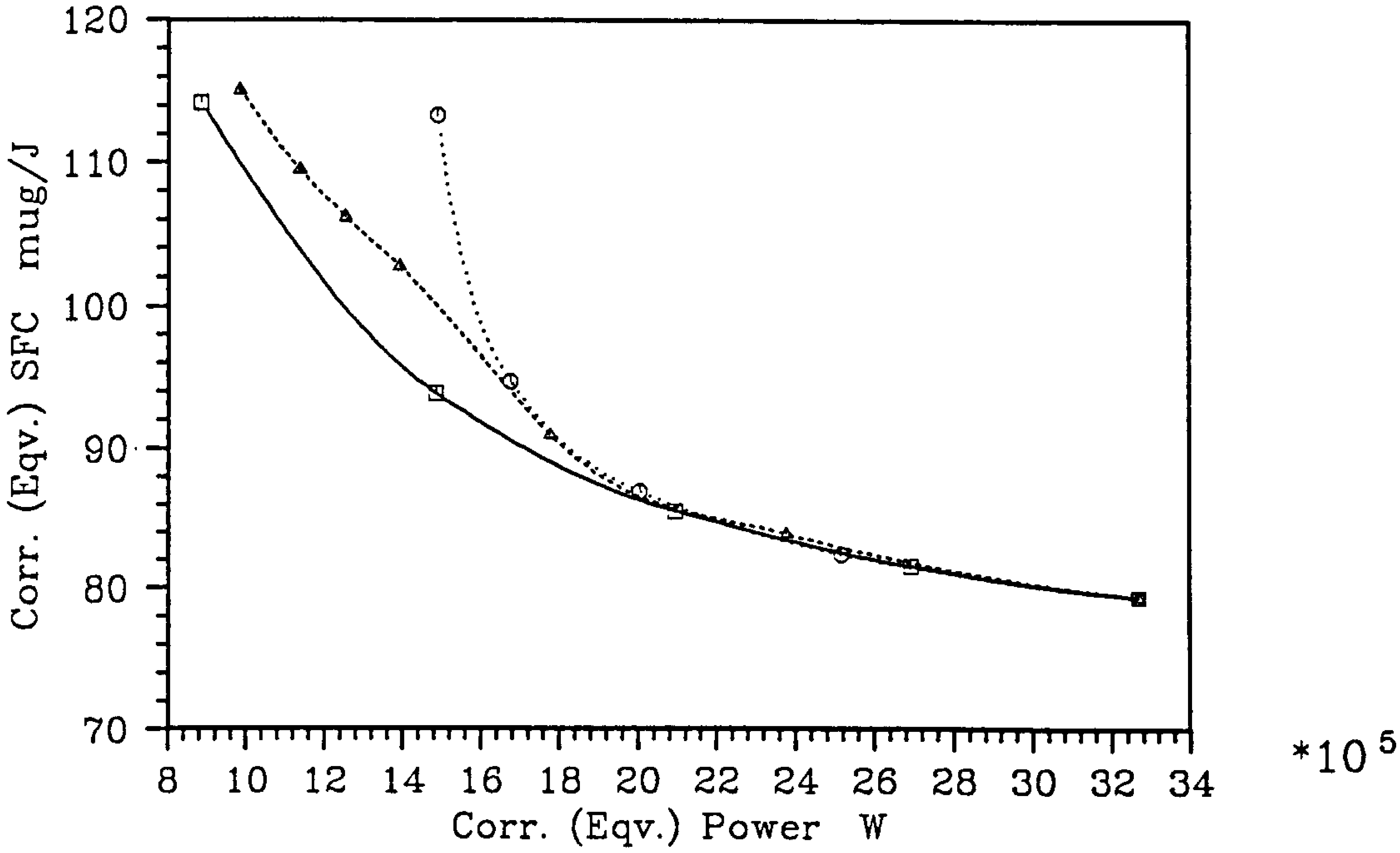



Fig. 7.25a Fuel Flow Requirement For Unit Power

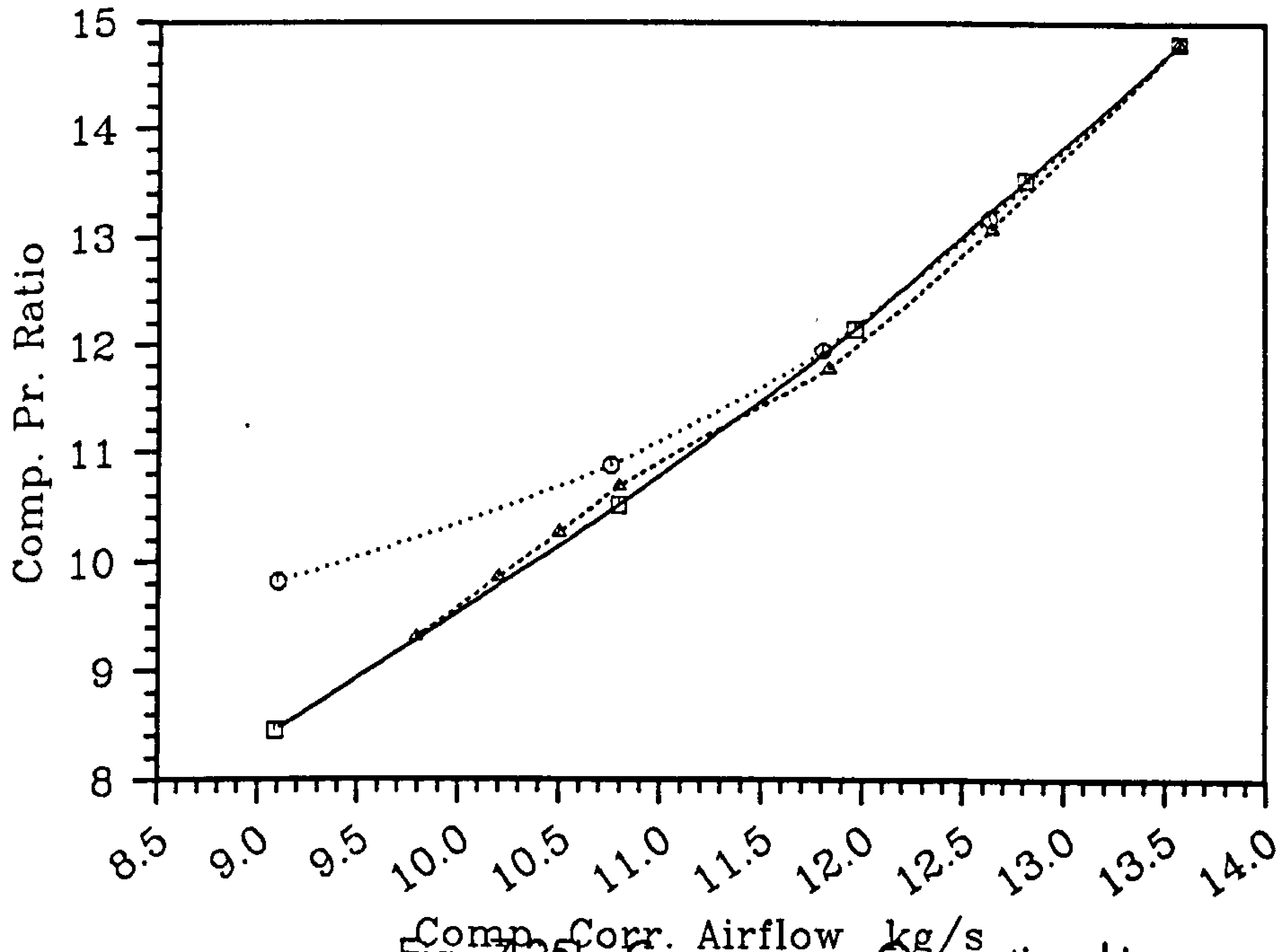



Fig. 7.25b Compressor Operating Line

Alt (m) 0.0
Mn 0.0

Fixed Comp. Vanes/ Fixed Turbine Area
Var. Comp. Vanes/ Fixed Turbine Area
Var. Comp. Vanes/ Var. LP Turbine Area

# CHAPTER 8

# AERONAUTICAL APPLICATIONS

Some of the problems encountered in operating aircraft gas turbines efficiently at off-design have been highlighted in Chapter 3. Some of these problems may be much more severe in fighter applications as these aircraft have to cover a wider flight envelope.

As was also discussed, the mixed flow turbofan is normally chosen for fighter application as a compromise to meet the conflicting requirements for low and high Mach number flights. The diverse duties to which future fighter propulsion systems will be put will cause the fuel burned in both subsonic and supersonic legs of any mission to be quite important and installation effects will greatly impact the final choice of cycle for a given application.

It was mentioned in an earlier chapter that the thrust and fuel burn characteristics of the turbofan differ from those of the turbojet so therefore, any alternative cycles or cycle change that will significantly improve the performance of an engine in any part of an aircraft's flight envelope is worth further investigation.

This chapter examines the possible performance improvements that can be gained by the use of variable geometry components in aircraft gas turbines. The use of the variable area turbine to improve principally sfc is first examined followed by the use of compressor variable geometry to improve the thrust response of lifting engines. Finally, a study is carried out to find out what the component loading requirements are and how they could be effected to accommodate a fully variable bypass ratio cycle.

## 8.1 The Constant Compressor Operating Point Cycle

It was shown in Chapter 3 that the aircraft gas turbine with fixed geometry components has its compressors and turbines matching such that overall pressure ratio and massflow, hence speed, decrease as thrust (TIT) is decreased. The resulting decrease in the cycle parameters coupled with component losses cause a deterioration in overall engine performance.

If an engine can be operated at constant compressor non-dimensional speed and turbine temperature ratio, then all engine non-dimensional variables will remain essentially fixed with the advantage that the compressors, turbines, etc., continue to operate at conditions close to design thus giving good component performance throughout the entire flight envelope. However, low thrust will be developed at low flight speeds as a result of the low TITs encountered. Flexibility of engine operation can be obtained without hurting engine performance by allowing the compressor to continue to operate at its design point while TIT is changed to modulate thrust. A variable area turbine is required to keep compressor pressure ratio fixed while a variable area propelling nozzle is desirable for constant speed operation. Since installation losses are directly related to changes in airflow with changes in thrust, it is to be expected that constant compressor design point operation will significantly reduce installation losses.

As was mentioned earlier, the turbojet engine gives good performance at high thrust levels, including supersonic flight, but is characterized by poor low thrust or subsonic performance. This large difference in performance between the low thrust, low speed operating conditions and the high thrust, high speed operating conditions makes this engine an attractive candidate for performance improvement. On the other hand, the turbofan provides a good compromise in meeting the conflicting performance requirements at both ends of the thrust spectrum by a suitable choice of bypass ratio. The performance of a turbojet having fixed compressor operating points will be compared with those of a fixed geometry turbojet and a fixed geometry turbofan to qualify the use of variable geometry in jet engines. Arguments will be presented to justify the use of variable geometry turbines in turbojets rather than turbofans for this mode of control.

The performances of these engines will be compared at four different flight conditions including that at acceleration. These are,

1. 9144 m/0.9 M/Part Dry
2. 9144 m/0.9 M/Max Power
3. 15240 m/1.6 M/Part A/B
4. 11000 m/0.6-2.2 M/Max Power

The first set of flight conditions is representative of flight at subsonic cruise whilst the second represents a typical subsonic combat sizing point. Supersonic cruise conditions are representative of the third set of flight conditions whilst acceleration during a supersonic dash is considered in the last set. All engines were sized to produce the same installed thrust at sea level static conditions. The design point parameters of the engines are given in Table 8.1. For the installed performance analyses, twin engine installations are assumed with the ratio of maximum nozzle exit area to maximum fuselage area taking a value of 0.4. Installation losses due to power extraction and air bleed are neglected. Twin-spool configurations are considered and the turbofan engine is of the unmixed type. All engines are optimized for maximum specific thrust at a TIT of 1600 K.

Design Point Performance (SLS ISA)

| | | | T/FAN ($\mu$=.7) | | T/JET |
|---|---|---|---|---|---|
| OPR | | | 25 | | 15 |
| Airflow | | (kg/s) | 100 | | 67 |
| TIT | | (K) | | 1600 | |
| Reheat | Temp. | (K) | | 2000 | |
| Thrust | Reheat/Max. Dry | (KN) | 80/62 | | 80/60 |
| sfc | Reheat/Max. Dry | (mg/Ns) | 38.4/21 | | 44.2/27.6 |
| Fan/LPC | Pressure Ratio | | 3.82 | | 3.4 |

Table 8.1 Performance of Study Engines

The design point performance variables are quite conservative in that they reflect the performance levels of present day state-of-the-art technologies and not that expected at the time period when variable area turbines will be in service. It was decided to choose this performance level to match the available variable area turbine data. Another set of design data will be considered which will reflect the performance levels of future technologies.

Each inlet should be sized to pass the maximum mass flow demanded by the cycle. In reality, each inlet-engine combination has a unique maximum airflow characteristic, and at the inlet sizing point, the reduced mass flow should be calculated and the inlet capture area obtained from the inlet characteristic. Although an inlet characteristic is provided internally in the program, this was not used. Various points in the flight envelope were considered and the maximum inlet area thus obtained was taken as the capture area. It is possible that at takeoff, auxiliary intake doors may be needed to pass the airflow demanded by the engine.

The curves shown in the following figures compare the performance of a turbofan with that of a turbojet engine with and without variable area turbines. Variable geometry was used in both turbines of the VAT turbojet to keep the operating point of the compressors fixed at the design point.

## Subsonic Performance

The installed performance curves for flight conditions 1 and 2 listed above are shown in Figs 8.1 through 8.7. A typical cruise power setting would lie in the range of 25-30 percent of the maximum available thrust whereas combat thrust would be set at about 90-95 percent. The turbofan is used as a reference.

As the augmentor fuel flow is reduced, there is no change in the operating point of the rotating components as it is assumed that the afterburner is separated from the gasifier, but thrust decreases as do the core nozzle throat and exit areas to accommodate the change in density of the exhaust gases. The main burner fuel flow will have to be reduced before the operating point of each gas generator component changes. The turbine inlet temperature schedule is shown in Fig. 8.1a where it is seen that at intermediate power, all the engines are able to operate at maximum TIT without exceeding spool speed limits, Fig. 8.1b. The variable area turbines in conjunction with the variable area nozzle are able to modulate the thrust of the variable geometry turbojet down to 35 percent keeping the compressors operating at their design point.

The fan and LP compressor airflow schedules are shown in Fig. 8.2a while the pressure ratio variation can be found in Fig. 8.2b. Because of the higher mass flow at which the variable geometry turbojet operates, its TIT is lower at a given thrust as is seen in Fig. 8.1a. Inherently, the VAT turbojet develops a higher thrust at intermediate power.

As TIT is reduced to decrease thrust, the HP turbine area of the variable geometry engine decreases in proportion to the root of the turbine inlet temperature to maintain choked conditions at turbine inlet, Fig. 8.3a. Since the mass flow into the LP turbine remains almost constant, the LP turbine area changes in proportion to the change in $\sqrt{T}/P$ if choked conditions are maintained. Both the total temperature and pressure at inlet decrease as thrust is reduced. Fig. 8.3b shows that the LP turbine area increases as output decreases indicating that the HP turbine outlet pressure decreases much faster that the outlet temperature. The other two engines operate with both these areas fixed. At low power settings, the VAT turbojet could not continue to operate with fixed compressor design points as the LP turbine work function got too large.

The plots of turbine pressure ratios can be found in Fig. 8.4 from where it is seen that both turbine pressure ratios of the variable geometry engine increase as thrust is reduced. Since both compressors are controlled to operate at their design point, both turbine work remain constant. As the inlet temperatures decrease with reduced thrust, the work function, $\Delta H/T$, for each turbine increases and since turbine pressure ratio and work function are equivalent expressions, the turbine pressure ratios increase with

Performance Comparison Of Fixed And Variable Geometry Jet Engines

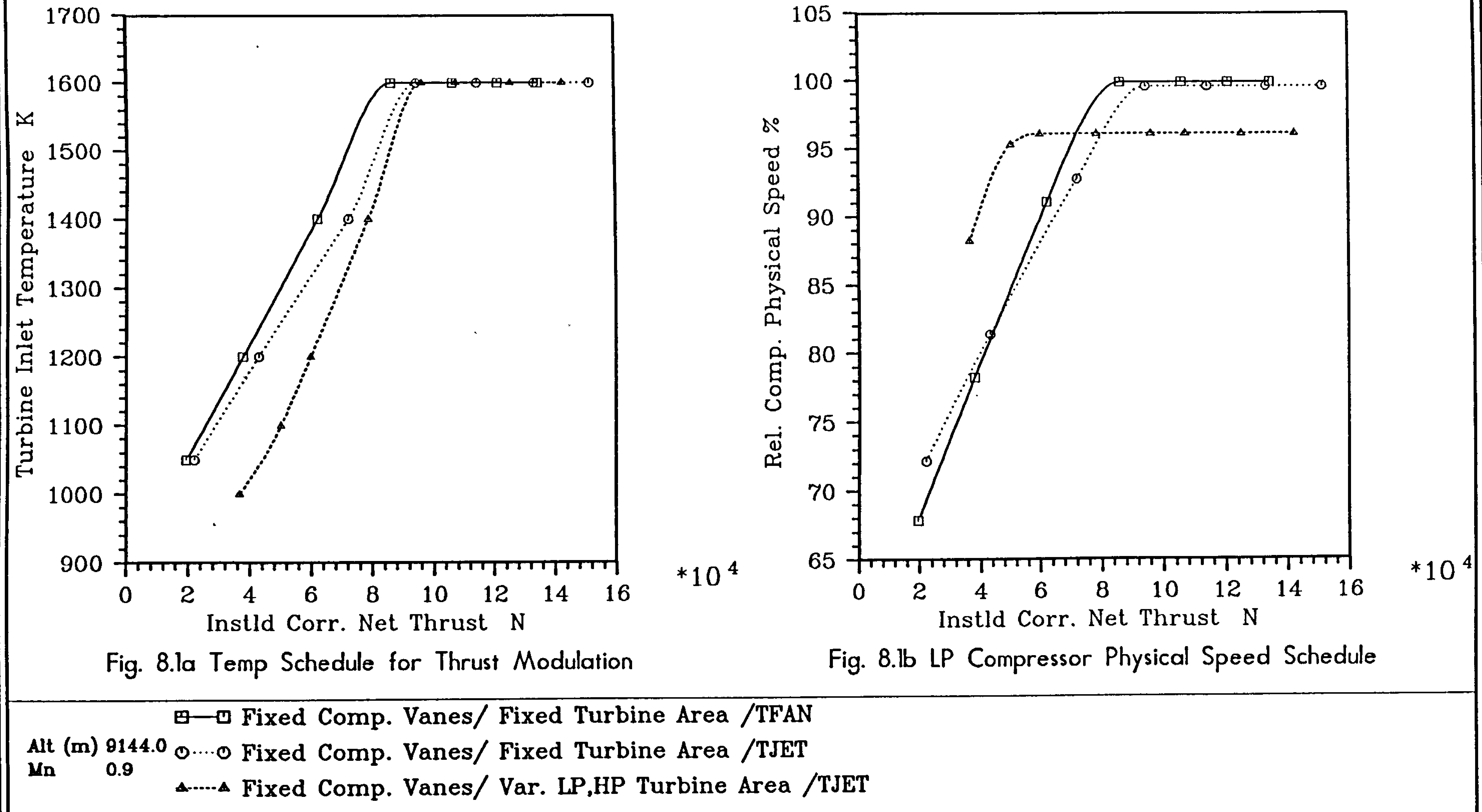



Fig. 8.1a Temp Schedule for Thrust Modulation

Fig. 8.1b LP Compressor Physical Speed Schedule

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

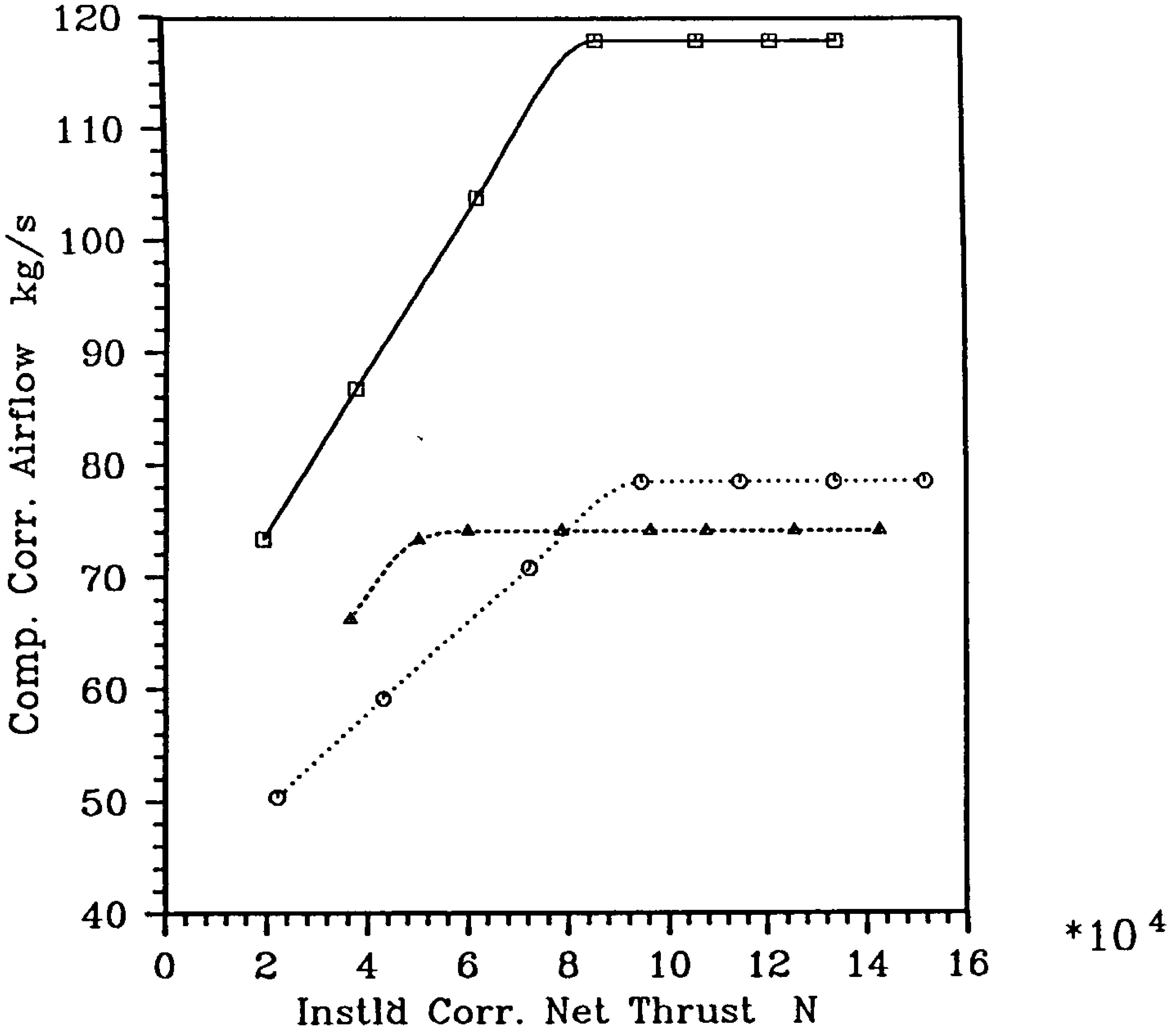



Fig. 8.2a LP Compressor Airflow Requirement

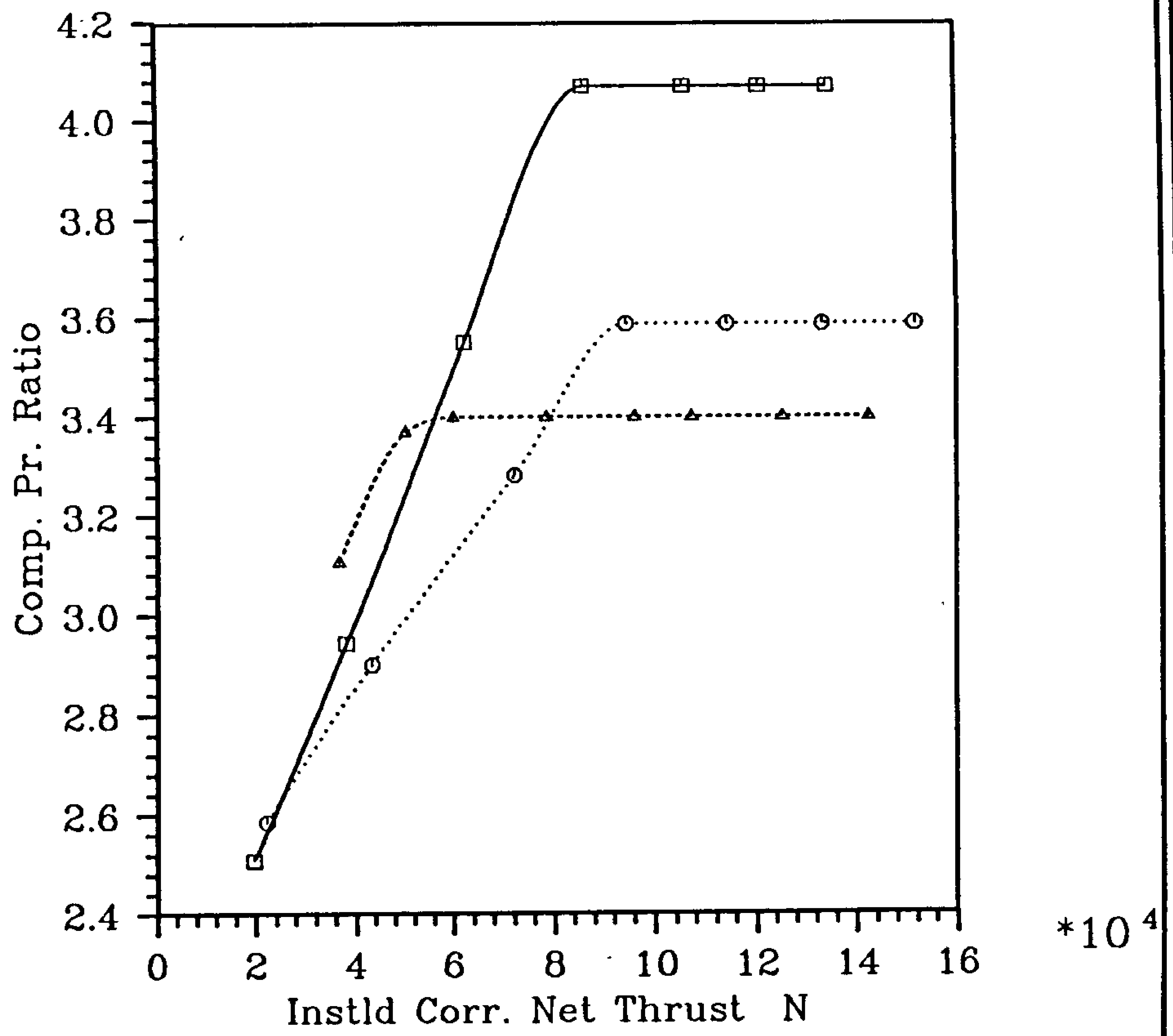



Fig. 8.2b LP Compressor Pr. Ratio

Alt (m) 9144.0
Mn 0.9

- Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
- Fixed Comp. Vanes/ Fixed Turbine Area /TJET
- Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

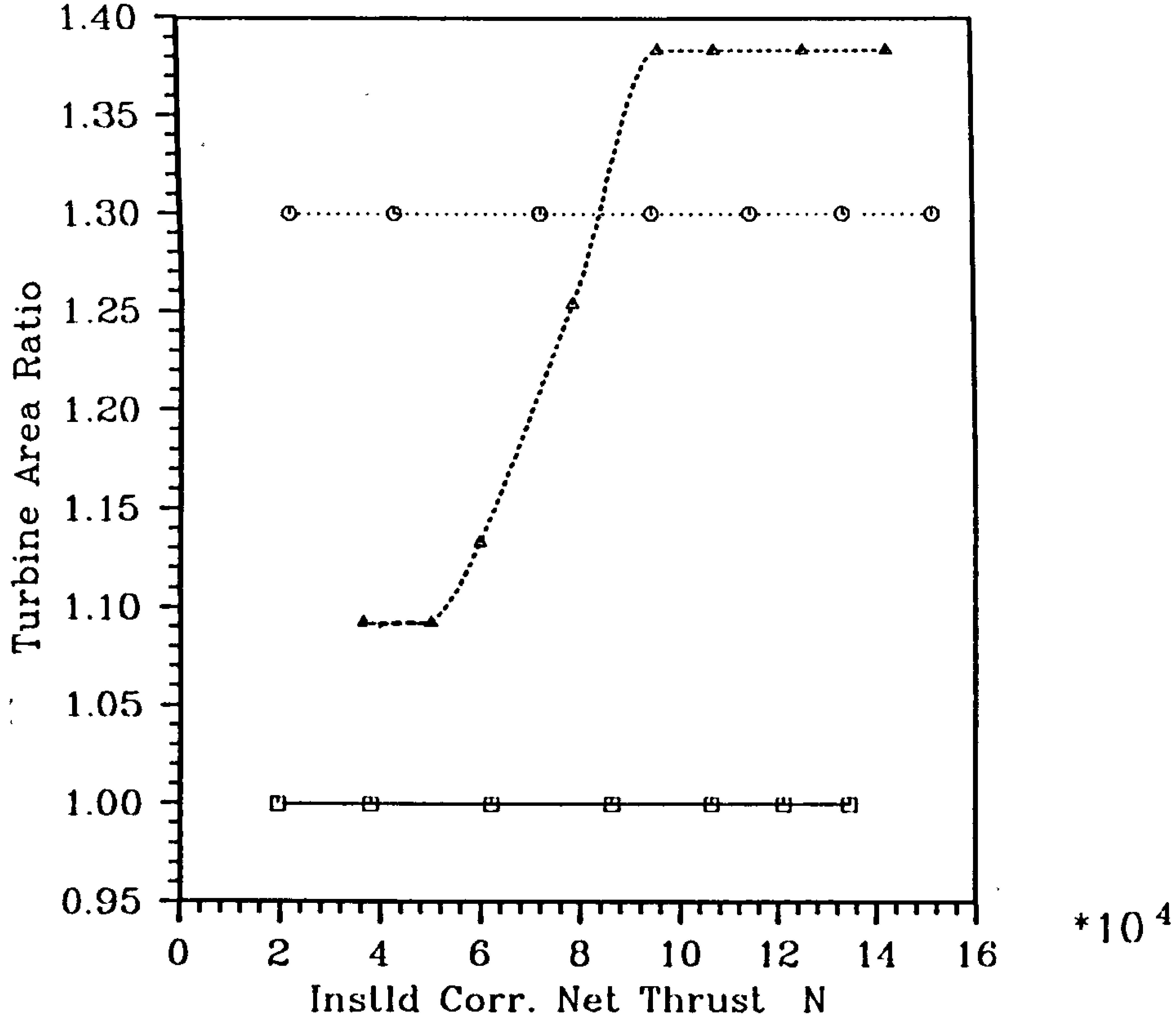



Fig. 8.3a HP Turbine Area Schedule

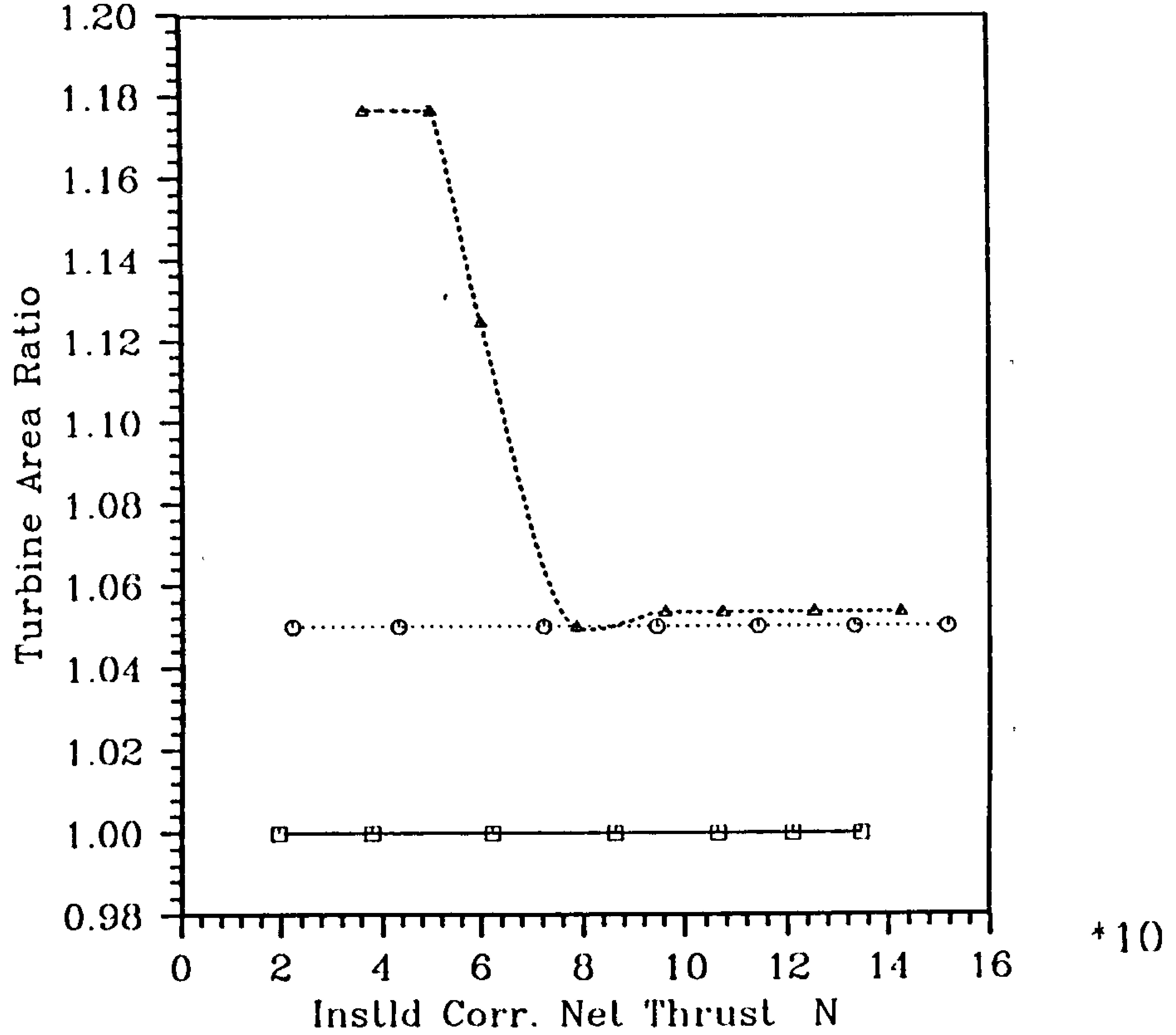



Fig. 8.3b LP Turbine Area Schedule

Alt (m) 9144.0
Mn 0.9

Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
Fixed Comp. Vanes/ Fixed Turbine Area /TJET
Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

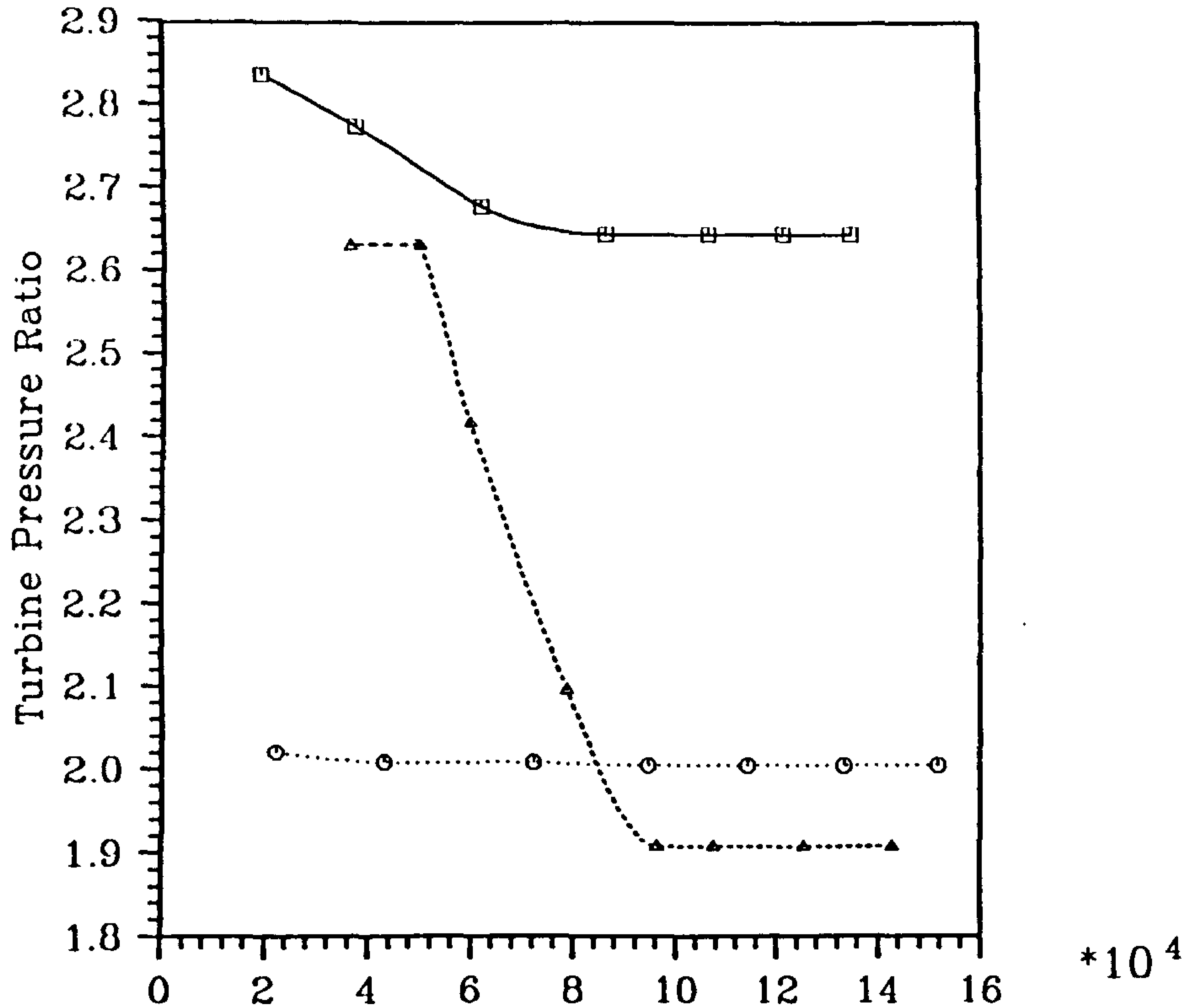



Fig. 8.4a HP Turbine Pr. Ratio

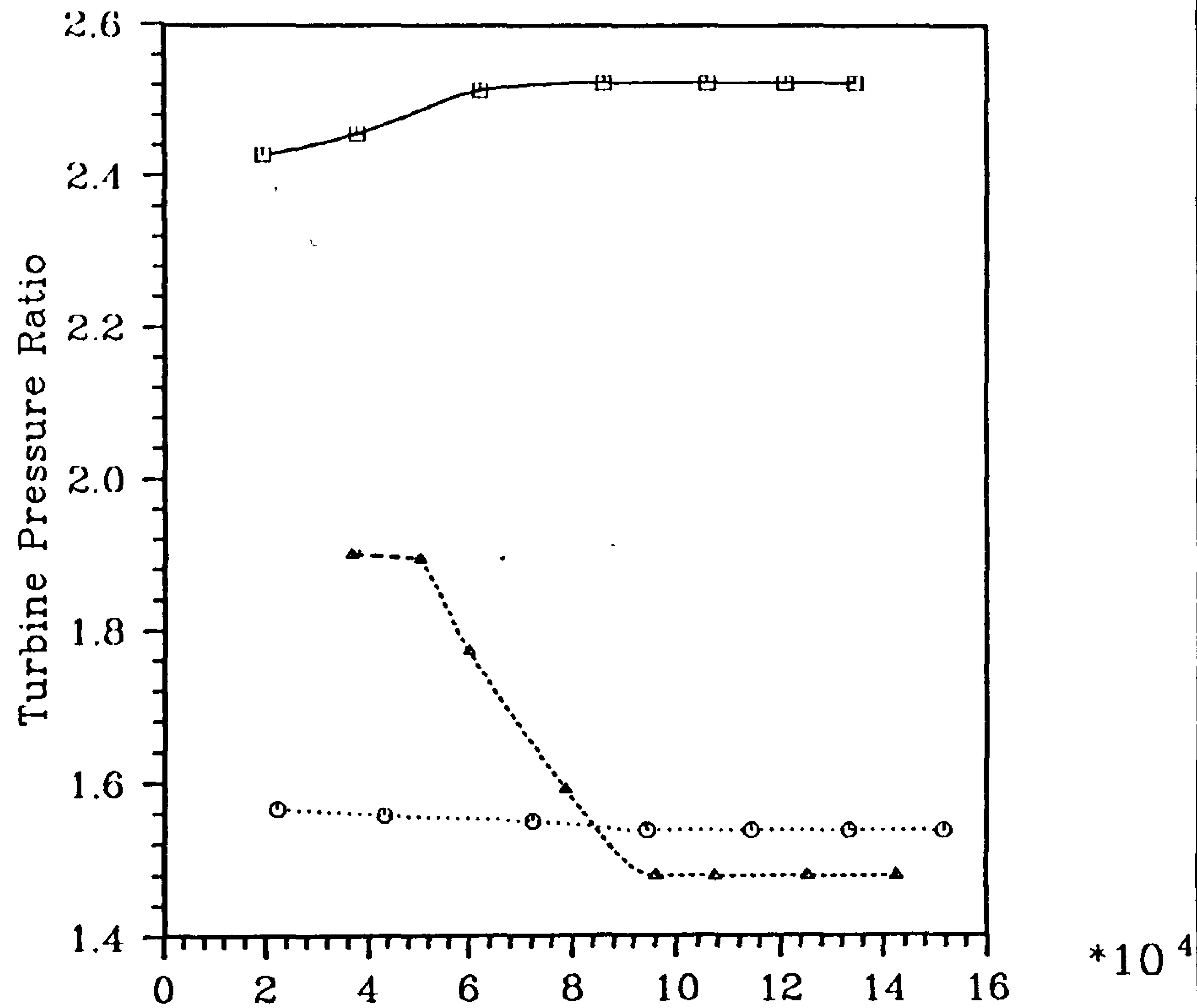



Fig. 8.4b LP Turbine Pr. Ratio

Alt (m) 9144.0
Mn 0.9

Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
Fixed Comp. Vanes/ Fixed Turbine Area /TJET
Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

reduced thrust. The two fixed geometry engines operate at essentially fixed pressure ratio across each turbine indicating that the propelling nozzle is operating choked.

While the turbines of the VAT engine vary area to give the required expansion ratio to maintain fixed compressor operating points at the specified thrust, the propelling nozzle opens up its throat area thereby reducing the back pressure so as to enable the turbines to operate at a higher pressure ratio as thrust is reduced. The core nozzle throat areas relative to the designed are plotted in Fig. 8.5a. Initially, the throat areas decrease as the afterburner fuel flow is reduced to reduce thrust. At intermediate power and below, the fixed geometry engines operate with fixed throat area and as explained above, the VAT engine's nozzle throat area increases as thrust is reduced. The nozzle exit areas decrease as the afterburning thrust is reduced and thereafter remain fixed for the fixed geometry engines while that of the VAT engine increases, in proportion, to permit full expansion of the gas, Fig. 8.5b. When constant LP compressor design point operation can no longer be maintained, the nozzle exit area decreases to maintain full expansion with fixed throat area.

The variation in core nozzle pressure ratio is shown in Fig. 8.6a. At intermediate power and below, both fixed geometry engines are operating under-expanded at constant pressure ratio since the ratio of throat to exit area remains fixed. The VAT engine operates at full expansion and with reduced pressure ratio for the reasons already explained above. When the reheat is lit, the pressure ratio across the nozzles of the fixed geometry engines increases as the nozzles are then operating at full expansion.

The Mach numbers at exit from the LP turbine are plotted in Fig. 8.6b. Ignoring the dummy value of 0.05 for the turbofan engine, the fixed geometry engines operate with almost constant Mach number while the VAT engine operates with increased Mach number as thrust is reduced. This is a crucial design consideration for LP turbines designed to drive fixed operating point compressors which must be given due attention. This is related to the phenomenon of "limiting blade loading" which was mentioned in an earlier chapter and will be discussed in the next chapter. It will suffice at this point to say that an exit Mach number not greater than 0.7 is allowed.

Both the uninstalled and installed sfc's are compared in Fig. 8.7, and as is clearly seen the fixed geometry engines exhibit the characteristics associated with their respective class of engines with respect to dry and reheated sfc's. For the uninstalled case, the VAT turbojet's dry sfc compares quite well with that of the turbofan but slightly loses its favourable reheated

## Performance Comparison Of Fixed And Variable Geometry Jet Engines

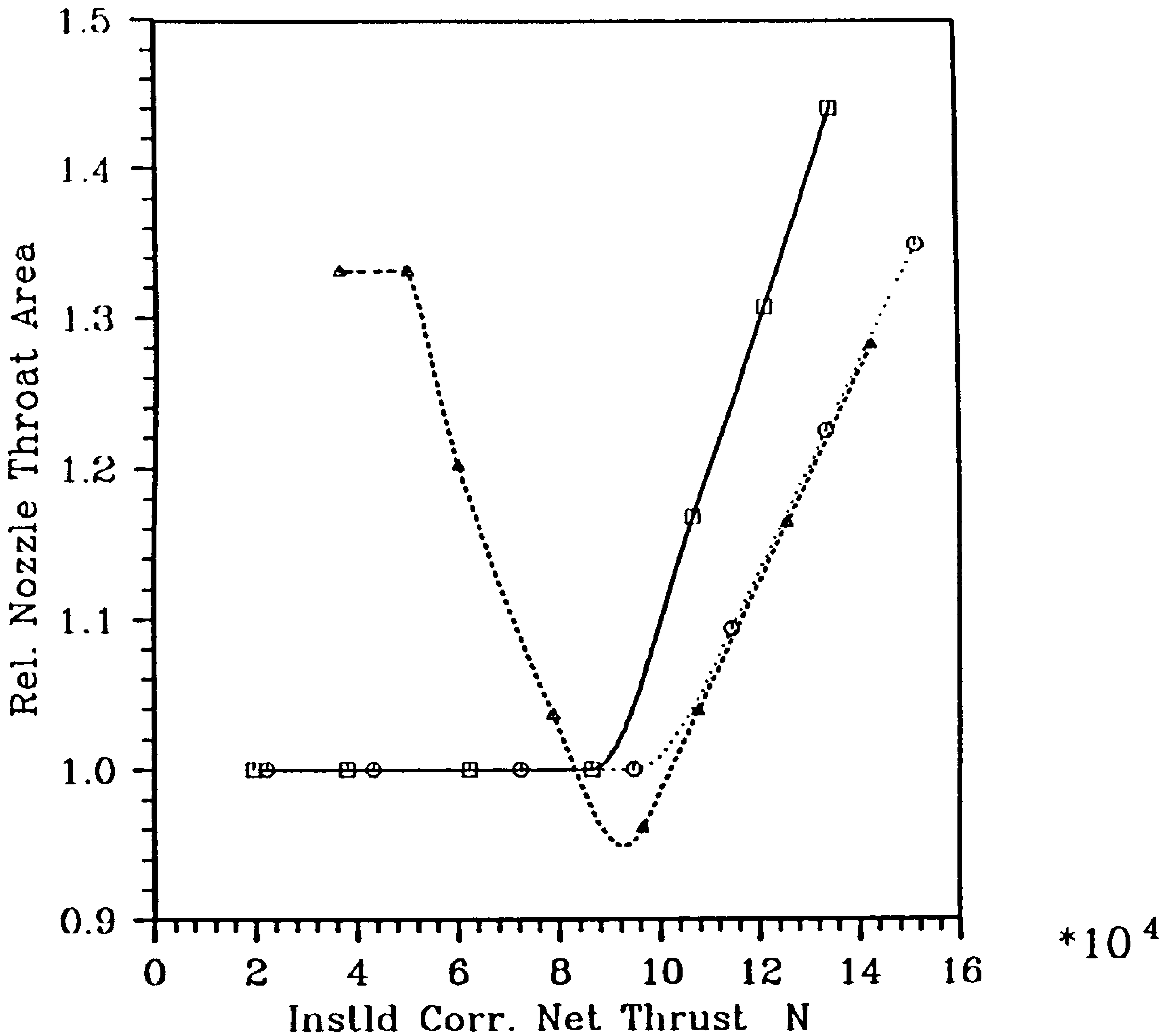



Fig. 8.5a Core Nozzle Throat Area Variation

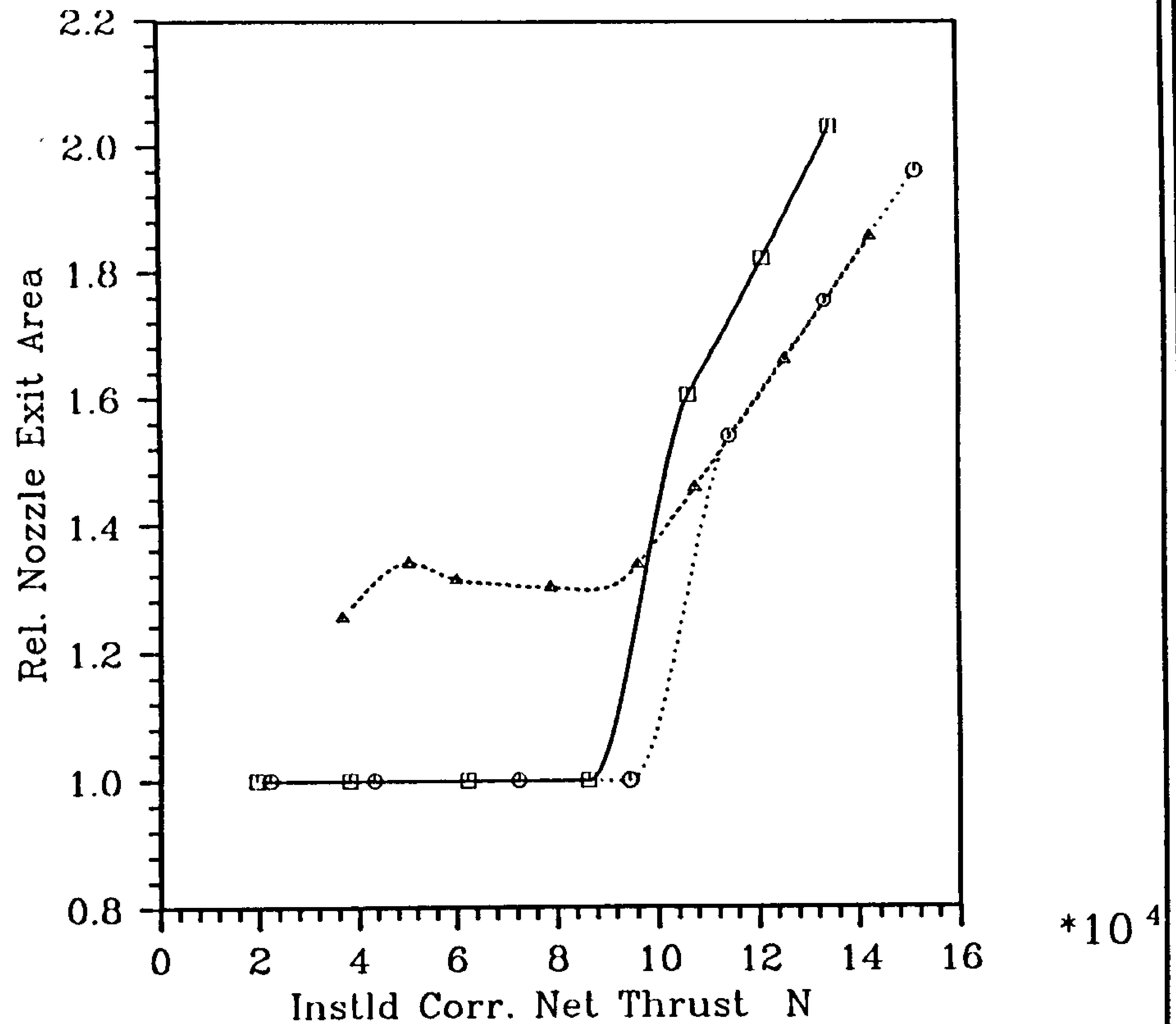



Fig. 8.5b Core Nozzle Exit Area Variation

Alt (m) 9144.0
Mn 0.9

- □—□ Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
- o····o Fixed Comp. Vanes/ Fixed Turbine Area /TJET
- ▲----▲ Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

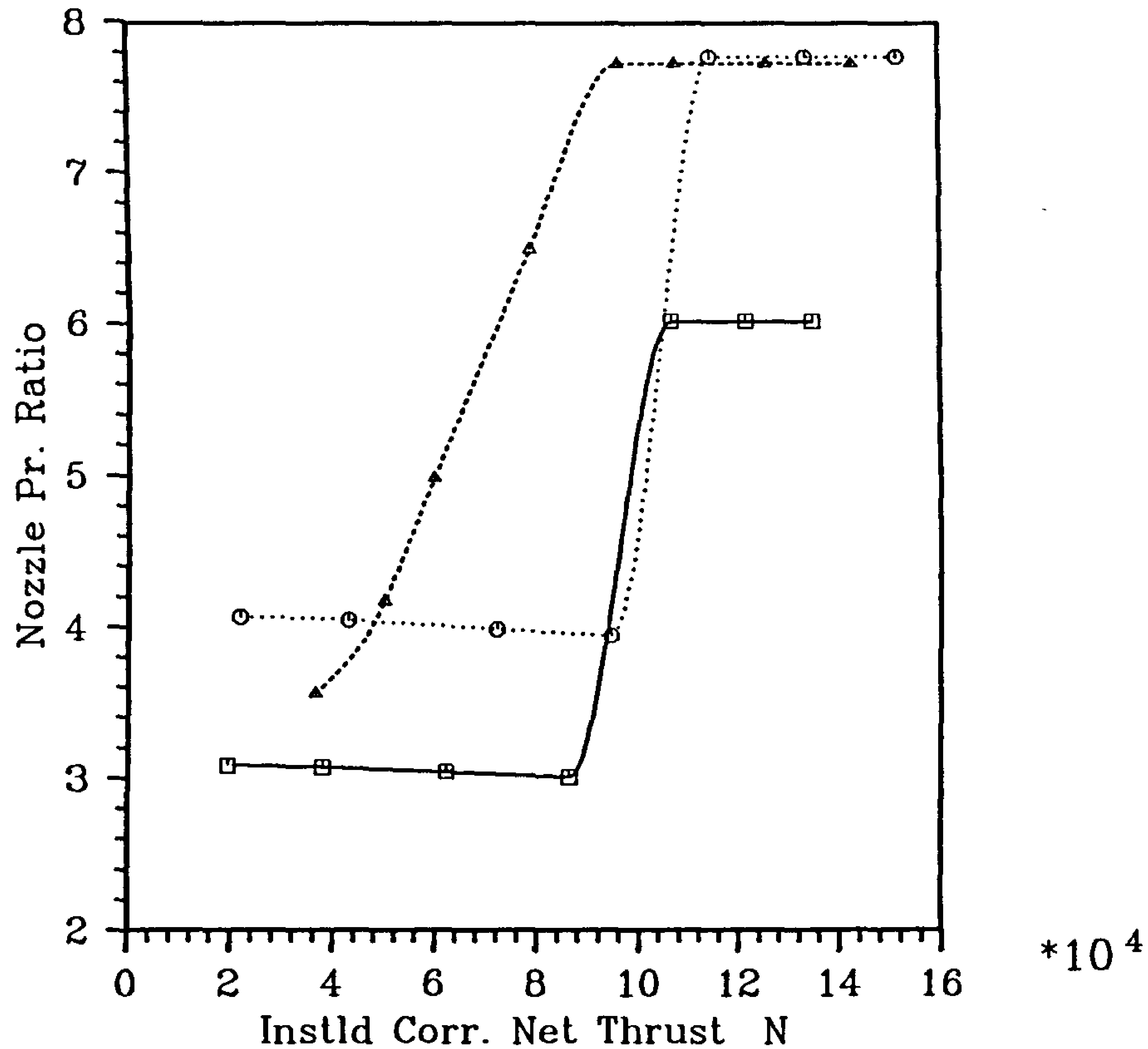



Fig. 8.6a Core Nozzle Pressure Ratio Variation

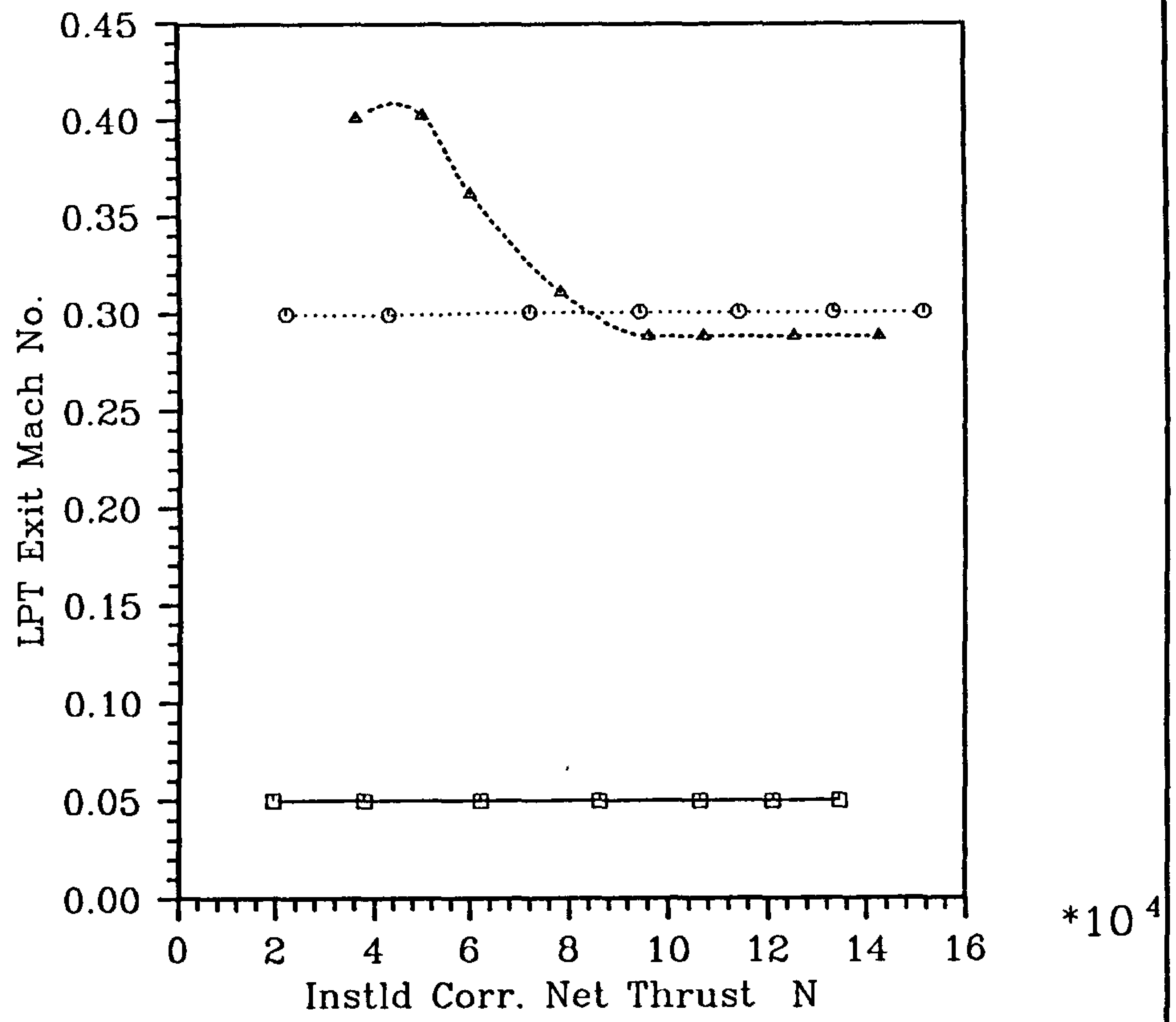



Fig. 8.6b LPT Exit Mach No. Variation

Alt (m) 9144.0
Mn 0.9

- Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
- Fixed Comp. Vanes/ Fixed Turbine Area /TJET
- Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

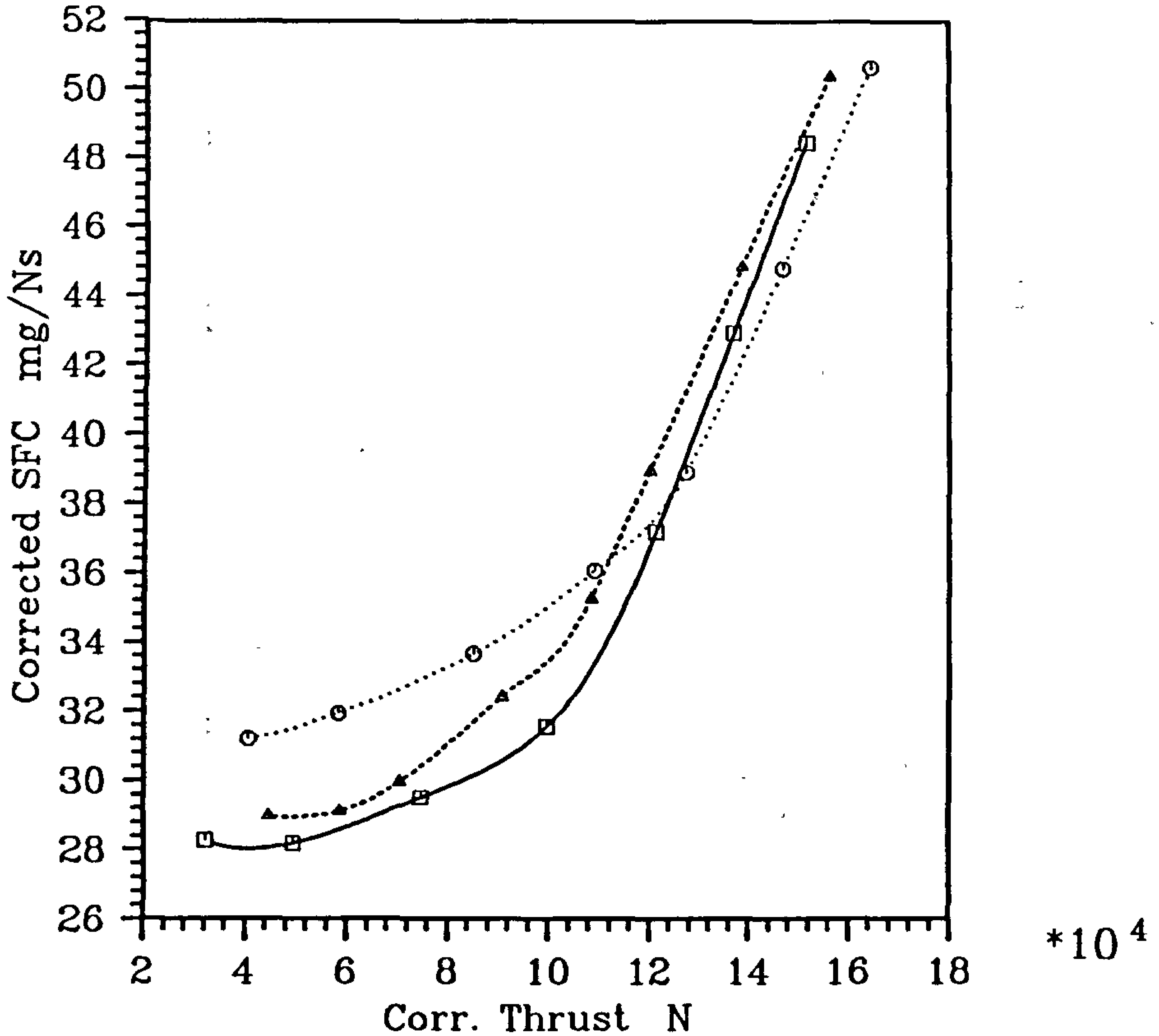



Fig. 8.7a Fuel Flow Requirement For Unit Thrust

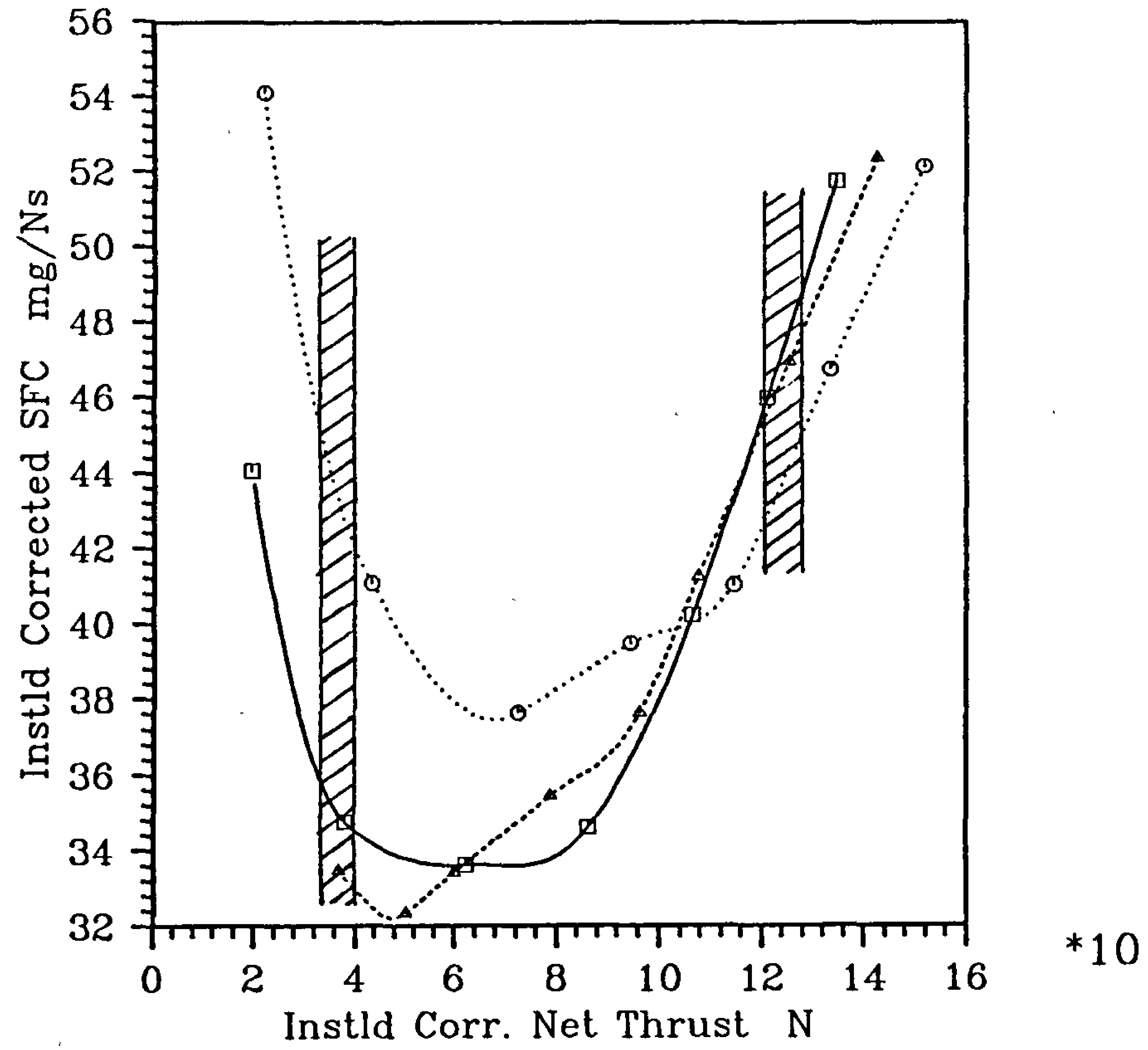



Fig. 8.7b Fuel Flow Requirement For Unit Thrust

Alt (m) 9144.0
Mn 0.9

- Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
- Fixed Comp. Vanes/ Fixed Turbine Area /TJET
- Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

performance, Fig 8.7a. However, its installed performance is even superior to that of the turbofan at low power settings while at the same time improving on its reheated performance, Fig. 8.7b.

The reason for a poorer reheated performance for the VAT engine is that the two fixed geometry engines were able to run at maximum TIT at this flight condition with a resultant higher non-dimensional speed than that of the variable geometry engine. Hence, the fixed geometry engines were passing a corrected mass flow higher than the designed, and with the fixed geometry turbojet having similar gas properties in the nozzle as the variable geometry turbojet, and with a much higher augmentation ratio for the turbofan, the thrust developed by these engines was much higher than that of the variable geometry engine. However, the ability of the VAT engine to pass a higher airflow reduced both the inlet and aft-end installation losses resulting in improved installed performance. It was observed that for flight conditions where the maximum thrust developed by the fixed geometry engines was spool speed limited, the VAT engine gave competitive uninstalled performance by being able to run at the maximum TIT.

A typical range of thrust at both the cruise and combat thrust settings is shown shaded in Fig. 8.7. For the installed case, the savings in cruise fuel consumption of the VAT engine over that of the turbofan and fixed area turbojet are about 1.6 and 9.9 percent respectively, while the figures for the combat point are 0.3 and -2.9 percent.

Supersonic Performance

The performances of the three engines at the specified flight conditions are compared in Figs 8.8 to 8.11. Unlike the subsonic flight conditions discussed above, both turbojet engines maintain the high Mach characteristics, with the VAT engine giving a superior performance both uninstalled and installed, Fig. 8.8. Over a typical range of power settings for supersonic cruise of the turbofan engine, the fixed geometry turbojet gives a fuel saving of about 4.7 percent over that consumed by the turbofan when installed, while the VAT engine gives a saving of 11.9 percent. Both turbojets are operating at part dry power over this thrust range.

The thrust of the turbofan is HP spool speed limited and therefore the engine cannot be run at maximum TIT, Figs 8.9a,b. Because of the higher temperature at compressor face, both rotors of the VAT engine run at a higher rotational speed compared to the designed to maintain non-dimensional speed fixed. Over the thrust range considered, there are no constraints on the VAT engine

## Performance Comparison Of Fixed And Variable Geometry Jet Engines

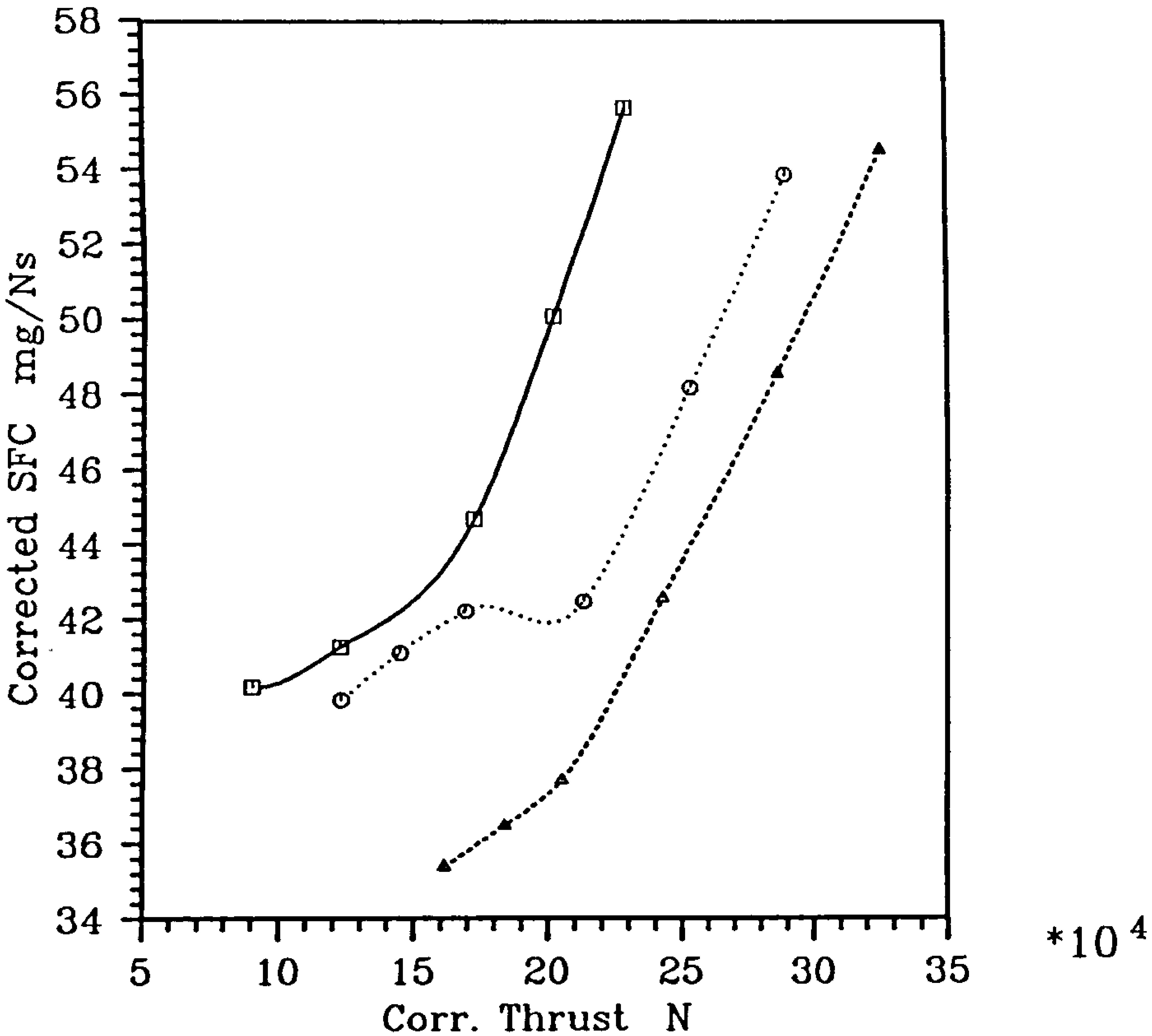



Fig. 8.8a Fuel Flow Requirement For Unit Thrust

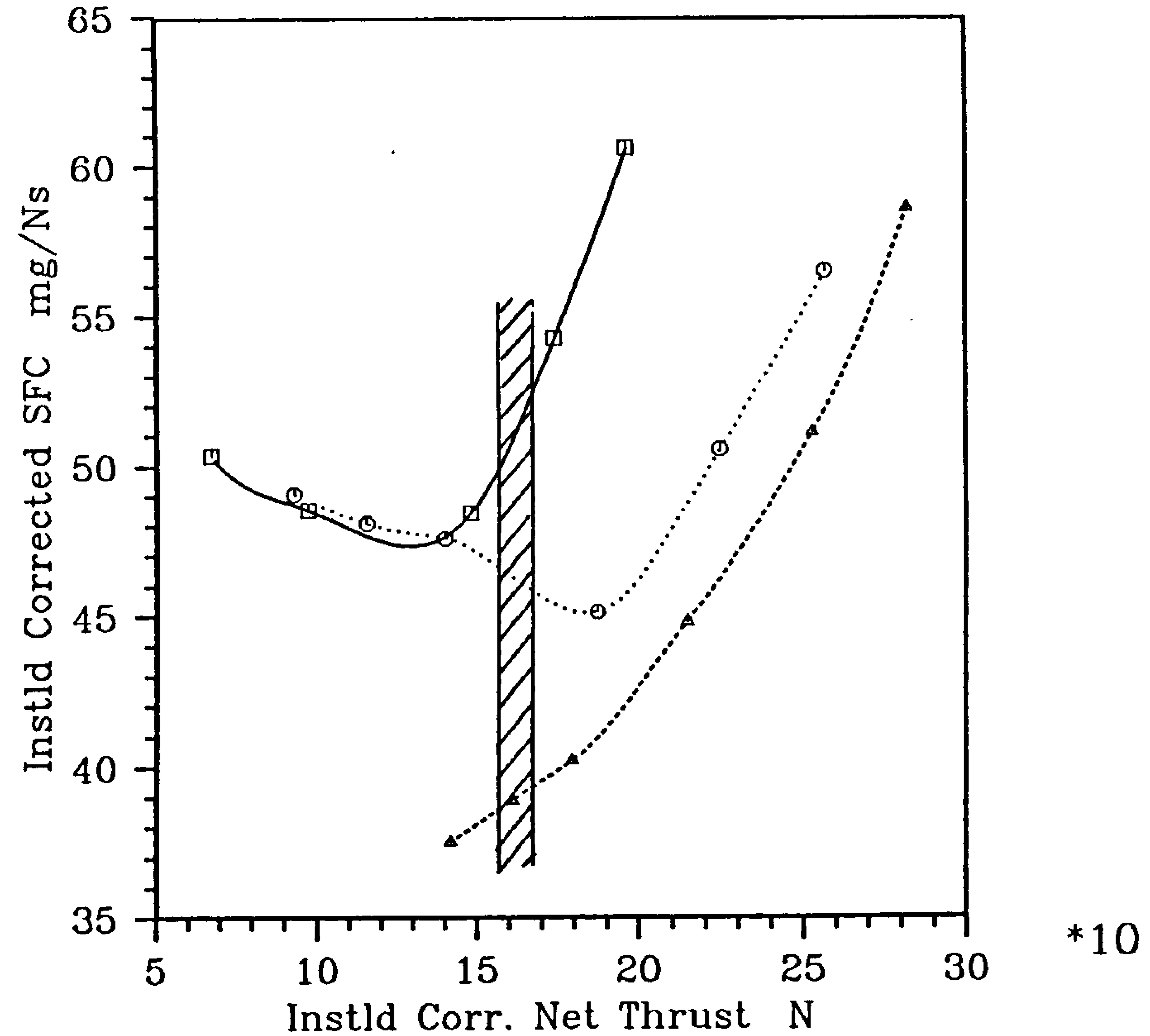



Fig. 8.8b Fuel Flow Requirement For Unit Thrust

Alt (m) 15240.0
Mn 1.6

- Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
- Fixed Comp. Vanes/ Fixed Turbine Area /TJET
- Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

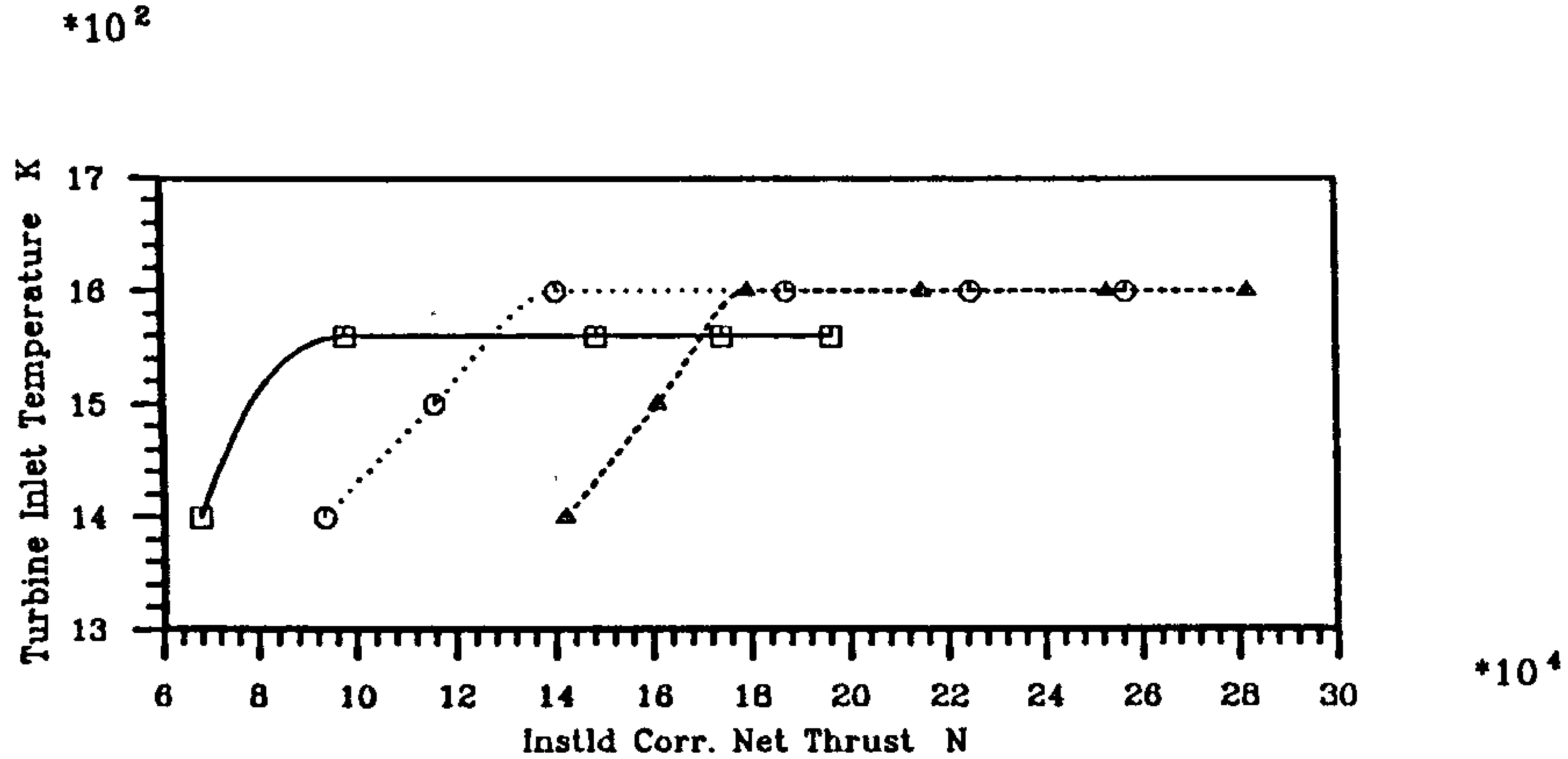



Fig. 8.9a Temp Schedule for Thrust Modulation

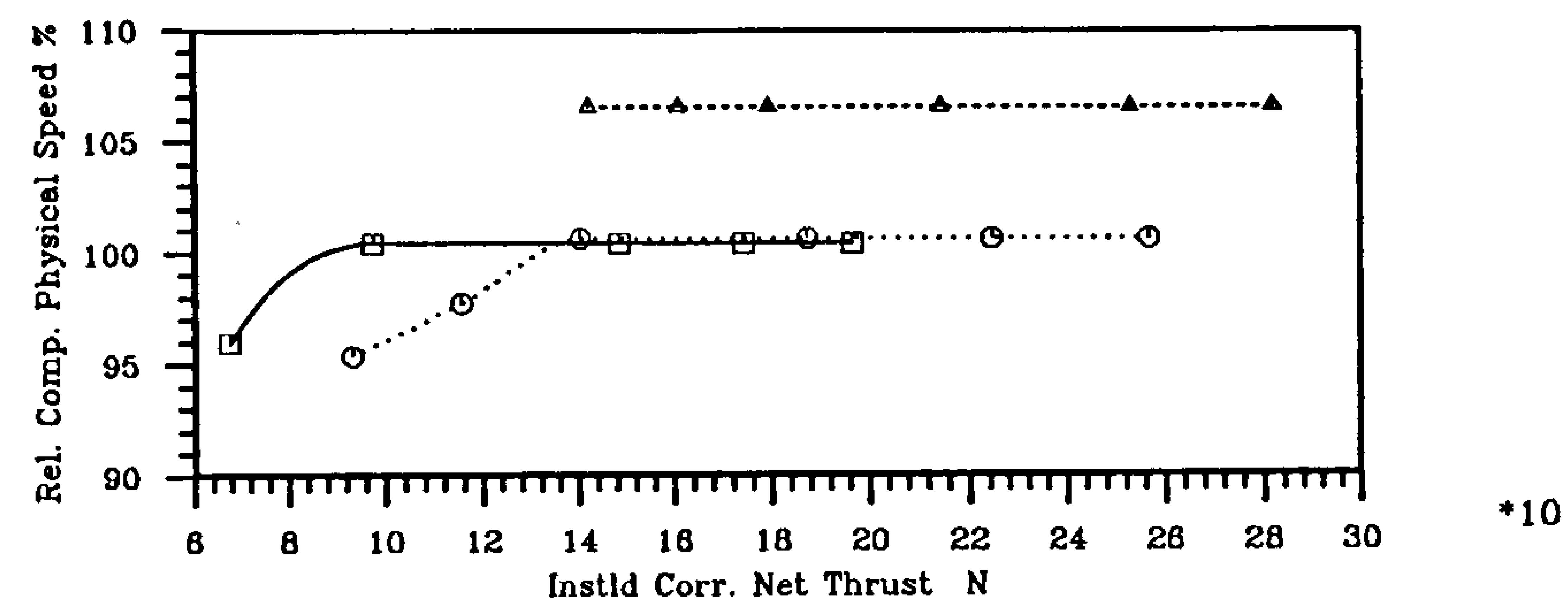



Fig. 8.9b HP Compressor Physical Speed Schedule

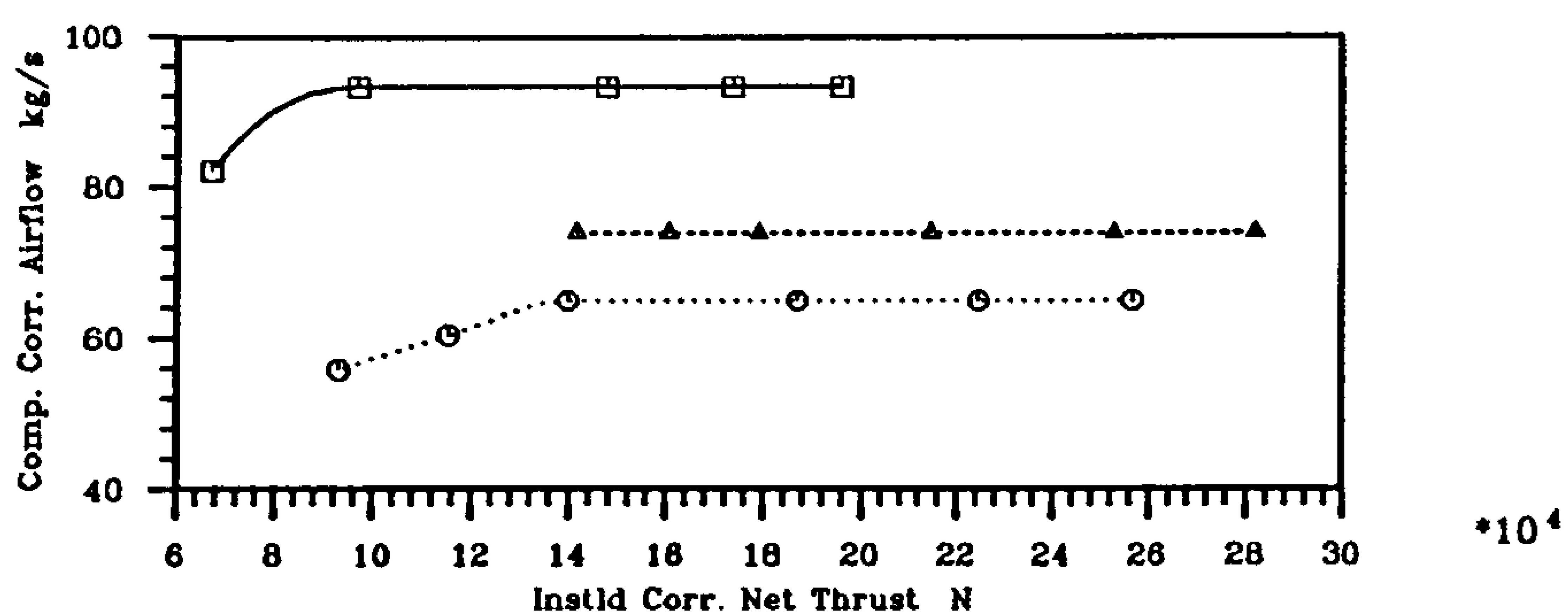



Fig. 8.9c LP Compressor Airflow Requirement

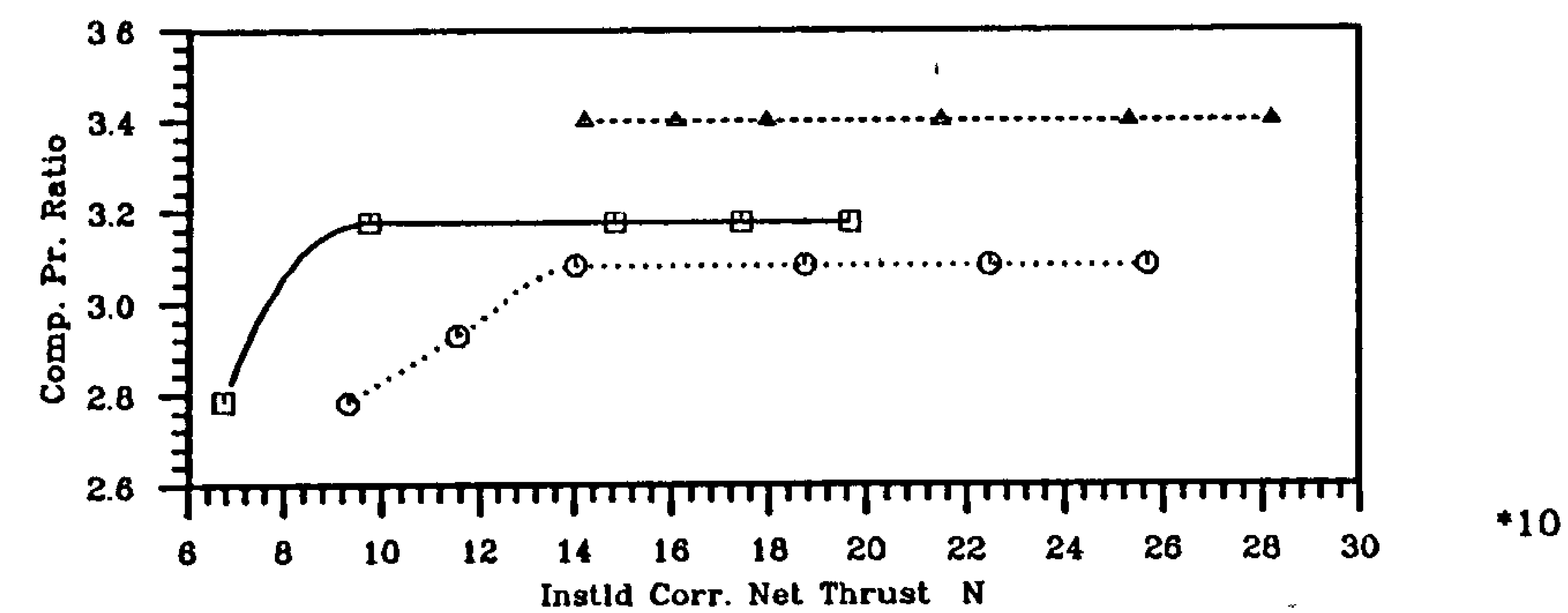



Fig. 8.9d LP Compressor Pr. Ratio

Alt (m) 15240.0
Mn 1.6

- Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
- Fixed Comp. Vanes/ Fixed Turbine Area /TJET
- Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

since both turbine areas continuously vary when the reheat is off, Fig. 8.10a,b, and as a result, both mass flow and pressure ratio remain fixed for both compressors. Figs 8.9c,d give the plots for the LP compressor. Since both fixed geometry engines are operating at low values of non-dimensional speed, their pressure ratio levels are much lower than that of the VAT engine.

The trends in turbine pressure ratios are similar to those observed for the subsonic flight conditions, however, the VAT engine operates with higher turbine pressure ratios than the other turbojet due to the increased compressor work required to maintain compressor operating points constant at a higher inlet temperature, Figs 8.10c,d. The nozzle throat area variations are shown in Fig. 8.11a and the trends are similar to those of the subsonic case. The trends in nozzle exit area variation, Fig. 8.11b, are however different. With the afterburner on, all the engines operate under-expanded with the exit area fixed at the maximum. At dry operation, the VAT engine continues to operate at the maximum exit area implying that the performance as shown in Fig. 8.8b can be further improved for the VAT engine as installation losses will be much smaller than those for the fixed geometry engines. The aft-end drag forces are not included in Fig. 8.8b as the available data does not include flight at this Mach number.

The core nozzle pressure ratio variation is shown in Fig. 8.11c, where it is seen that this is fixed for the constant throat to exit area operation. The LP turbine exit Mach number variations are given in Fig. 8.11d, and once again, limit loading of the LP turbine rotors is not encountered.

Performance at Acceleration

The performances at acceleration from Mach 0.6 to 2.2 at an altitude of 11 km are compared in Figs 8.12 to 8.16. Acceleration takes place at maximum afterburning thrust and it is assumed that maximum thrust is limited by either TIT or rotational speed.

The transonic and supersonic performances of the turbojets in particular are completely distinct. At supersonic speeds the thrust of the turbojets exceeds that of the turbofan and the discrepancy gets higher with higher flight speeds. At Mach numbers of up to 1.2 the fixed geometry turbojet gives a higher thrust than the VAT engine because of the following reasons. Over this speed range, the fixed geometry turbojet is operating at LP spool speed limit, Fig. 8.13a, and hence at a higher corrected speed, Fig. 8.13b. The HP spool is also operating at a higher speed, Figs 8.13c,d, and therefore the engine mass flow is higher. Since both engines are operating at their maximum TIT, Fig. 8.14a, the fixed geometry turbojet gives a higher thrust.

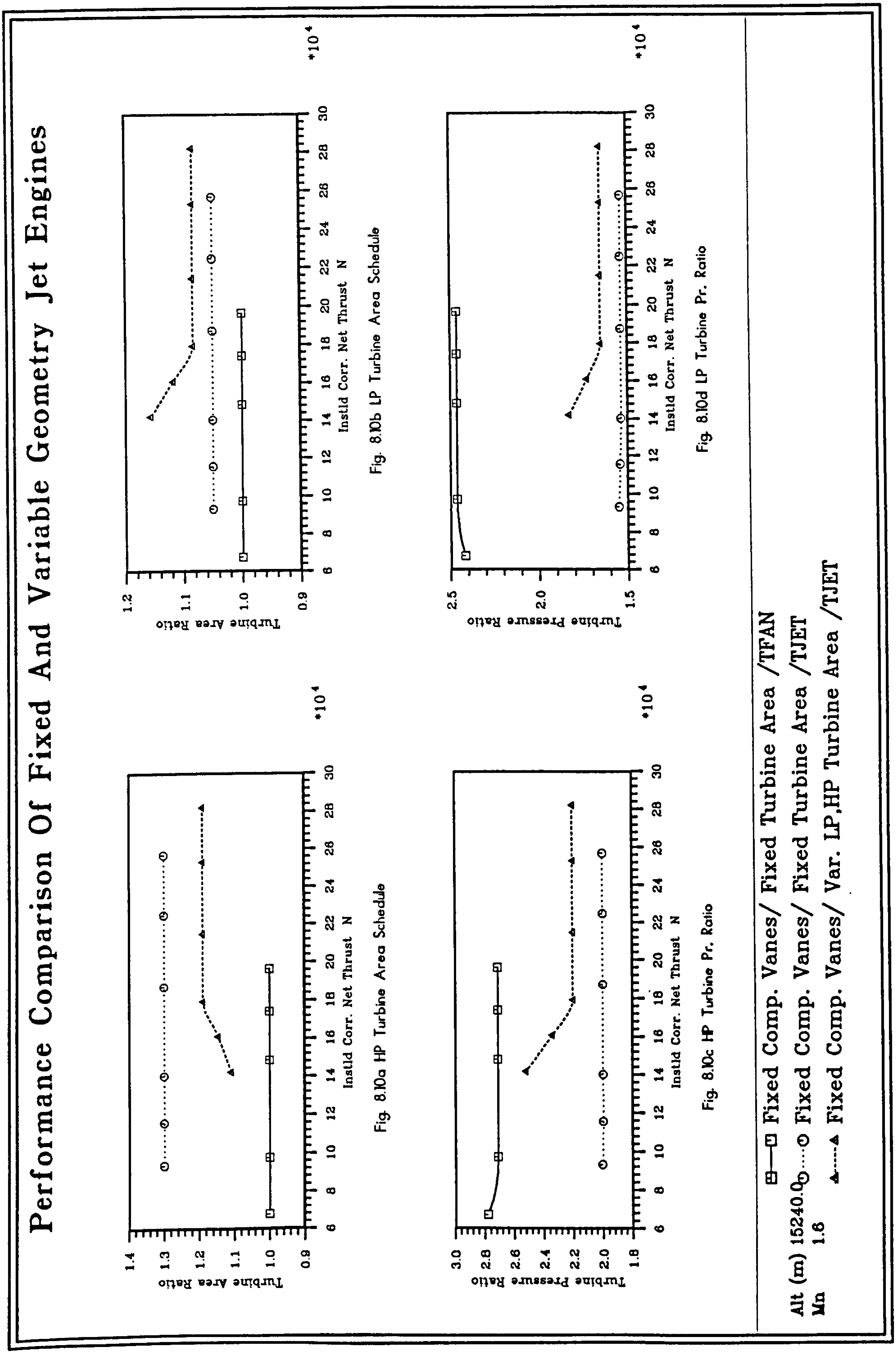



Fig. 8.10a HP Turbine Area Schedule

Fig. 8.10b LP Turbine Area Schedule

Fig. 8.10c HP Turbine Pr. Ratio

Fig. 8.10d LP Turbine Pr. Ratio

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

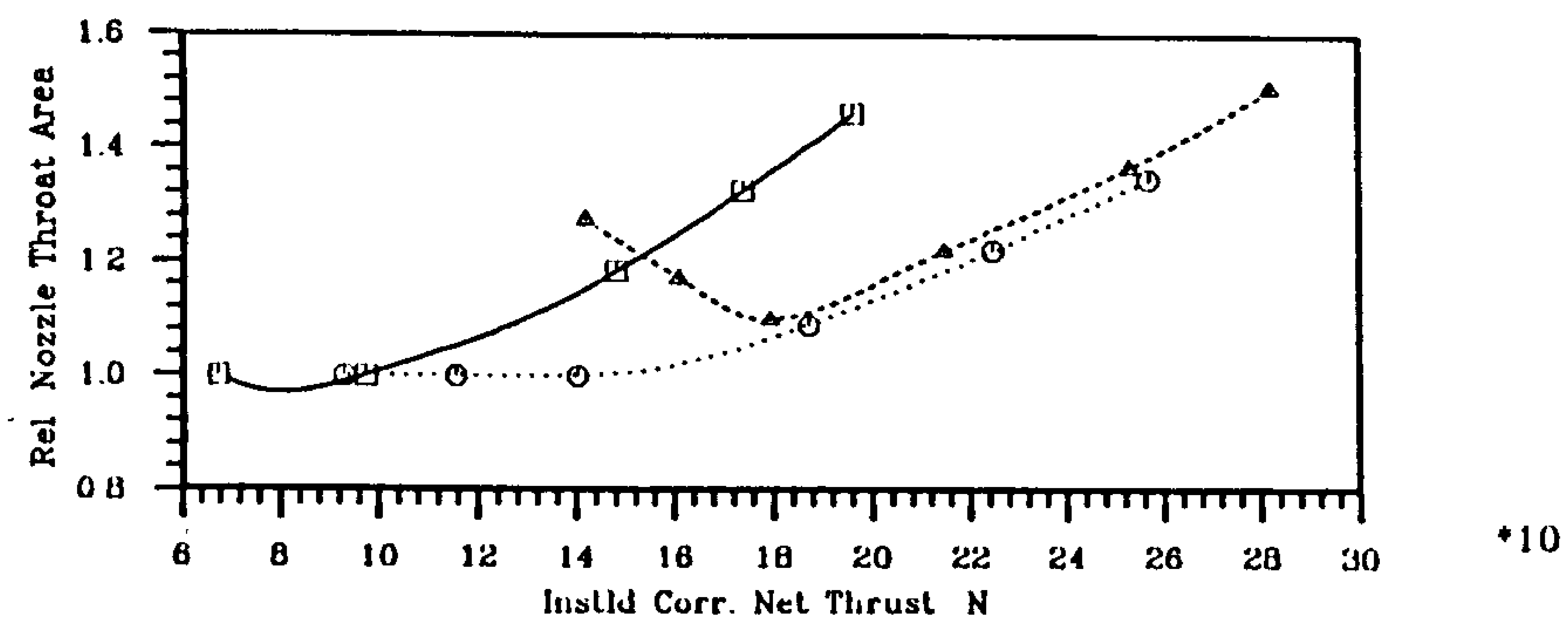



Fig 8.11a Core Nozzle Throat Area Variation

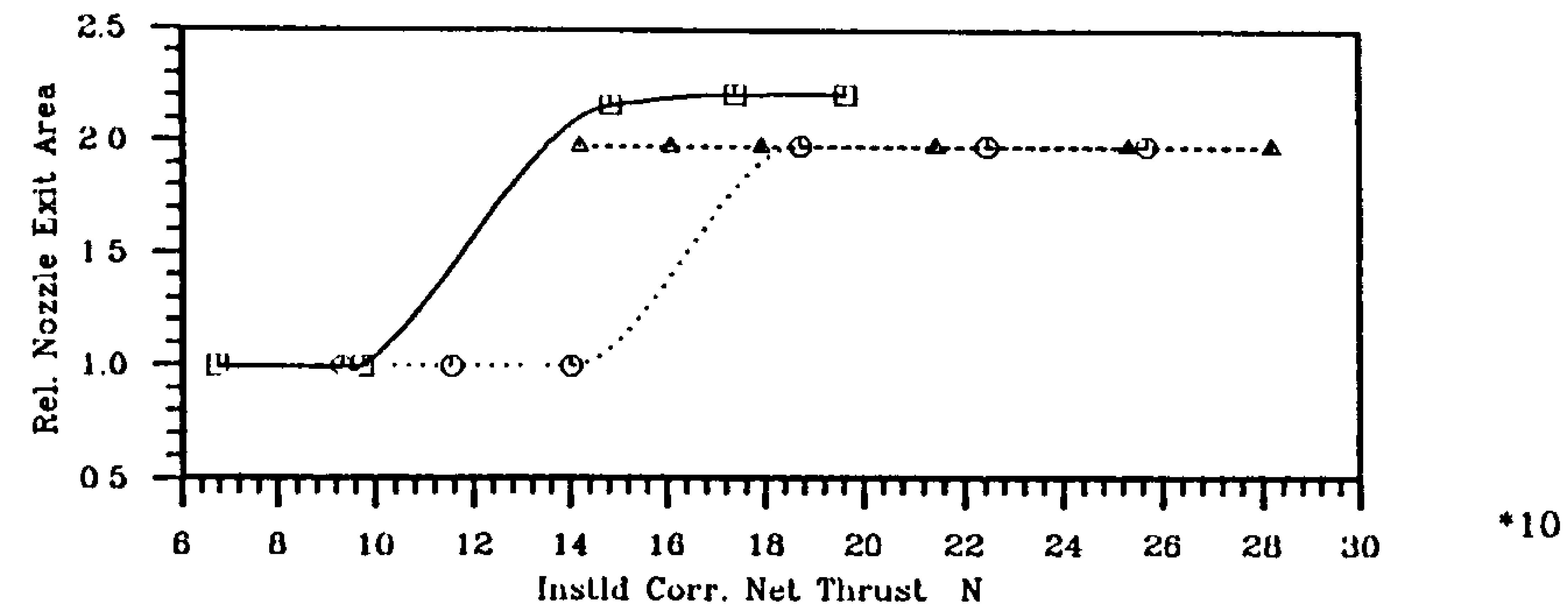



Fig. 8.11b Core Nozzle Exit Area Variation

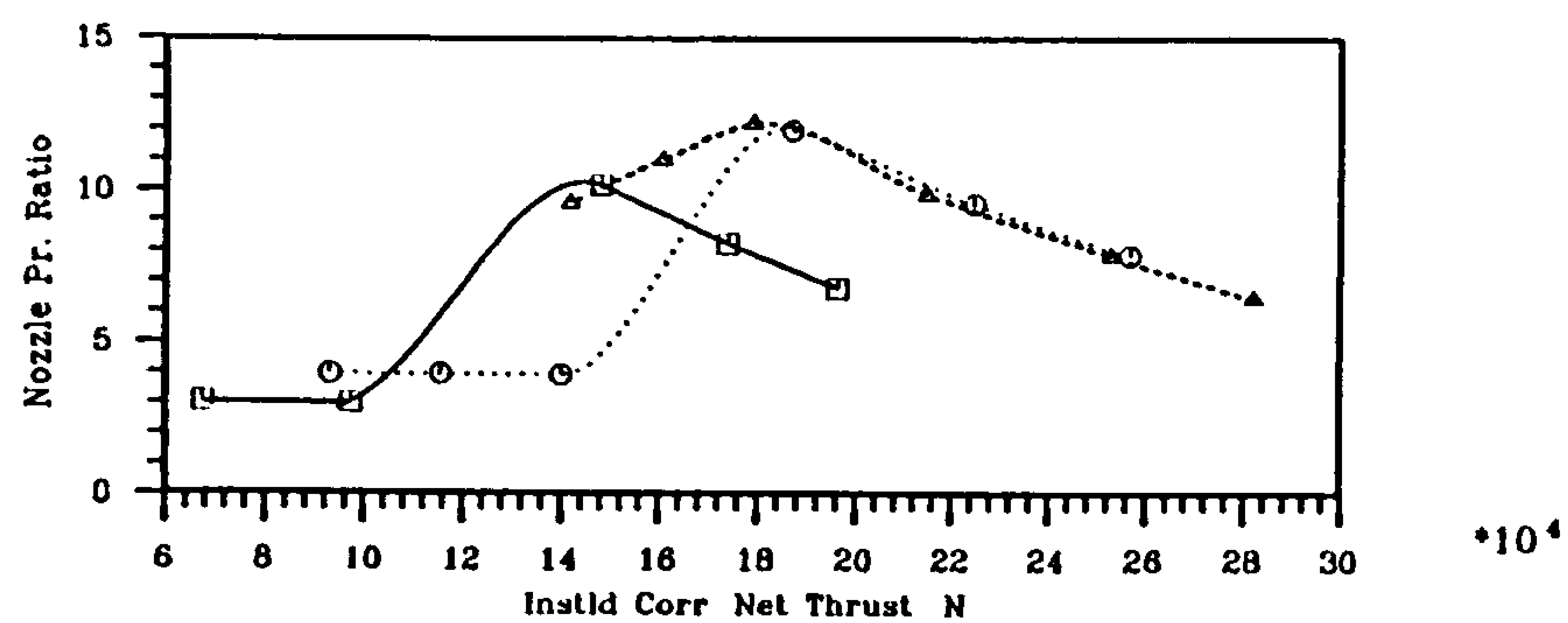



Fig. 8.11c Core Nozzle Pressure Ratio Variation

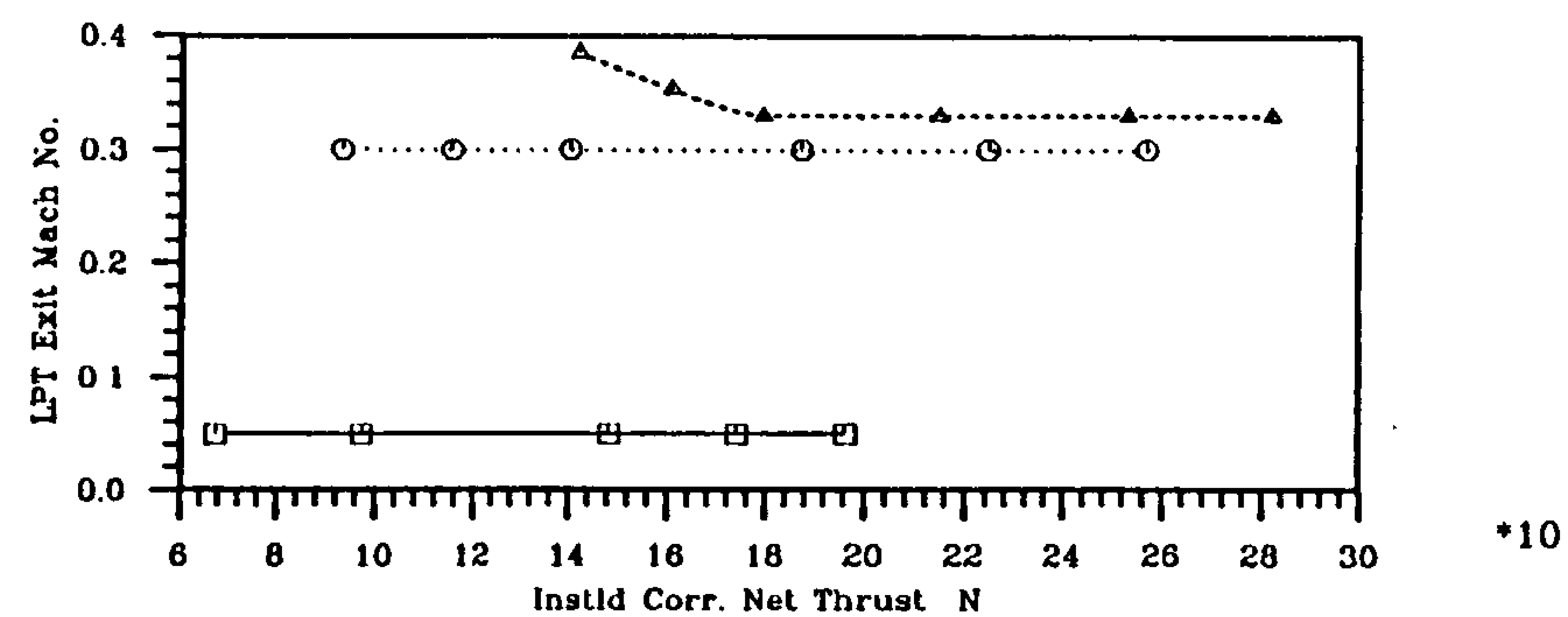



Fig. 8.11d LPT Exit Mach No. Variation

Alt (m) 15240.0
Mn 1.6

- Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
- Fixed Comp. Vanes/ Fixed Turbine Area /TJET
- Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

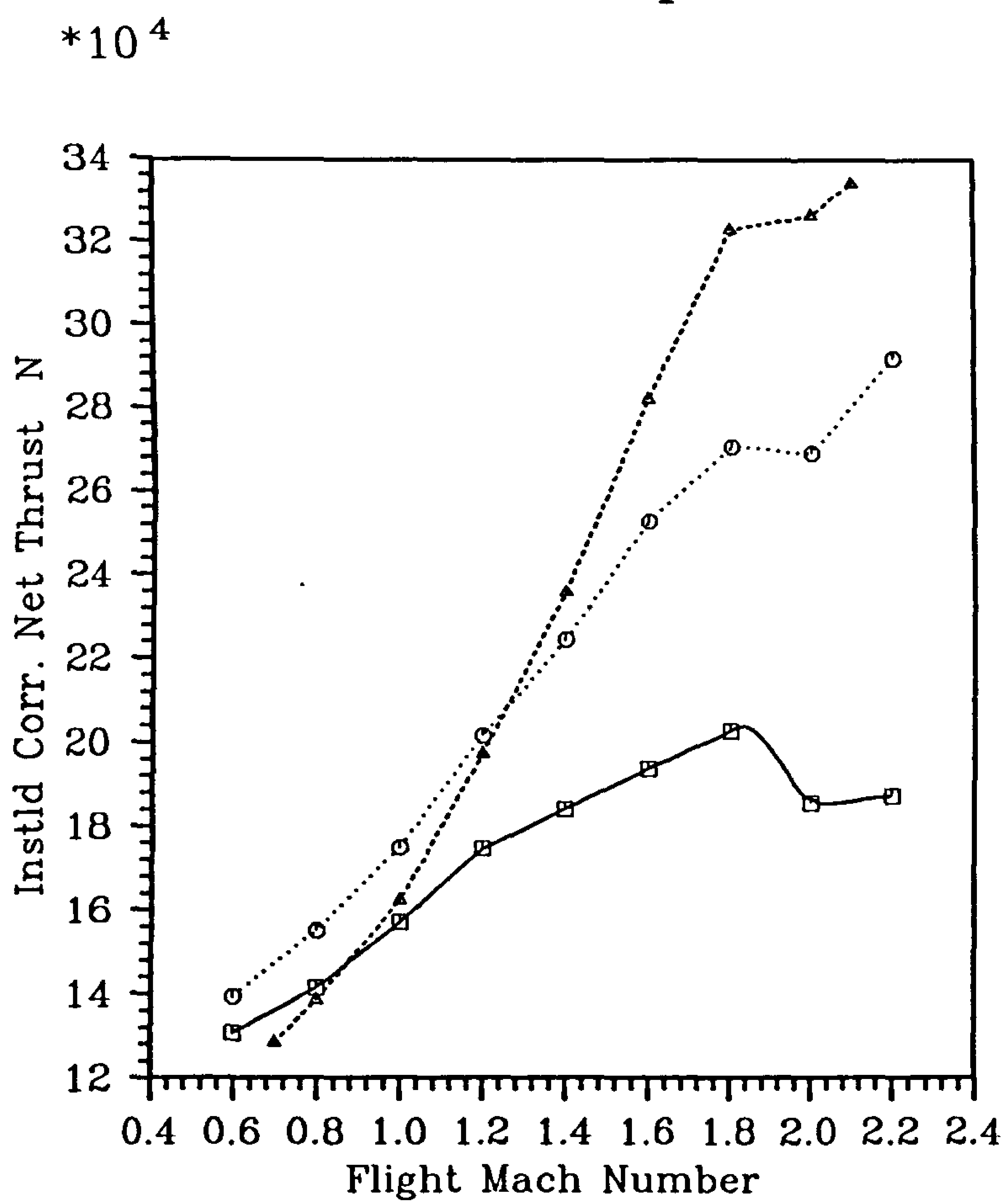



Fig. 8.12a Thrust Requirement at Acceleration

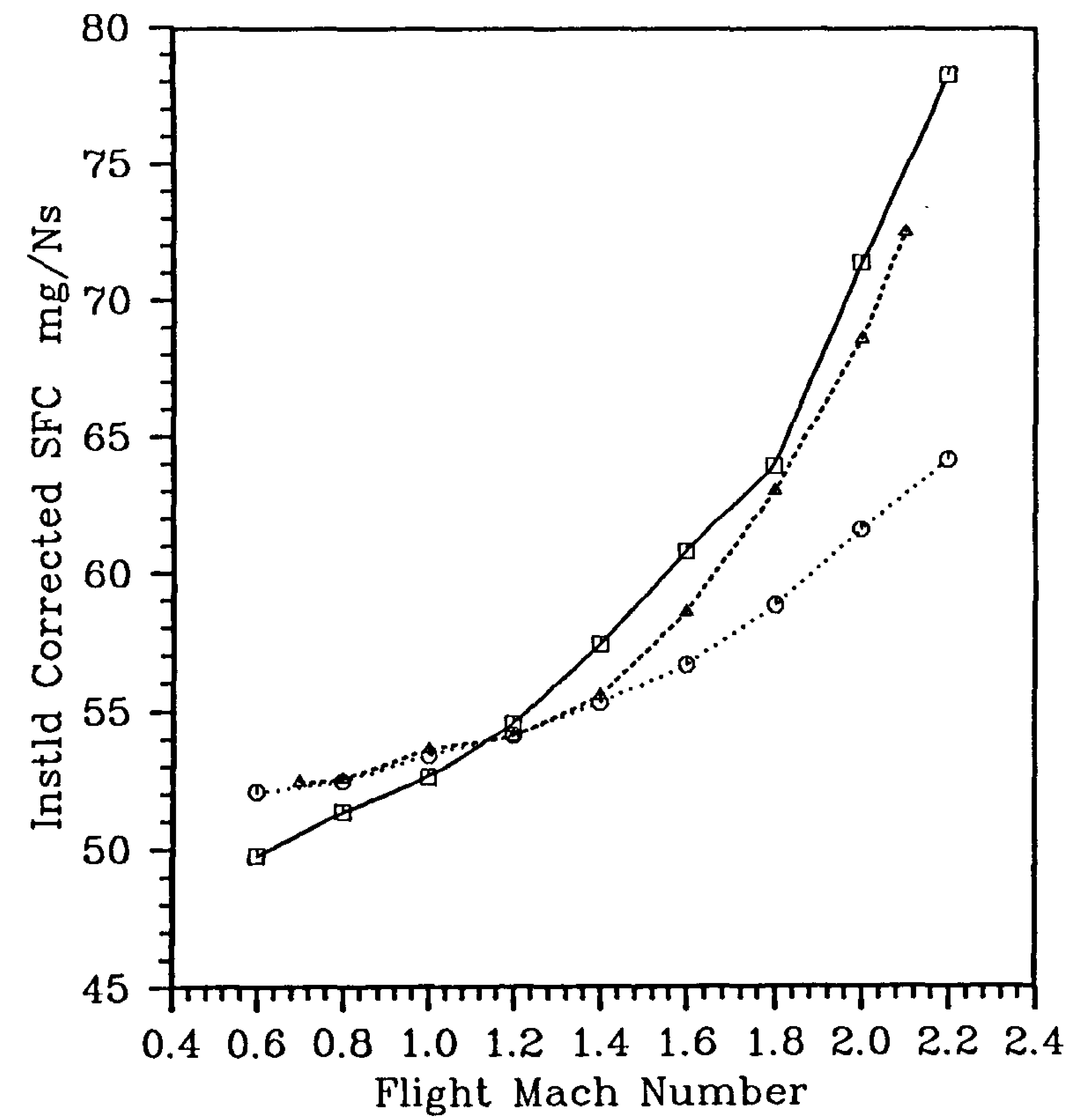



Fig. 8.12b Fuel Flow Requirement For Unit Thrust at Acceleration

Alt (m) 11000.0

- Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
- Fixed Comp. Vanes/ Fixed Turbine Area /TJET
- Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

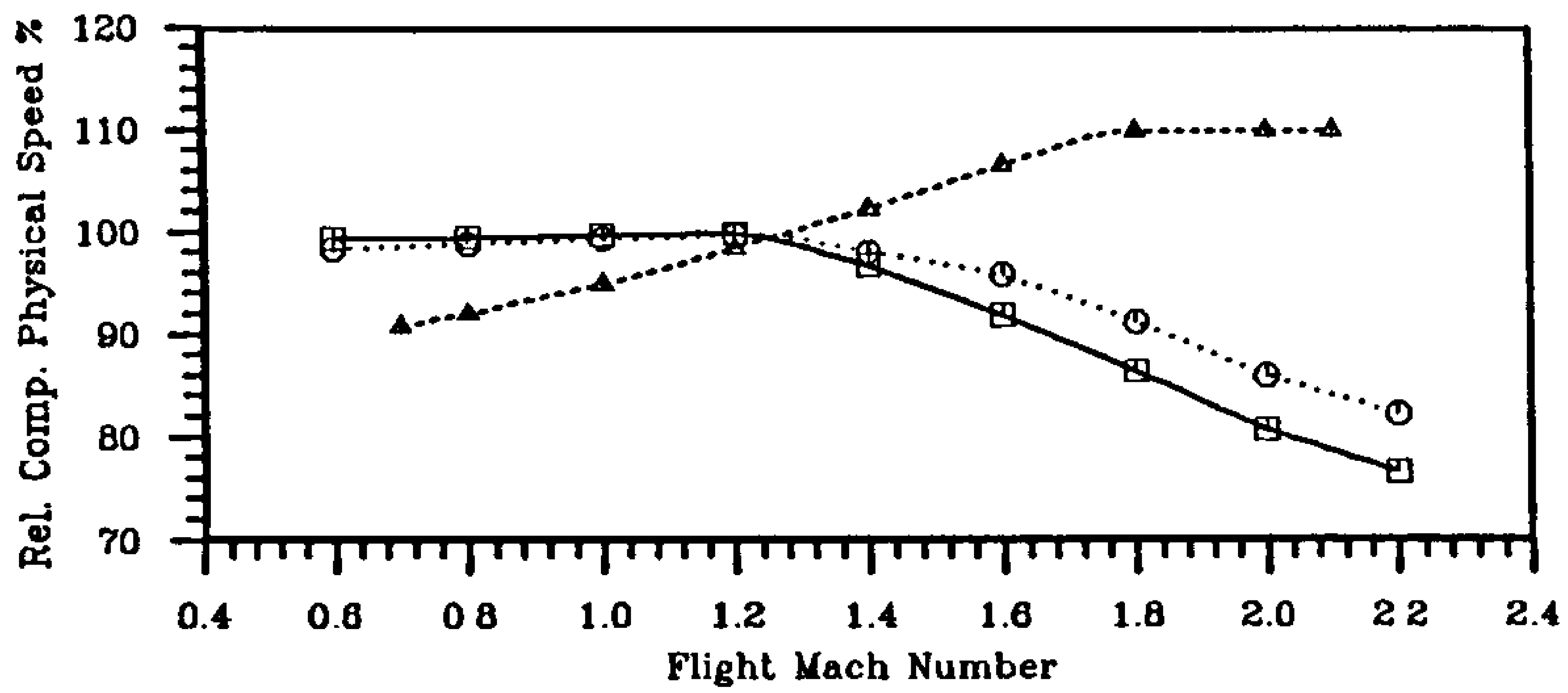



Fig. 8.13a LP Compressor Physical Speed Schedule at Acceleration

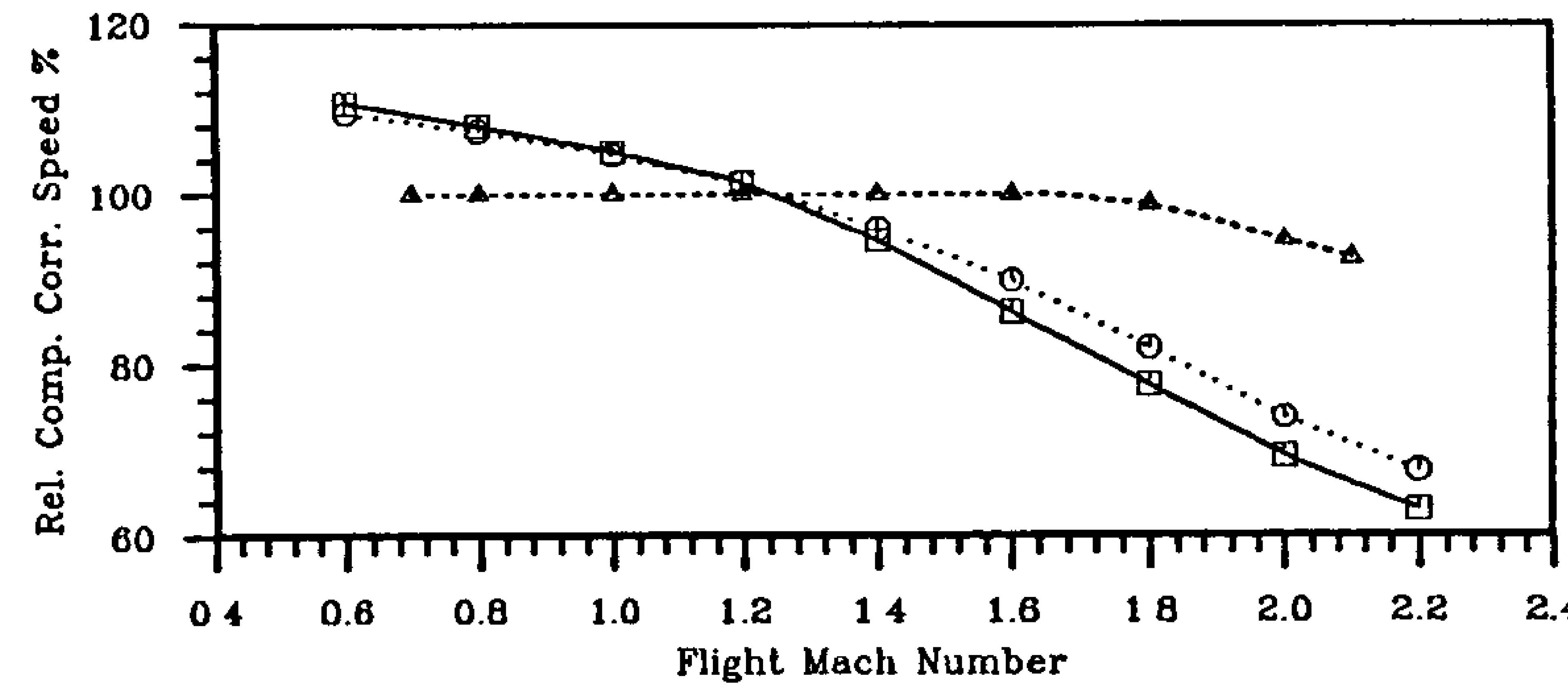



Fig. 8.13b LP Compressor Corrected Speed Schedule at Acceleration

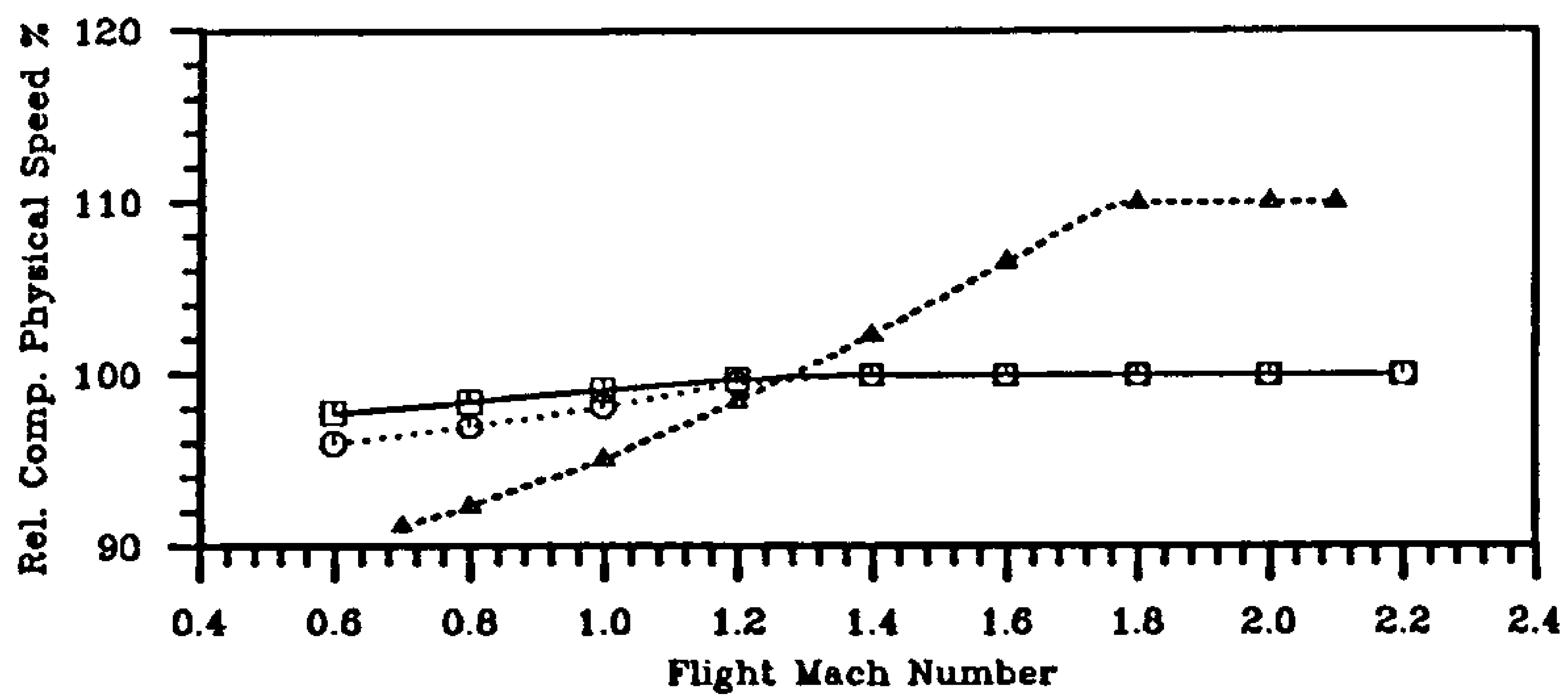



Fig. 8.13c HP Compressor Physical Speed Schedule at Acceleration

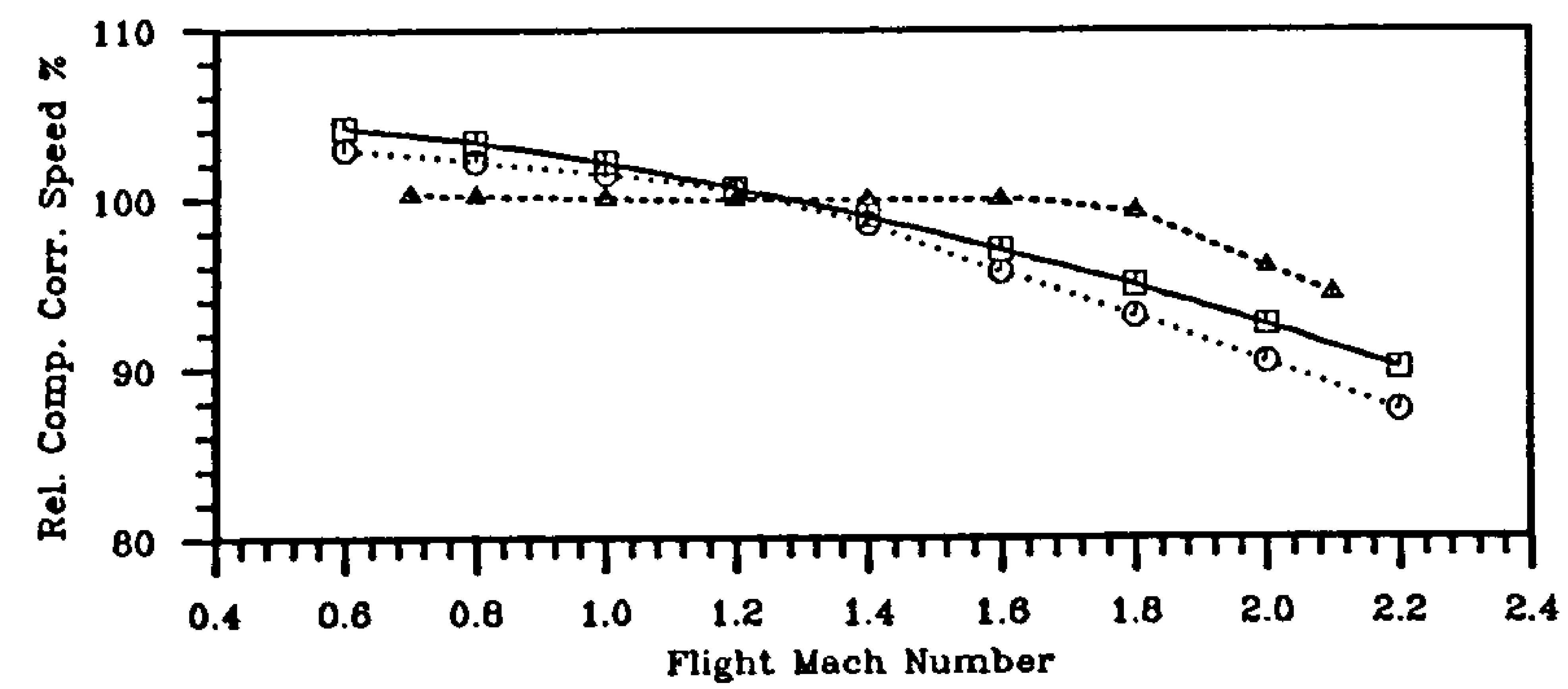



Fig. 8.13d HP Compressor Corrected Speed Schedule at Acceleration

Alt (m) 11000.0
Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
Fixed Comp. Vanes/ Fixed Turbine Area /TJET
Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

The turbofan is also operating at maximum TIT in this range but with a lower mass flow going through the core, Fig. 8.14b, and a higher expansion ratio across the turbines, Figs 8.15c,d, giving rise to a lower thrust.

With the VAT engine, the HP compressor cannot operate at its design point as the HP turbine is operating at its upper area limit up to a Mach number of about unity, Fig. 8.15b. Since speed limit was not reached on any of the spools, it was decided to operate the engine at its TIT limit. The high TIT caused the HP compressor to operate at a higher pressure ratio and with reduced surge margin limit of about 4 points at most. A more realistic operation would be to reduce TIT in this speed range thereby keeping the HP compressor operating with reduced surge margin on the constant non-dimensional mass flow line. Norvaisis [33], in his analysis of a twin-spool unmixed turbofan, was unable to operate the engine at maximum TIT when the HP turbine reached its maximum area as he considered corrected speed to also limit engine operation. In this investigation, the HP corrected speed exceeded the designed, at low Mach numbers.

As both HP turbines in the turbojets operate at the same inlet temperature in the subsonic and transonic speed range, the HP turbine work function and hence, pressure ratio, of the fixed geometry turbojet is higher as a result of having a higher HP compressor work; see Fig. 8.15d and Figs 8.14b and c. The LP turbine of the fixed geometry turbojet develops a higher power at a lower inlet temperature to drive the LP compressor and therefore, its pressure ratio is greater than that of the LP turbine in the VAT engine, Fig. 8.15c. As the aircraft accelerates, the pressure ratio across each turbine of the VAT engine increases due to increased compressor work at the same or lower temperature at inlet to the relevant turbine. The fixed geometry engines operate at constant pressure ratio due to choking of the nozzle immediately downstream.

As the aircraft accelerates further, both the stagnation pressure and temperature at engine face increase as does the mass flow. In order to keep non-dimensional mass flow fixed at the face of the VAT turbojet, the percentage increase in total pressure must exceed that of mass flow and total temperature, hence, the HP turbine of the VAT engine continuously operates at a much higher inlet pressure than mass flow, and since TIT is constant, its area continuously decreases to maintain choked conditions, Fig. 8.15b. Obviously, if the HP turbine area of the fixed geometry turbojet operates choked at the same TIT, then its area would be lower because of the higher mass flow and pressure ratio. Therefore, the VAT engine operates with a higher area up to the start of the supersonic leg.

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

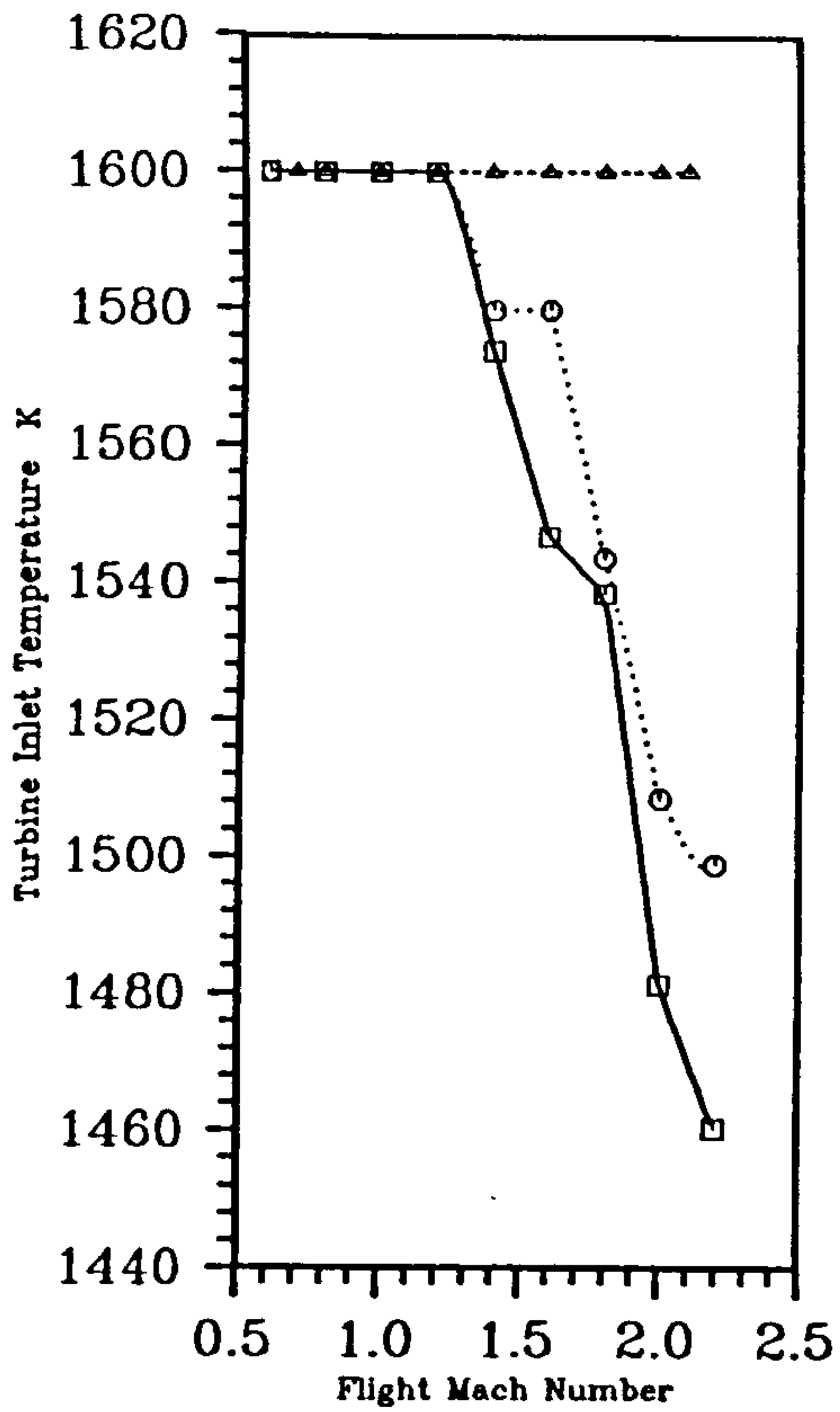



Fig. 8.14a Temp Schedule for Thrust Modulation at Acceleration

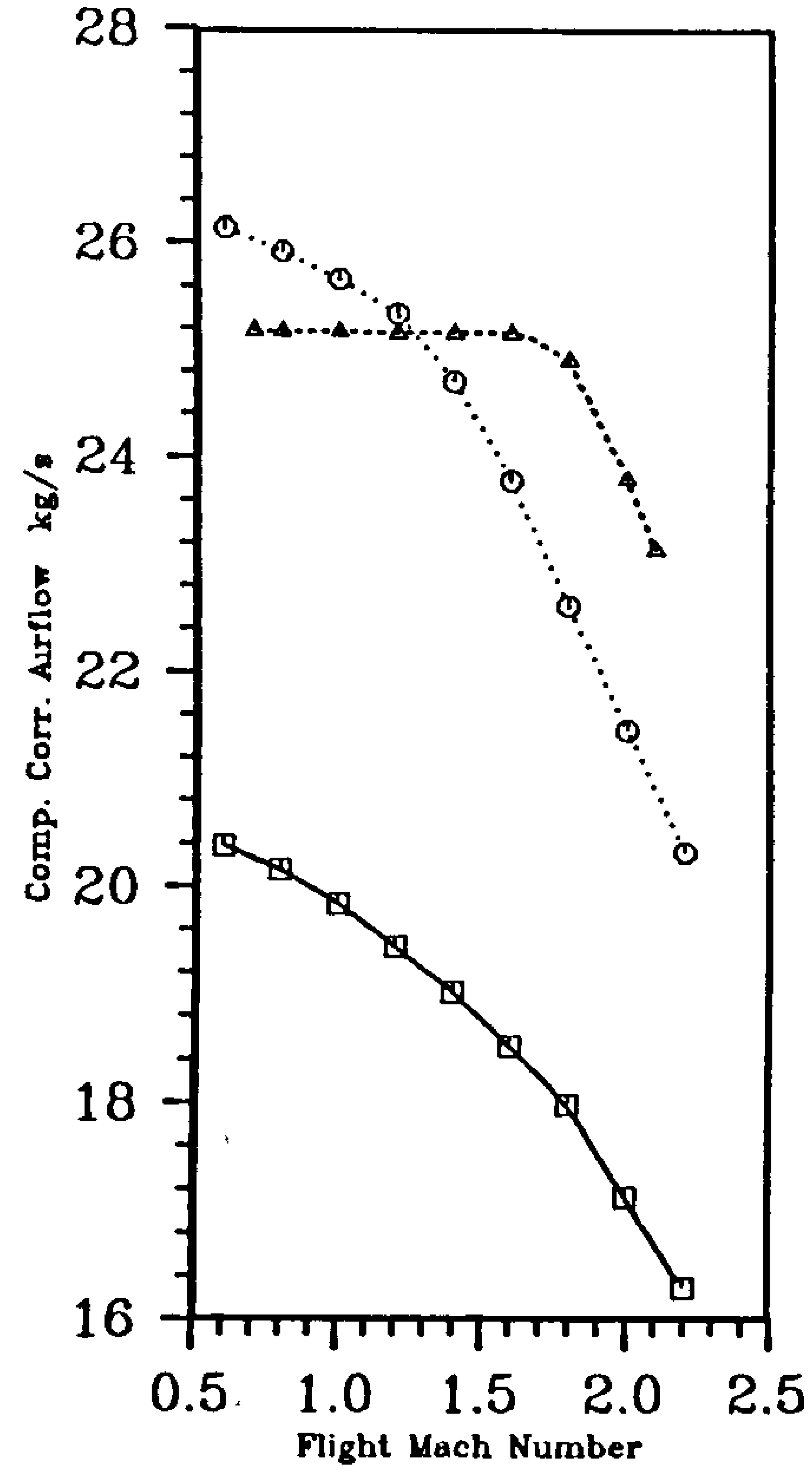



Fig. 8.14b HP Compressor Airflow Requirement at Acceleration

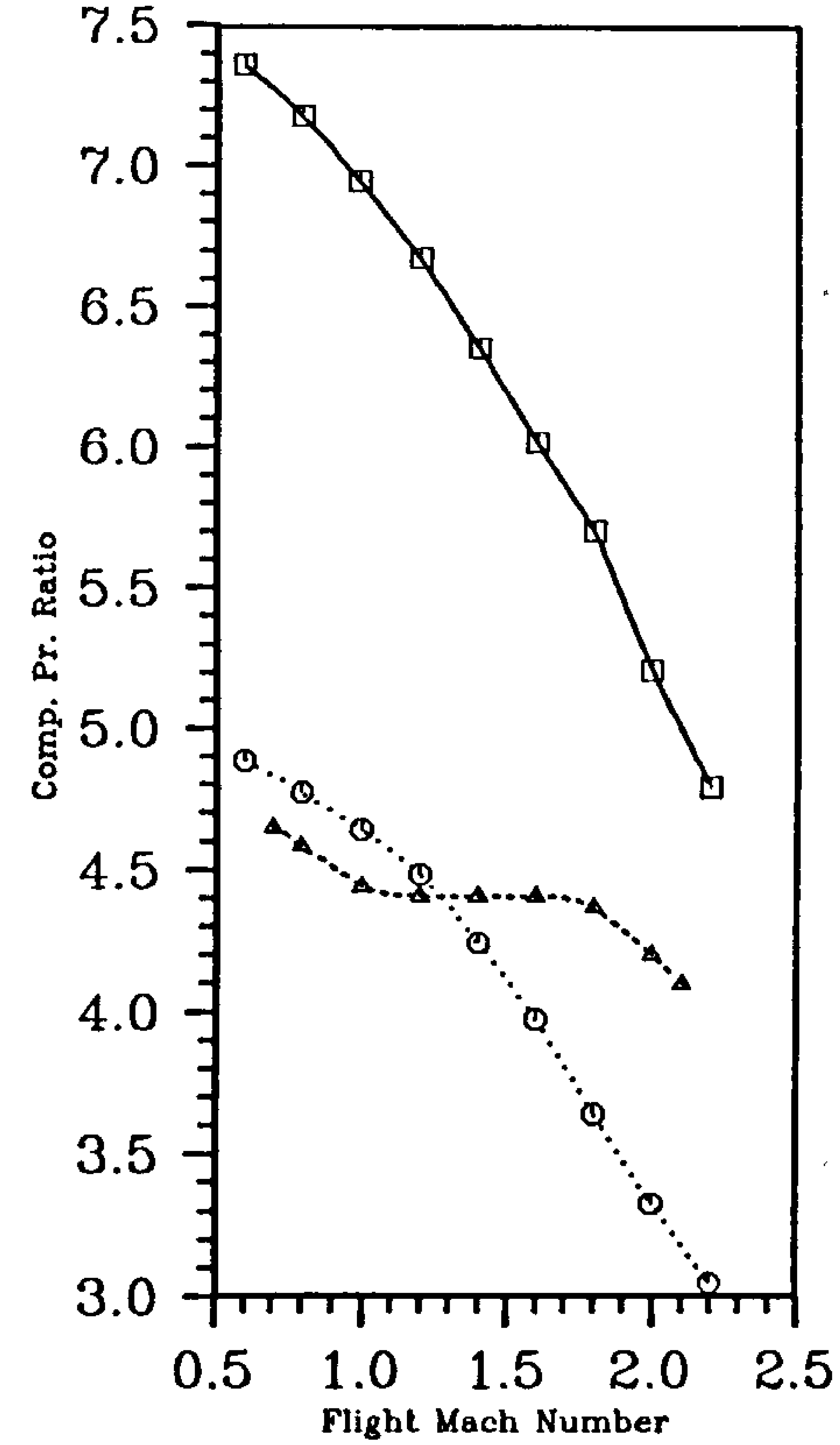



Fig. 8.14c HP Compressor Pr. Ratio at Acceleration

Alt (m) 11000.0

□—□ Fixed Comp. Vanes/ Fixed Turbine Area /TFAN

○·····○ Fixed Comp. Vanes/ Fixed Turbine Area /TJET

▲·····▲ Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

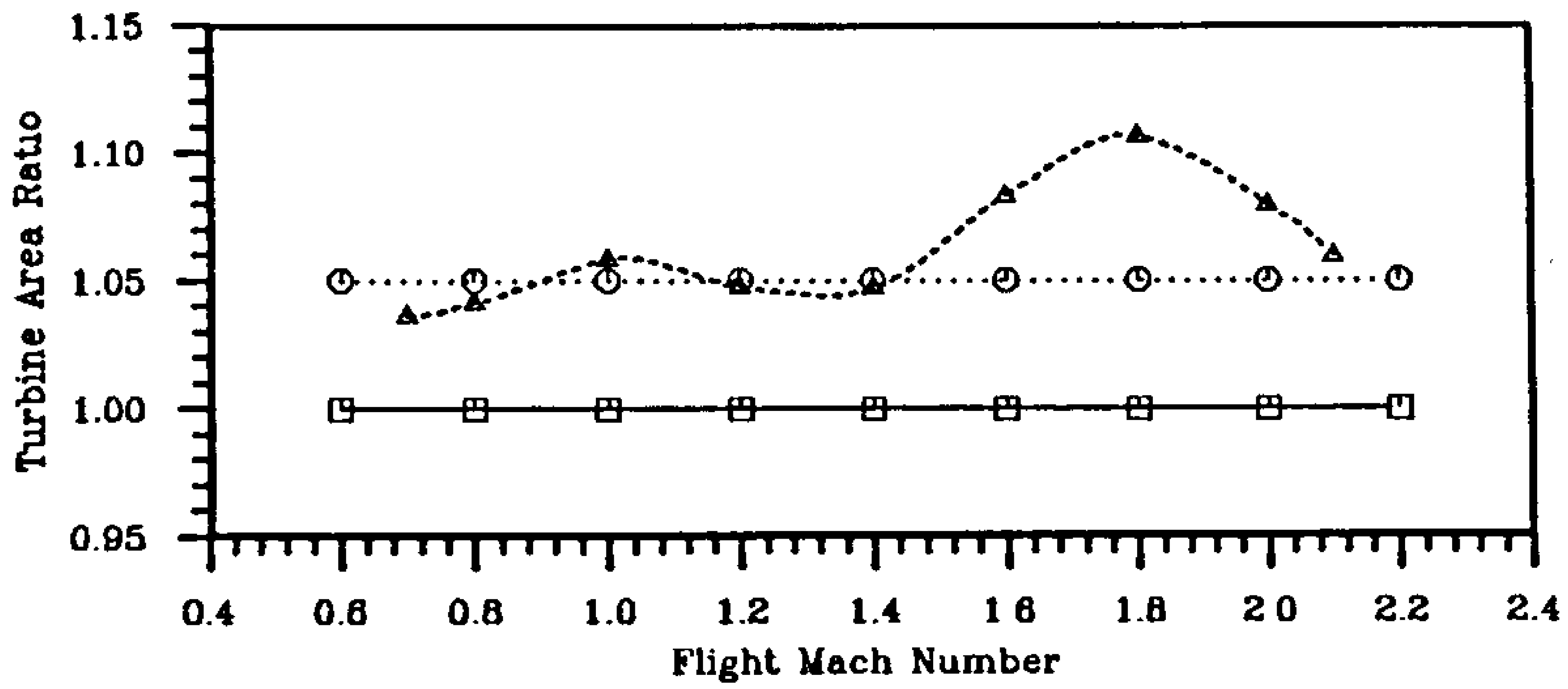



Fig. 8.15a LP Turbine Area Schedule at Acceleration

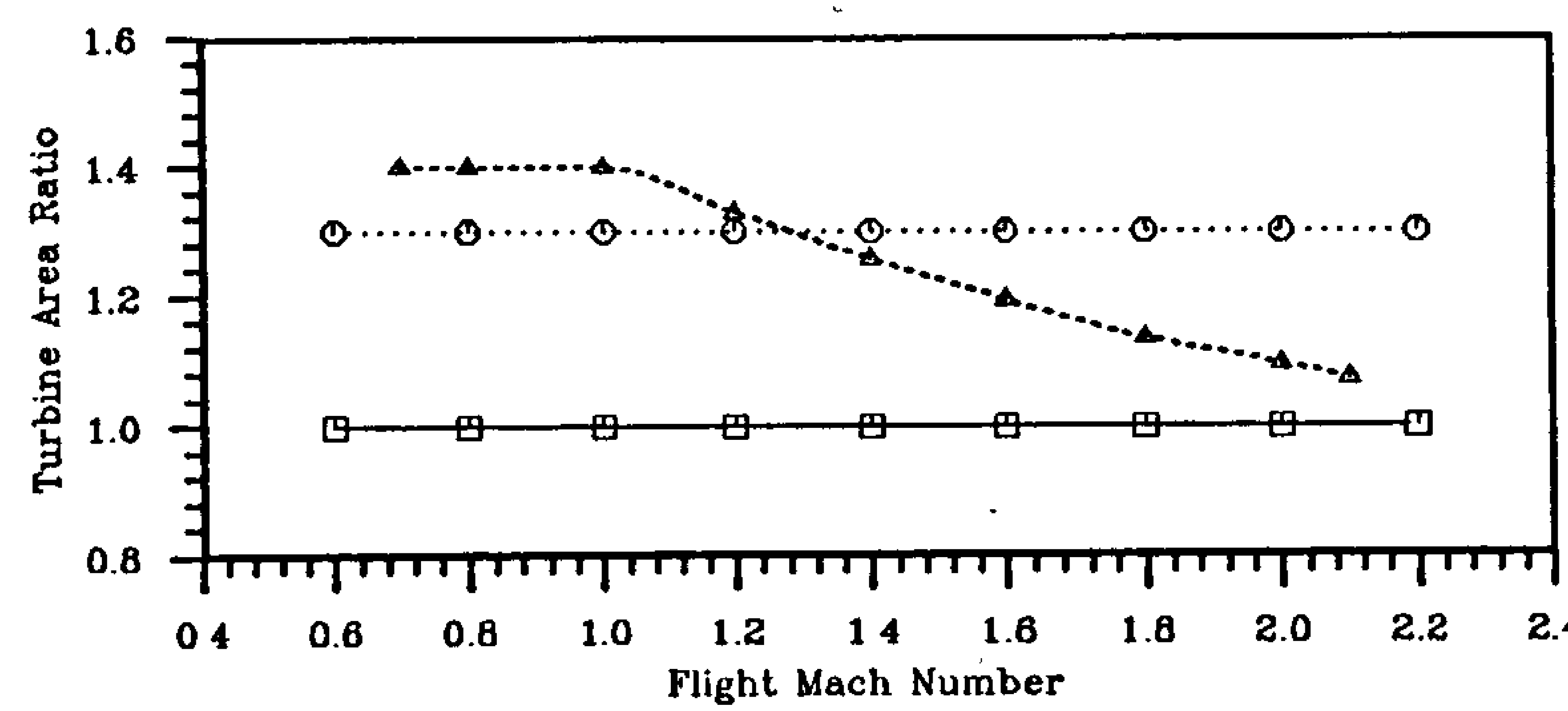



Fig. 8.15b HP Turbine Area Schedule at Acceleration

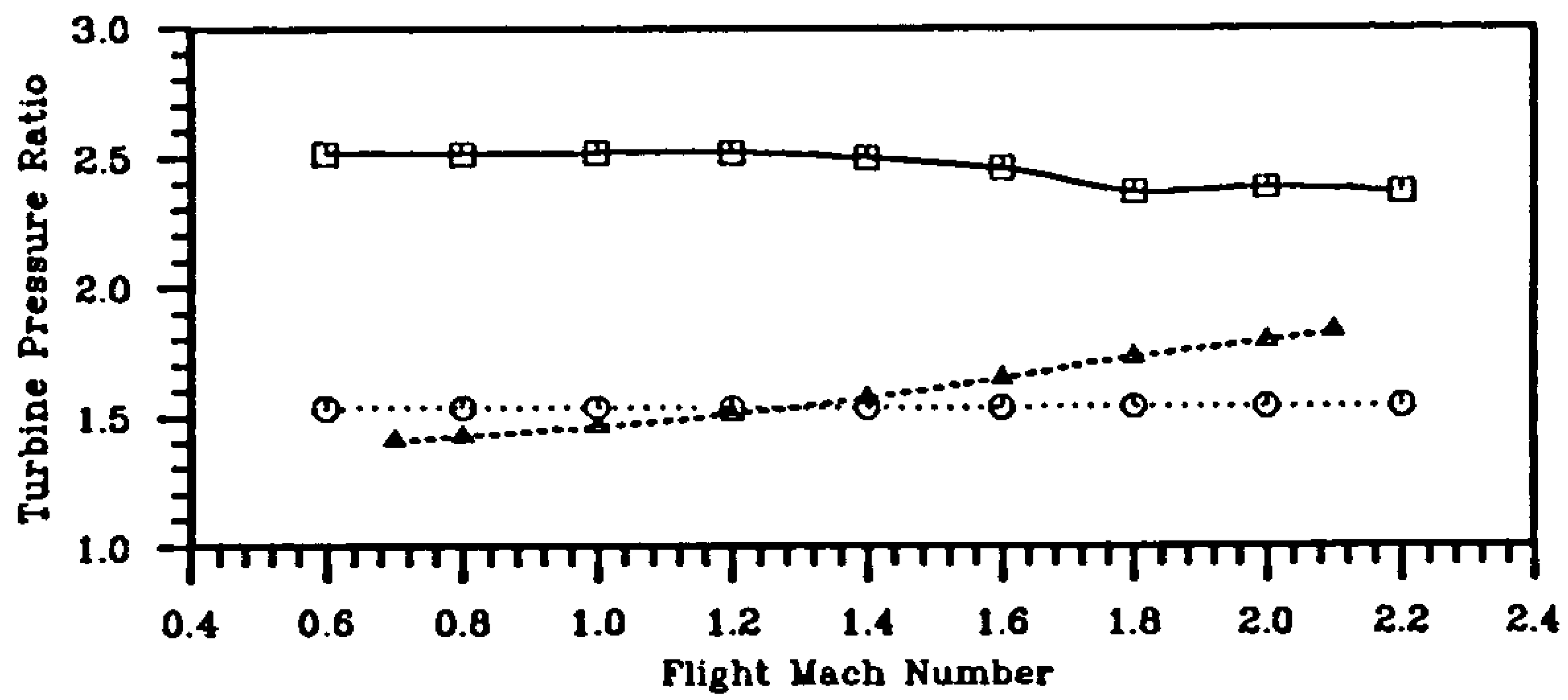



Fig. 8.15c LP Turbine Pr. Ratio at Acceleration

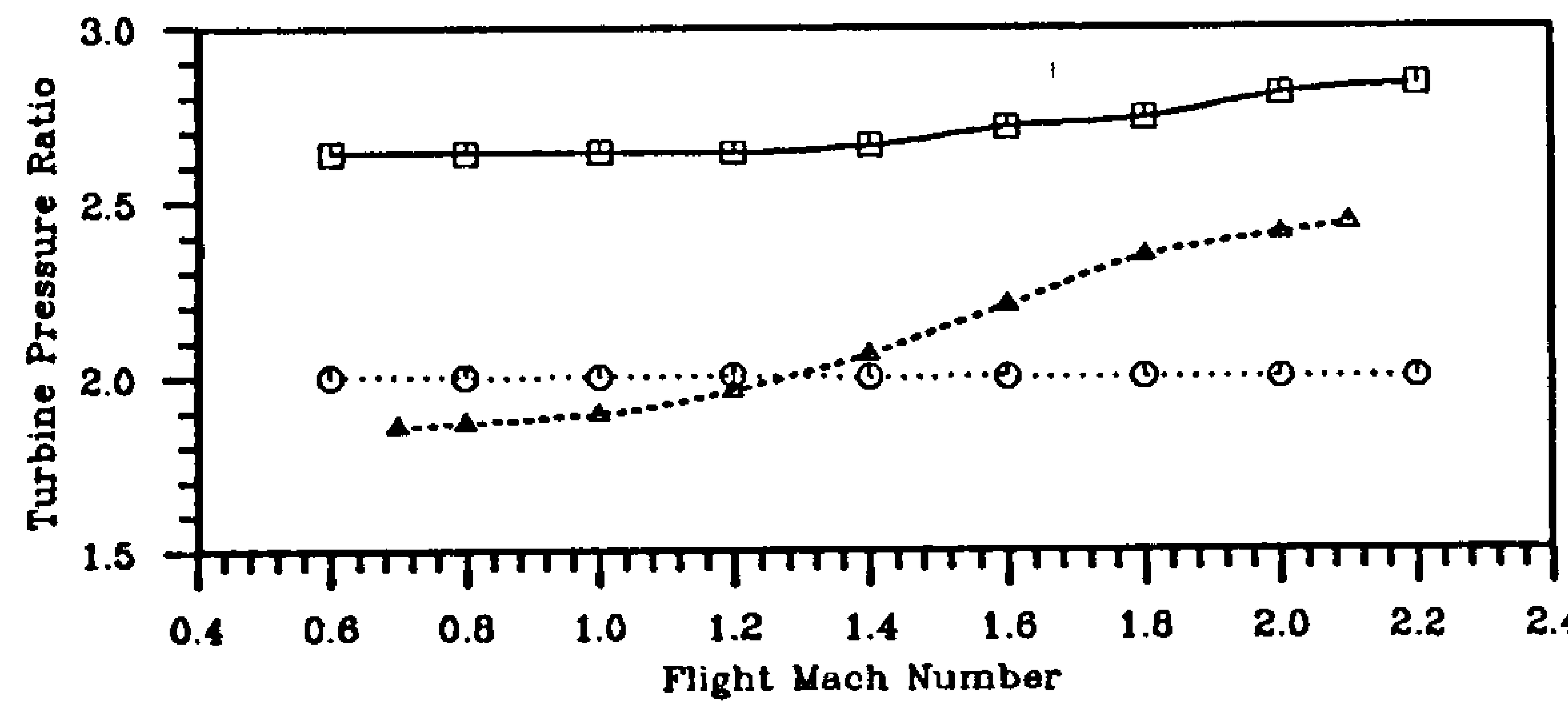



Fig. 8.15d HP Turbine Pr. Ratio at Acceleration

Alt (m) 11000.0

□—□ Fixed Comp. Vanes/ Fixed Turbine Area /TFAN

○····○ Fixed Comp. Vanes/ Fixed Turbine Area /TJET

▲----▲ Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

The variation of the LP turbine area shown in Fig. 8.15a is a bit more difficult to explain. What is happening, it is believed, is that as the flight Mach number increases from a low subsonic value, the airflow into the turbine increases but as the HP compressor pressure ratio decreases, Fig. 8.14c, the HP compressor work decrease should cause an increase in the temperature at turbine inlet since TIT remains fixed. But it was observed that the temperature at the inlet to the turbine remained almost constant probably due to increased compressor efficiency. The LP turbine does not benefit much from the increase in total pressure at compressor face as Mach number increases because of the decrease in HP compressor pressure ratio. Hence, the percentage decrease in pressure causes an increase in turbine area. As the Mach number increases, the HP compressor is able to operate at its design point and the temperature at the LP turbine inlet decreases. The pressure at turbine inlet continues to increase but the effect of the increase in mass flow predominates thereby causing the turbine area to increase. At the high Mach number end where both compressors are spool speed limited, the corrected speeds drop and the percentage decrease in pressure predominates causing the turbine area to decrease.

The propelling nozzle throat areas of both fixed geometry engines remain almost constant as aircraft speed increases, Fig. 8.16a, while that of the VAT engine increases because of a reduced increase in jet pipe pressure due to increased turbine expansion ratio. At low and medium flight speeds, the VAT nozzle operates with a smaller throat area compared with the fixed geometry turbojet as it is passing less mass flow, but as the fixed geometry turbojet operates at lower TITs due to spool speed limit at high Mach number, it swallows less airflow and therefore operates with a smaller throat area. All the core nozzles operate at higher exit area to maintain full expansion as aircraft speed increases, Fig. 8.16b. When the maximum area is reached, the nozzles operate under-expanded. The turbofan engine operates with larger throat and exit areas because of the lower jet pipe pressure. The nozzle pressure ratio of the fixed geometry engines initially rises with Mach number due to the increase in jet pipe pressure and then stays almost constant when maximum exit area is reached, Fig. 8.16c. The VAT turbojet takes the same trend initially but operates at reduced expansion ratio when the exit area reaches the maximum as a result of an increase in area ratio.

The LP turbine exit Mach number of the VAT engine increases with aircraft speed as can be seen in Fig. 8.16d, but there is no danger of approaching limit loading. Since turbine pressure and area ratios increase with flight Mach number, the exit non-dimensional mass flow increases as can be shown from mass flow balance. Therefore, the exit Mach number increases. At low flight speeds, the exit Mach

# Performance Comparison Of Fixed And Variable Geometry Jet Engines

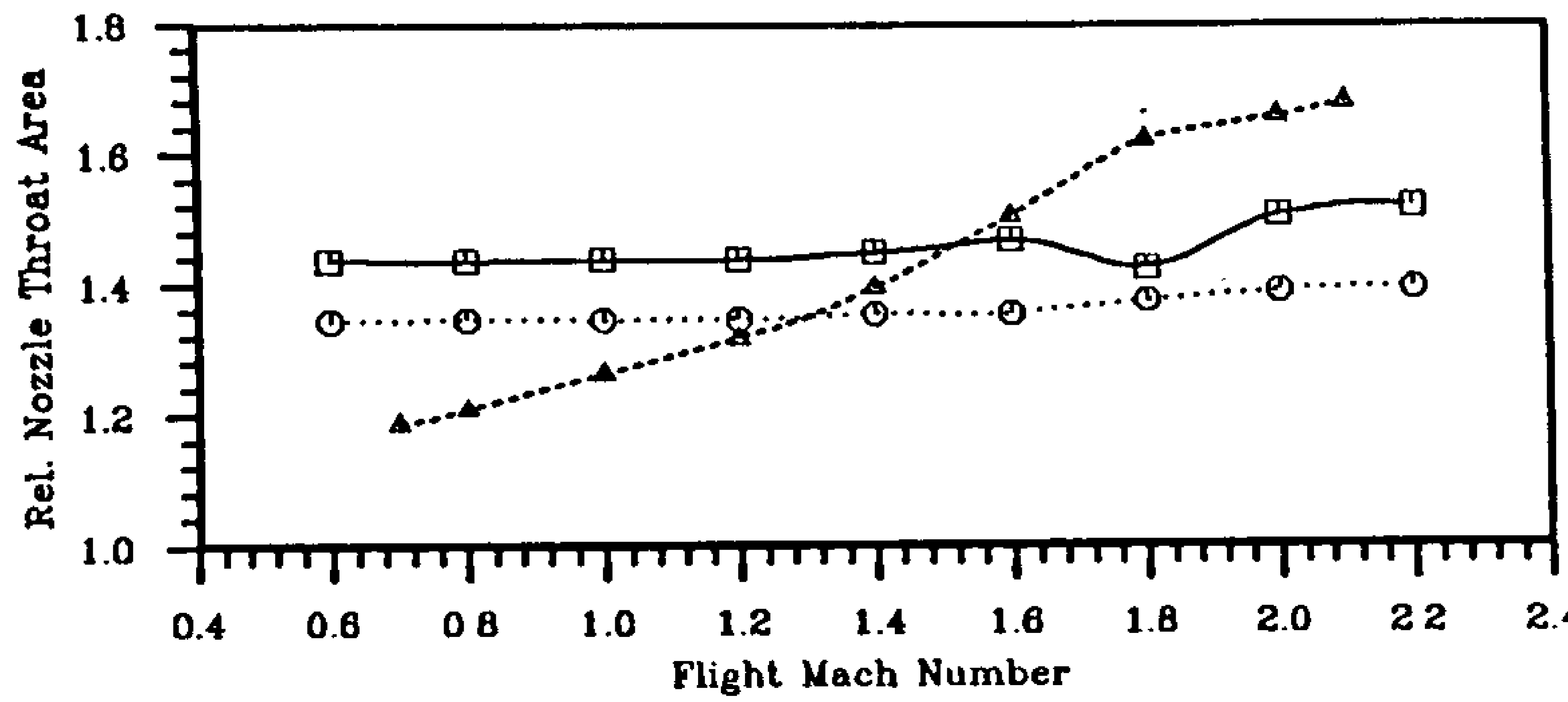



Fig. 8.16a Core Nozzle Throat Area Variation at Acceleration

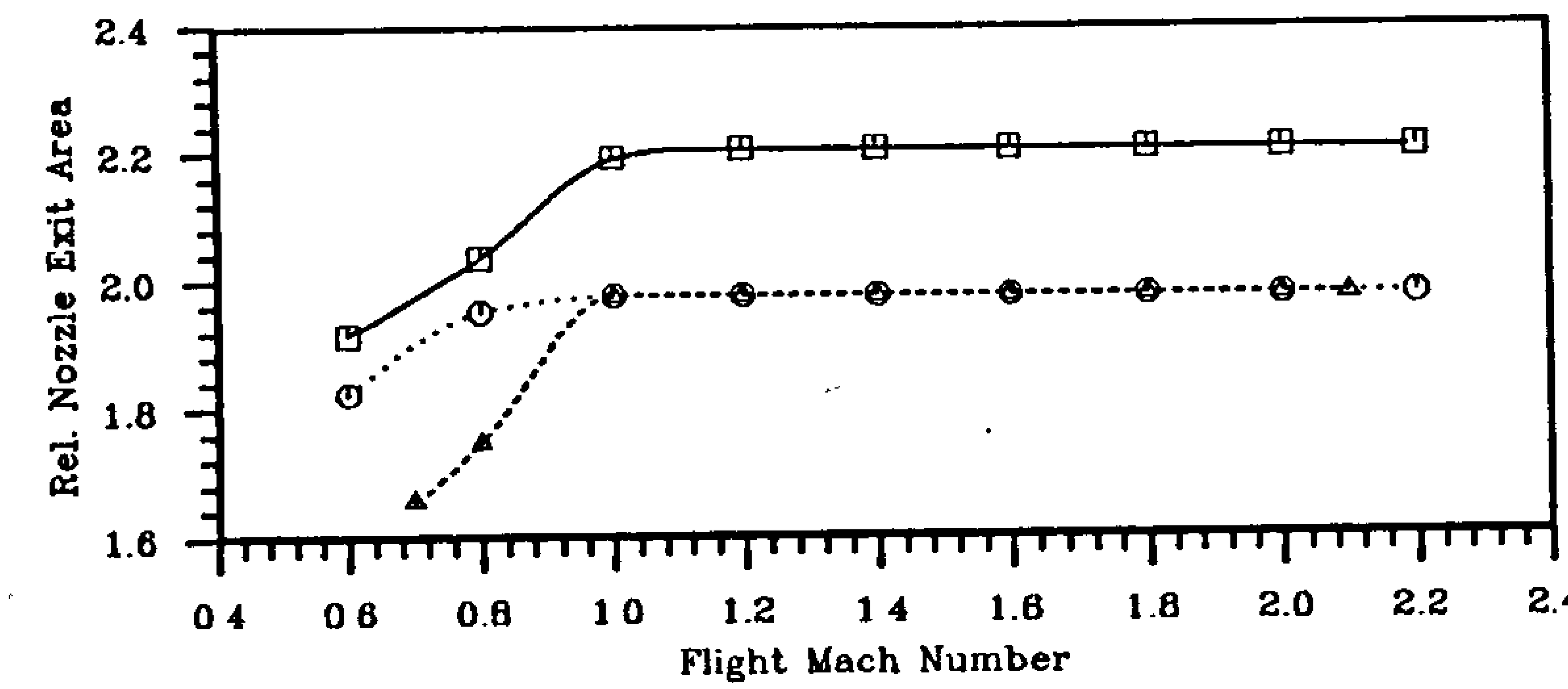



Fig. 8.16b Core Nozzle Exit Area Variation at Acceleration

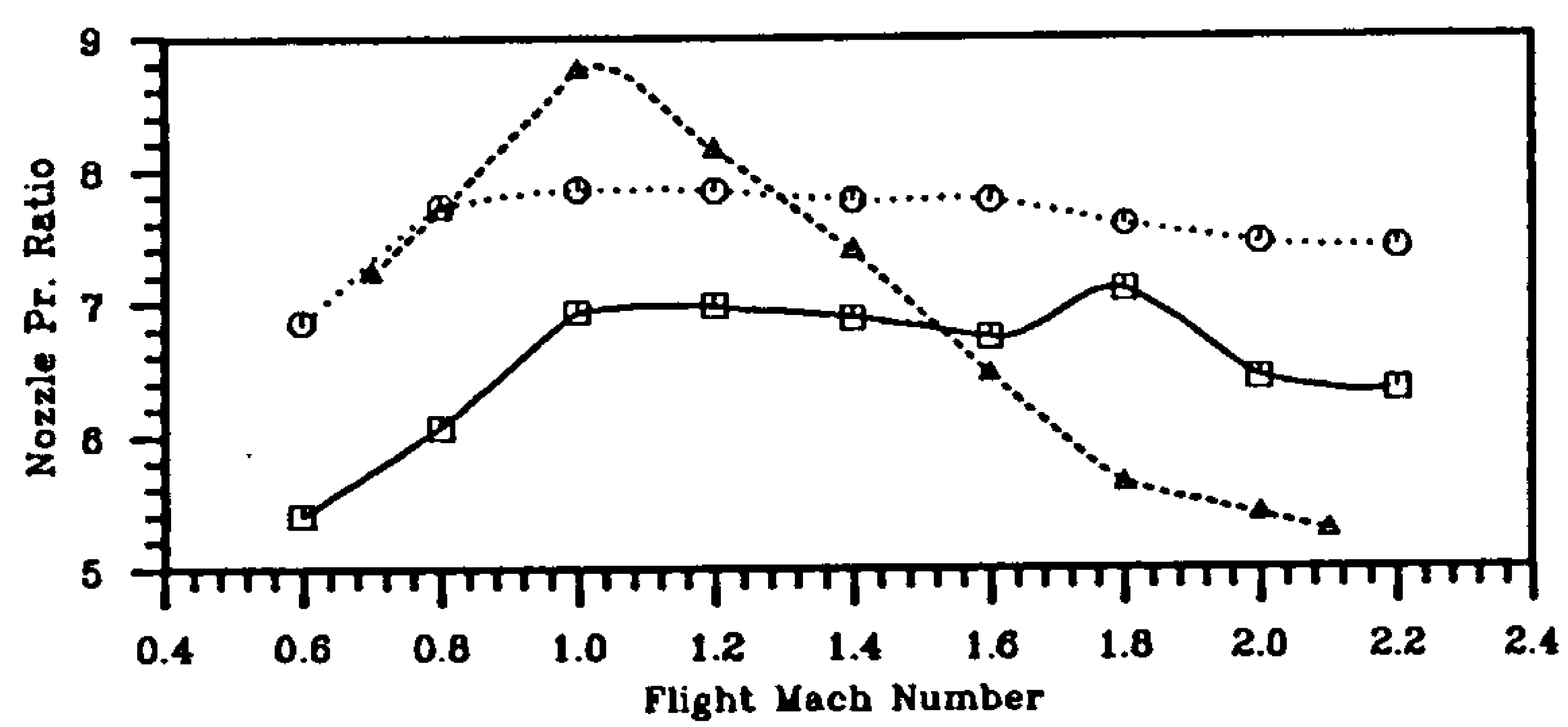



Fig. 8.16c Core Nozzle Pressure Ratio Variation at Acceleration

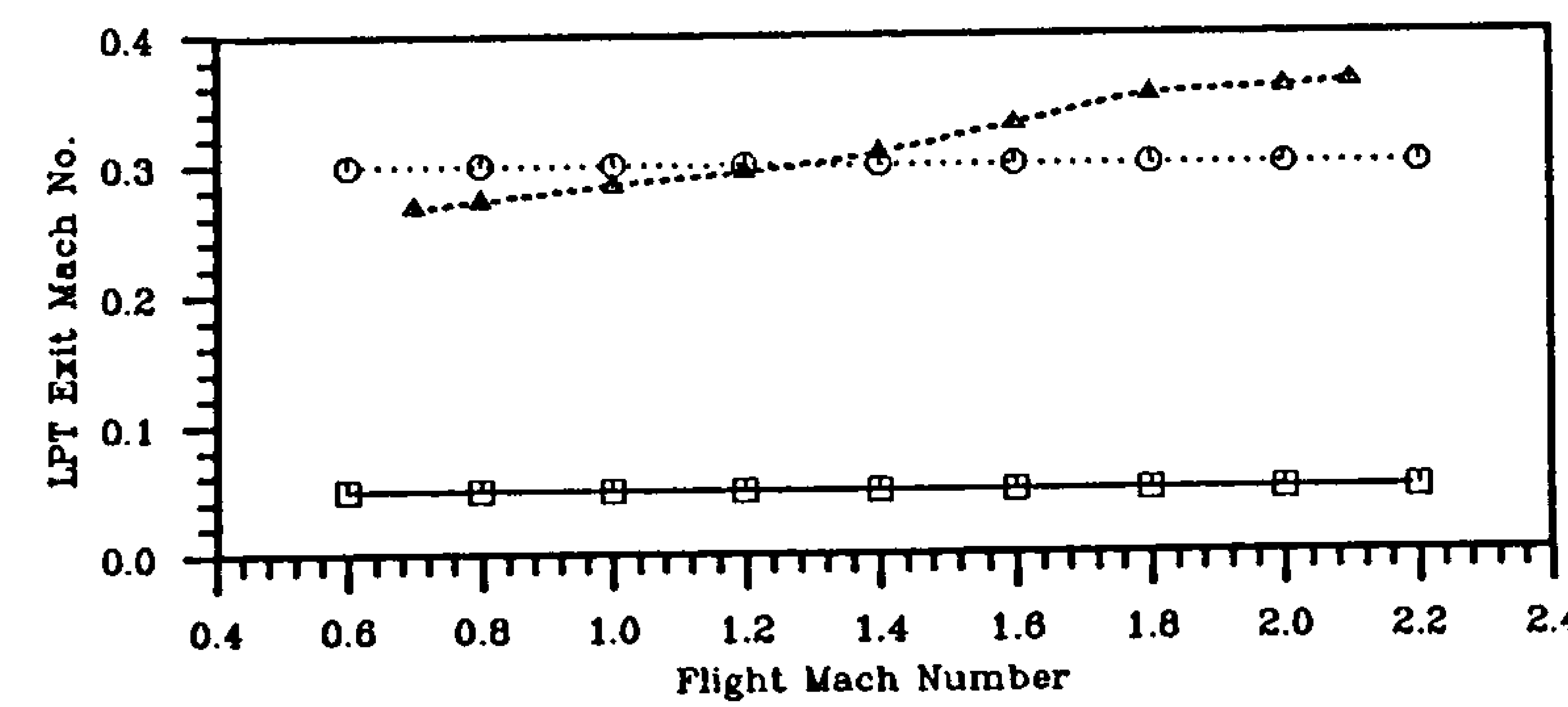



Fig. 8.16d LPT Exit Mach No. Variation at Acceleration

Alt (m) 11000.0

Fixed Comp. Vanes/ Fixed Turbine Area /TFAN

Fixed Comp. Vanes/ Fixed Turbine Area /TJET

Fixed Comp. Vanes/ Var. LP,HP Turbine Area /TJET

number is lower than that of the fixed area turbojet as the VAT engine operates at a lower LP turbine pressure ratio and inlet area, Fig. 8.15c,a.

At supersonic speeds, the fixed geometry engine cannot operate at the maximum TIT due to HP spool speed limit, Fig. 8.13c, therefore, both HP and LP compressor speeds roll off, Fig. 8.13b,d. As a result, the mass flow decreases while the VAT engine is able to maintain compressor design point performance and this gives rise to a lower thrust as compared with that of the VAT engine, Fig. 8.12a. The turbofan is a non competitor. Referring to Fig. 8.12b, both turbojets give similar fuel burn characteristics up to about the start of the supersonic leg. The fixed area turbojet produces a higher thrust at a higher mass flow resulting in comparable specific thrust with that of the VAT. The turbofan gives a better performance due to its higher propulsive efficiency.

At the high speed end, the VAT turbojet exhibits a higher sfc than its fixed geometry counterpart as it operates at a much higher airflow giving rise to a lower specific thrust. The aft-end drag was not included in this analysis as no data was available at the Mach numbers considered, but since the nozzles were operating at their maximum exit area for most of the speed range, the level of aft-end drag will be determined by the exit static pressure and since that for the VAT is greater, Fig. 8.16c, the attendant aft-end drag will be lower and so it is possible that the performance of the VAT engine may be better in terms of increased thrust and reduced sfc. Therefore, it cannot be concluded that the fixed area turbojet gives a better aircraft acceleration performance in general.

## Engine Performance with HP Compressor Operating Point Varying

In the above analyses, both the HP and LP compressors of the VAT engine were controlled to operate at their design point within the given constraints. The compressor controls were coupled such that if the LP compressor could no longer keep its operating point fixed, the HP compressor too would stop operating in this mode whether or not it has violated a constraint. The reverse is not true; that is, the LP compressor will continue to operate at fixed operating point when the HP compressor can no longer operate in this fashion as was observed during the investigation of acceleration performance, at low Mach numbers. The engine could have been controlled such that only the LP compressor operated with fixed operating point while the HP spool speed varied to give the required matching as thrust is modulated. The performance of the engine with both types of control is presented in Figs 8.17 and 8.18 for dry power operation.

Fig. 8.17a shows that having both compressors operating at their design point gives a better engine performance in terms of fuel consumption. There is the tendency that at very high power settings quite close to intermediate, controlling the engine such that only one compressor operating point is fixed results in a lower fuel burn. The general trend of performance can be explained by taking a look at the other figures in Figs 8.17 and 8.18. The performance of the engine with just the LP compressor operating point fixed is described below. Operation with both compressor operating points fixed is used as a reference and the comparisons are made at the same thrust setting.

Since the LP compressor operates at its design point, the HP compressor is constrained to operate at a fixed inlet non-dimensional mass flow, that is, the operating line is vertical with speed being changed slightly as thrust changes. The HP compressor pressure ratio therefore changes, Fig. 8.17b. At high power settings quite close to intermediate, there is the tendency for the HP compressor to operate at a higher pressure ratio, and depending on the position of the design point, compressor efficiency may be higher. The higher pressure ratio at the high thrust end improves the thermal efficiency and this gives rise to an improvement in sfc. As thrust is reduced, the HP compressor pressure ratio falls and the loss of efficiency which may occur will both lead to a higher fuel burn.

Since the LP compressor operating point is fixed for both cases, the LP turbine pressure ratio variations are identical, Fig. 8.18a, and therefore allowing the HP compressor operating point to change causes the HP turbine to operate at a lower pressure ratio. Owing to identical TIT profiles, Fig. 8.18b, the temperature at inlet to the LP turbine is much higher as the HP turbine work is less, therefore, to maintain choked LP turbine conditions, the LP turbine opens up its area more, Fig. 8.17c, and as was observed, the pressure at this station does not change much from that obtained with both compressor operating points fixed. Both mass flows are the same, neglecting the small difference in fuel flow.

It can be shown from mass flow balance that a larger LP turbine inlet area gives a larger non-dimensional exit mass flow, and since non-dimensional mass flow is a measure of flow Mach number, the LP turbine exit Mach number is higher, Fig. 8.18c. If the propelling nozzle is to continue to operate choked at the higher LP turbine exit flow function, then its throat area must increase to pass the mass flow, Fig. 8.18d, and the exit area also has to increase to maintain optimum area ratio for full expansion, Fig. 8.18e. The propelling nozzle pressure ratio is lower

# Effect Of HP Turbine Area Variability

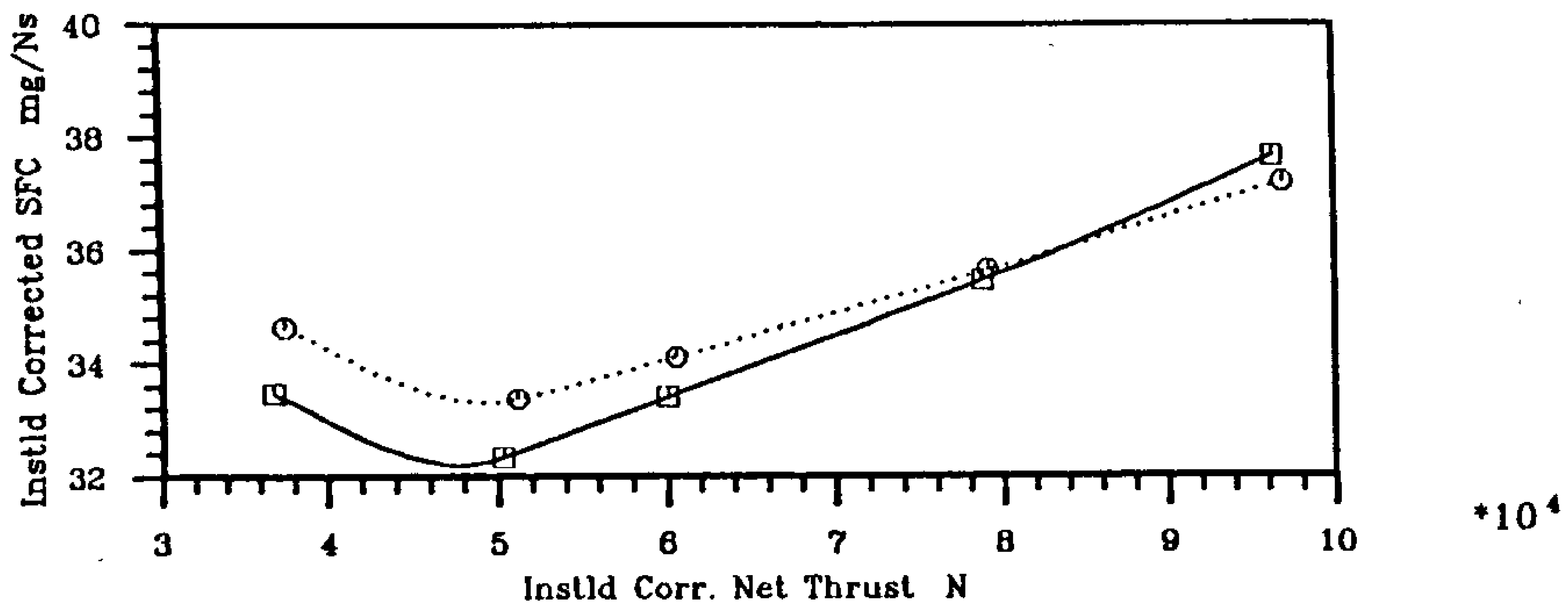



Fig. 8.17a Fuel Flow Requirement For Unit Thrust

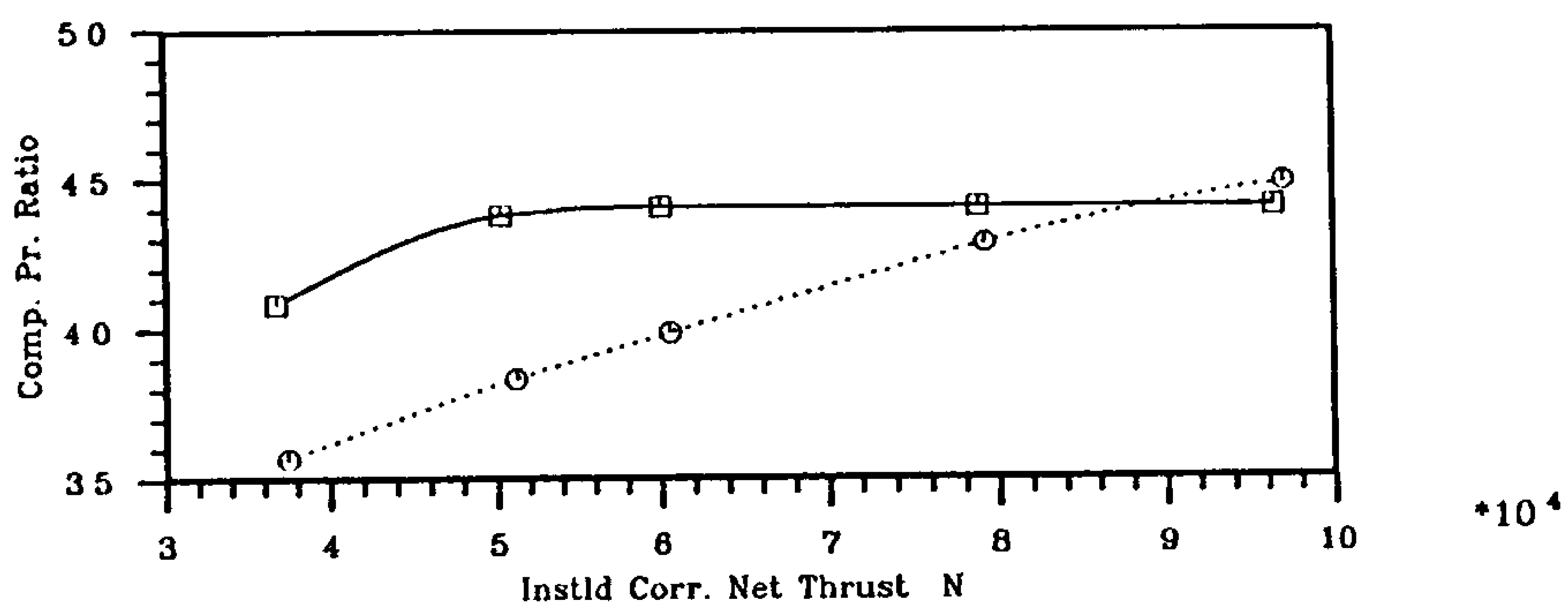



Fig. 8.17b HP Compressor Pr. Ratio

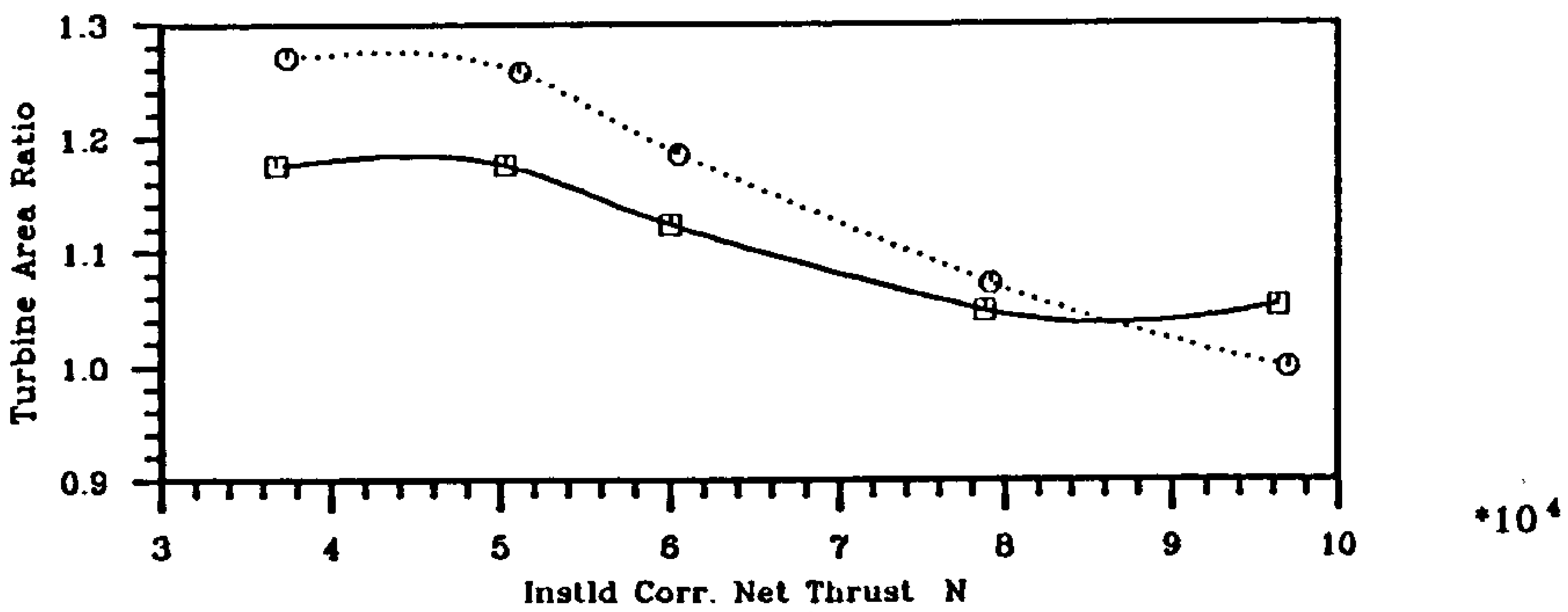



Fig. 8.17c LP Turbine Area Schedule

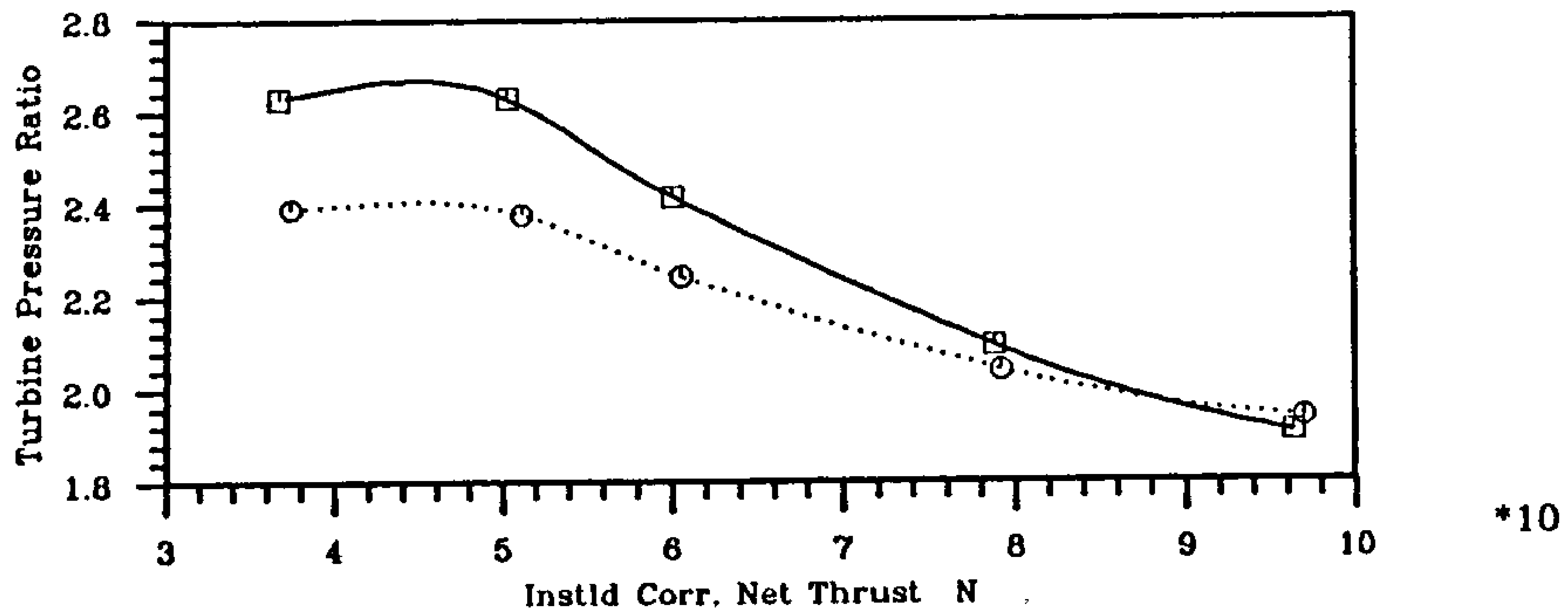



Fig. 8.17d HP Turbine Pr. Ratio

Alt (m) 9144.0
Mn 0.9

□—□ Fixed Comp. Vanes/ Var. LP,HP Turbine Area
○····○ Fixed Comp. Vanes/ Var. LP Turbine Area

# Effect Of HP Turbine Area Variability

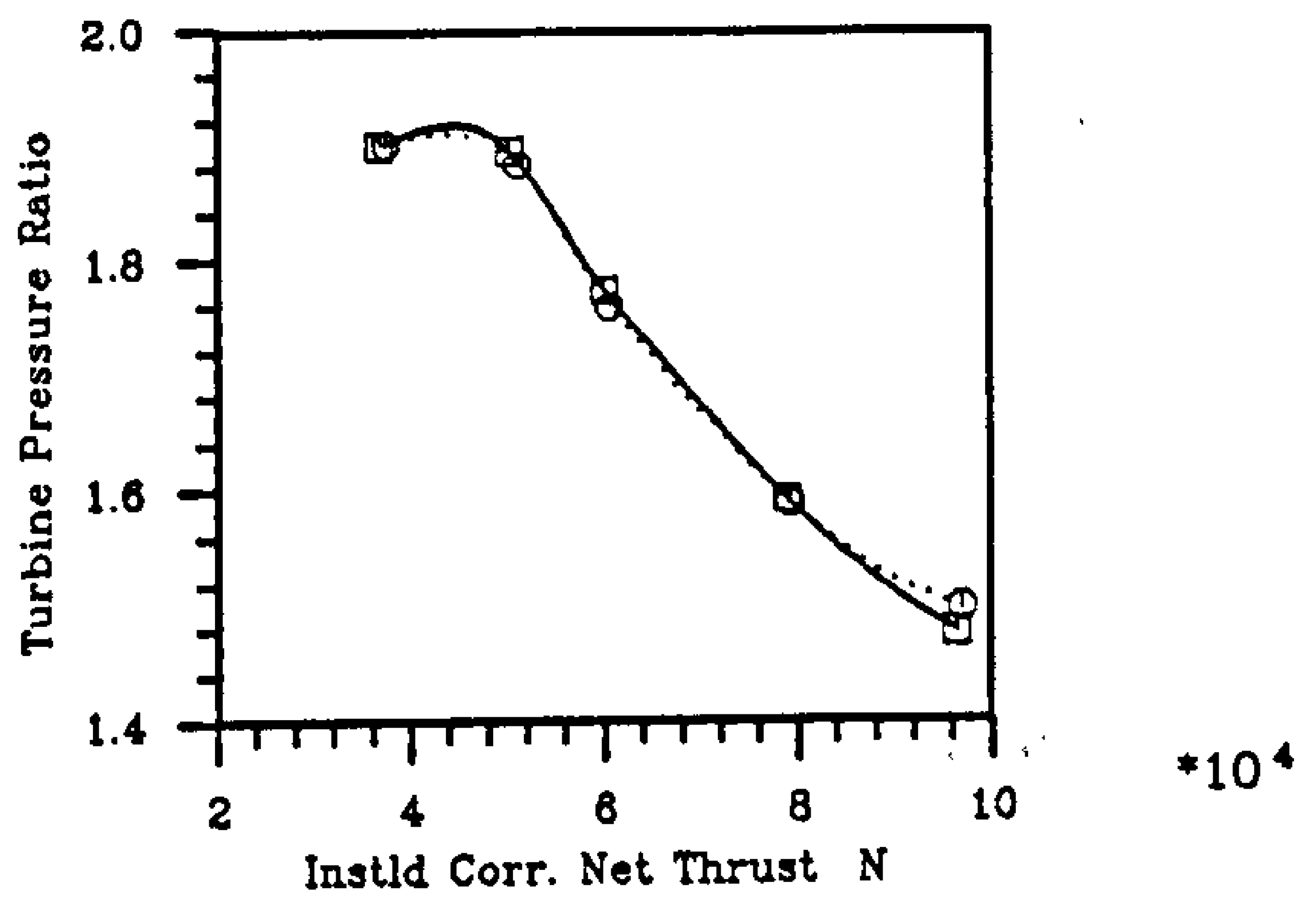



Fig. 8.18a LP Turbine Pr. Ratio

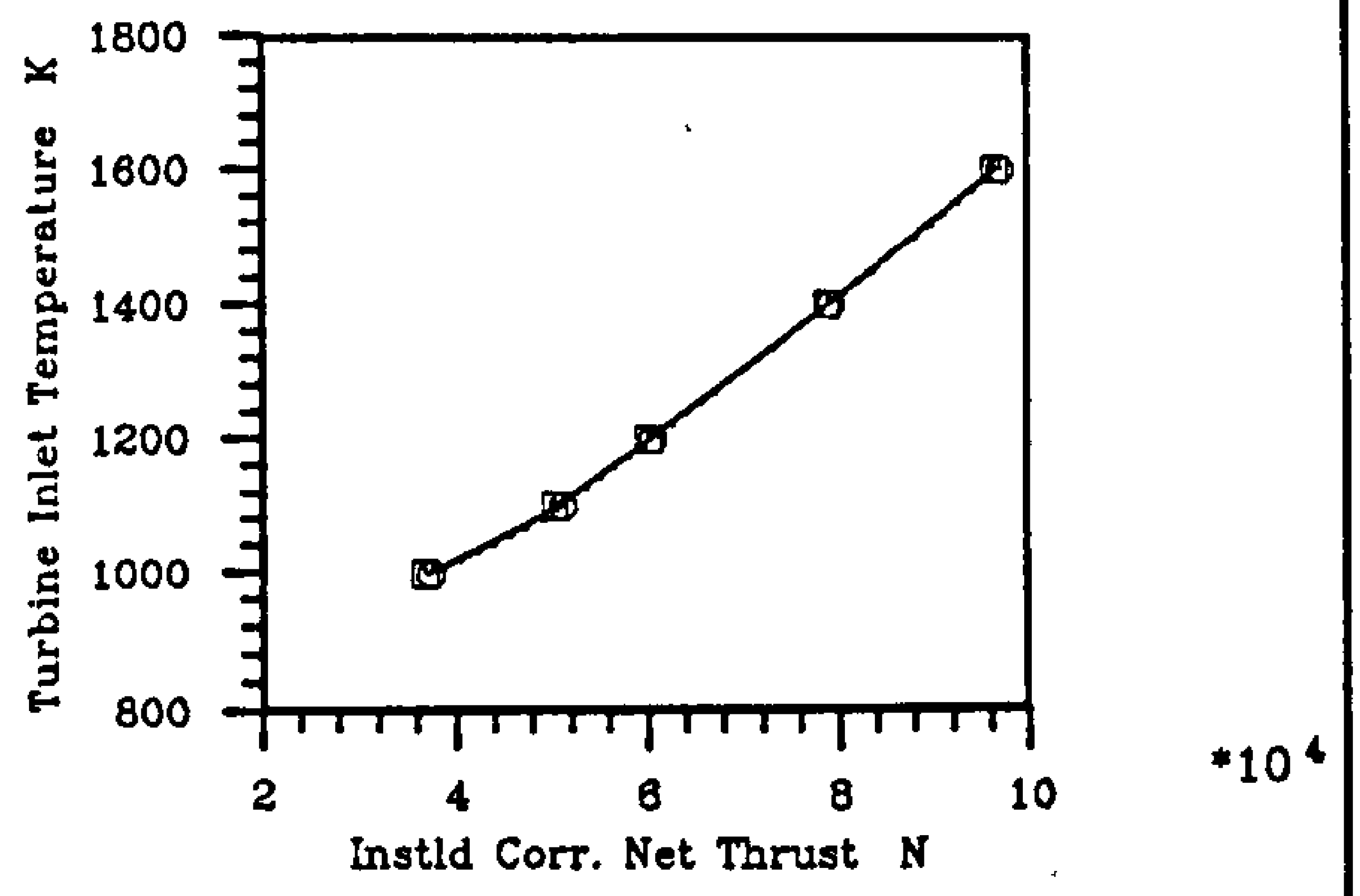



Fig. 8.18b Temp Schedule for Thrust Modulation

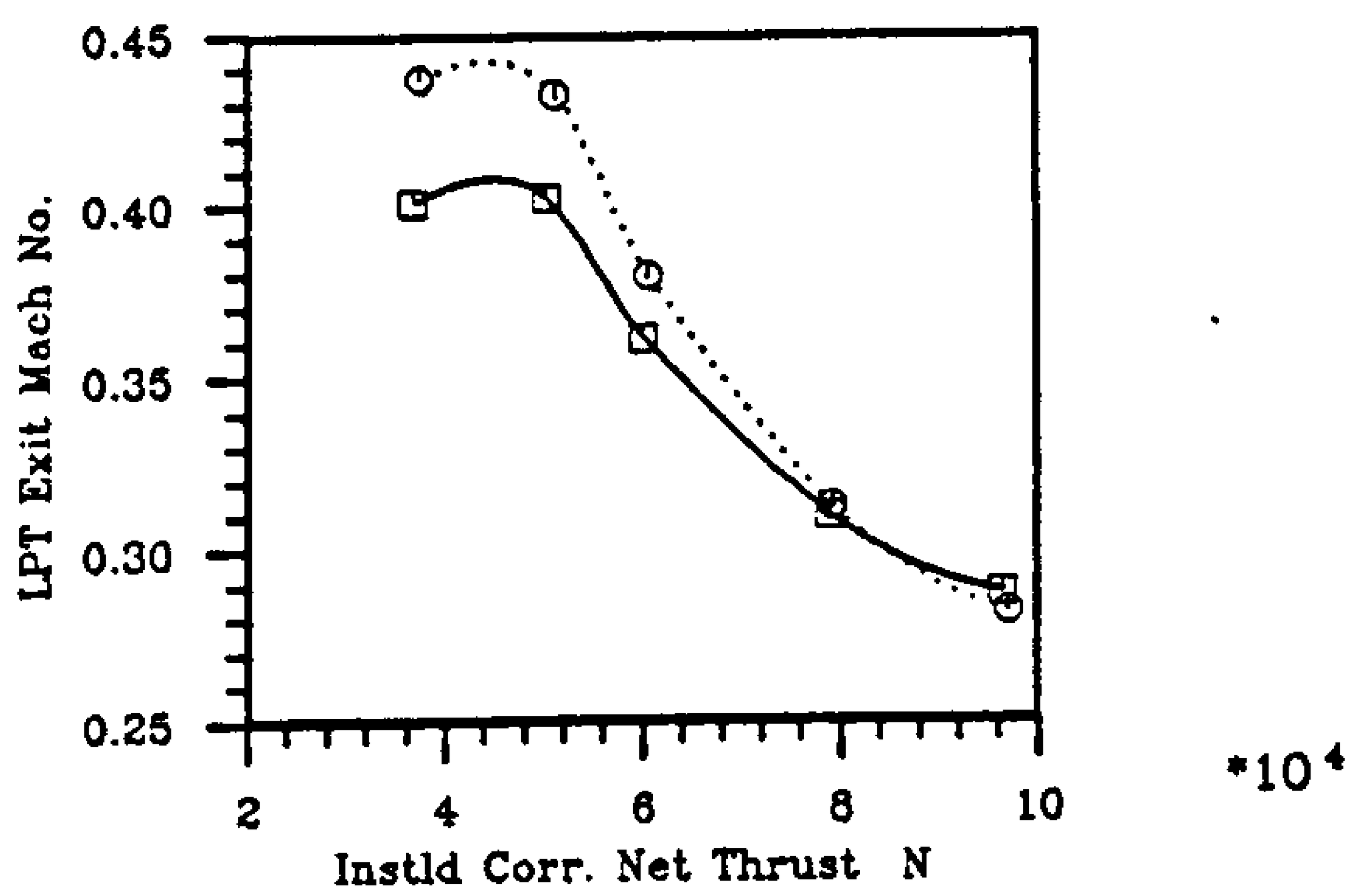



Fig. 8.18c LPT Exit Mach No. Variation

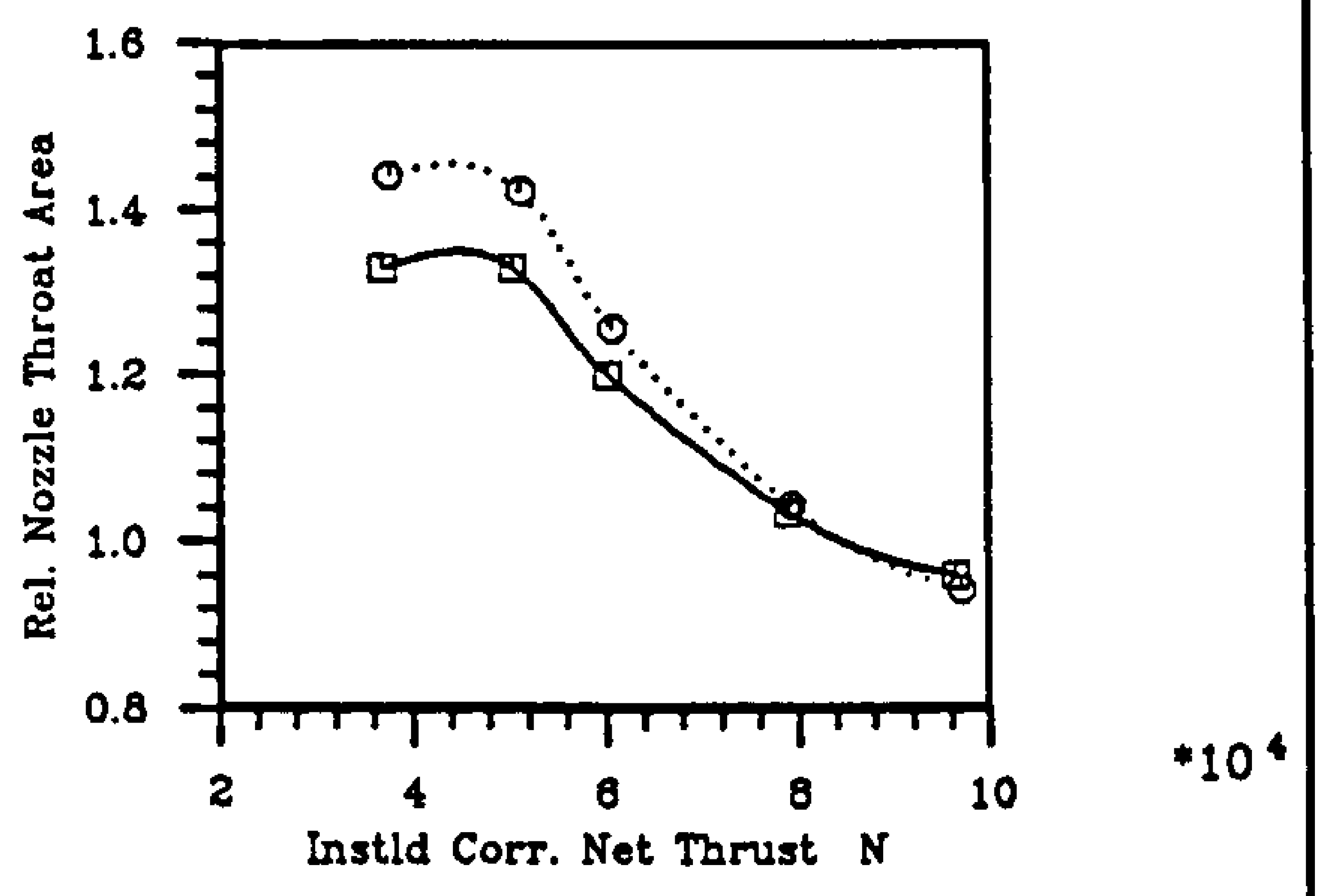



Fig. 8.18d Nozzle Throat Area Variation

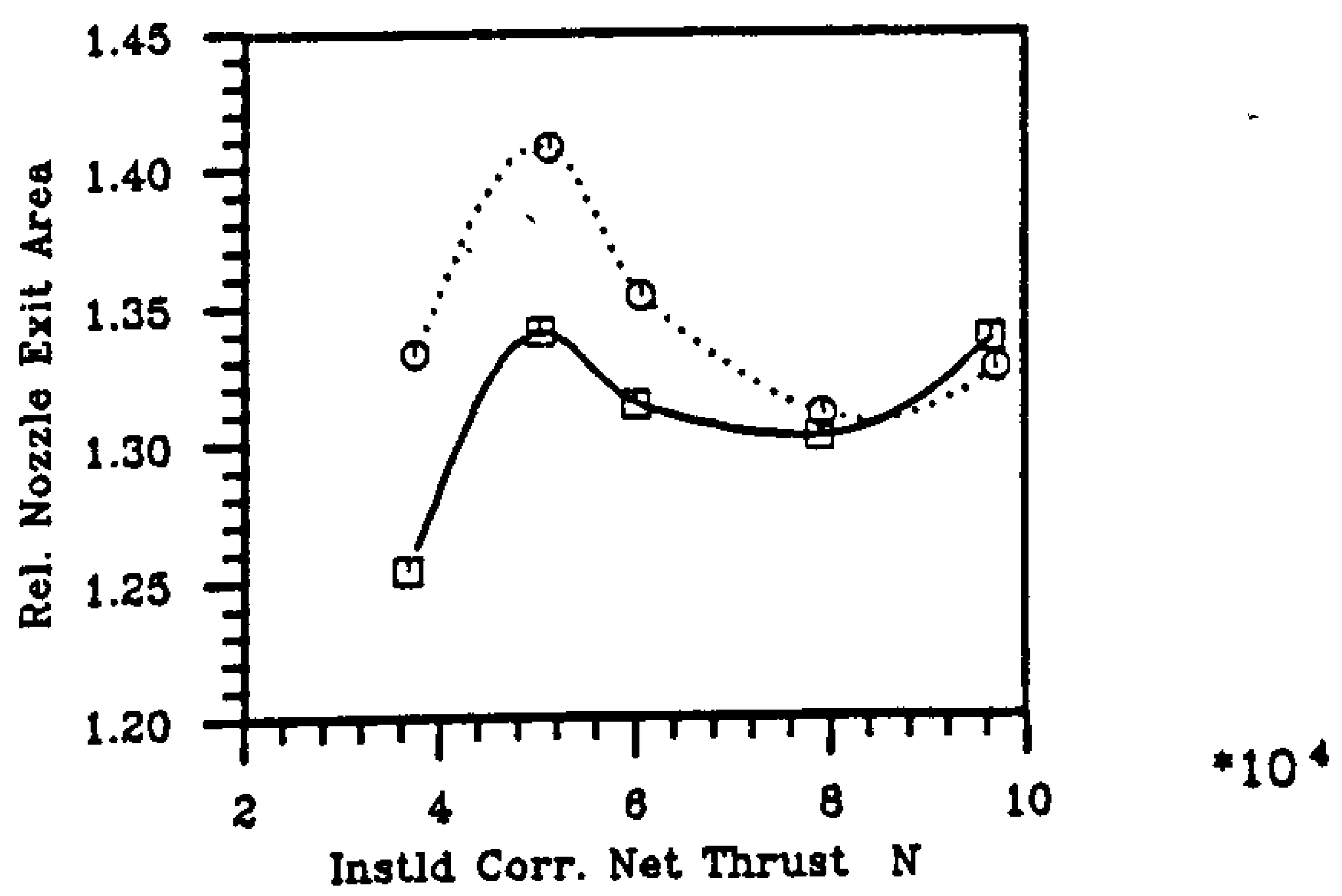



Fig. 8.18e Nozzle Exit Area Variation

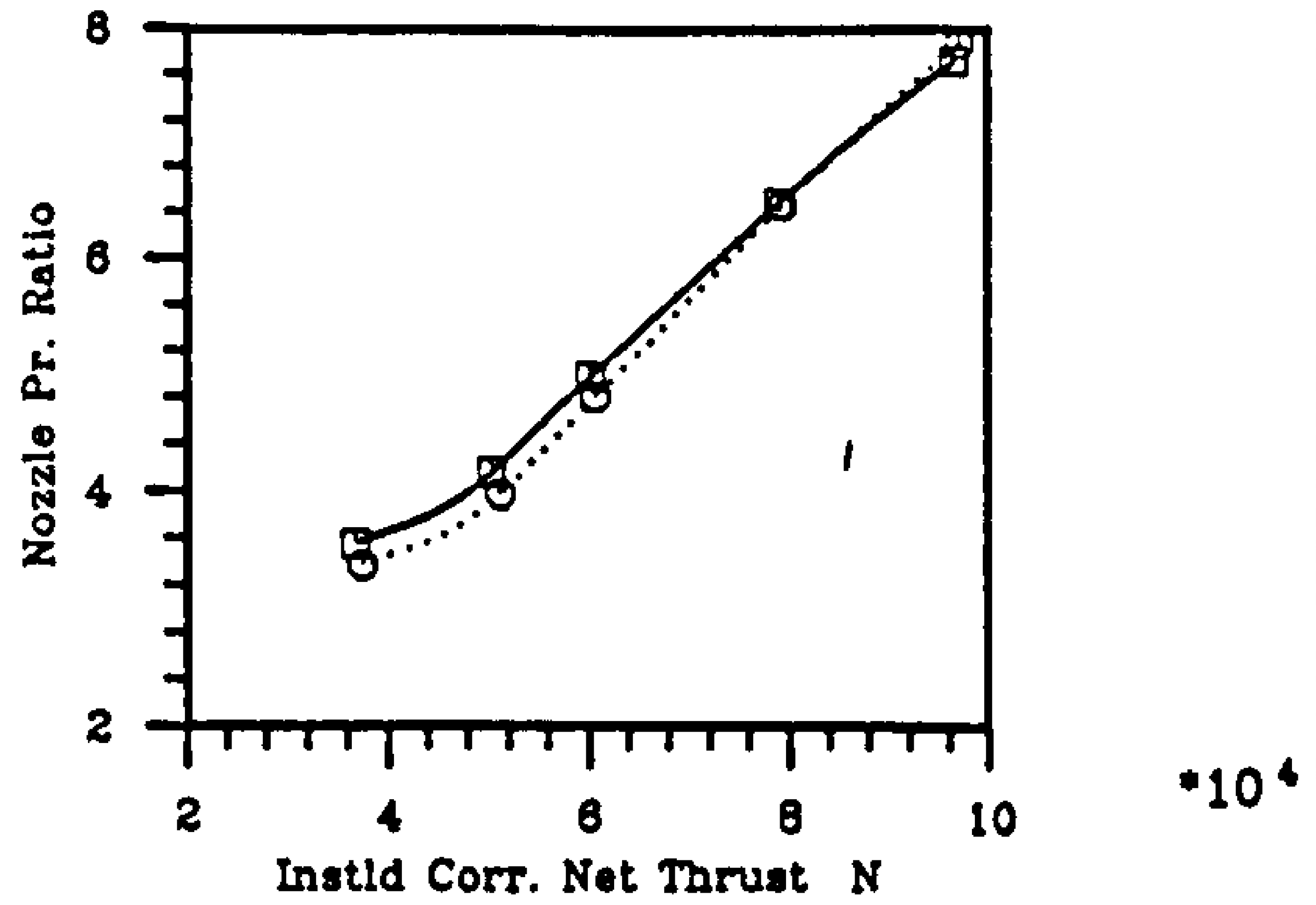



Fig. 8.18f Nozzle Pressure Ratio Variation

⊞—⊡ Fixed Comp. Vanes/ Var. LP,HP Turbine Area

Alt (m) 9144.0 ⊙····⊙ Fixed Comp. Vanes/ Var. LP Turbine Area

Mn 0.9

because of the lower jet pipe pressure, a result of lower overall pressure ratio, Fig. 8.18f.

The saving in fuel consumption is at most 3.7 percent when both compressor operating points are fixed. This is a significant figure and therefore additional performance can be gained by controlling the engine thus. However, the additional control loop may complicate engine control and coupled with cooling problems, it may be better to forego these performance gains and control just the LP compressor to operate at its design point.

### 8.1.1 Performance with Advanced Components

Engine performance variables at the design point were chosen for both the turbojet and turbofan engines which would reflect the level of performance of future high performance gas turbines. The performance variables at the design point are given in Table 8.2.

Design Point Parameters (SLS ISA)

| | | | T/FAN ($\mu$=.7) | T/JET |
|---|---|---|---|---|
| OPR | | | 25 | 27.7 |
| Airflow | | (kg/s) | 79.6 | 71.3 |
| TIT | | (K) | 1800 | 1800 |
| Reheat Temp. | | (K) | 2100 | 2100 |
| Fan/LPC | PR | | 4.71 | 3.5 |
| Thrust | RH/Dry | (KN) | 100/62.6 | 100/74.1 |
| sfc | RH/Dry | (mg/Ns) | 45.4/20.5 | 40.7/25.9 |

Table 8.2 Performance of Advanced Technology Engines

The fan pressure ratio was optimized at the chosen TIT and overall pressure ratio while the turbojet's pressure ratio was the optimum for maximum specific thrust at the chosen TIT. The airflows were chosen to give an installed afterburning thrust of 100 KN. The turbofan is of the mixed flow type and the variable geometry turbojet incorporates a variable area LP turbine and propelling nozzle to facilitate running the LP compressor at its design point while TIT changes to modulate thrust. It was decided not to use variable geometry in the HP turbine because of the reasons given at the end of the previous section. All engines are twin-spool.

The key design variables, airflow, OPR, and TIT are quite comparable, but the performances are distinct. The engines are compared at the same flight conditions as considered earlier with the turbofan being used as a reference. The noticeable performance differences from what were observed in the previous study are commented on below.

At an altitude of 9144 m and a speed of Mach 0.9, the trends in the performance of the various components are similar to what were observed in the previous study. The VAT engine could not operate at very low power because the LP turbine operating point went off the map. At the low thrust end, the VAT engine was swallowing a higher mass flow than the turbofan because the LP compressor was able to hold the high airflow down to very low thrust settings, cf. Fig. 8.2a. The trends in LP turbine area variation are similar, with the VAT engine's area rising from a lower value from that of the fixed geometry turbojet at intermediate thrust, as thrust is reduced. The explanation of this has already been given when Fig. 8.17c was being described.

With reference to Fig. 8.4a, the turbofan's HP turbine pressure ratio remains constant throughout the thrust range but at a lower level than those of the turbojet engines. This is to be expected as the HP compressor pressure ratio of the turbojets far exceeds that of the turbofan at the design point, and with mass flows being of the same order, the HP compressor work of the turbojets is therefore higher. Unlike the previous study engines whose nozzle exit area increased as the afterburning temperature increased, the present study turbojets operate at maximum exit area throughout while the turbofan's exit area increases to a maximum at a high afterburning thrust. The nozzle pressure ratio of the turbojets therefore decreases with increase in afterburning temperature rather than remain constant as noted earlier.

The uninstalled and installed sfc-thrust curves are shown in Fig. 8.19. In comparing these with those of Fig. 8.7, it is seen that the discrepancy in dry installed sfc between the fixed geometry engines is not that great. As a result, the variable geometry engine gives competitive dry and reheated performances.

The discrepancy in supersonic performance of the turbojets relative to that of the turbofan is quite marked when Fig. 8.20 is compared with Fig. 8.8. The fixed geometry turbojet does not give any appreciable saving in fuel consumption over that of the turbofan. It appears that the saving in fuel consumption due to a higher overall pressure ratio is offset by the increase in compressor work. The variable geometry engine gives a better performance but not as great as the low technology variable geometry engine studied earlier.

Once again, the trends in performance of the components are in general similar to what were observed for the previous engines. The turbofan could be run at maximum TIT in this case. With reference to Fig. 8.9c, in the present study,

## Performance With Advanced Components

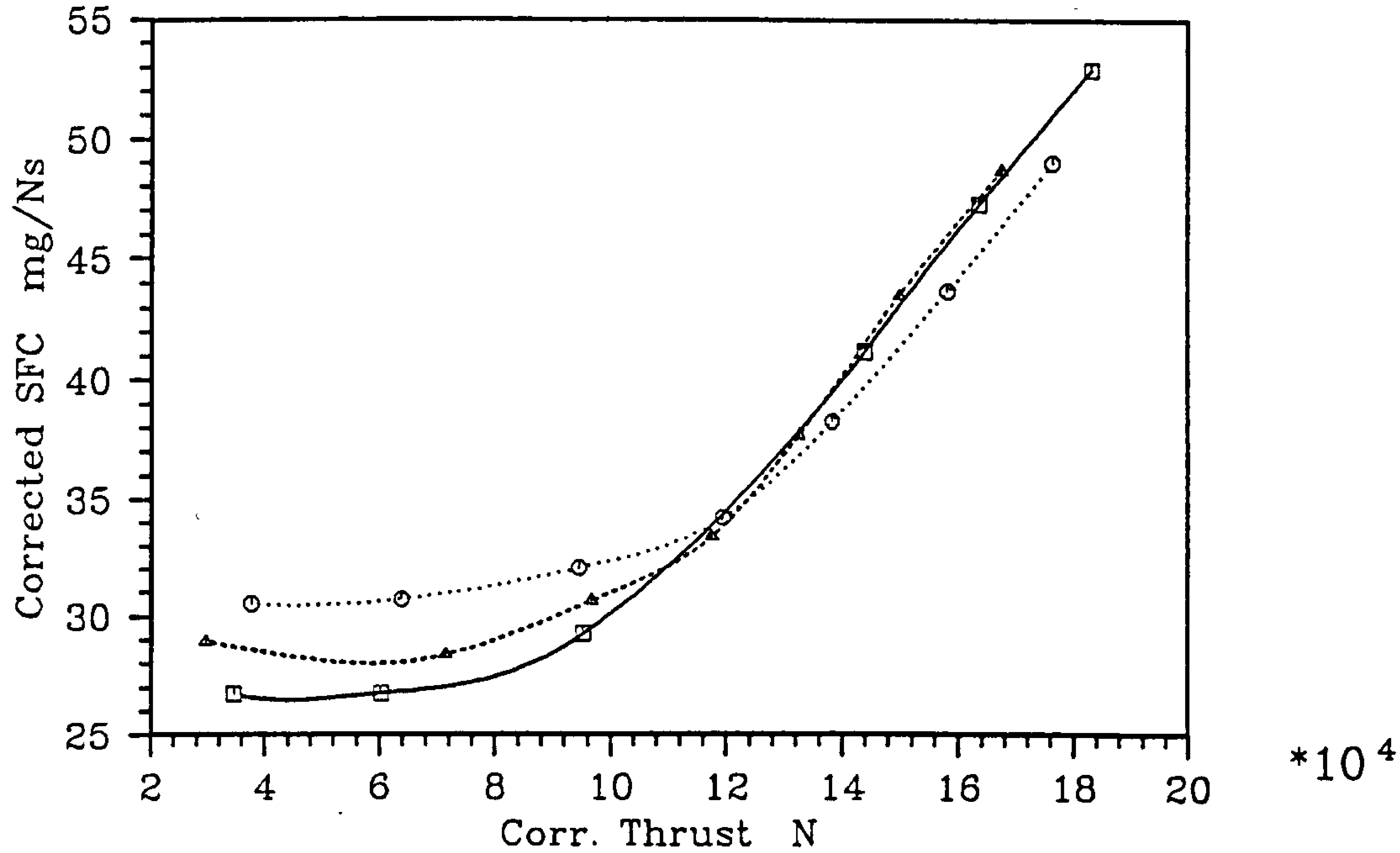



Fig. 8.19a Fuel Flow Requirement For Unit Thrust

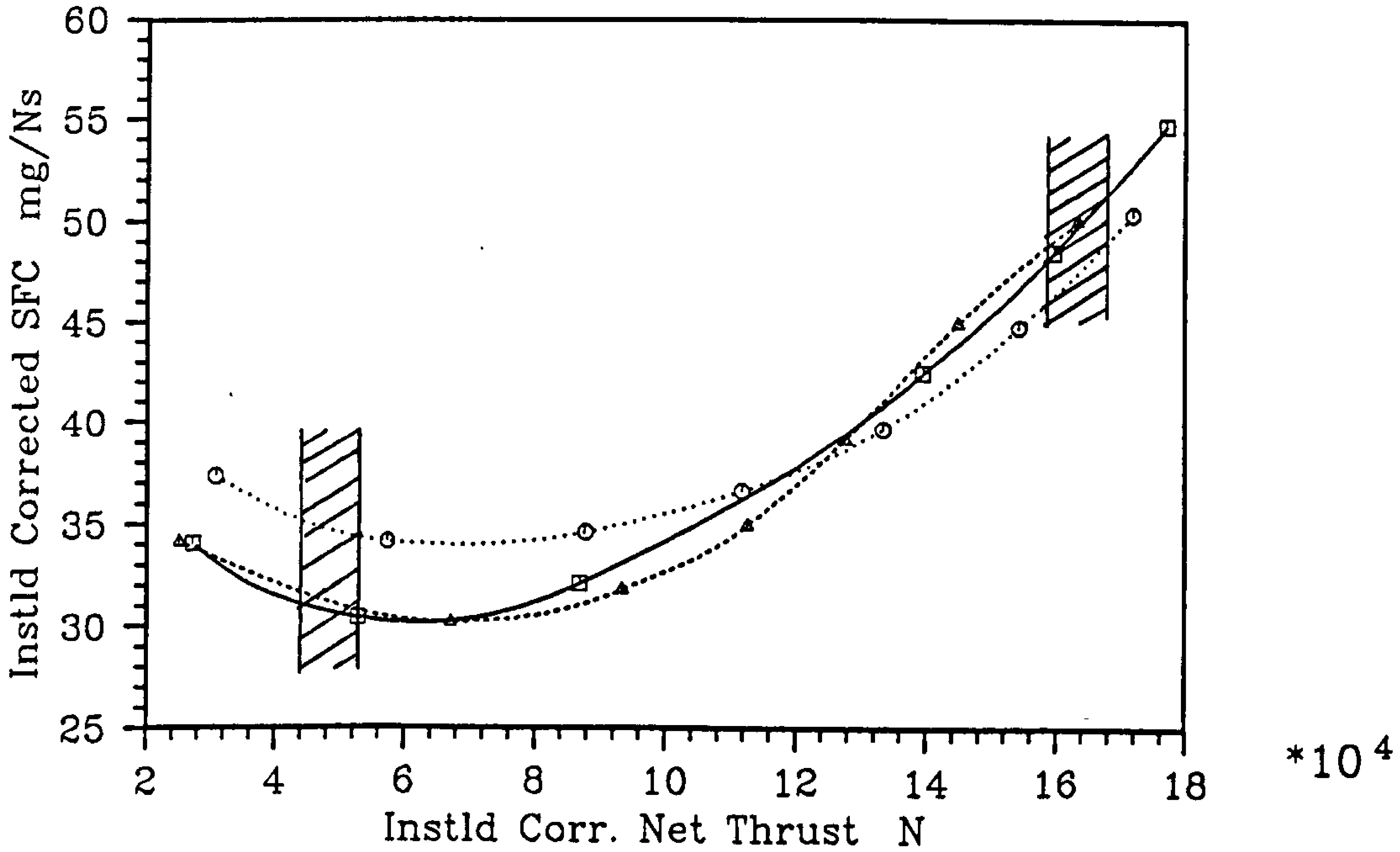



Fig. 8.19b Fuel Flow Requirement For Unit Thrust

Alt (m) 9144.0
Mn 0.9

□—□ Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
○····○ Fixed Comp. Vanes/ Fixed Turbine Area /TJET
▲----▲ Fixed Comp. Vanes/ Var. LP Turbine Area /TJET

## Performance With Advanced Components

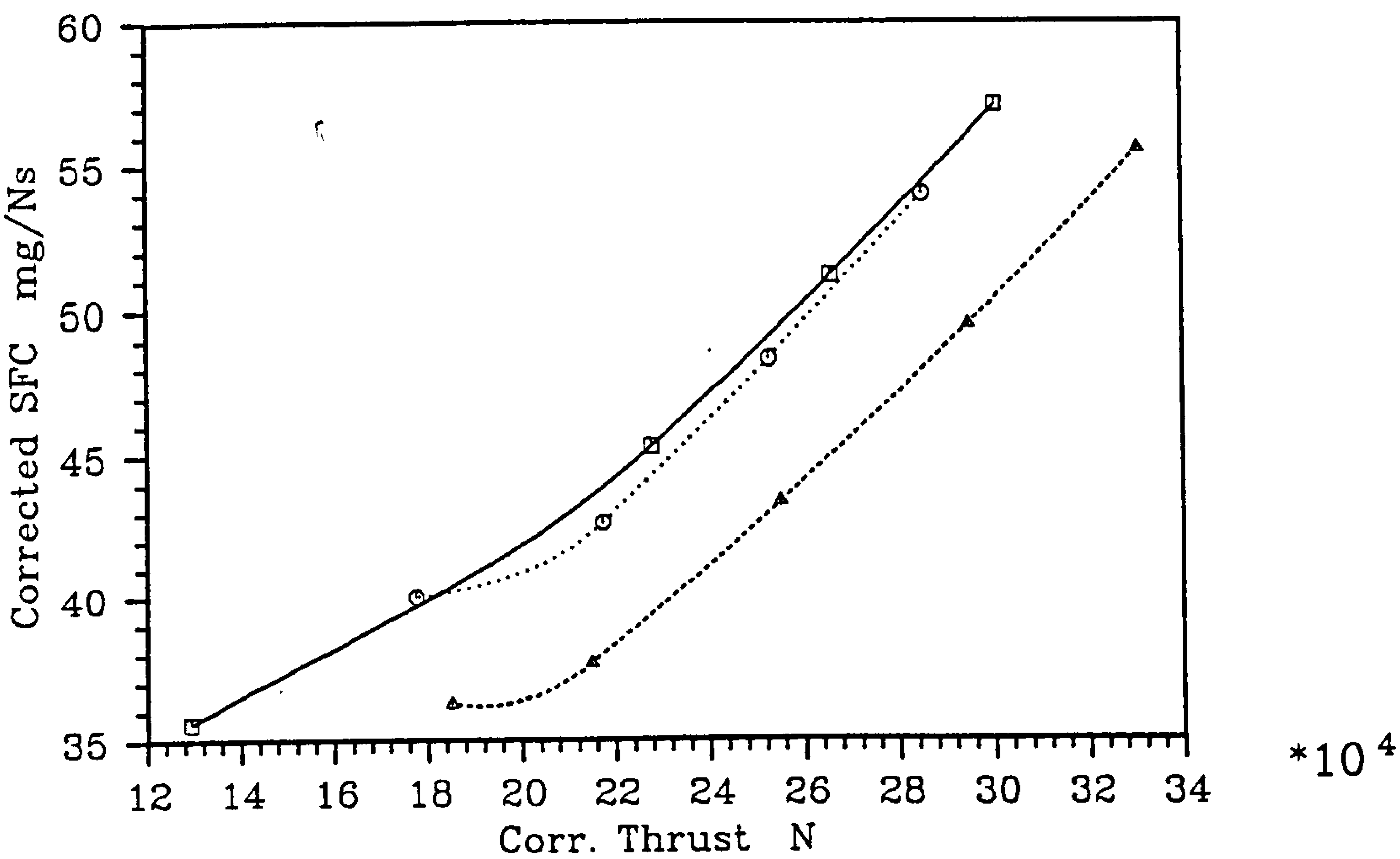



Fig. 8.20a Fuel Flow Requirement For Unit Thrust

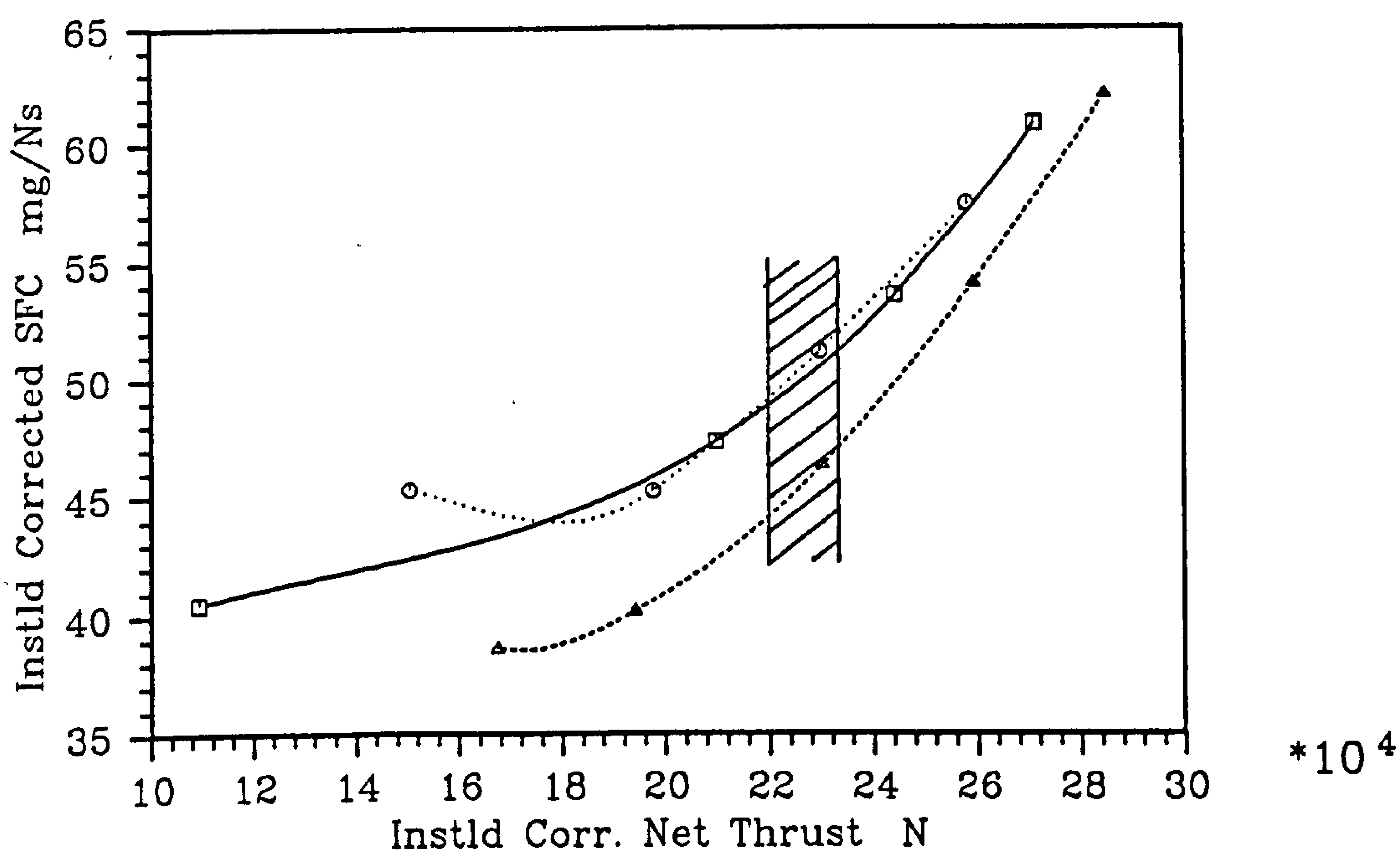



Fig. 8.20b Fuel Flow Requirement For Unit Thrust

Alt (m) 15240.0
Mn 1.6

- □—□ Fixed Comp. Vanes/ Fixed Turbine Area /TFAN
- ○····○ Fixed Comp. Vanes/ Fixed Turbine Area /TJET
- ▲----▲ Fixed Comp. Vanes/ Var. LP Turbine Area /TJET

the airflow swallowed by the variable geometry engine is higher than that swallowed by the turbofan but the turbofan's fan pressure ratio is higher in order to satisfy the mixing requirements. As a result, the HP turbine pressure ratio falls below that of the VAT engine, cf. Fig. 8.10c. Both the turbofan and the variable area turbojet operate at identical afterburning pressure ratio. The fixed geometry turbojet has a higher afterburning pressure ratio attributed to the higher jet pipe pressure. The trends in afterburning nozzle pressure ratio are different for the previous study engines as seen in Fig. 8.11c.

The acceleration performance of the present study engines is also different from what was observed for the previous ones. As can be seen from Fig. 8.21, the fixed geometry engines give almost the same thrust at all flight speeds as they have similar mass flows. The turbojet has a lower sfc due to its higher thermal efficiency. The variable geometry engine develops lower thrust up to the start of its supersonic leg giving similar fuel burn as its fixed geometry counterpart. At supersonic speeds, its thrust and specific fuel consumption are higher than those of the fixed geometry engines, a consequence of its higher specific thrust.

The fixed geometry engines can operate at maximum TIT at all speeds whereas the VAT engine has to drop TIT at very high Mach numbers as its operation is HP spool speed limited. These trends were reversed for the previous study engines as was noted in Fig. 8.14a. Once again, the HP compressor pressure ratio of the turbofan is lower to enhance mixing resulting in lower HP turbine pressure ratio. The nozzle exit areas are at their maximum throughout the acceleration.

It appears that the performance gained by controlling a turbojet such that constant airflow is maintained as thrust is modulated diminishes with a higher level of design point performance. There is an improvement in both the uninstalled and installed performances with a more significant improvement at low power and cruise conditions. A sacrifice is made on maximum power supersonic fuel consumption but the thrust is significantly higher. This engine could meet the thrust and fuel requirements of a mixed mission aircraft and it may be necessary to limit the level of design thrust.

### 8.1.2 The Variable Area Turbine Turbofan

The same investigations as those carried out for the VAT turbojet can be done for a turbofan engine to find out what sort of performance gains can be obtained when equipped with one or more variable area turbines to enable constant

# Performance With Advanced Components

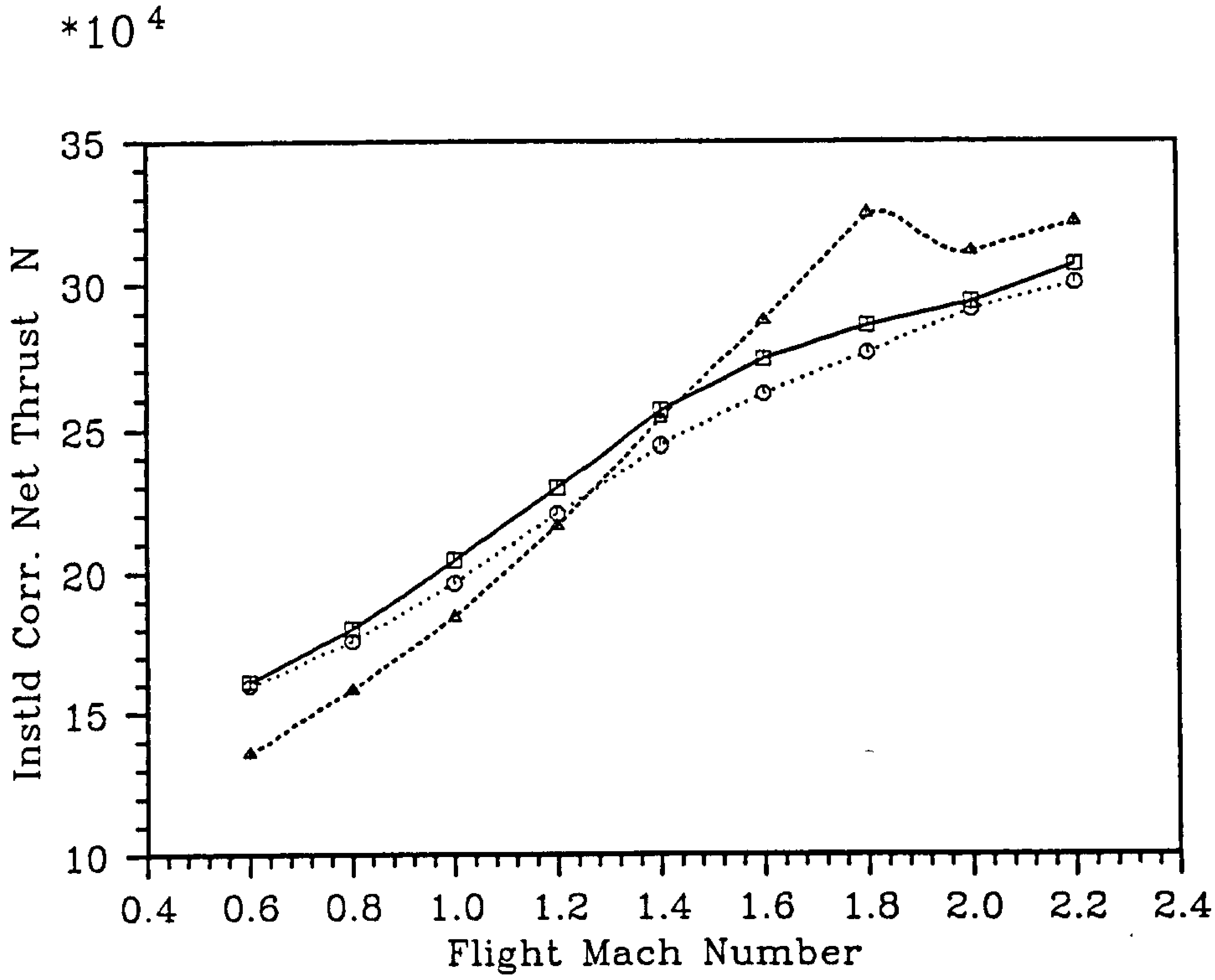



Fig. 8.21a Thrust Requirement at Acceleration

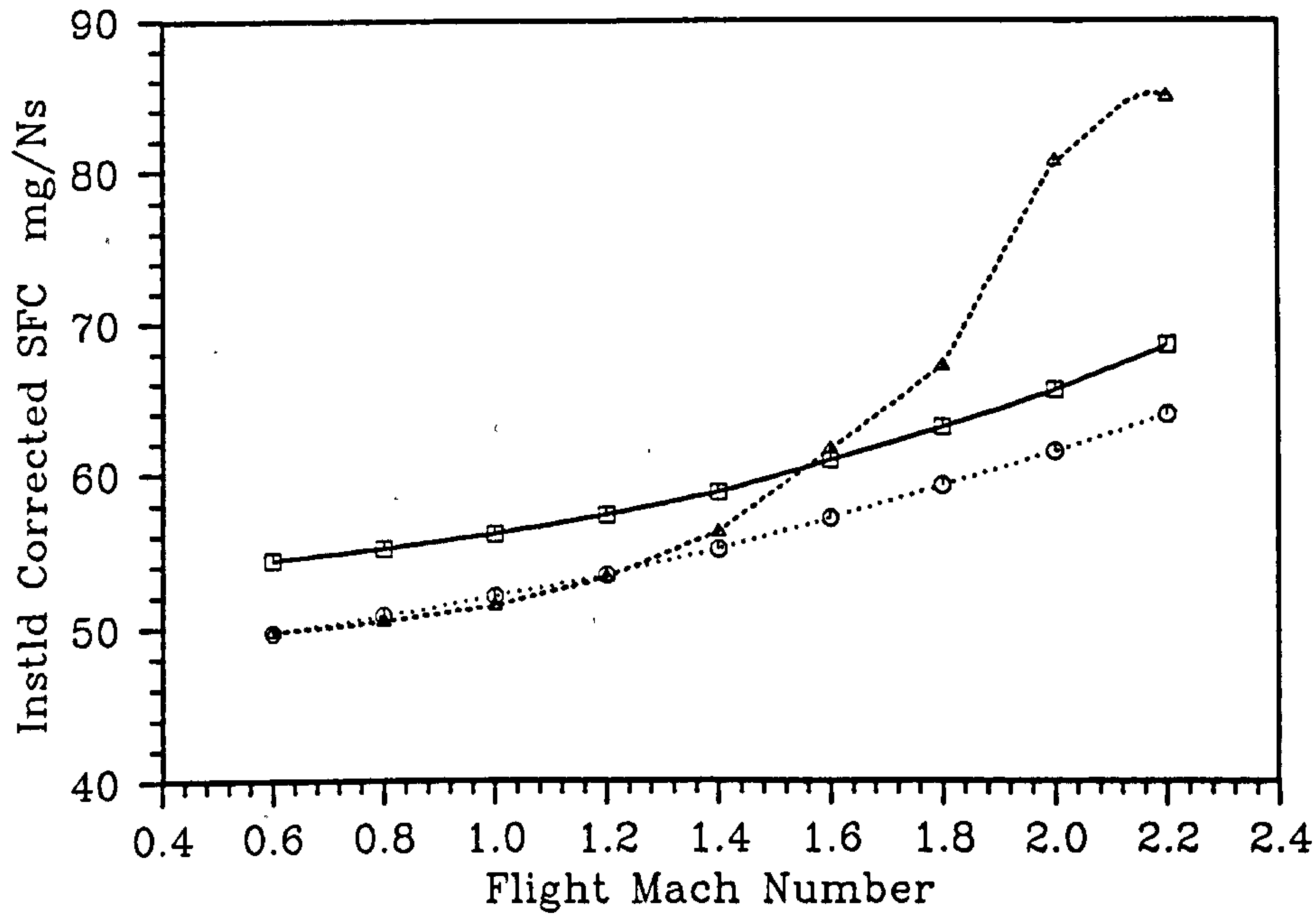



Fig. 8.21b Fuel Flow Requirement For Unit Thrust at Acceleration

Fixed Comp. Vanes/ Fixed Turbine Area /TFAN

Alt (m) 11000.0 Fixed Comp. Vanes/ Fixed Turbine Area /TJET

Fixed Comp. Vanes/ Var. LP Turbine Area /TJET fan and/or compressor design point operation as thrust is modulated. Such an analysis has been carried out for a twin spool separated-flow turbofan [33] and the results indicated an improvement in performance over that of the conventional engine. This was also investigated here at Cranfield for both a mixed and an unmixed two-spool turbofan with either the fan or both fan and compressor operating points fixed.

Both types of engine have an inherent disadvantage when controlled thus. The unmixed turbofan is more suited for this type of control as there is no static pressure matching requirement that will limit its operational ceiling. Since the conditions of the gas remain almost constant in the bypass stream, the cold thrust is fixed and therefore only the hot thrust can be modulated. A limit on thrust attenuation would occur when thrust reduces to that developed by the cold stream. A complex control system would have to be designed which would revert the engine back to the fixed mode of control when this thrust limit is sensed. It was found out that the higher the bypass ratio, the lower is the percentage of thrust modulation achieved as more thrust is produced by the fan stream.

The static pressure balance requirement of the mixing process in the mixed flow turbofan imposes a limit on the maximum altitude and minimum power at which the aircraft can fly. Since the total pressure in the bypass duct decreases as the altitude increases at a given flight Mach number due to the constant pressure ratio at which the fan operates, it becomes more and more difficult to reduce the static pressure of the gases at LP turbine exit to a level that will ensure mixing. Also, as TIT is reduced at constant flight conditions the Mach number at LP turbine exit increases resulting in reduced static pressure, and at certain power levels it may become impossible to keep the exit Mach number at a value which will give a static pressure equal to that in the bypass stream which remains constant.

These problems can be alleviated or solved by throttling the flow in the bypass duct or by using a variable area mixer, or both. Whatever method employed will certainly add some degree of complexity to the control system. We may be able to get away without using any of these methods if only the fan is controlled to operate at its design point. If the compressor too is controlled for fixed design point operation, then it is a must that a means be provided to reduce the cold stream static pressure as this can no longer remain fixed as thrust is changed. The area of the cold stream should decrease as thrust is reduced in order to raise the Mach number of the airflow which then allows the static pressure to fall. When this area is choked, the engine can no longer be operated with compressor and fan operating points fixed.

The turbojet on the other hand does not encounter these problems and therefore is more suited to this type of control.

## 8.2 Variable Geometry for V/STOL Operation

The successful design of a propulsion system for vertical and short take-off and landing operation is one of the most tasking problems that gas turbine propulsion engineers will ever seek a solution to. There is an increasing need for such aircraft systems and the requirements for successful operation are getting all the more stringent. In the military arena, the Navy's interest lies in operations from smaller destroyer-sized vessels whilst the Air Force is interested in maintaining operation from a bombed or damaged runway. Civilian applications include operation from city centers, rescue missions, and transportation into undeveloped areas.

Regardless of the application, the propulsion system and flight controls most provide the required thrust and thrust control during takeoff, landing, and transitional flight from engine support to wing support and vice versa. Various types of propulsion system, each with its complex controls, have been proposed for the three types of V/STOL aircraft, viz., rotorcraft, subsonic cruise, and supersonic cruise. The supersonic cruise arena has not yet been entered but some promising cycles have been looked at [22,32]. The free turbine turboshaft is universally used as the propulsion system for helicopters; some successful designs have been proved for subsonic cruise. However, future applications will require that thrust changes should take place within shorter time periods for emergency and manoeuvre.

Several methods have been looked at which can produce large changes in thrust over a short time period. These methods are described in [61]. The two most promising methods for fast thrust response are the variable inlet guide vane control and the variable pitch rotor (VPR) controls.

### Performance Requirements

The performance requirements that will be looked at are those laid down by the US Navy for their V/STOL Type A aircraft. The requirements are basically described in Fig 8.22 where it is assumed that the aircraft is approaching to land from a speed of 64 m/s. At takeoff and landing, a high response thrust is needed to maintain attitude and altitude, for example, in large scale air turbulence. A magnitude of $\pm 25$ percent of the takeoff thrust was prescribed for pitch and/or roll control, and to achieve this thrust control quickly, fan speed must be maintained constant. In addition to this, there is the requirement that during transition from forward to vertical flight, that is, on the approach to landing, a thrust control of ±25 percent of the takeoff thrust must also be obtained for pitch and/or roll control. The approach with control should also be accomplished at constant fan speed and at the same speed as the takeoff/landing with control. If the nominal approach thrust is taken as 50 percent of the takeoff thrust, then the required thrust range varies from 25 to 125 percent of the takeoff thrust at constant speed.

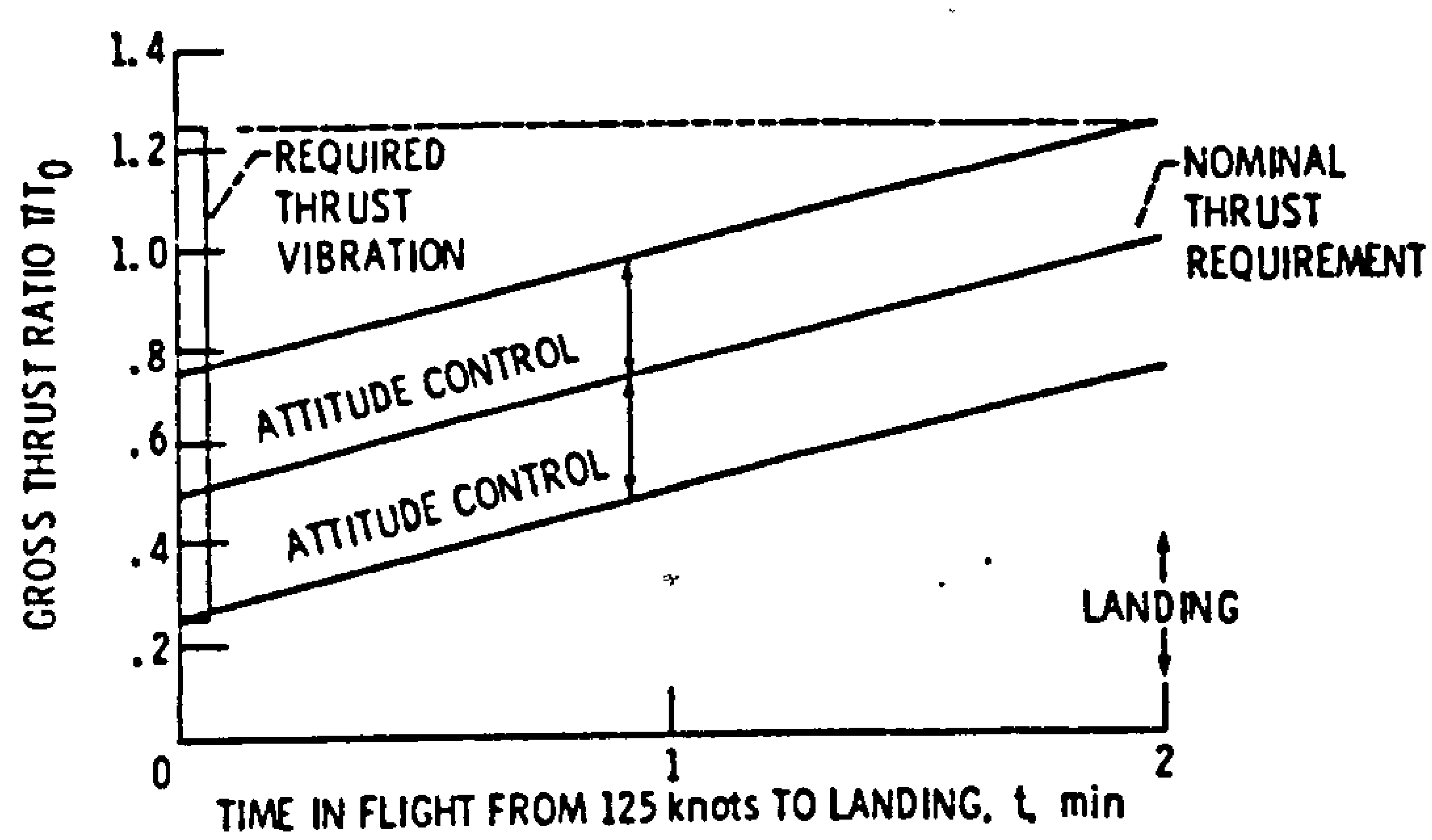



Fig. 8.22 Landing thrust requirement of a V/STOL aircraft [61]

A mission constraint is that in the case of one engine inoperative (OEI) in a twin engine installation, the core and the LP turbine of the operable engine should also be able to drive the fan of the inoperable engine to maintain balanced thrust to achieve a vertical landing. Therefore, the core size is determined by this OEI requirement.

## Rapid Thrust Modulation Methods

As was mentioned earlier, both the VPR and VIGV controls have been proposed to meet the rapid thrust requirements of V/STOL aircraft. The effectiveness of these two methods have been investigated and their relative merits compared [16]. It was found out that the variable pitch rotor is more effective in covering the required thrust range. However, since no data was available for a VPR fan, only the VIGV fan will be considered here. Also for other reasons which will be mentioned in the next chapter, a VIGV fan is more desirable for this type of application.

## The Variable Inlet Guide Vane Fan

Variable inlet guide vanes have been used for almost three decades as a means of improving the off-design performance of gas turbines. The vane row, and in some cases one or more stator rows, are scheduled with corrected speed to improve the surge characteristics of an engine. This is a proven technology and therefore, the use of VIGVs to provide thrust control at constant speed is an extension of the art and may not present great risks or development problems.

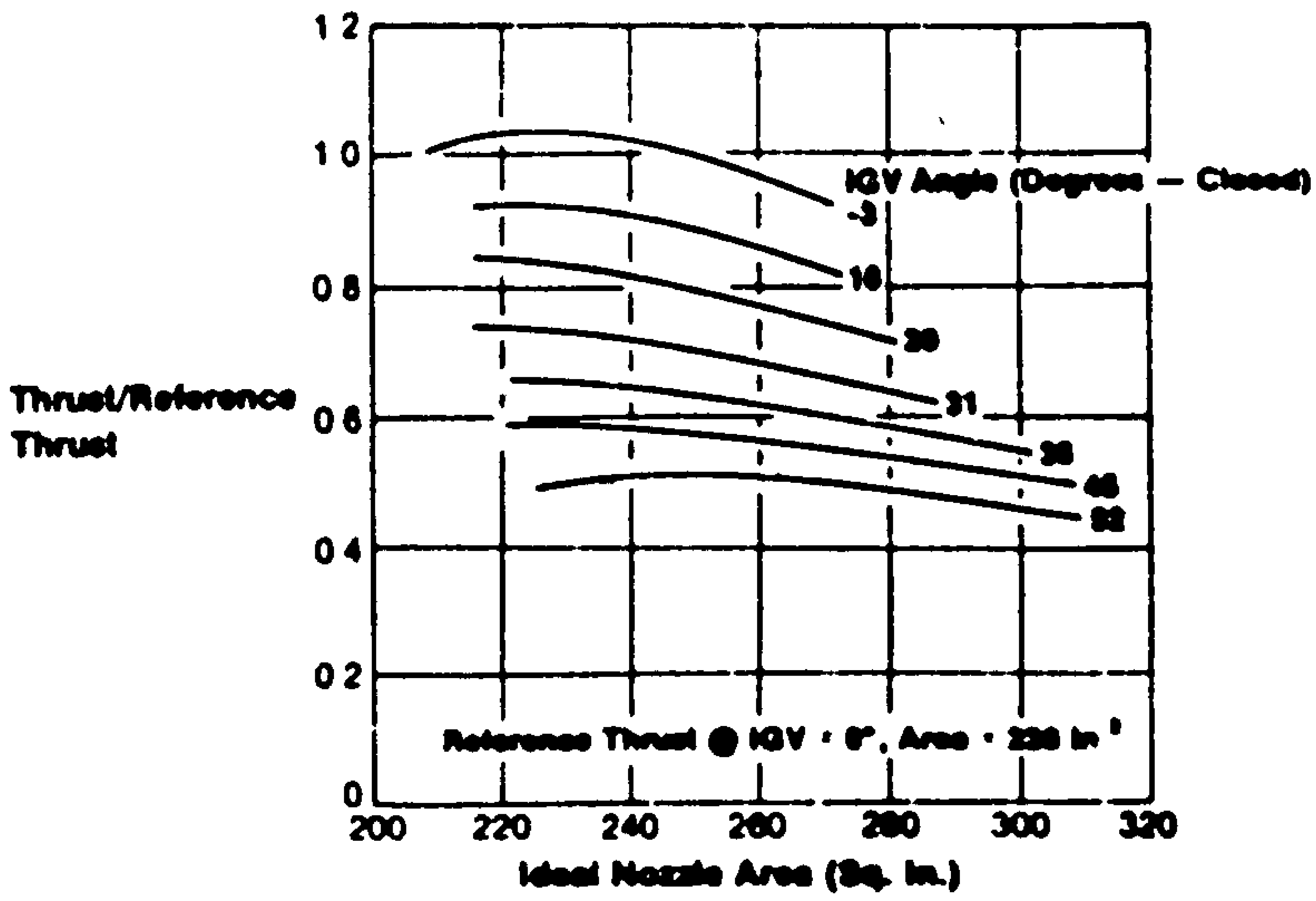



Fig. 8.23 Thrust modulation capability of a VIGV fan engine [46]

The use of VIGVs to modulate thrust at constant speed has been successfully demonstrated and it may be necessary to schedule some other variables such as nozzle throat area or even make small changes in speed to cover the entire thrust range. Fig. 8.23 depicts the possible thrust variation that can be obtained by the use of VIGVs and nozzle area change while Fig. 8.24 illustrates the effect of spool speed on VIGV thrust attenuation. Time constants of about 0.4 seconds are generally regarded as being sufficient to provide the desired thrust change. Typical transient response characteristics of a VIGV controlled turbofan are shown in Fig. 8.25 and it is seen that the rapid thrust response could easily meet the requirements imposed by aircraft demands.

### 8.2.1 Performance of Study Engines

Table 8.3 shows the design point performance of the study engines. Both engines have identical component and overall performance. With the VIGV engine, thrust is modulated at constant speed until a vane angle limit is reached whereupon some form of engine control is deployed to further modulate thrust. The compressor speed is allowed to vary. The effect of keeping core speed fixed as well will be commented on later. The conventional engine uses speed

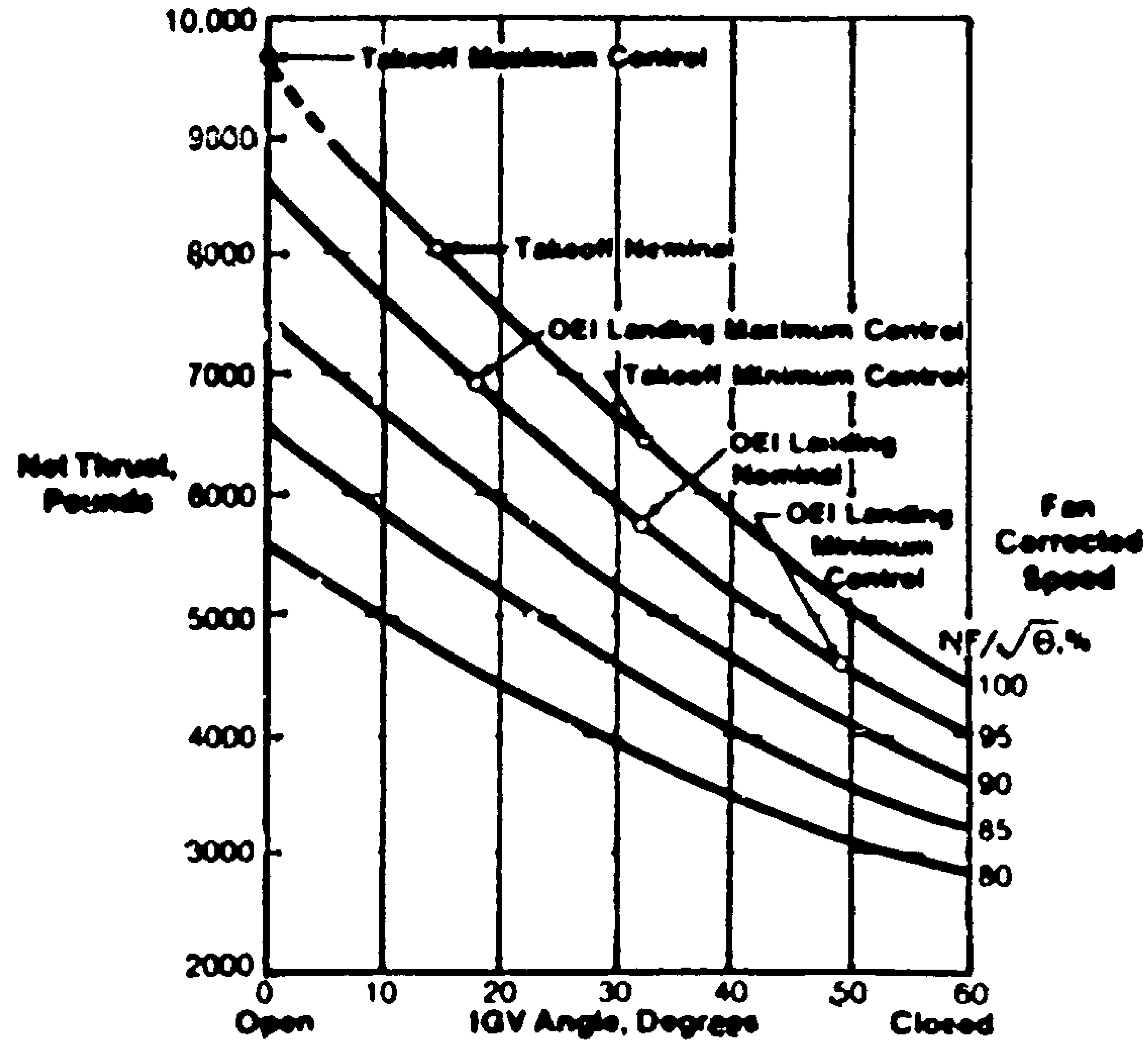



Fig. 8.24 Effect of speed and VIGVs on thrust attenuation [46]

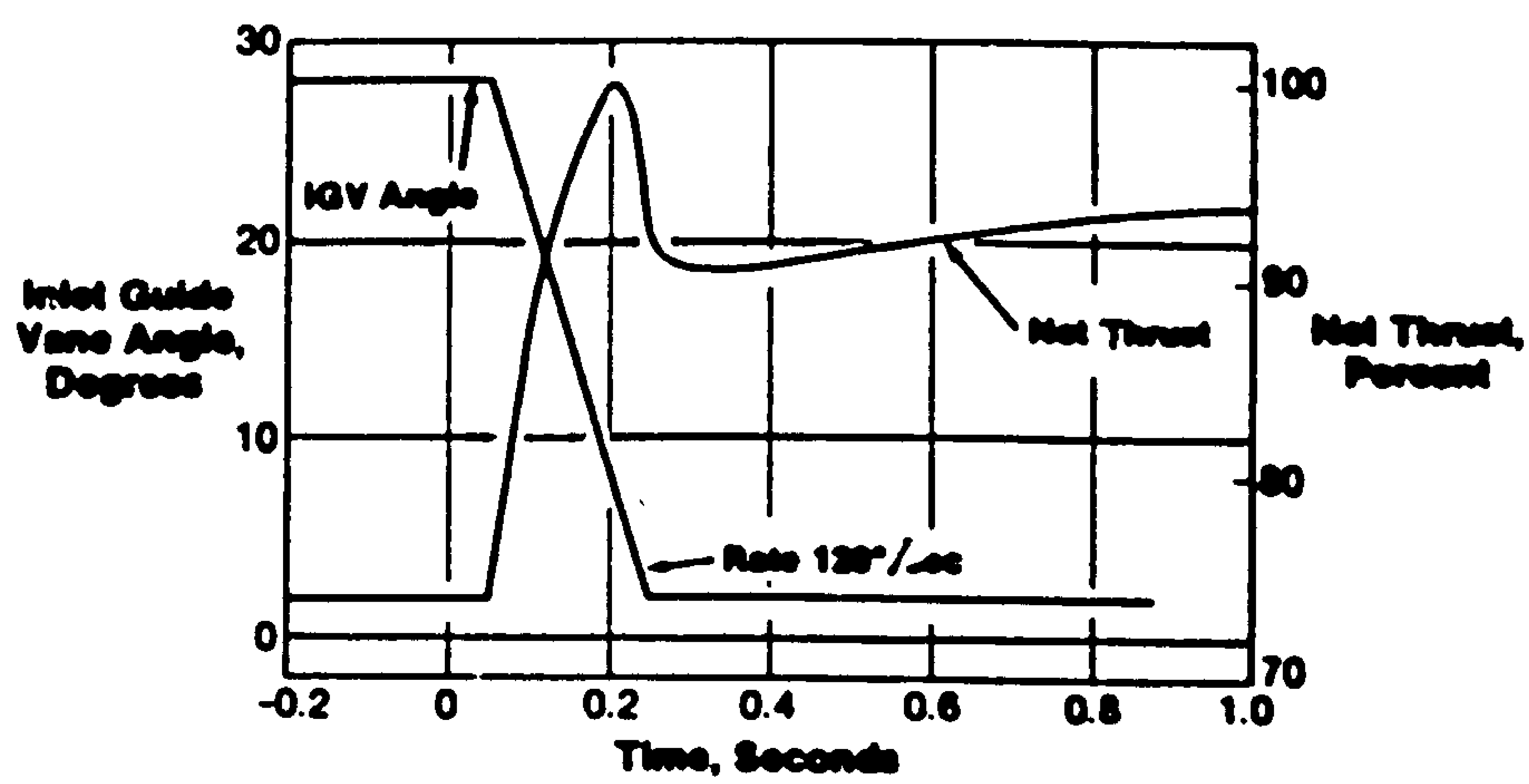



Fig. 8.25 Transient thrust response capability with variable inlet guide vanes [46]

control to effect thrust change. Spool speeds and vane angles were chosen such that maximum excess thrust conditions for vertical flight would occur at the maximum limit of spool speed and vane angle. No overspeeding of shaft was considered even though this may be acceptable for the short time periods associated with vertical flight. A ceiling of 1800 K was placed on TIT.

Table 8.3 V/STOL Performance at Design Point (SLS ISA)

| | | VIGV | | Conventional |
|---|---|---|---|---|
| Bypass Ratio | | | 1.5 | |
| Airflow | (kg/s) | | 191 | |
| TIT | (K) | | 1600 | |
| Fan PR | | | 3.0 | |
| OPR | | | 25 | |
| Thrust | (KN) | | 100 | |
| sfc | (mg/Ns) | | 17.0 | |
| $N/N_{max}$ fan | (%) | 100 | | 91 |
| $N/N_{max}$ comp | (%) | 93 | | 95 |
| $IGV_{fan}$ | (deg) | 2 | | 0 |

The landing thrust requirements were easily met by both engines. The conventional engine required a spool speed change of about 20 percent to cover the nominal ±25 percent thrust range, Fig. 8.26a, while the VIGV fan had to rotate its vanes through 40 degrees to cover the same thrust range, Fig. 8.26b. Both fans operated along the same operating line, Fig. 8.26c, but the lower efficiencies at which the VIGV fan operated resulted in higher TITs, Fig. 8.27a, the consequence being a higher sfc, Fig. 8.27b.

The approach thrust requirements were more difficult to meet. The flight conditions at the onset of the approach are an altitude of 500 meters and a Mach number of 0.18. The thrust requirements stipulate a thrust range of 25-75 percent of the nominal landing thrust. None of the engines could develop the minimum thrust of 25 KN. At the minimum speed of 60 percent of the designed as dictated by fan characteristics, the conventional engine produced a thrust of 30 KN, Fig. 8.28a, while the VIGV fan produced 49 KN of thrust when the vanes were fully closed, Fig. 8.28b. These values are much higher than the required minimum and so it was decided to vary core nozzle area to further reduce thrust. Fan speed of the conventional engine was held fixed at the minimum while an area increase of 70 percent was required to reduce thrust to the minimum, Fig. 8.28c.

The thrust of the VIGV fan engine could be further decreased by using some amount of variable geometry or shaft speed change. Since it was intended to modulate thrust at constant speed, variable geometry was considered in both the core nozzle and LP turbine. This exercise did not change the minimum thrust situation much. The change in thrust as a result of LP turbine area change was insignificant while a nozzle area increase up to the maximum allowable dropped the thrust down to only 33 KN. The maximum nozzle exit area was assumed to be equal to the value of the inlet area defined by the Mach number in the jet pipe at design. As was noted in Fig. 8.23, the change

# Performance Of A Rapid Response Turbofan

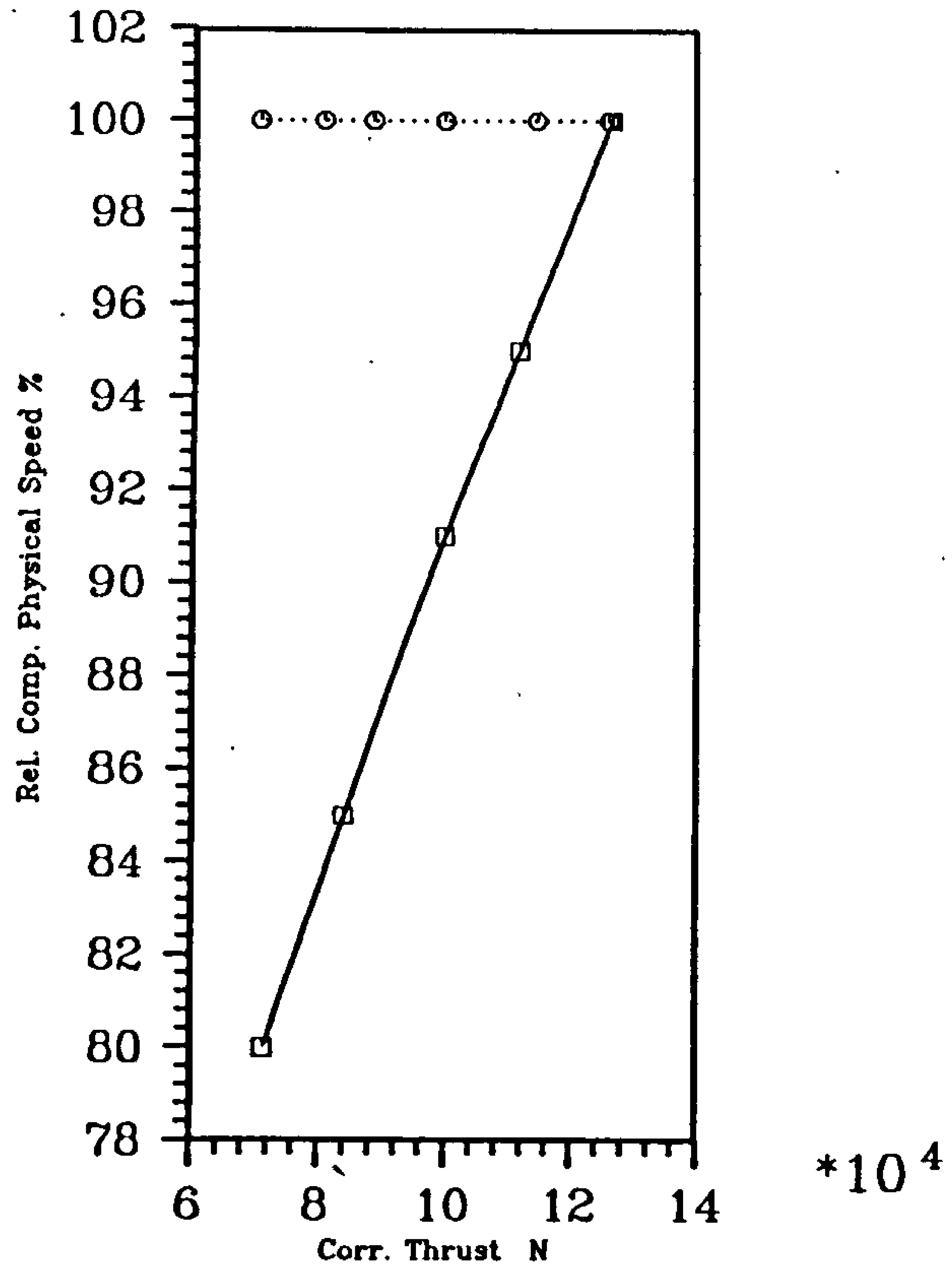



Fig. 8.26a LP Compressor Physical Speed Schedule

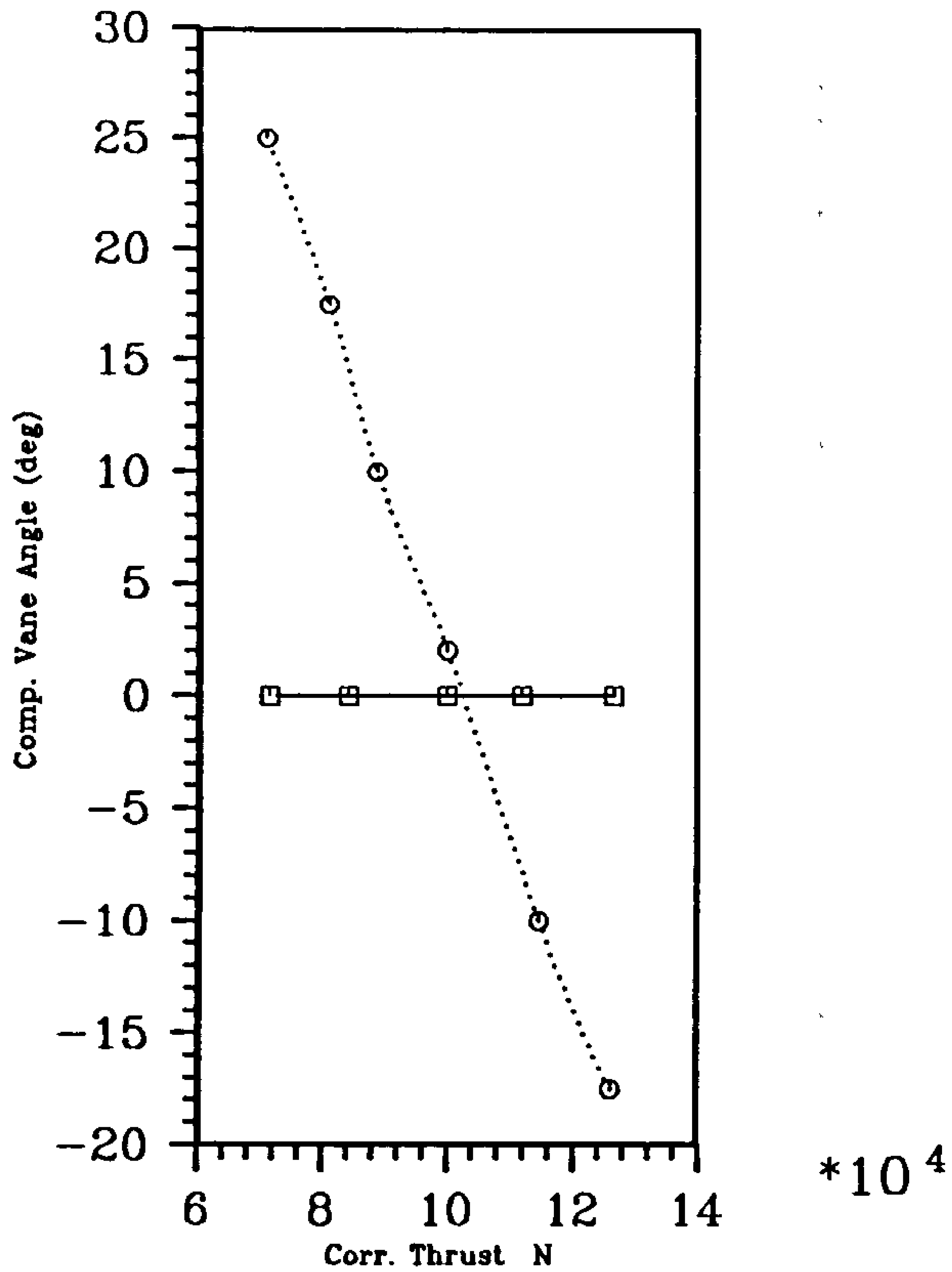



Fig. 8.26b LP Compressor Vane Angle

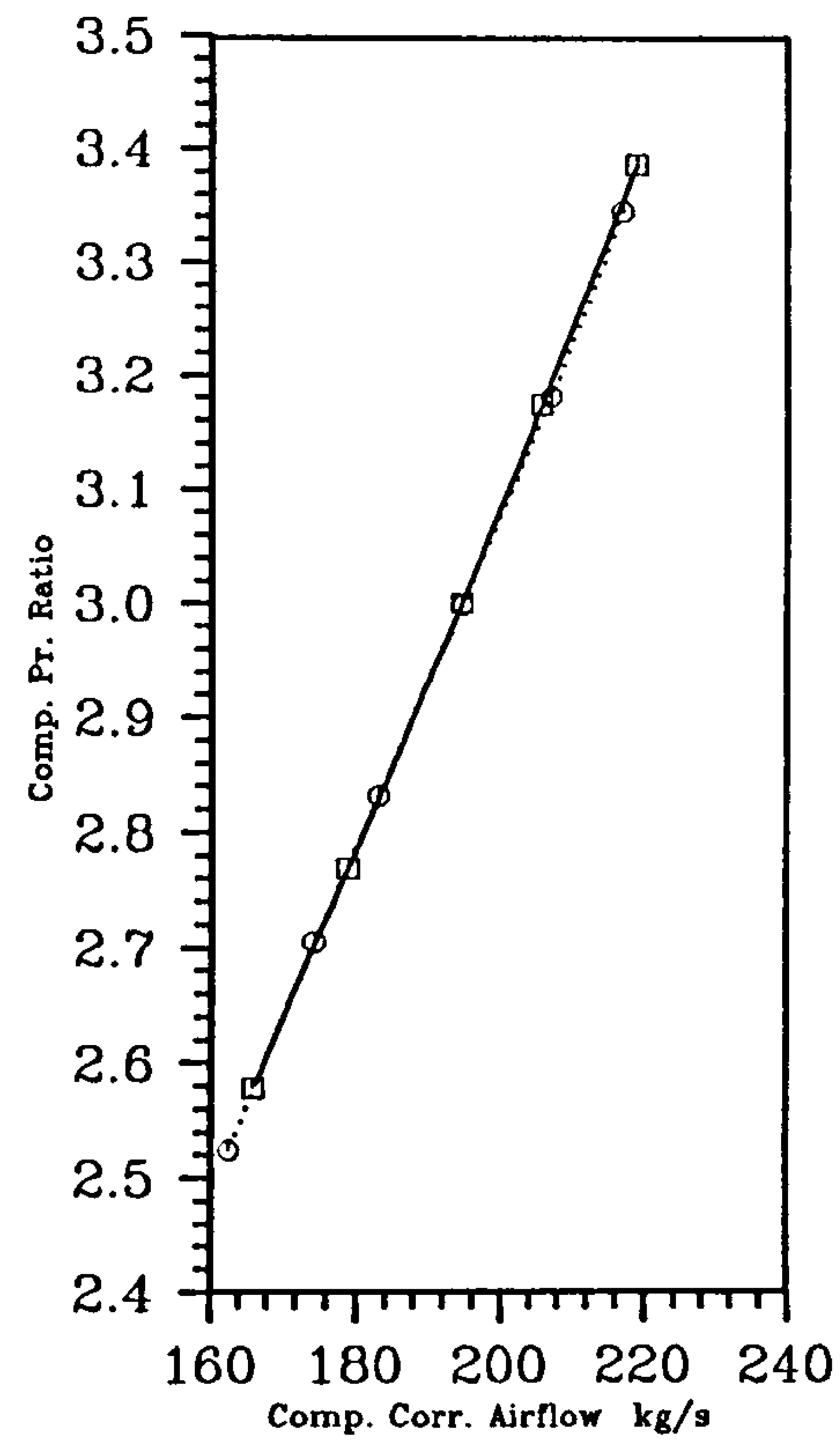



Fig. 8.26c LP Compressor Operating Line

Alt (m) 0.0
Mn 0.0

Fixed Comp. Vanes/ Fixed Turbine Area

Var. LP Comp. Vanes/ Fixed Turbine Area

# Performance Of A Rapid Response Turbofan

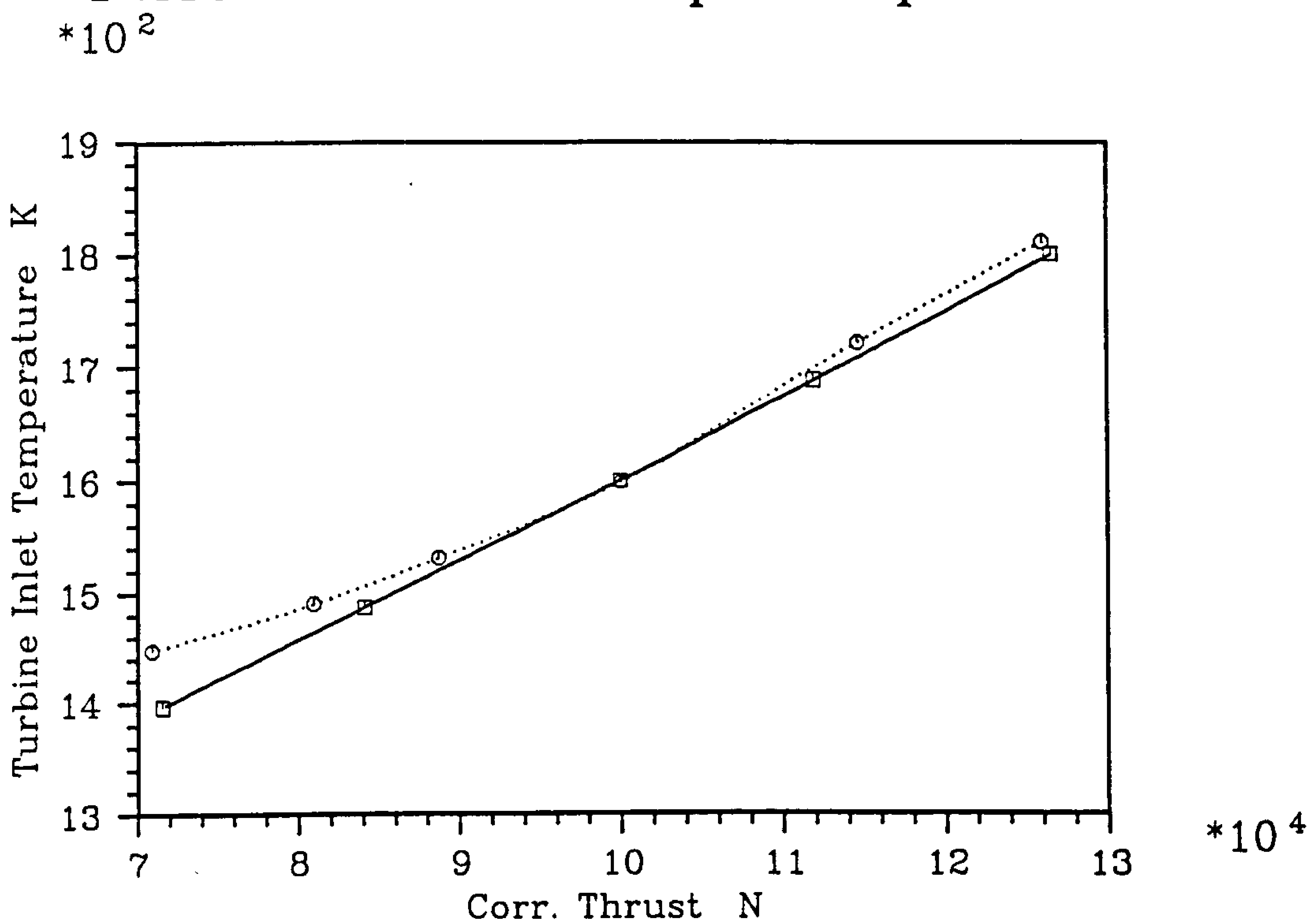



Fig. 8.27a Temp Schedule for Thrust Modulation

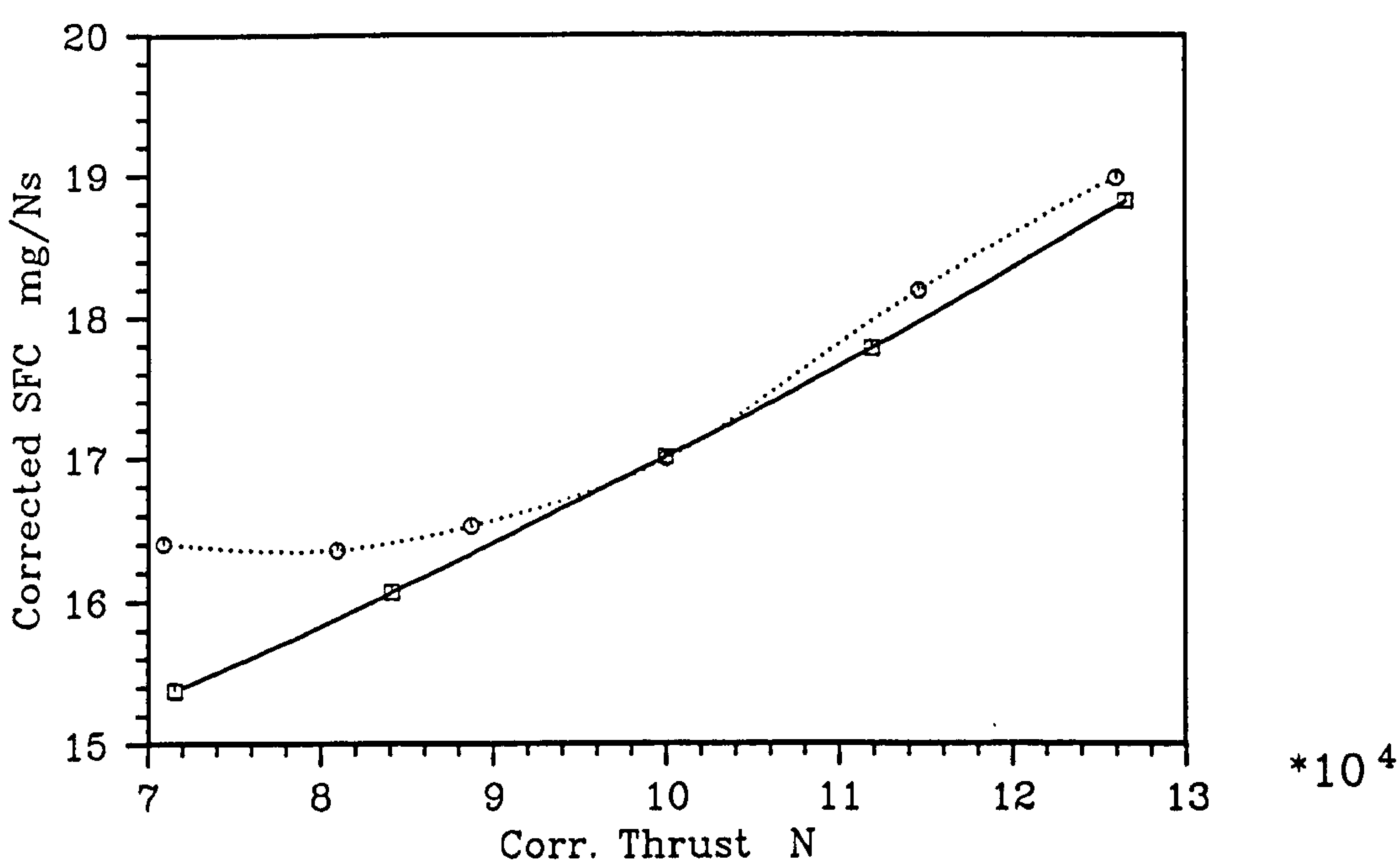



Fig. 8.27b Fuel Flow Requirement For Unit Thrust

Fixed Comp. Vanes/ Fixed Turbine Area

Var. LP Comp. Vanes/ Fixed Turbine Area

Alt (m) 0.0
Mn 0.0

# Performance Of A Rapid Response Turbofan

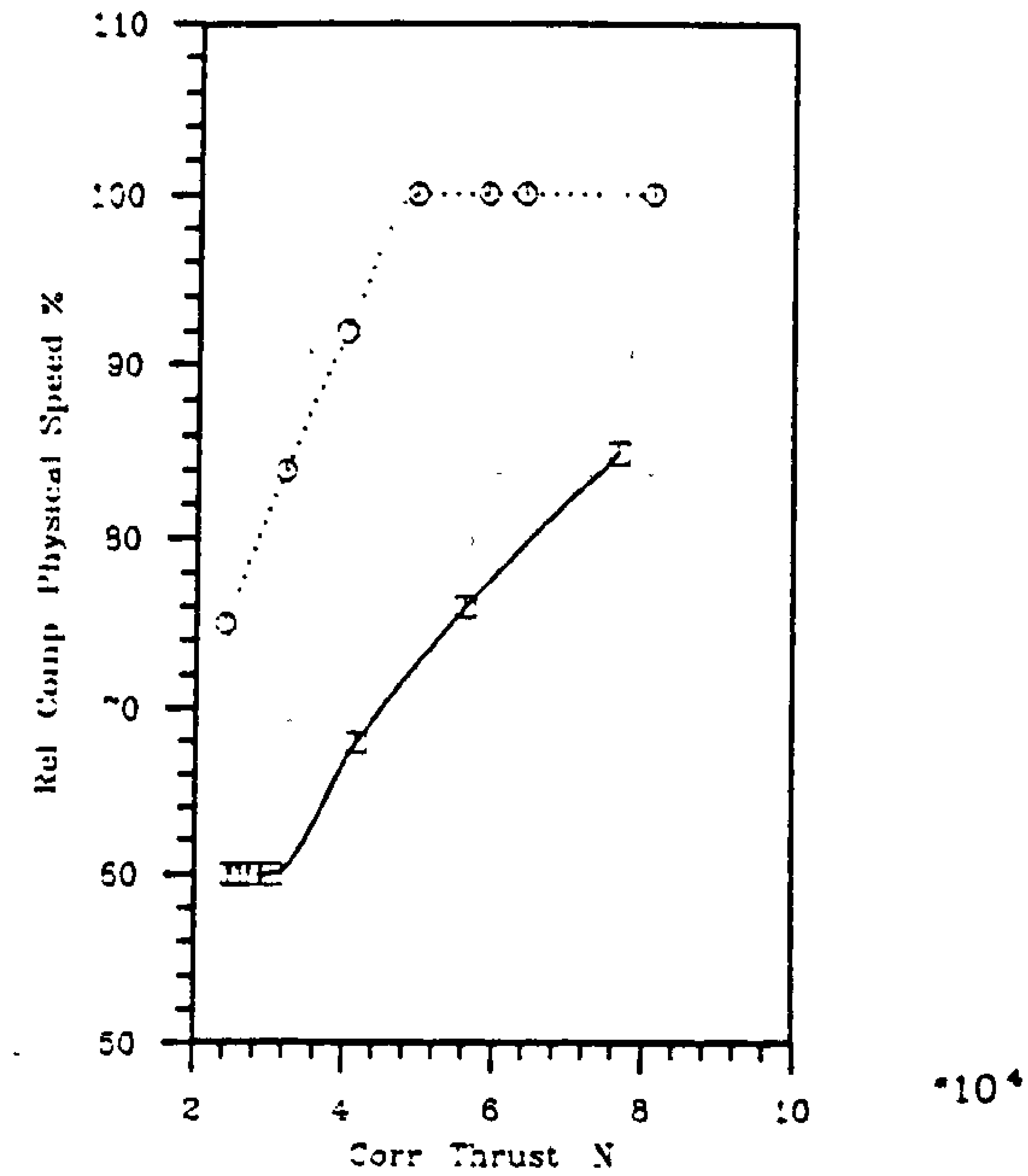



Fig. 8.28a LP Compressor Physical Speed Schedule

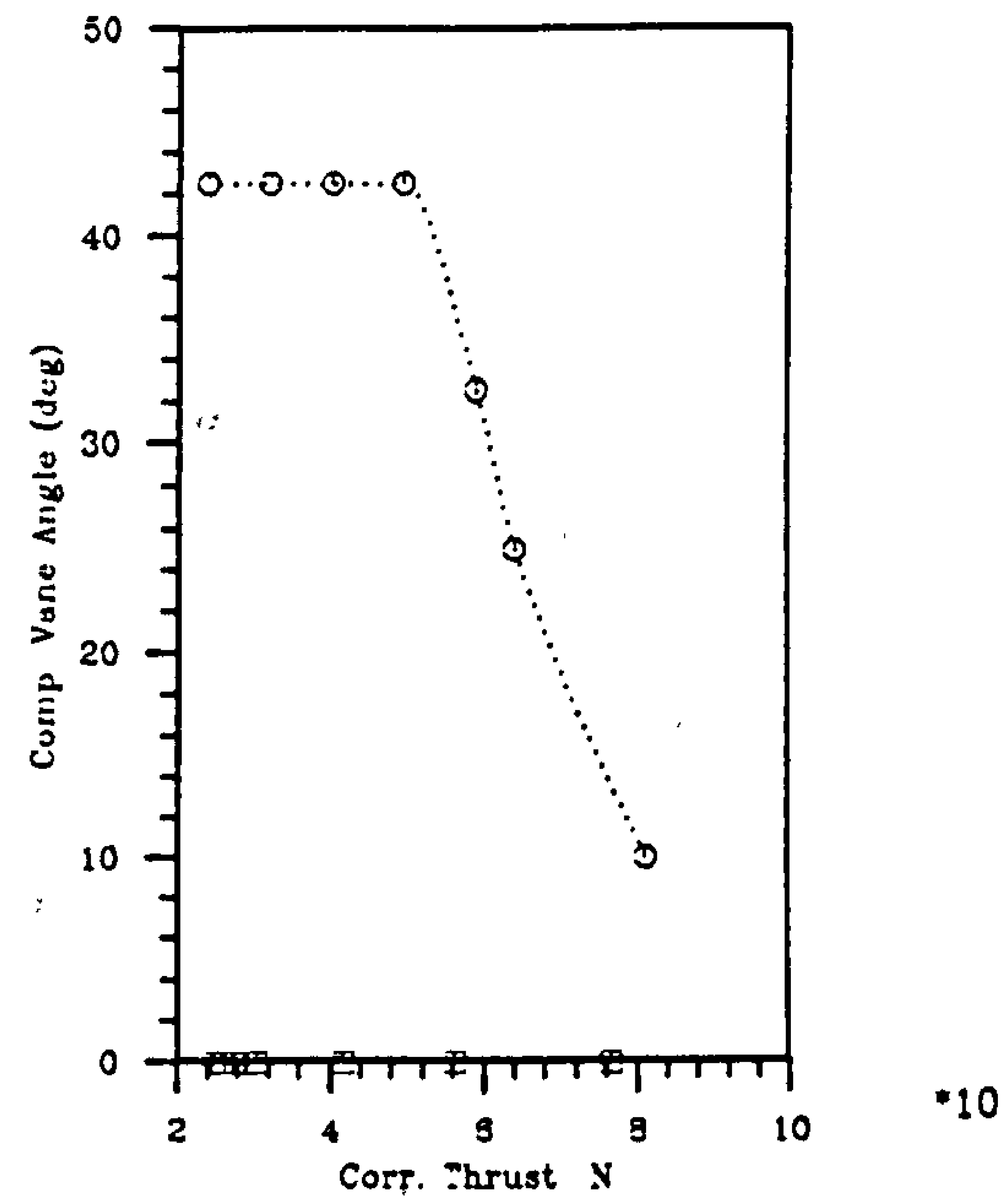



Fig. 8.28b LP Compressor Vane Angle

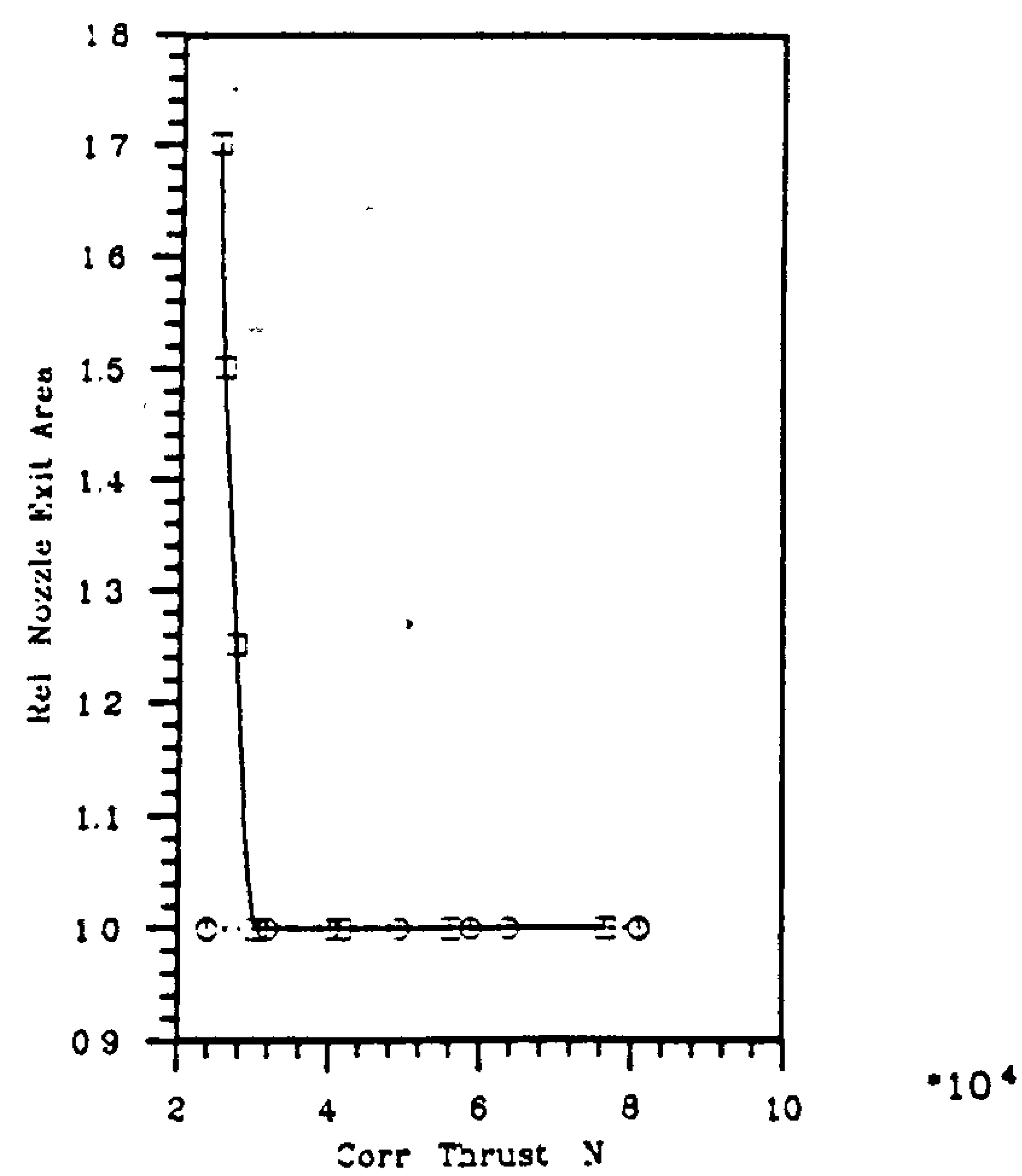



Fig. 8.28c Core Nozzle Exit Area Variation

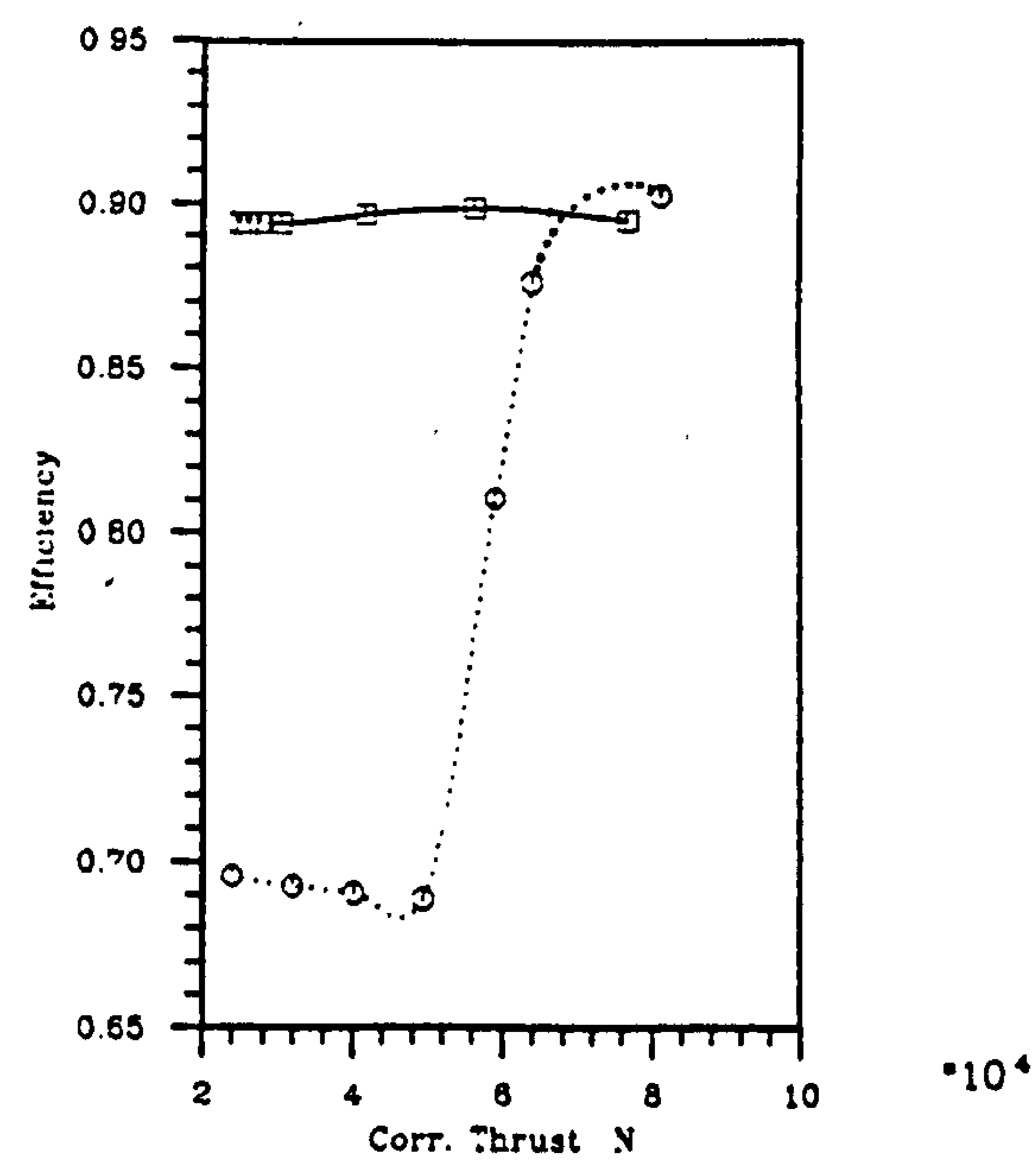



Fig. 8.28d Fan Efficiency Variation

Alt (m) 500.0
Mn 0.2

Fixed Comp. Vanes/ Fixed Turbine Area

Var. LP Comp. Vanes/ Fixed Turbine Area

in thrust with nozzle area change at a given IGV angle is quite small; hence, the results obtained were not surprising.

Owing to the low gradient of thrust change with nozzle area, a spool speed change was considered as Fig. 8.24 depicts a high thrust gradient with this mode of thrust modulation. A spool speed change of 25 percent was sufficient to produce the minimum thrust and as this speed change was considered to be too large, a nozzle area change was considered in conjunction. This exercise also turned out to be unsuccessful. Even a nozzle area increase to the maximum could not produce the minimum thrust with a speed change of 10 percent which was considered to be reasonable. A speed change of 18 percent was required with an area change of 50 percent to give the required thrust. Since engine control will become very complicated with a variable area propelling nozzle control loop added to a control system incorporating both speed and IGV control, it was decided to use speed change only to obtain the minimum thrust, Fig. 8.28a.

The very low efficiencies at which the fan operated, Fig. 8.28d, resulted in the engine operating at very high TITs as is seen in Fig. 8.29a. When variable speed was activated, the efficiency remained almost constant with TIT falling to reduce thrust. The fuel consumption for unit thrust is significantly much higher than that for the conventional engine and it takes a trend in relation to TIT variation, Fig. 8.29b.

As VIGVs closed, surge margin was eroded and it may therefore be necessary to take evasive action if the running line closely approaches the surge line. The bypass nozzle area could be used to modulate thrust while at the same time adjusting fan surge margin. However, bypass nozzle area was not varied in this investigation as it did not give a large thrust attenuation and also, the control system would be made more complicated.

<u>The Effect of Keeping HP Spool Speed Fixed</u>

When the HP spool speed is allowed to vary, time is needed to change HP shaft speed when thrust changes and therefore engine response could be further improved if the engine were controlled so as to keep the HP spool speed fixed as well. An investigation was carried out to find out how the performance of the engine is affected when both shaft speeds are allowed to remain constant.

It was expected that any loss in compressor efficiency occurring as the vanes are rotated would cause an increase in compressor delivery temperature which would result in a higher fuel flow for the same thrust as that developed when

# Performance Of A Rapid Response Turbofan

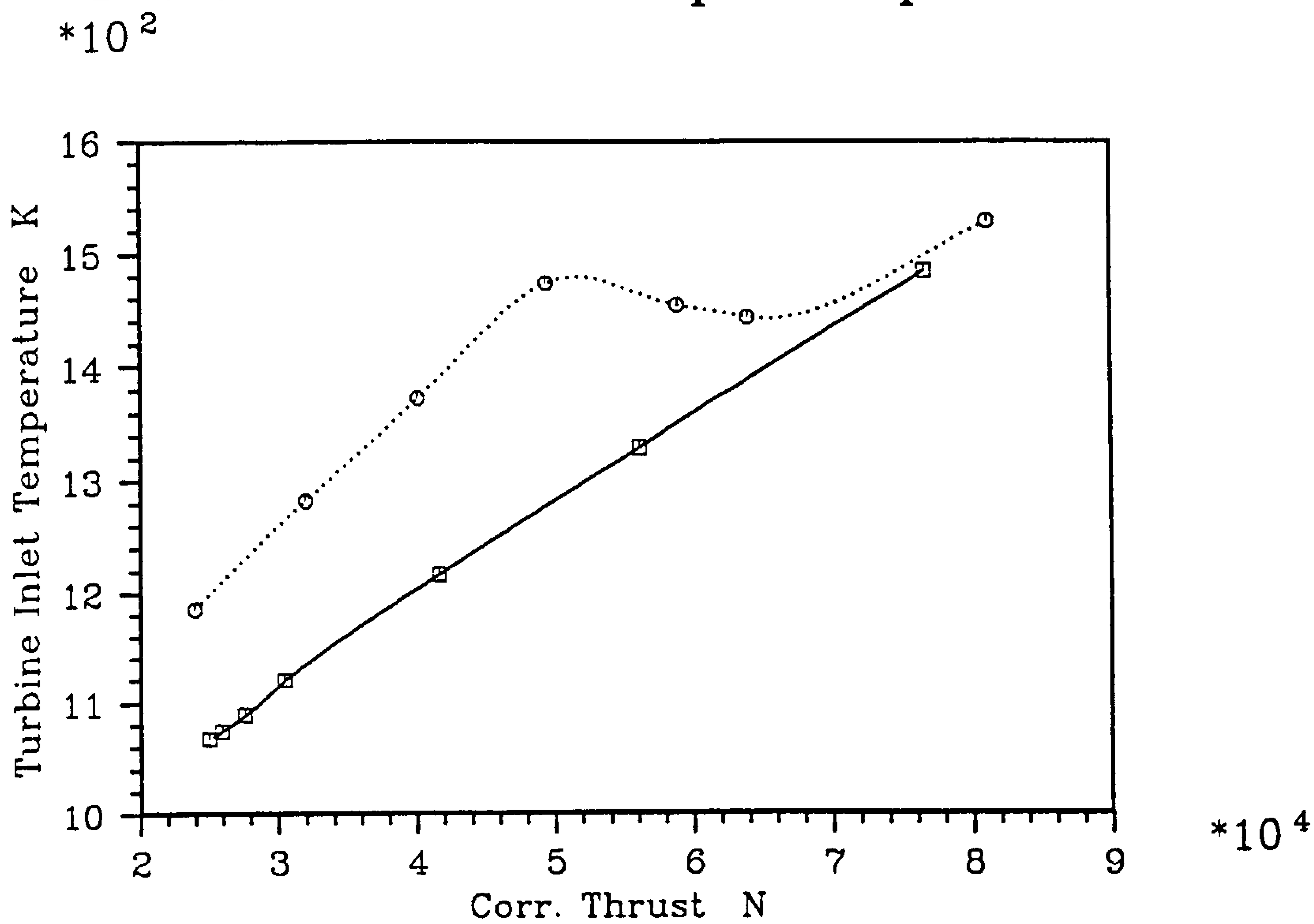



Fig. 8.29a Temp Schedule for Thrust Modulation

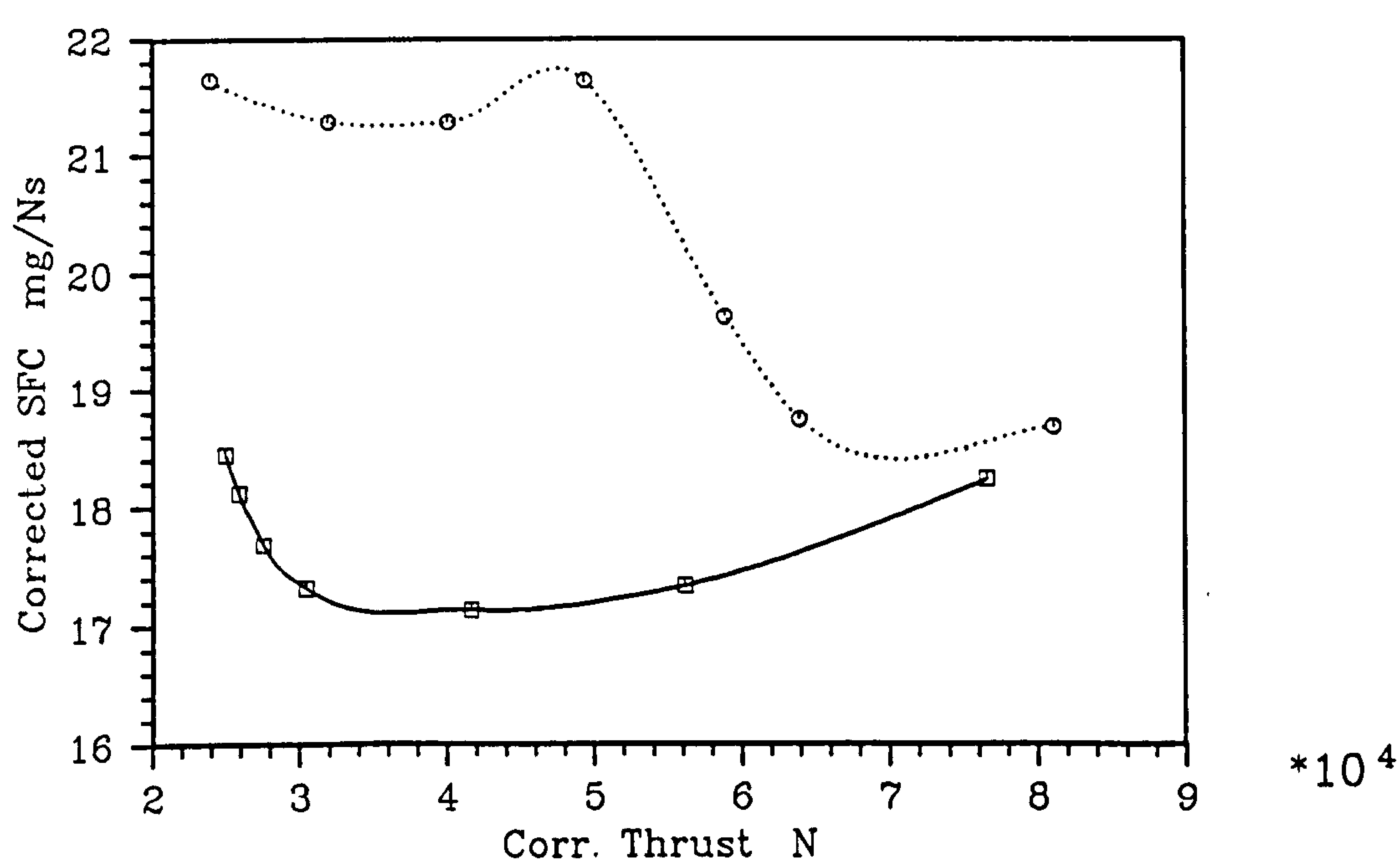



Fig. 8.29b Fuel Flow Requirement For Unit Thrust

Fixed Comp. Vanes/ Fixed Turbine Area

Var. LP Comp. Vanes/ Fixed Turbine Area

Alt (m) 500.0
Mn 0.2

the HP shaft speed is allowed to vary. Fig. 8.30 compares the performances of the two types of control for the landing thrust requirement.

Fig. 8.30a shows that at the high power end, the specific fuel consumption is higher when controlled with both shaft speeds fixed whereas the reverse is true at the low thrust end. The higher TIT encountered at the high thrust end was responsible for this, Fig. 8.30b, and this in turn was due to the low efficiencies at high power settings as can be seen in Fig. 8.30c. The efficiencies are higher at low thrust and therefore an improvement in fuel consumption was realized. The shape of the efficiency curves for the compressors will determine the relative sfc's but it is expected that in general the sfc will be higher when both shaft speeds are held constant as there will be an appreciable loss of compressor efficiency over a wide thrust range.

## 8.3 Multi-Cycle Engines

It has been mentioned several times that the thrust and sfc characteristics of the turbofan and turbojet differ considerably at both subsonic and supersonic speeds and therefore one may be more suited than the other for a particular application. Aircraft that have to cruise for short or extended periods at supersonic speeds impose severe performance requirements on the propulsion system and because of the conflicting performances exhibited by the turbofan and turbojet engines in either flight phase, supersonic propulsion systems may give quite a poor performance in certain portions of its flight envelope.

The turbofan is almost universally used for jet propulsion, with the exception of high Mach cruise application, and therefore, a versatile propulsion system would be a turbofan with the potential of varying its bypass ratio to suit any current flight conditions and also power setting. Such an engine, a variable cycle engine, would then be equipped with one or more components that can vary their geometry to provide the required matching that may allow for a significant variation of thermodynamic cycle, within limits.

The effective control of the bypass ratio to effect improvement in thrust or specific fuel consumption entails movement of one or more operating points to positions in the relevant component performance maps that will change the aerodynamic loading of the components thereby effecting the desired change in engine mass flow, fuel flow, etc., and even though component losses may not be improved, the beneficial effects of improved specific thrust, or thermal or propulsive efficiency, may outweigh the detrimental

# Effect Of Keeping HP Spool Fixed

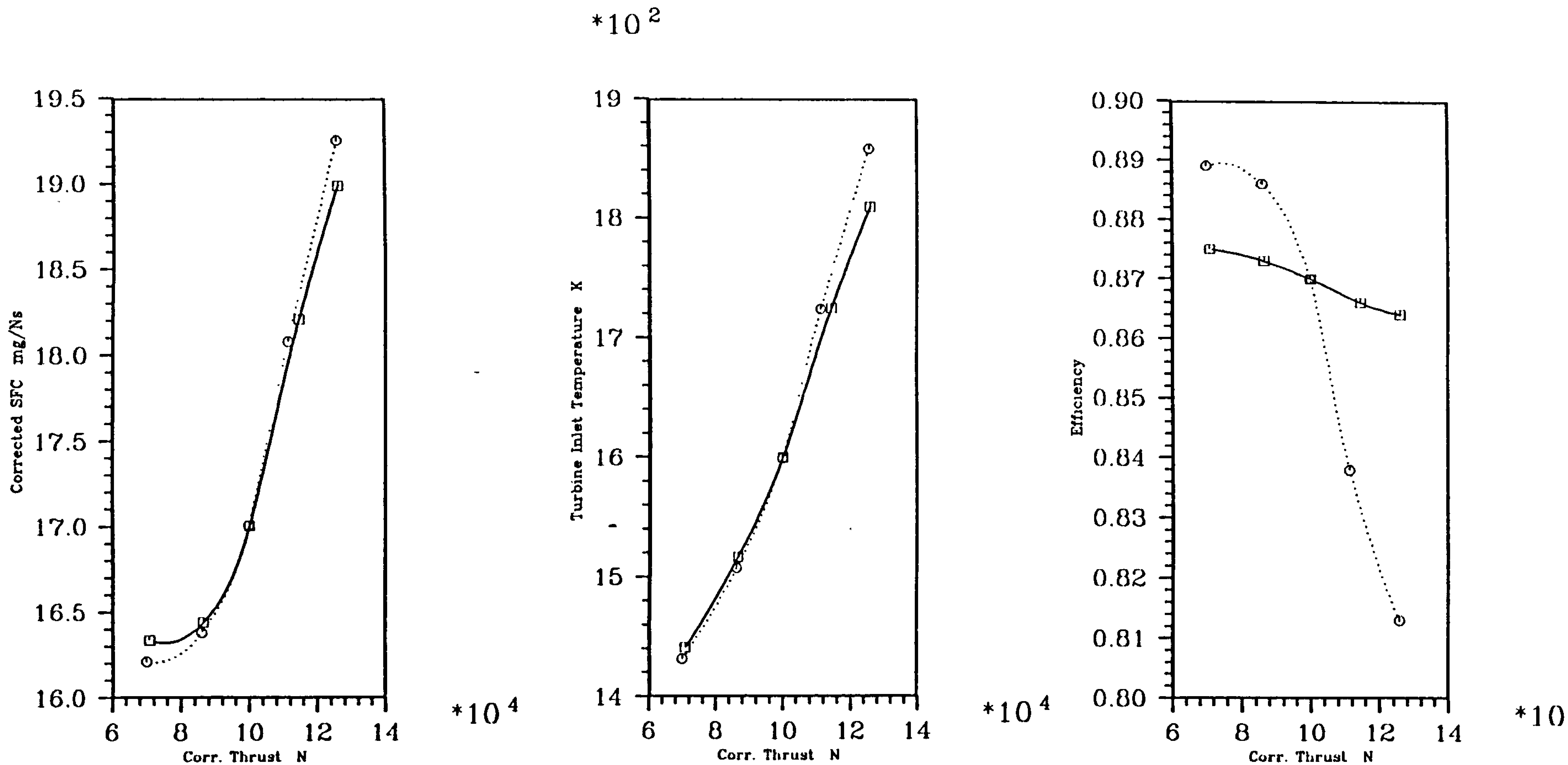



Fig 8.30a Fuel Flow Requirement For Unit Thrust

Fig. 8.30b Temp Schedule for Thrust Modulation

Fig 8.30c Comp. Efficiency Variation

Var. LP Comp. Vanes/ Fixed Turbine Area

Var. LP,HP Comp. Vanes/ Fixed Turbine Area

Alt (m) 0.0
Mn 0.0

effects of losses that may tend to deteriorate cycle performance.

A cycle study was carried out to find out the characteristics of a variable bypass ratio engine and to determine what level of performance improvement can be obtained by cycle variation.

### 8.3.1 Characteristics of A Variable Bypass Ratio Cycle

The purpose of this exercise is not to study the performance of a variable cycle engine but rather to investigate what components require variable geometry and how a change in geometry effects control of bypass ratio. The design point performance of the study engine is given in Table 8.4; a schematic of a typical twin-spool mixing turbofan is shown in Fig. 8.31

Design Point Parameters
(9144 m/ M = 0.9 ISA)

| | | | |
|---|---|---|---|
| Bypass Ratio | | | 0.25 |
| OPR | | | 16.0 |
| Fan PR | | | 3.88 |
| TIT | | | 1400 |
| Thrust | (Corrected) | (KN) | 46.7 |
| sfc | (Corrected) | (mg/Ns) | 31.2 |

Table 8.4 Performance of A Variable Cycle Turbofan

The possible components for variable geometry are the fan, compressor, turbines, propelling nozzle (throat and exit areas), mixer, and splitter. Variable geometry could be

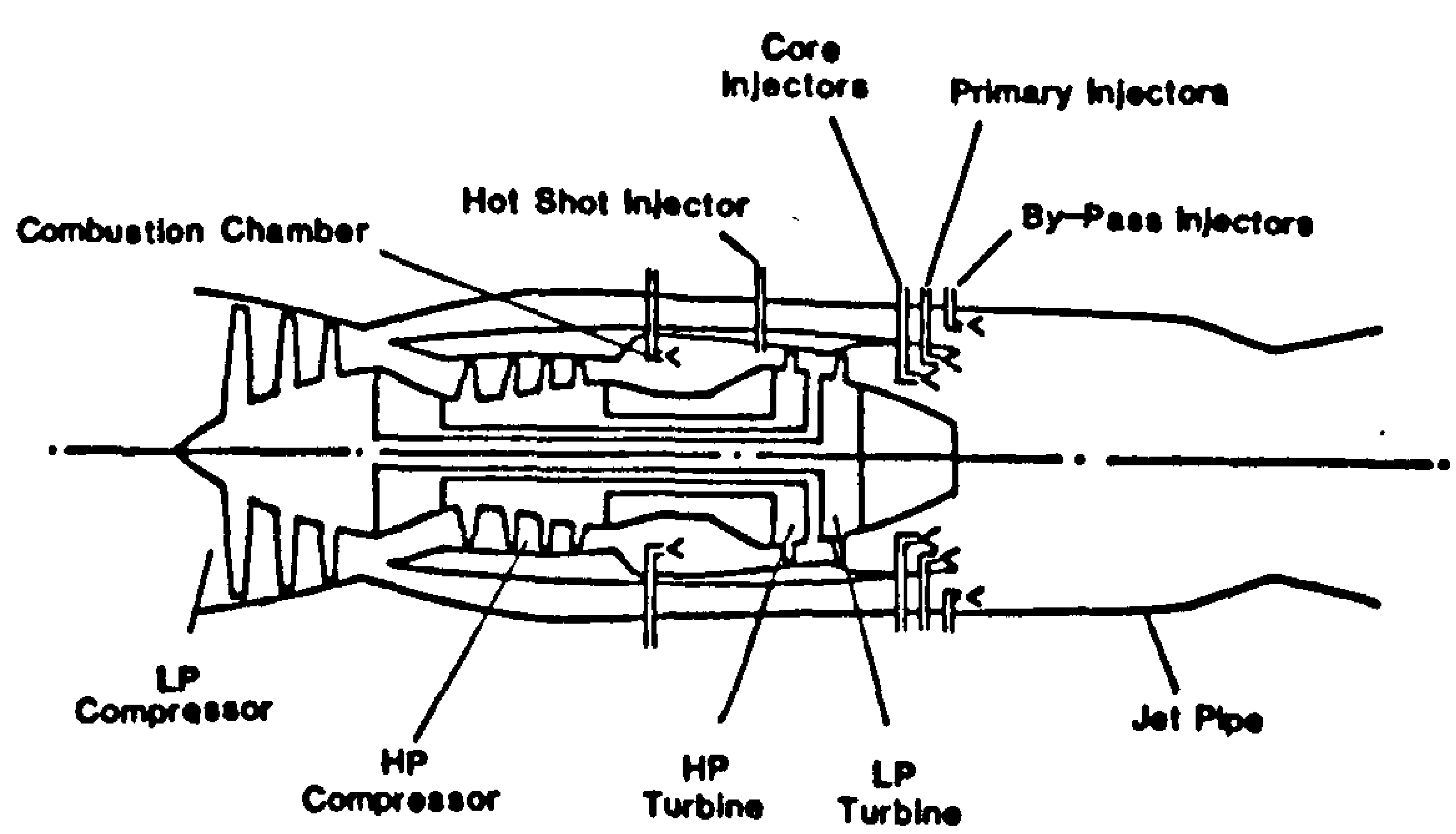



Fig. 8.31 Schematic of a typical mixed-flow turbofan

incorporated in the inlet as well for improved inlet-engine matching; this is a special case which does not have an effect on bypass ratio and therefore will not be mentioned any further.

If variable geometry is incorporated in the bypass splitter, then it can be considered as a valve whose area determines the bypass ratio. Although this area can be scheduled or specified in the program which then gives an indication of the level of bypass airflow, it does not in fact affect the matching of the components to effect a change of bypass ratio. It is used in the program more as a two-position mode selector valve which can be opened or closed to denote turbofan or turbojet operation respectively. The splitter area is defined at the design point either by specifying it or a Mach number at the inlet and when this is reduced to zero at off-design, turbojet mode is effected, otherwise, turbofan analysis is performed. It may be possible to vary the bypass ratio continuously over a wide range of thrust by varying the appropriate geometries but in this investigation, the extreme is considered which is that of a bypass ratio of zero, the turbojet.

Remembering that for a fixed TIT and OPR the design fan pressure ratio of a mixing turbofan, which is fixed by the requirement of static pressure balance in the mixer, goes up as bypass ratio is reduced, it can be guessed that when the bypass duct is closed at constant fan speed, the fan pressure ratio will rise tremendously with an accompanying decrease in engine airflow. Because of the reduced non-dimensional flow at compressor inlet, the HP spool speed will drop and so will core power and therefore, turbine inlet temperature will fall accordingly to satisfy HP turbine work requirement. Also, the increased fan work will cause the LP turbine power to rise and coupled with reduced inter-turbine temperature, the LP turbine work function and hence, pressure ratio, will increase. The lower jet pipe temperature will cause the specific thrust to fall and since mass flow falls as well, thrust will decrease. It is possible that thrust may fall in proportion to fuel flow to a level such that sfc does not rise much.

There is scope for increasing thrust by raising HP spool speed and hence, core power, by the use of variable geometry in one or more components. An increase in TIT will increase specific thrust and this can be brought about by increasing HP or LP turbine area, or decreasing propelling nozzle throat area, or opening compressor vanes, or closing fan vanes, or a combination of these. HP turbine geometry change will not be considered here in order to maintain simplicity as much as possible; also, fan IGVs will be assumed to follow an implicit schedule based on fan corrected speed, if these vanes are variable. Therefore, only compressor vanes, LP turbine and propelling nozzle areas are allowed to vary in order to obtain the required matching.

In order to restore fan operation, fan power should be decreased by unloading the LP turbine. As compressor non-dimensional mass flow increases, HP spool speed increases as well and it is possible that speed limit may be exceeded. Therefore, it may be desirable to vary some compressor vane rows to maintain speed within limits, and in this case, the vanes are to be opened. Compressor variable geometry is therefore used to match the flow requirements of the two compression components while keeping compressor speed at bay. The increase of compressor work therefore necessitates a loading of the HP turbine. A physical understanding of how variable geometry in the LP turbine and propelling nozzle can be used to change the load on the components can be easily sought by considering the matching of two turbines in series.

Referring to Fig. 7.18 and remembering that thermodynamically the mass flow characteristics of a propelling nozzle and power turbine are similar, a decrease in the choked propelling nozzle throat area will reduce the flow function at LP turbine exit annulus and therefore decreases the pressure ratio across the turbine thereby unloading it. By similar arguments, HP turbine loading can be increased by increasing LP turbine area.

### 8.3.2 Performance Results

The performance of the engine when the switch over is done at constant TIT is not readily amenable to analysis as that for fixed fan point operation. In this case, the fan speed increases tremendously while compressor aerodynamic speed decreases as a result of the increase in compressor inlet temperature. The thrust also increases greatly with sfc remaining almost constant. As with the constant speed operation described above, fan operating point should be optimized by unloading the fan while at the same time increasing the loading on the compressor to increase core power.

In Figs 8.32 through 8.34, the performances of the turbofan when in both modes of operation, that is, conventional and turbojet, are compared at an altitude of 9144 m and a speed of Mach 0.9. Maximum dry power of the conventional engine is limited by LP rotor spool speed and therefore any further increase in thrust can only be obtained by raising TIT.

Fig. 8.32a shows that the transition is made at constant fan operating point and the new operating line runs at a higher pressure ratio level. As was explained earlier, core power has to increase in order to drive the fan and therefore, compressor power increases, Fig. 8.32b. Since

# Performance Of A Variable Bypass Ratio Engine

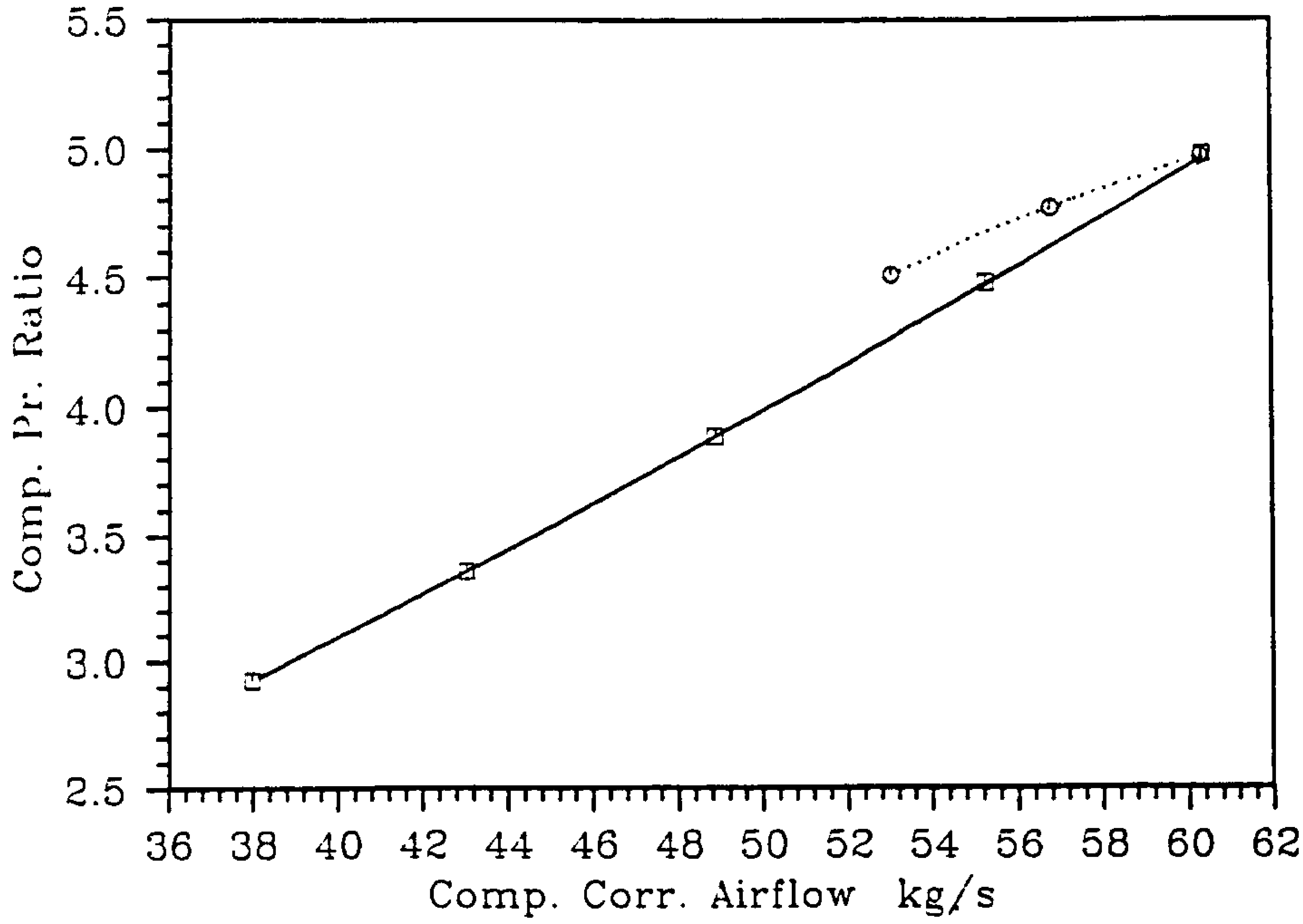



Fig. 8.32a LP Compressor Operating Line

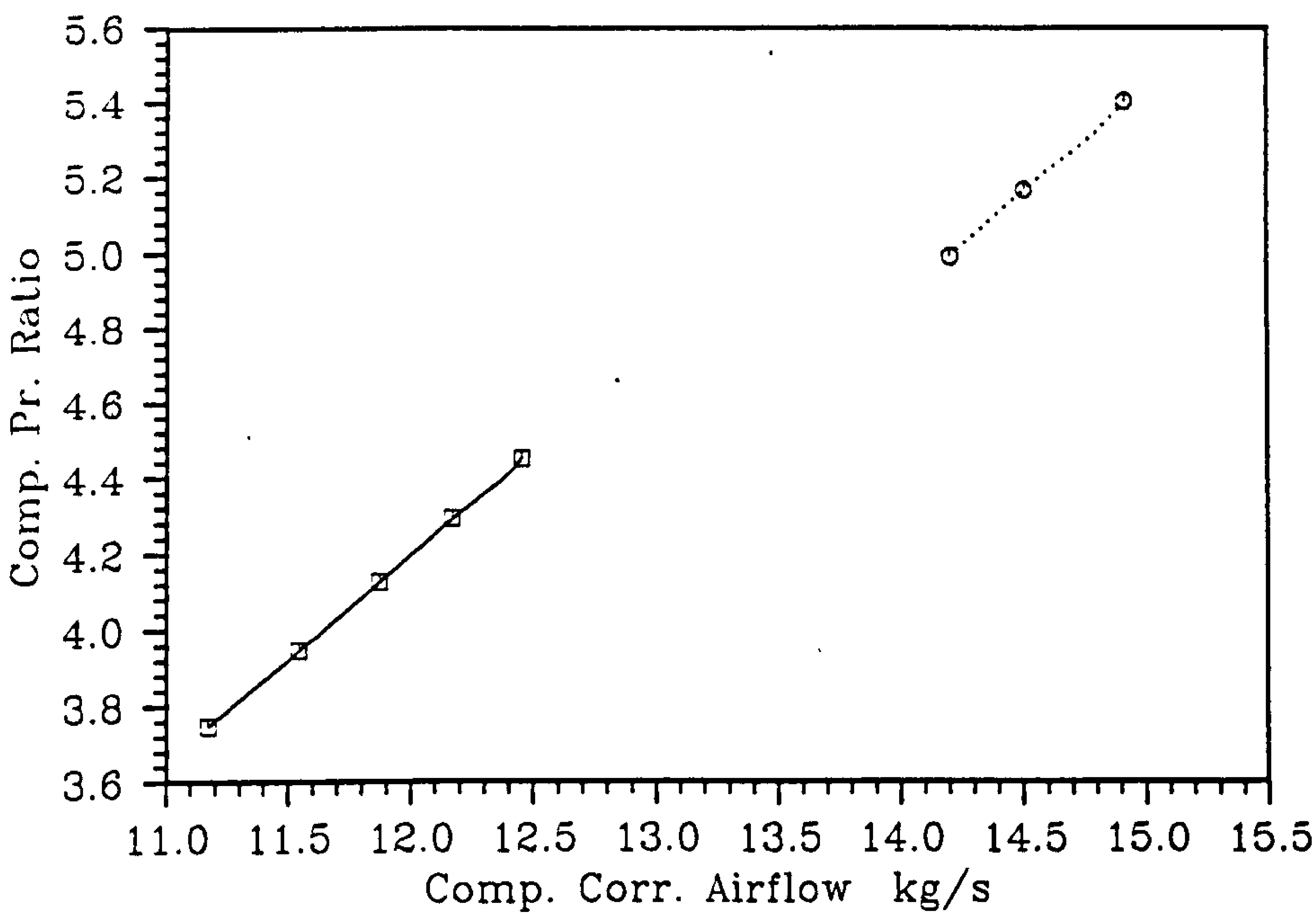



Fig. 8.32b HP Compressor Operating Line

Fixed Comp. Vanes/ Fixed Turbine Area
Alt (m) 9144.0 Var. HP Comp. Vanes/ Var. LP Turbine Area
Mn 0.9

the compressor passes a higher mass flow at the same inlet temperature and pressure at the transition point, the vanes are to be opened up. Ideally, the setting of the compressor vanes should vary with the amount of bypass ratio required for proper matching with the fan. In this case, the vanes should be fully opened since all the fan flow goes through the compressor with the exception of the bleed air but was rather opened to the design setting with the aim of keeping compressor speed almost constant when the transition is made.

The propelling nozzle throat area was decreased by 6 percent to unload LP turbine while LP turbine area was increased by 12.5 percent to load the HP turbine. The work function of the HP turbine increases accordingly, Fig. 8.33a, whereas that of the LP turbine decreases, Fig. 8.33b. It is not clear why the LP turbine work function increases as thrust decreases when in the turbojet mode of operation. The pressure ratio across the propelling nozzle increases as the transition is made, Fig. 8.33c, as a result of the increase in overall pressure ratio. An increase in TIT was necessary to give the 16 percent increase in dry thrust that was realized, Fig. 8.34. At a typical combat thrust setting, the saving in fuel consumption ranges from about 6.5 to 9 percent.

At the supersonic flight condition of Mach 1.6 at 15240 m, maximum dry power is limited by turbine inlet temperature. In this case, engine thrust can be increased by increasing spool speeds. The transition to the turbojet mode was carried out such that the maximum dry power of the turbojet was obtained at maximum rotational fan speed. Since compressor delivery pressure increases tremendously compared with the ambient static pressure, the amount of spool speed increase and hence, thrust increase, may be dictated by the limit on pressure levels in the combustor and compressor delivery temperature. The trends in component performance improvement are similar to those observed in the subsonic case discussed above. In this case there is a 33 percent increase in dry thrust with a saving in fuel consumption of about 16 percent at a typical supersonic cruise power setting, Fig. 8.35.

The increase in thrust and subsequent improvement in sfc when the engine is in the turbojet mode is explained by the curves in Fig. 8.36. At the transition point which occurs at constant turbine inlet temperature, both LP and HP compressors are operating at a higher non-dimensional speed with increased mass flow and pressure ratio. The operating lines for the LP compressor are shown in Fig. 8.36a. Even though the compressor powers increase, the increase in compressor delivery pressure along with the unloading of the LP turbine result in an increase in nozzle pressure ratio, Fig. 8.36b, which combines with a higher jet pipe

## Performance Of A Variable Bypass Ratio Engine

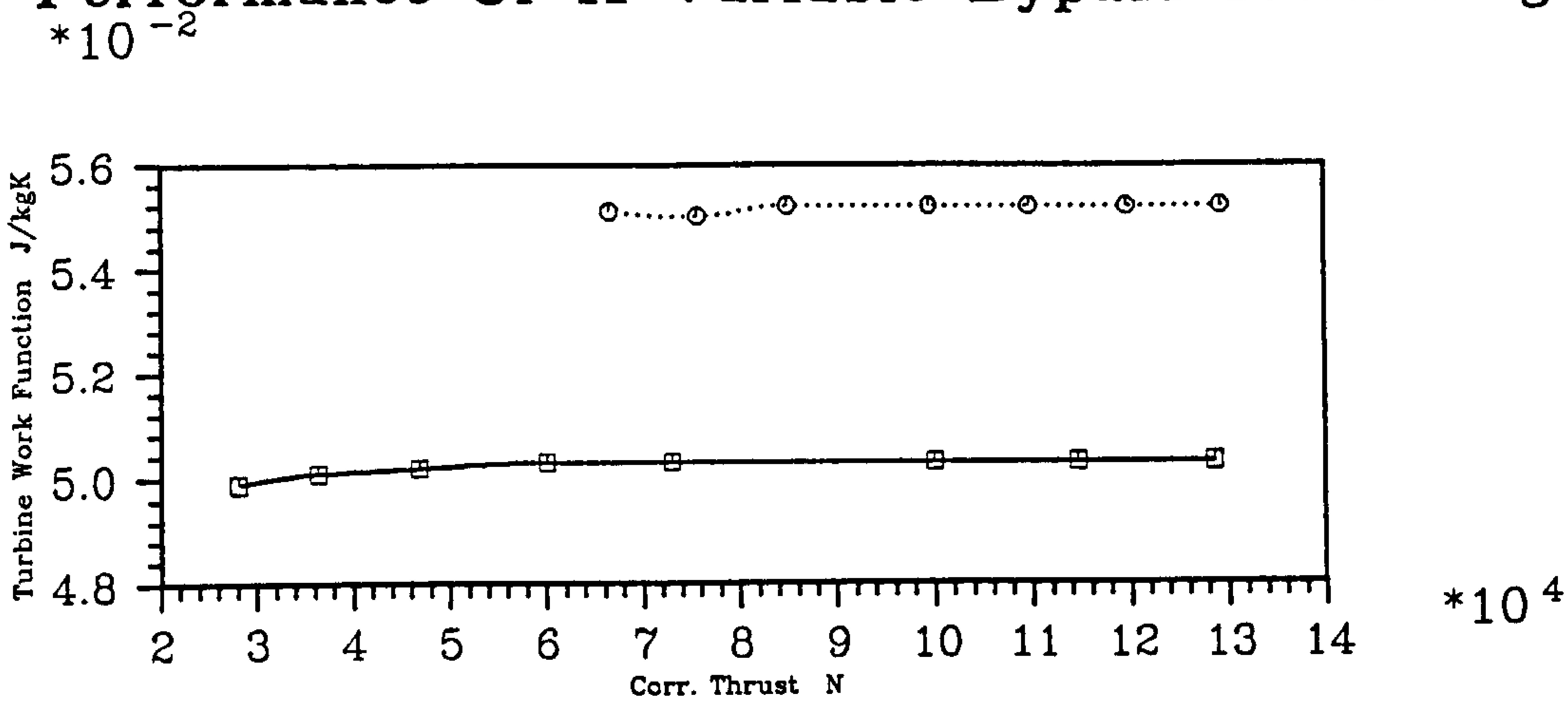



Fig. 8.33a HP Turbine Work Requirement

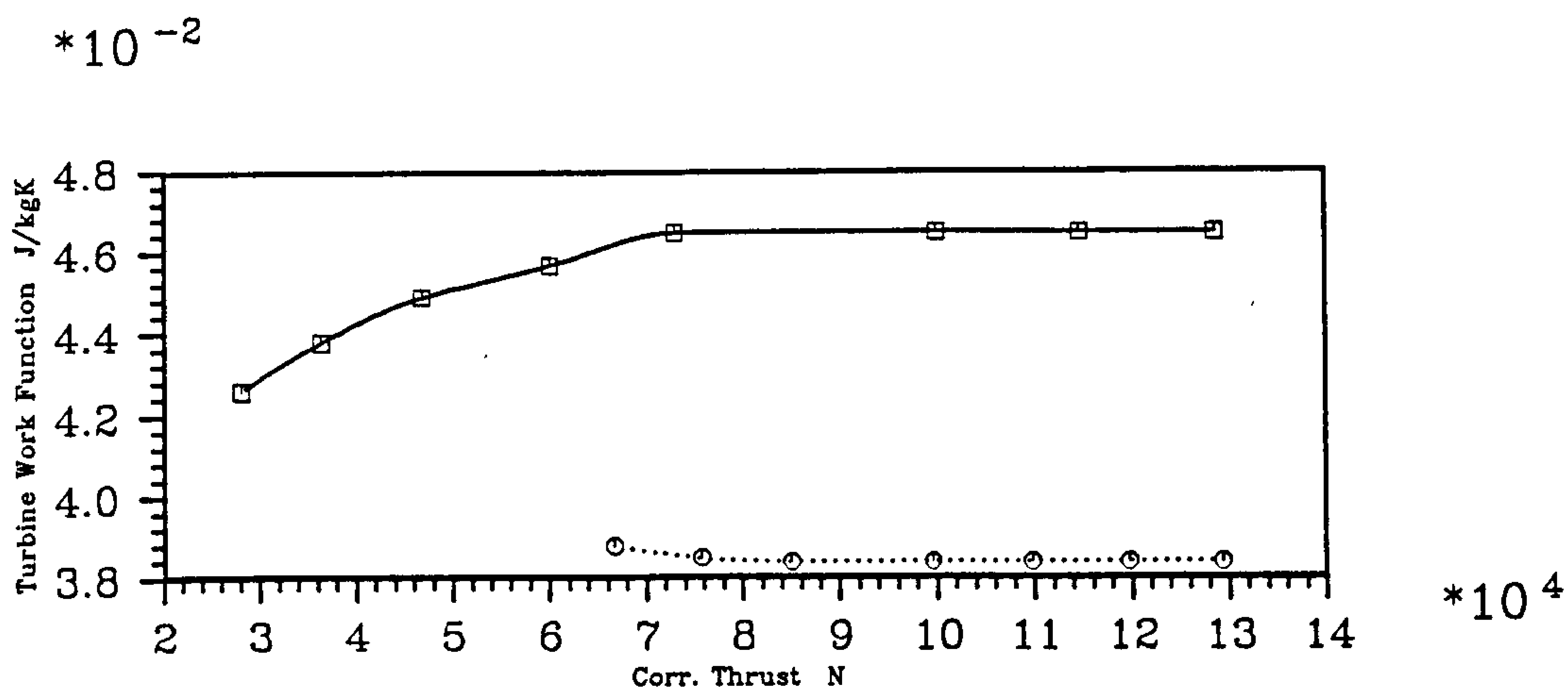



Fig. 8.33b LP Turbine Work Requirement

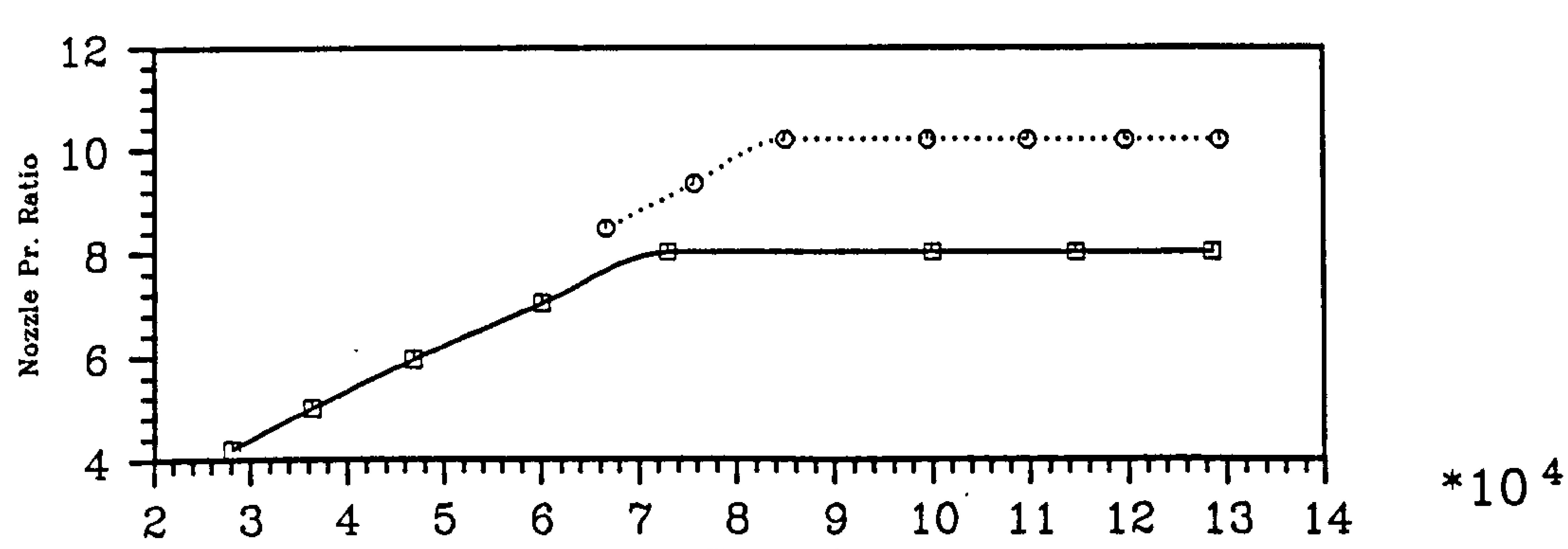



Fig. 8.33c Nozzle Pressure Ratio Variation

⊞—⊟ Fixed Comp. Vanes/ Fixed Turbine Area

Alt (m) 9144.0 ⊙····⊙ Var. HP Comp. Vanes/ Var. LP Turbine Area

Mn 0.9

## Performance Of A Variable Bypass Ratio Engine

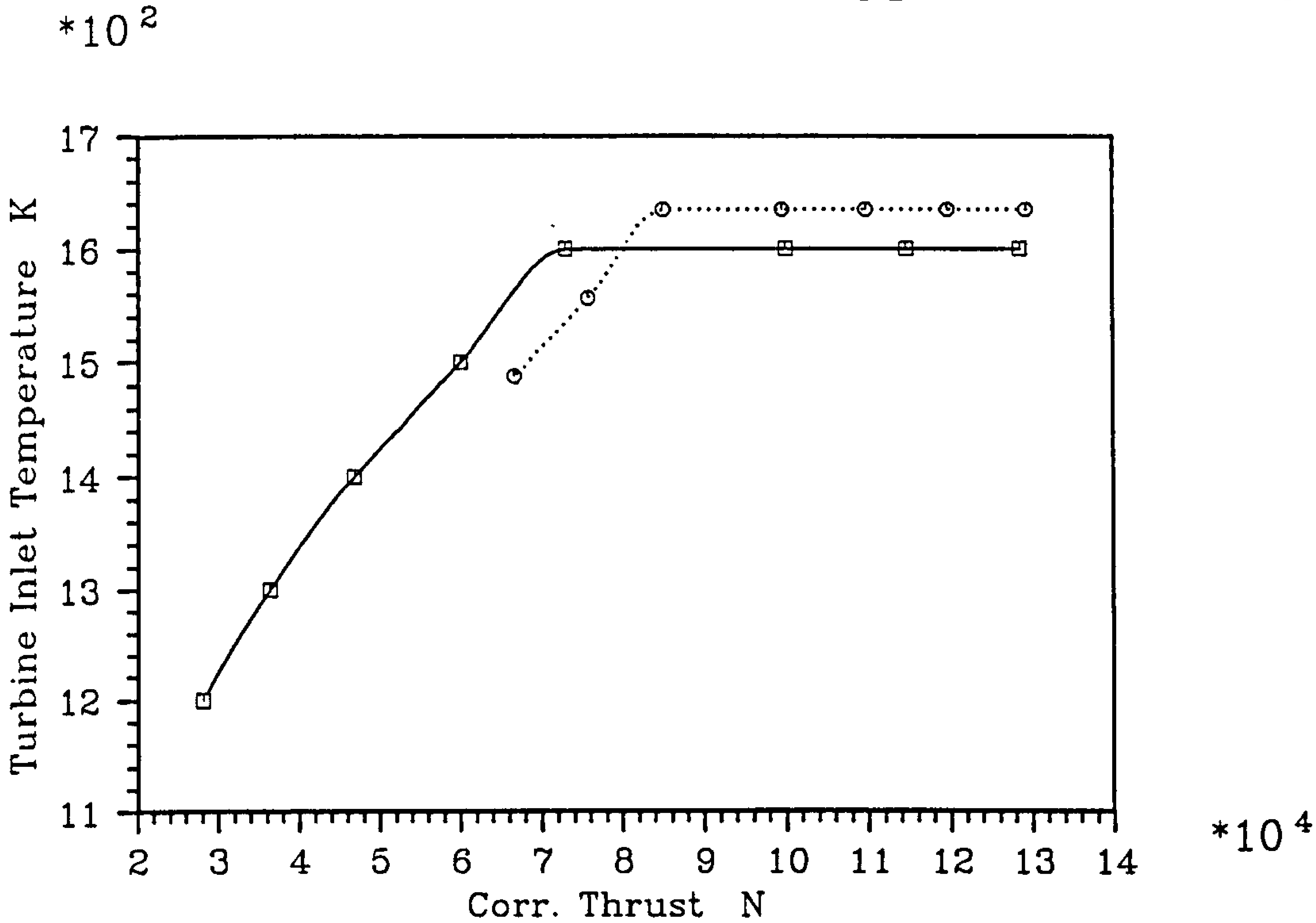



Fig. 8.34a Temp Schedule for Thrust Modulation

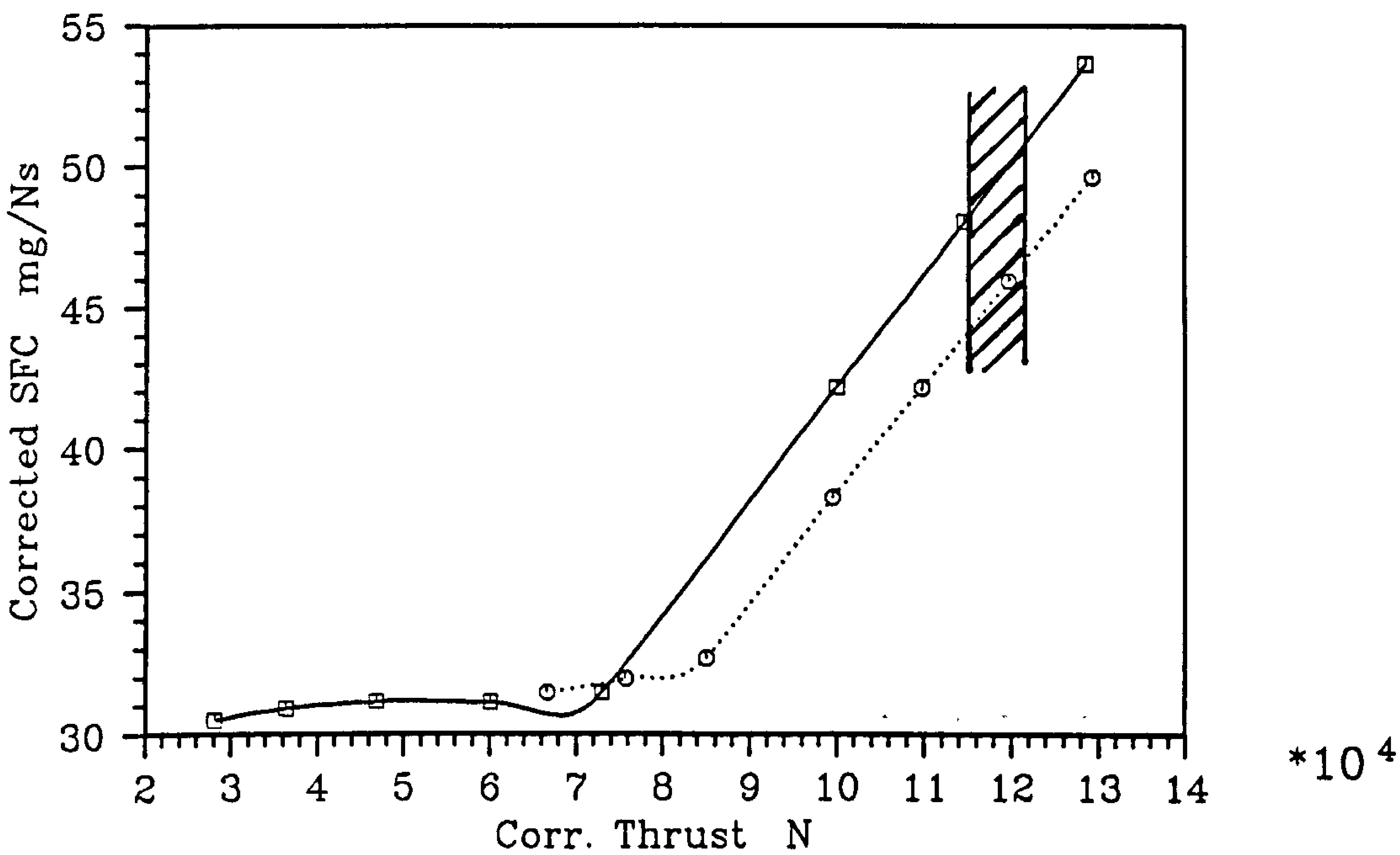



Fig. 8.34b Fuel Flow Requirement For Unit Thrust

□—□ Fixed Comp. Vanes/ Fixed Turbine Area

Alt (m) 9144.0 ⊙····⊙ Var. HP Comp. Vanes/ Var. LP Turbine Area

Mn 0.9

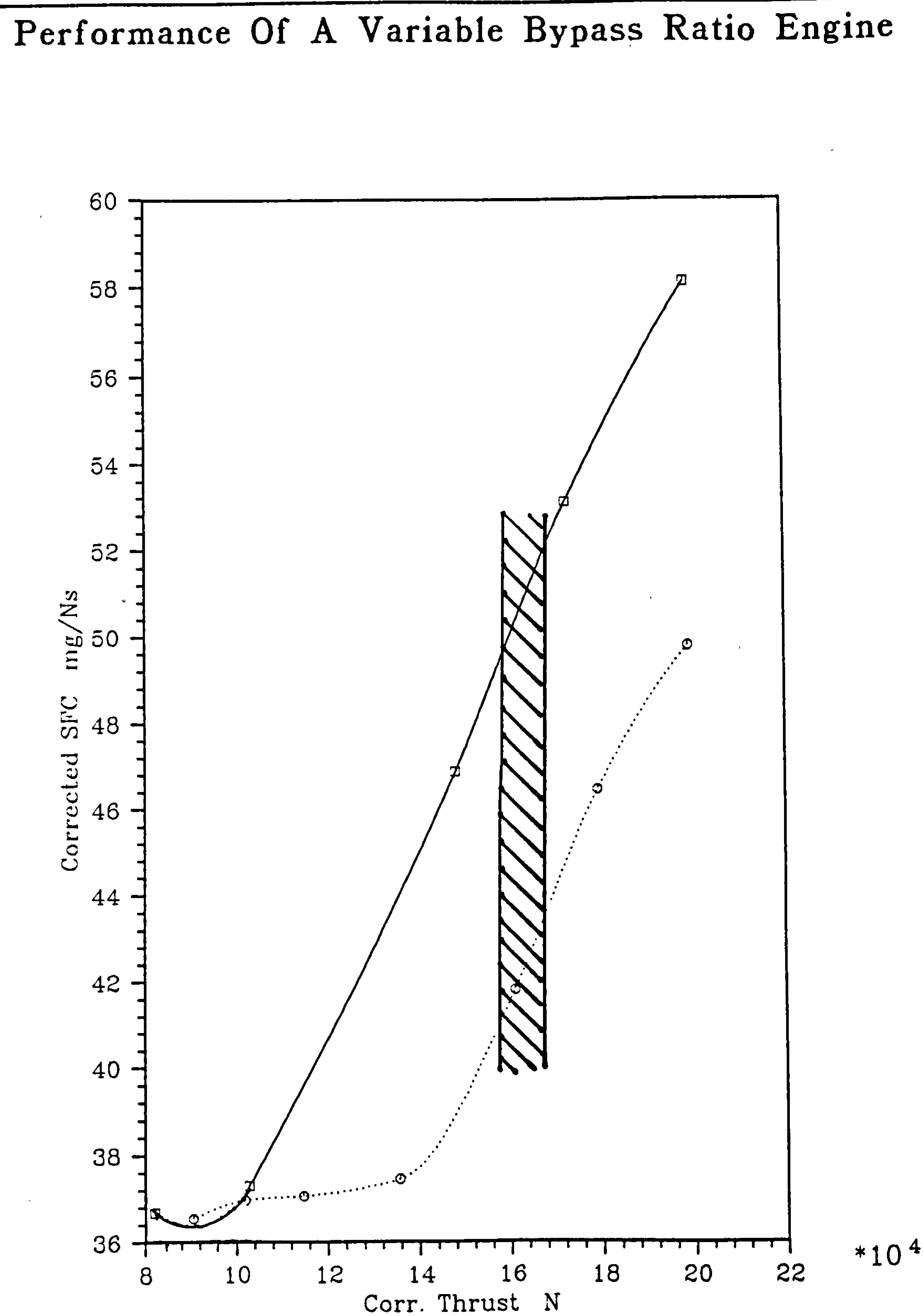



Fig. 8.35 Fuel Flow Requirement For Unit Thrust

Fixed Comp. Vanes/ Fixed Turbine Area

Alt (m) 15240.0 Var. HP Comp. Vanes/ Var. LP Turbine Area

Mn 1.6

# Performance Of A Variable Bypass Ratio Engine

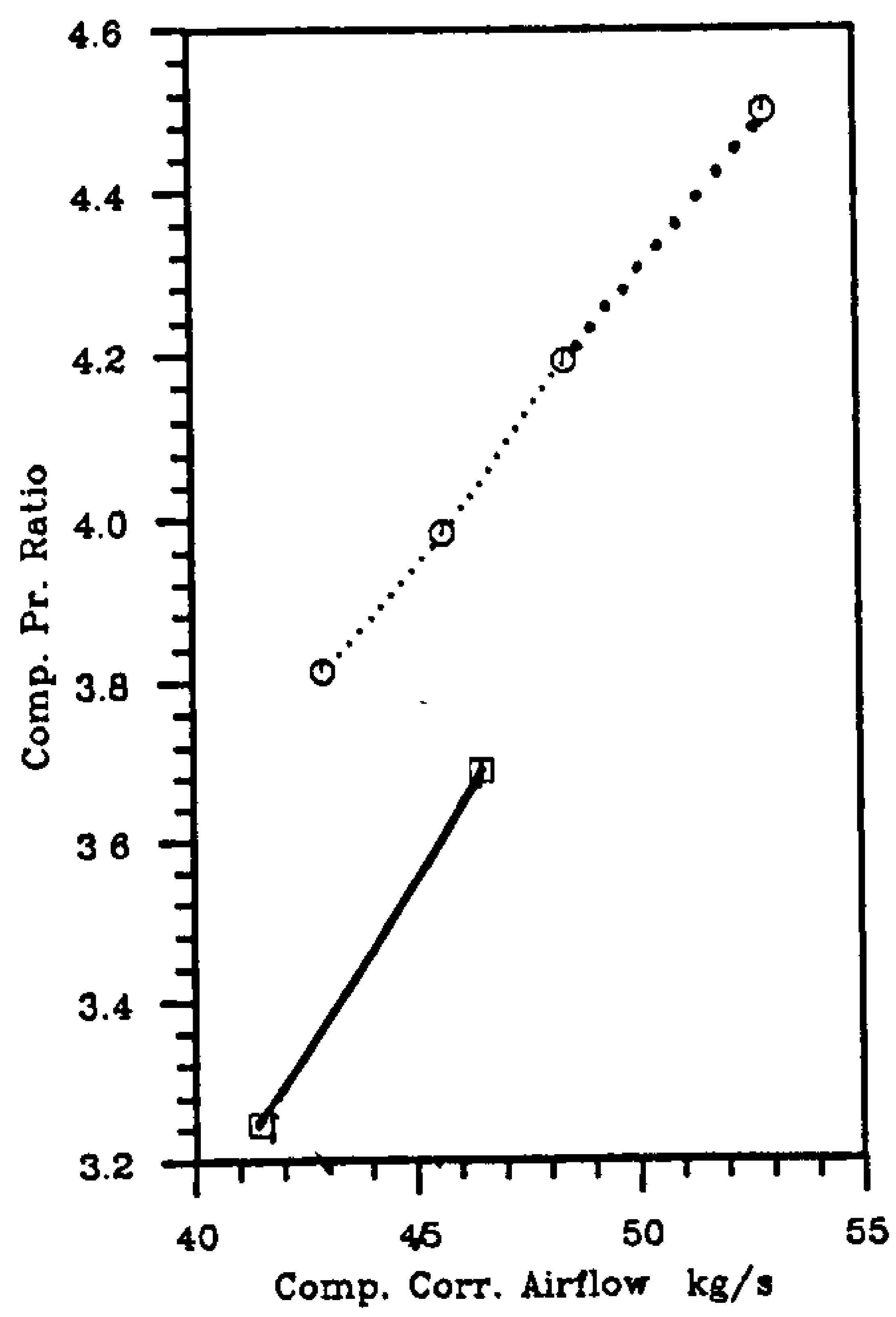



Fig. 8.36a LP Compressor Operating Line

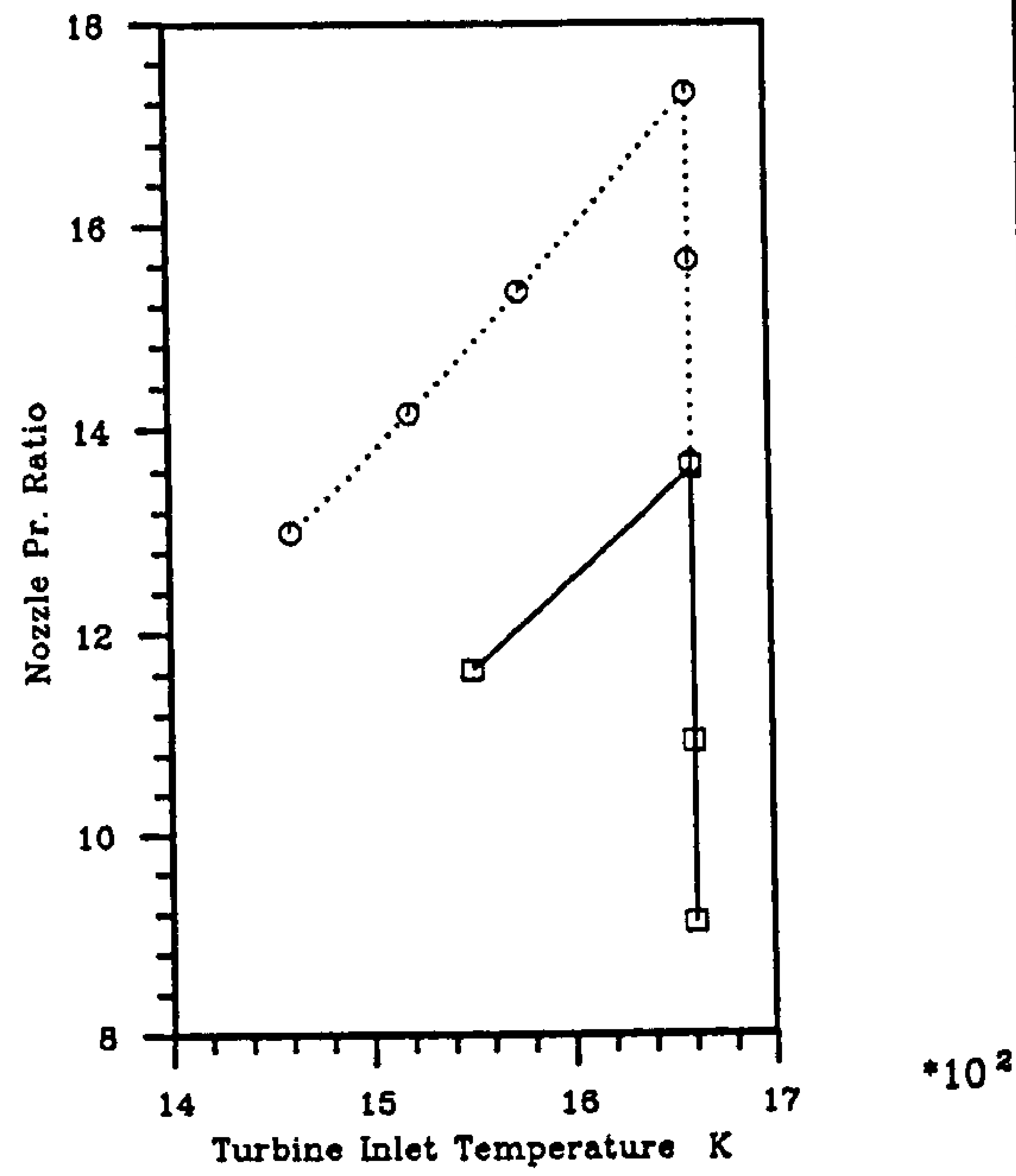



Fig. 8.36b Var. of Nozzle Pr Ratio With TIT

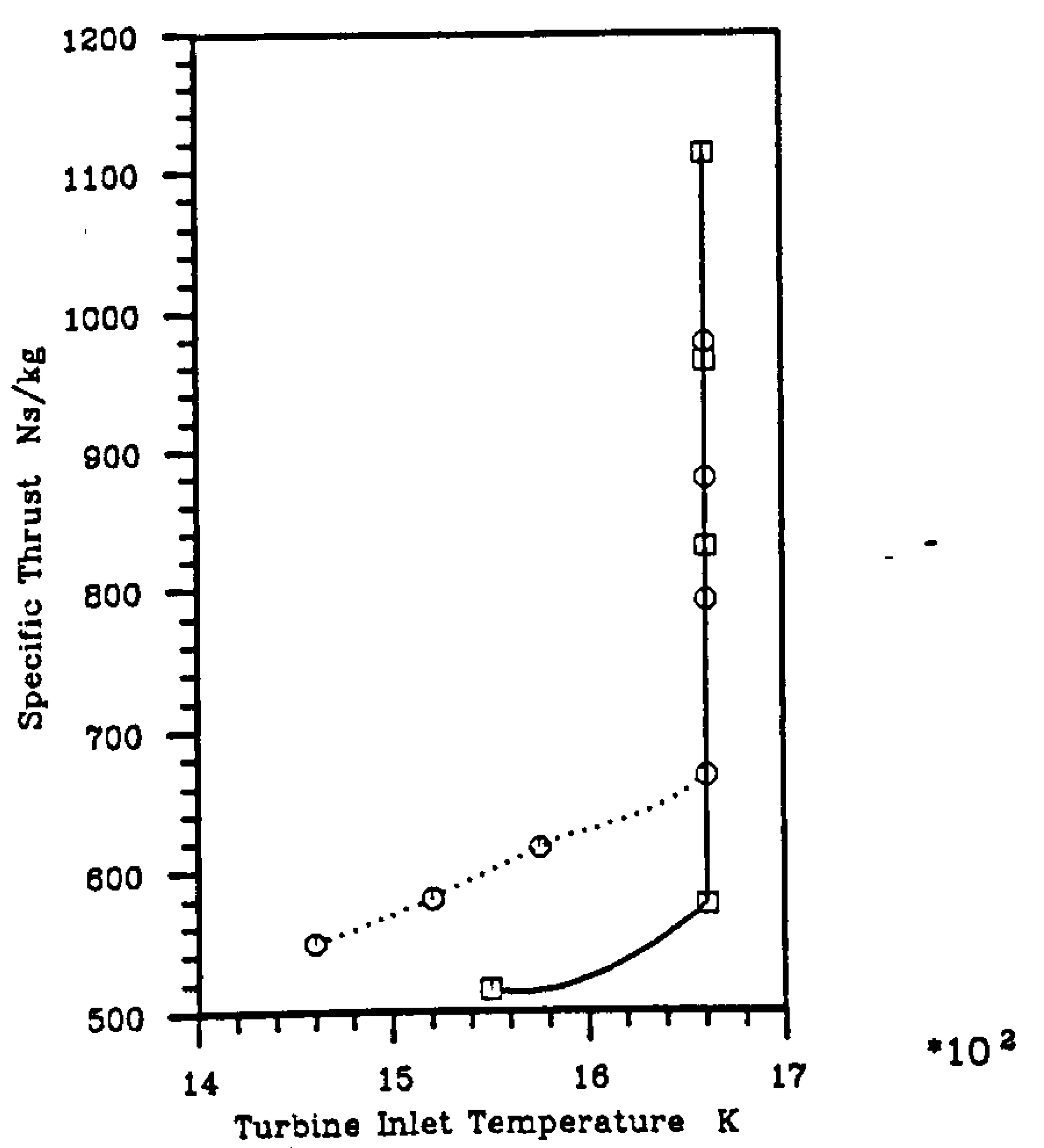



Fig. 8.36c Var. of Sp. Thrust With TIT

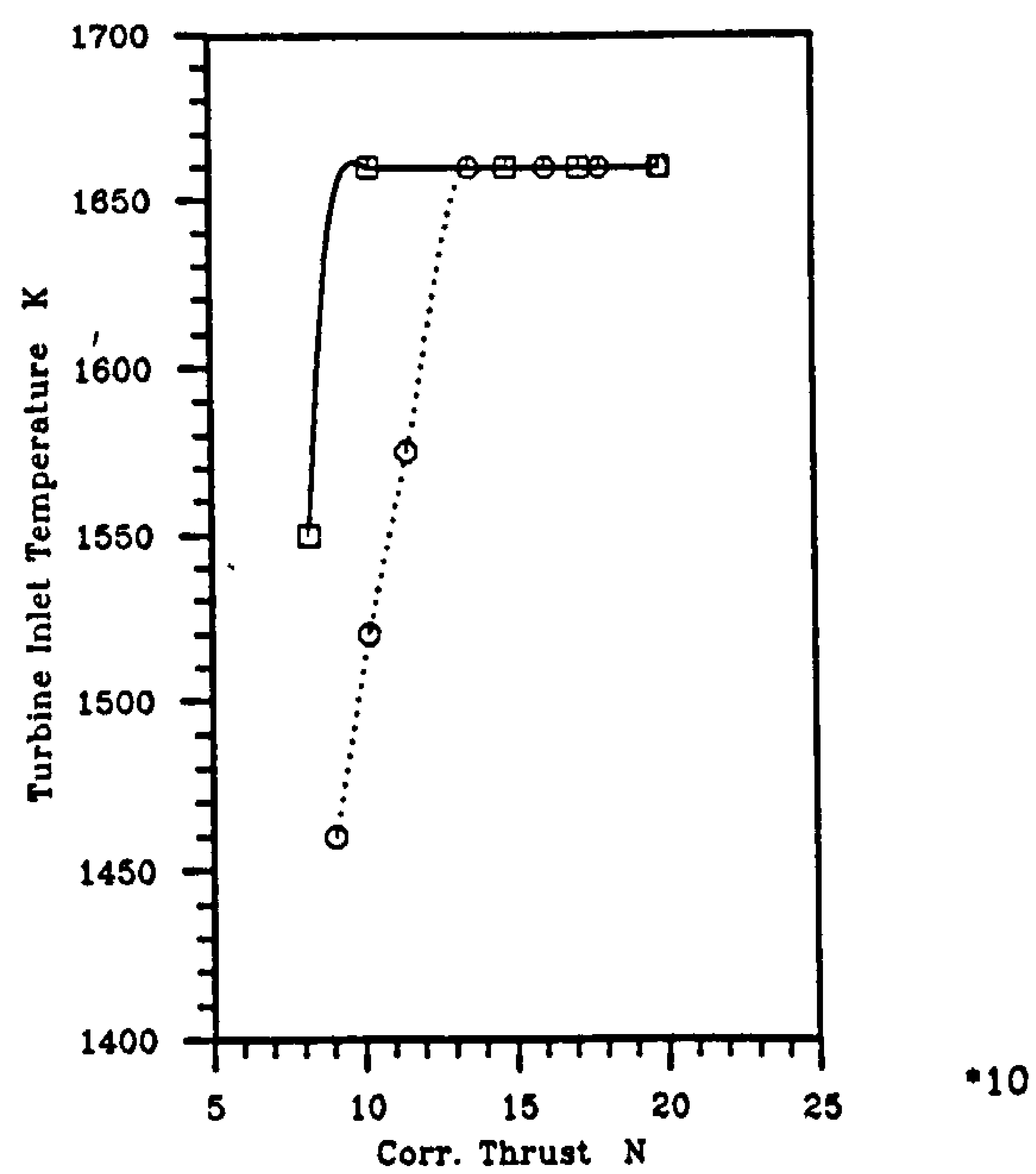



Fig. 8.36d Temp Schedule for Thrust Modulation

⊟—⊡ Fixed Comp. Vanes/ Fixed Turbine Area

Alt (m) 15240.0 ⊙····⊙ Var. HP Comp. Vanes/ Var. LP Turbine Area

Mn 1.60

temperature to give a higher specific thrust as shown in Fig. 8.36c. Since both mass flow and specific thrust increase, a large increase in thrust results, Fig. 8.36d. The increase in thermal efficiency due to a higher OPR slightly outweighs the reduction in propulsive efficiency leading to an increase in overall efficiency. Over the afterburning thrust range when in the turbojet mode, the lower specific thrust results in improved propulsive efficiency at the same thrust setting as the turbofan giving the improvement in performance as seen in Fig. 8.35. In practice, this amount of performance improvement cannot be attained as the fan speed and TIT will be set at a level below the maximum for the required engine life.

At takeoff conditions, the turbofan will be operating at maximum TIT and rotational speed and therefore no improvement in thrust can be obtained as neither spool speed nor TIT can be further increased, and also, it is necessary to maintain operation in the turbofan mode to keep noise down. Turbojet operation is required for flight where a high specific thrust is required such as transonic acceleration and supersonic cruise.

If bypass ratio is to be varied continuously to meet the various thrust demands, that is, from the nominal to zero, then variable geometry is required in the mixer. This facilitates mixing of the two flows by varying the Mach number and hence, static pressure, in the bypass stream. In order to control fan pressure ratio and speed, both the LP turbine and propelling nozzle areas should then be scheduled in addition to compressor vanes in such a manner that the optimum bypass ratio for the required performance is obtained. Such a variable cycle engine will then give a great flexibility of operation with possible significant savings in fuel consumption.

# CHAPTER 9

# DISCUSSION

Both the variable geometry compressor and turbine will be subjected to aerodynamic and mechanical loads which may not be encountered by their fixed geometry counterparts under the same operating conditions and therefore during design, due consideration has to be given to certain characteristics and limitations of these components, and also to certain phenomena, aerodynamic or mechanical, associated with either the variable geometry component or another component or the engine system as a whole, when variable geometry is deployed.

Since the deployment of variable geometry at off-design conditions entails losses in addition to those present at the design point, careful attention must be paid to the degree to which efficiency changes such that any loss of efficiency does not defeat the purpose for which variable geometry was deployed. Also, each component should be able to pass the required mass flow, and the required power should be developed to ensure unrestricted operation in the flight envelope or power spectrum. A knowledge of the operating characteristics of these components is required if full advantage is to be taken of their potential to improve engine performance.

## 9.1 Variable Area Turbine Design Considerations

The satisfactory performance of a variable area turbine requires that the turbine be able to handle a wide range of inlet non-dimensional mass flow over a wide range of non-dimensional speed while at the same time maintaining acceptable efficiency levels. The ability of the turbine to accomplish this requires a study of the flow conditions within the turbine for the particular application, and it may be necessary to conservatively design the turbine to allow for satisfactory or acceptable performance at conditions widely removed from the designed.

The compromises to be made on turbine design for the particular application may require a design range rather than a design point with the operating conditions for which engine performance is superlative or most critically sensitive to turbine performance be given utmost consideration in the design process while the other conditions in the range will generally only restrict or qualify the design [10]. The characteristics and limitations of the variable area turbine impose undue design considerations which are absent in the design of fixed geometry turbines.

## Blade Limit Loading

One of the problems that may have to be tackled in the design of the variable area turbine is that of limiting blade loading. This condition arises when the forces acting on the turbine rotor blades reach a maximum. If the conditions at inlet to a turbine are held fixed as the turbine pressure ratio is gradually increased at fixed turbine speed, then a pressure ratio will be reached above which any further increase in pressure ratio will produce no further increase in turbine work. The blades are then operating at their loading limit; the design of the turbine to preclude this phenomenon may significantly impact engine output per unit weight.

Limit loading has a pronounced effect on the designed annulus at turbine exit and hence, on turbine tip frontal area. Limit loading has been found to prevail when the turbine exit axial Mach number is of the order of 0.7. Since the exit non-dimensional mass flow is directly related to exit axial Mach number for low values of exit tangential Mach number, the limiting exit Mach number is therefore a measure of the minimum value of exit annulus area, and a Mach number of 0.7 gives a mass flow which is about 10 percent less than the choking value, that is, an exit annulus area about 10 percent greater than the critical.

The blade centrifugal stress is directly proportional to the square of the blade hub speed and the exit annulus area, and once the exit annulus area is fixed the turbine inlet annulus area is also obtained since this is directly proportional to the exit area. The engine thrust per unit tip frontal area for a given stress level and TIT will therefore influence engine size or the number of engines required to satisfy the takeoff thrust requirements; so, it is absolutely necessary to keep the exit annulus area to as low a value as is necessary.

The authors of [10] have devised a concept which gives a first hand knowledge of whether or not a particular turbine design for a given duty will cause blade limit loading to be approached at off-design. This concept, the break even point, determines the turbine conditions at which the rate of change of turbine exit non-dimensional mass flow with work diminishes. Analyses of the flow conditions within two turbine designs for different applications led to the conclusion that the necessary adjustments to the internal flow conditions required to satisfy the off-design operating requirements greatly depend on the location of the design point relative to the break even point and if at off-design the deviations from the design conditions are too great, the turbine may not be able to pass the required mass flow or attain the required work output. Therefore, the extent to which the turbine flow conditions are varied is greatly determined by the operating conditions at the design point and therefore careful design of the turbine is essential if unrestricted operation is to be achieved.

## Efficiency

A variable area turbine does have the potential to significantly improve engine performance but as the vanes are rotated to change effective area, losses are incurred and therefore it is important that loss of efficiency should not negate the advantages to be gained by the use of such a turbine. If maximum performance benefits are to be gained from the use of these turbines, it is necessary to understand what factors of variable geometry affect turbine efficiency in order that performance penalties be minimized. In order to accomplish this, the total loss should be broken down into component losses and an analysis performed to find out the effects of variable geometry on the various loss components.

Figure 9.1 shows a possible breakdown of turbine losses at a specified speed and pressure ratio as the vanes are rotated from the design setting at 0 degree; negative angles denote closure. For simplicity, the losses can be

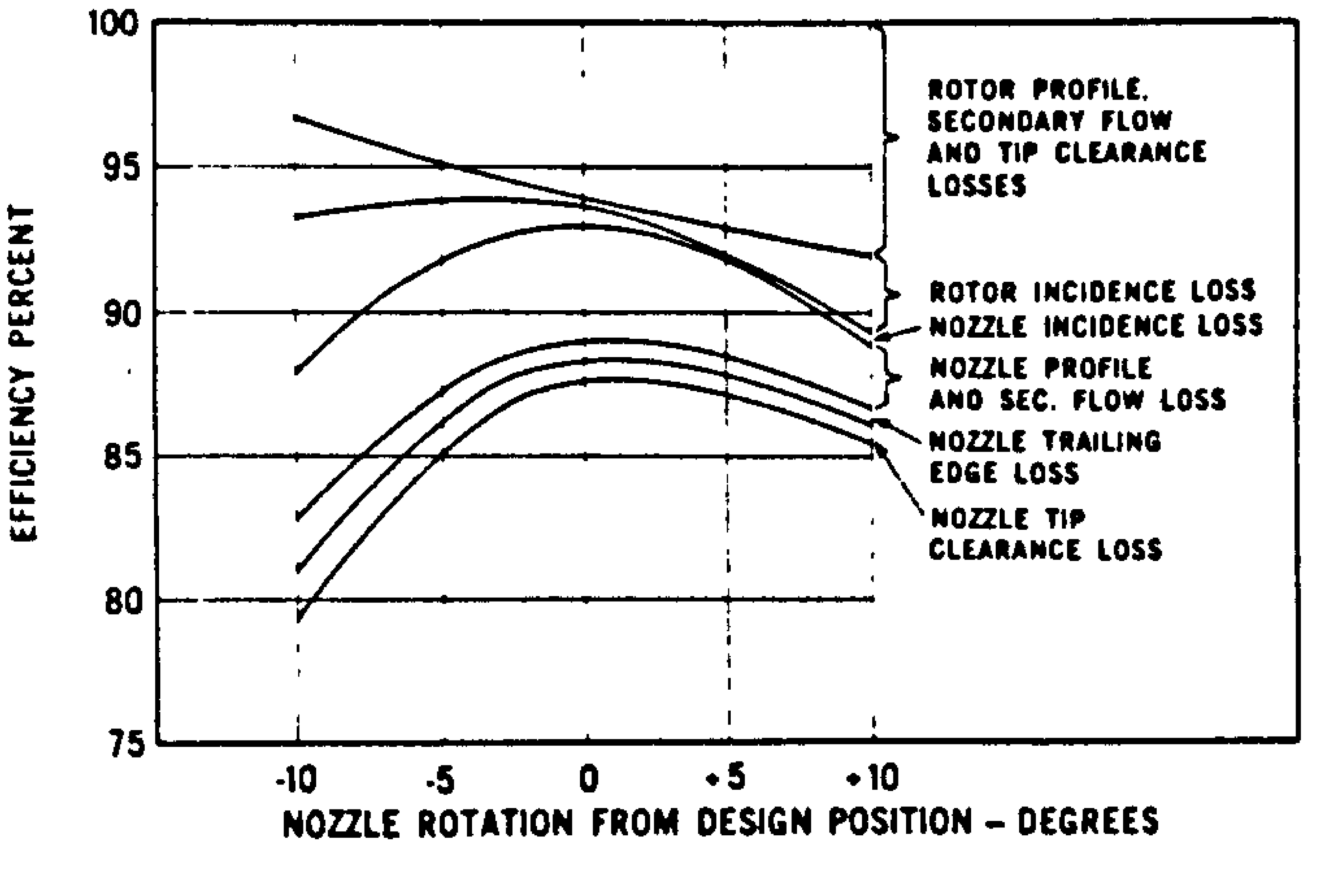



Fig. 9.1 Variation of losses for a variable geometry turbine stage [45]

charged to either the stator or rotor with the rotor incidence loss being accounted for separately to quantify the effect of stator exit swirl on stage performance. Since the incidence of the flow at stator inlet changes as a result of a change in operating conditions and vane setting, an attempt should be made to optimize the vanes for the most important operating range with the leading edge being designed to be insentitive to incidence changes. This also applies to the rotor blades.

In investigating the effect of stator angle change on stage performance, a criterion for comparing the performances of the stage at various angle settings should be chosen but the general trend of efficiency variation with speed or pressure ratio, say, will give an indication of the effect of stator area change on efficiency. It should be expected that a change in stator exit angle will change not only the rotor inlet Mach number but also rotor incidence and both of these affect turbine loss with one being more pronounced than the other depending on whether the stators are opened or closed. The variation in rotor entrance Mach number affects rotor hub reaction and over a certain range of pressure ratios unfavourable reaction may be the dominant contributing factor to stage loss.

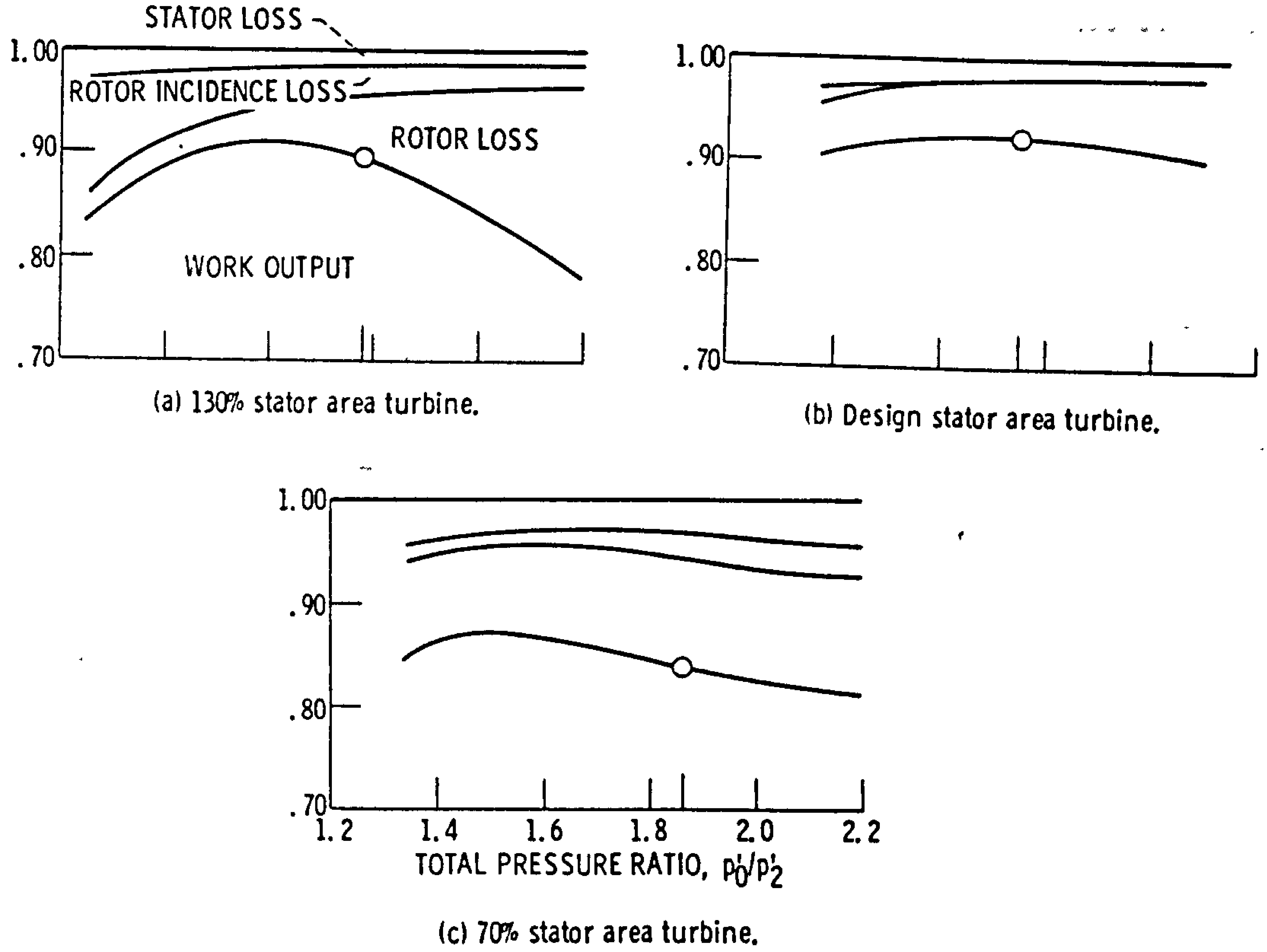



Fig. 9.2 Effect of turbine area on stage losses [30]

Figure 9.2 shows the variation of efficiency on a kinetic energy basis as a function of pressure ratio from an open position through the designed, to a closed position at the design speed. At low and medium pressure ratios, the efficiency of the closed stator is lower than that of the open stator and both are lower than the designed. Rotor losses are more pronounced for the closed stator whereas the open stator's performance is strongly affected by rotor incidence loss. These observations are due to the fact that for the open stator, the exit velocity and hence, Mach number, are low at a reduced flow angle whereas the closed stator increases the exit Mach number with a possible over-expansion of the flow leading to unfavourable reaction at the rotor hub which may cause the flow to separate with high rotor losses occurring [30]. At higher pressure ratios, the reaction at the rotor hub becomes favourable for the closed stator and therefore the rate of change of loss with pressure ratio is reduced whereas the increase in velocity at both the rotor inlet and outlet causes the rotor loss to increase further when the stator is opened. Therefore if good efficiency is to be obtained over the required range of stator settings, control over the various losses and design point reaction must be exercised. The importance of design point reaction on turbine efficiency is also highlighted in [45].

The efficiency of the closed or open turbine is generally at a higher level at lower speeds than at higher speeds; therefore, the engine will benefit from improved sfc when operated at low idling speed.

### Turbine Non-Dimensional Mass Flow

At any given area, the turbine inlet non-dimensional mass flow increases with an increase in speed until a pressure ratio is reached at which choking occurs. The mass flow also increases with an increase in area and the choking mass flow is reduced at reduced pressure ratio as area is increased.

The choked mass flow is directly proportional to the effective turbine area and due to the reduction in nozzle flow coefficient (defined as the ratio of effective area to actual area) as the stators are opened, the change in effective area is lesser than the change in actual area, and if choking occurs somewhere else within the turbine other than the stator, the nozzle coefficient is even much lower. As a result, it is possible that for certain duties, the desired mass flow cannot be passed due to the excessive stator swing that may be required. Therefore, a knowledge of the flow requirements at off-design is required to ensure that a particular turbine does not restrict operation of the engine.

## 9.2 Acceleration Characteristics

As has been mentioned in earlier chapters, the variable area turbine can be used to improve the starting, acceleration, and braking characteristics of a gas turbine unit by suitably changing stator angle to control power split.

During starting, the turbine nozzles can be rotated to a position that will move the operating point on the compressor map from a position close to surge to a position of improved compressor efficiency and higher pressure ratio, with a resultant increase in accelerating torque which then gives self-sustained operation at lower engine speeds. With aircraft engines, altitude starting can also be improved if the stators can be positioned such that the mass flow through the engine is reduced at a point of improved compressor efficiency which results in higher windmilling speeds and hence, higher burner inlet pressures, while maintaining low burner inlet velocities to give conditions favourable for ignition. Depending on the type of engine control employed, higher blow out altitudes can be obtained when a variable area turbine is used.

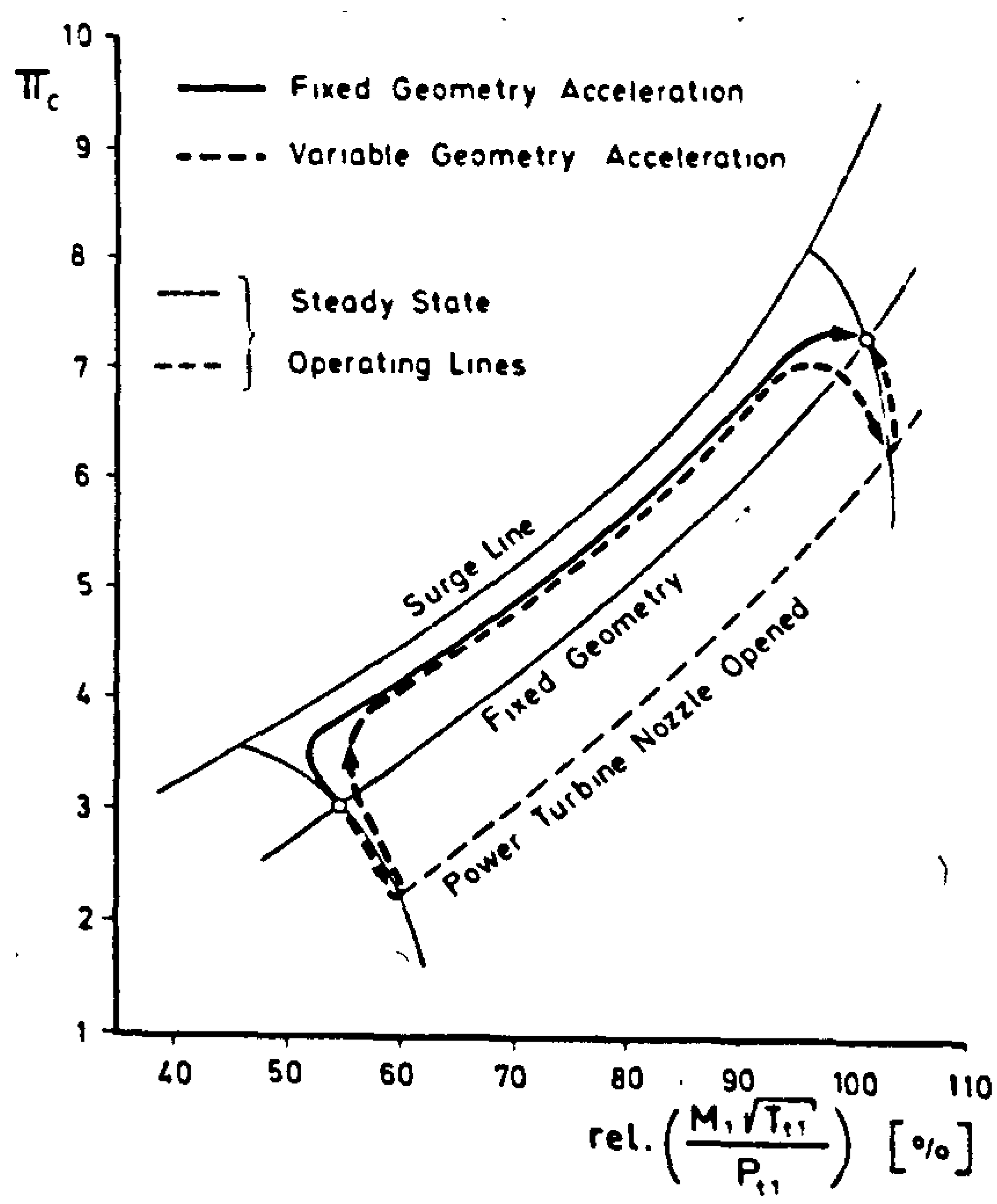



Fig. 9.3 Effect of turbine area change on transient running line [17]

In jet engines, improved engine acceleration is obtained by changing turbine area to lower the relevant compressor pressure ratio in order to increase surge margin which then permits the addition of more fuel to provide a faster acceleration. This is also true for a free turbine engine where opening up the power turbine stators controls the work split between the gasifier and power turbines such that a larger excess power is available to accelerate the gasifier shaft, Fig. 9.3.

Fig. 9.4 shows the maximum acceleration characteristics of a single spool turbojet with a variable area turbine. The

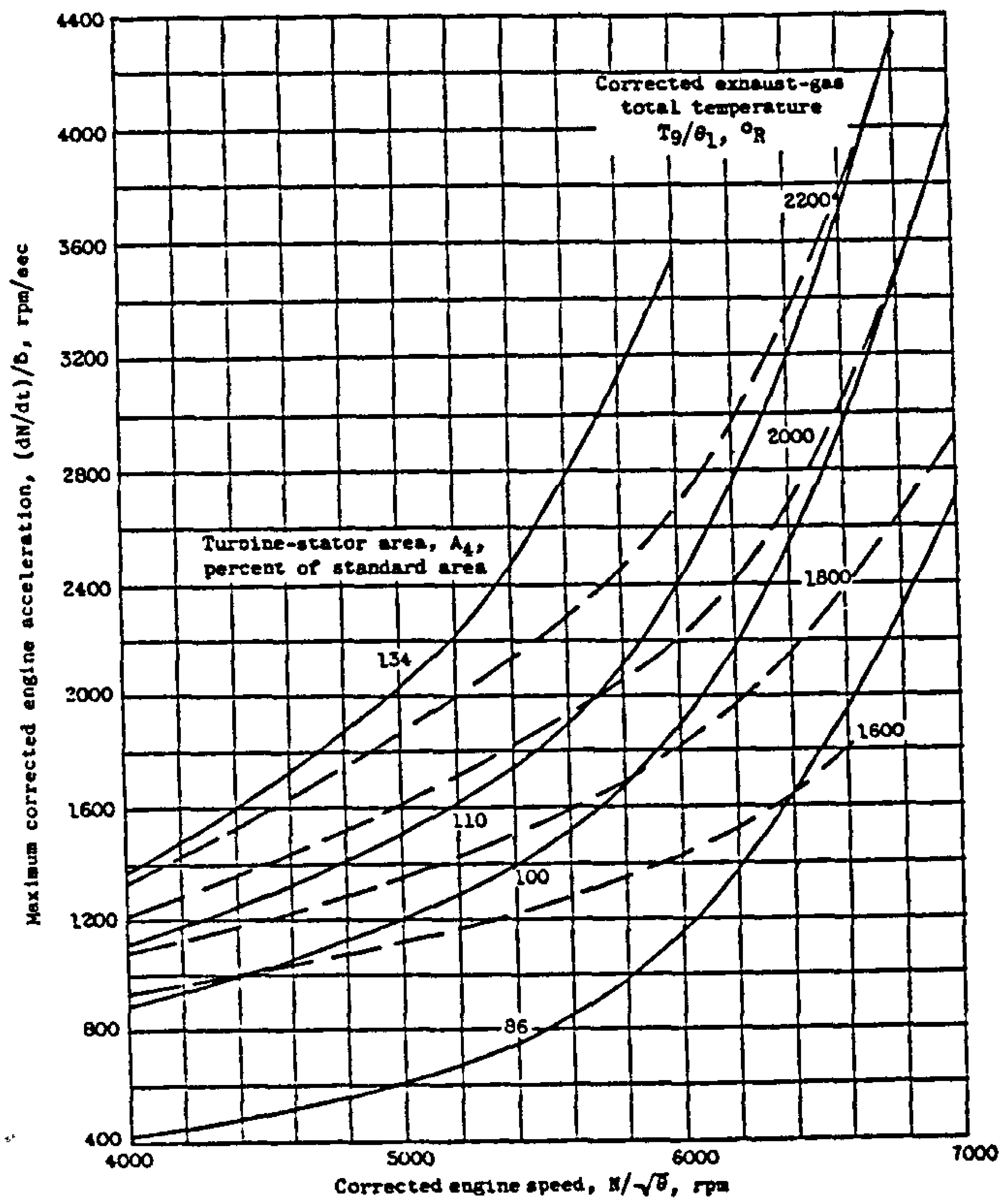



Fig. 9.4 Improvement of turbine acceleration due to variation of turbine area [75]

maximum acceleration is assumed to be limited by compressor surge unless a transient temperature limit is first

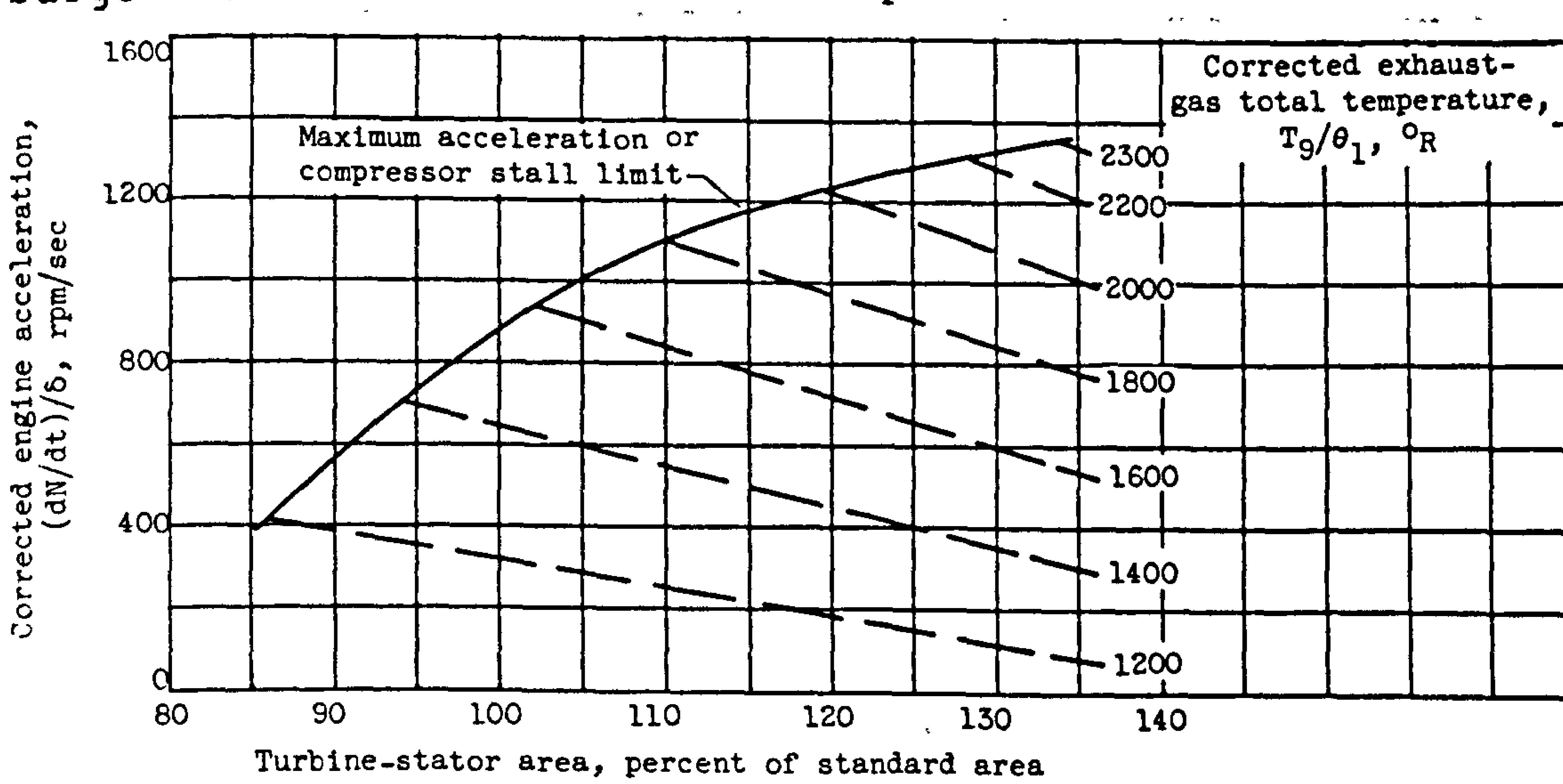



Fig. 9.5 Maixmum acceleration performance of a variable area turbojet [75]

reached. It is seen that opening up the turbine area increases the maximum acceleration at any speed but if a temperature limit is reached, the maximum acceleration is reduced if the stators are opened further as is clearly seen in Fig. 9.5. The transient performance of a variable area power turbine engine is shown in Fig. 9.6 and as expected, the acceleration time is improved. The acceleration time is here defined as the time required to accelerate the shaft from idle speed, that is, 50 percent speed, to full speed.

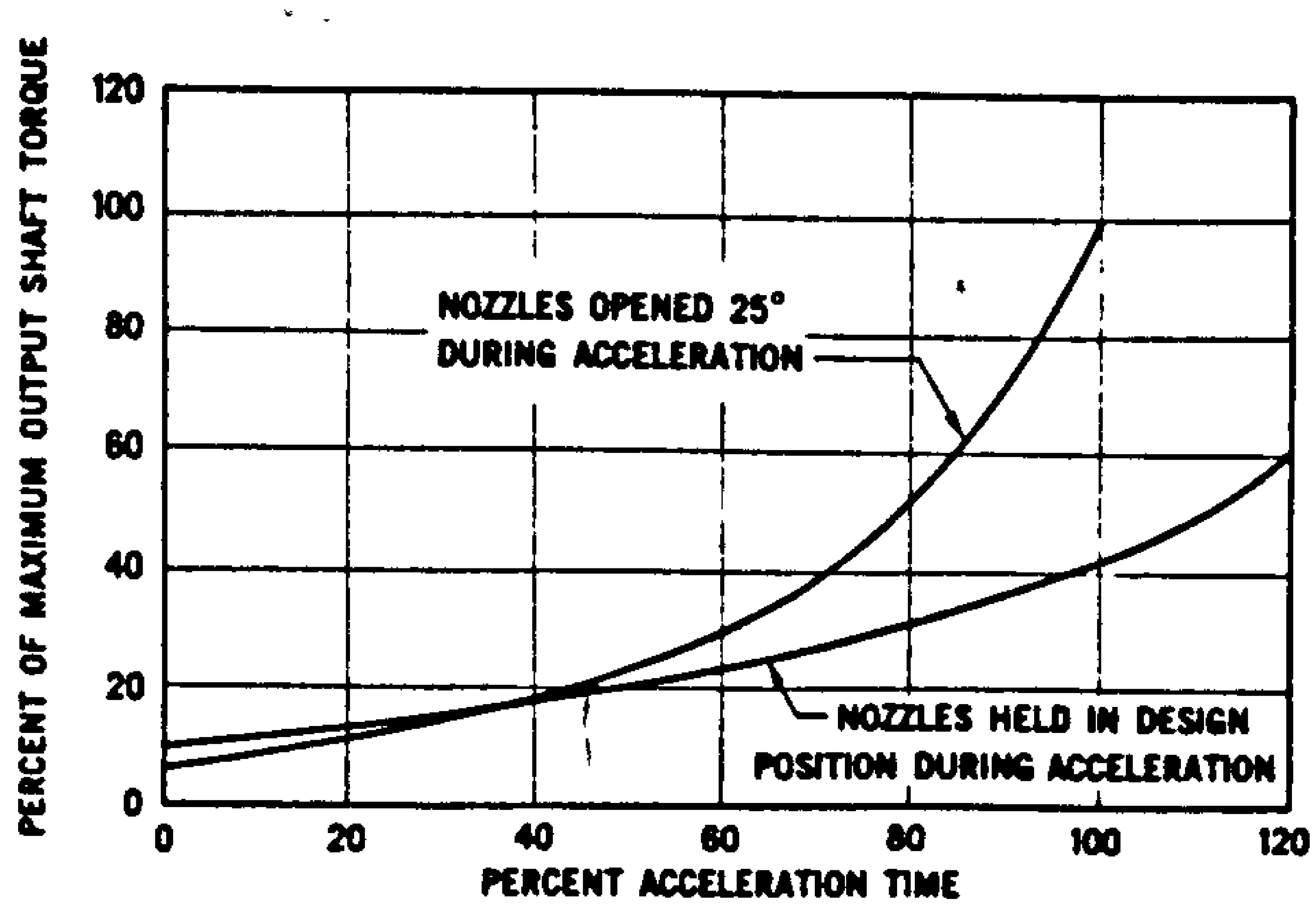



Fig. 9.6 Acceleration performance of a variable area free turbine engine [45]

If the stators are rotated through a large angle such that the flow is directed to impinge on the back side of the rotors then a substantial degree of engine braking can be obtained as a result of negative turbine efficiency. This is illustrated in Fig. 9.7 for a variable area free turbine engine. When the engine is developing full power with both

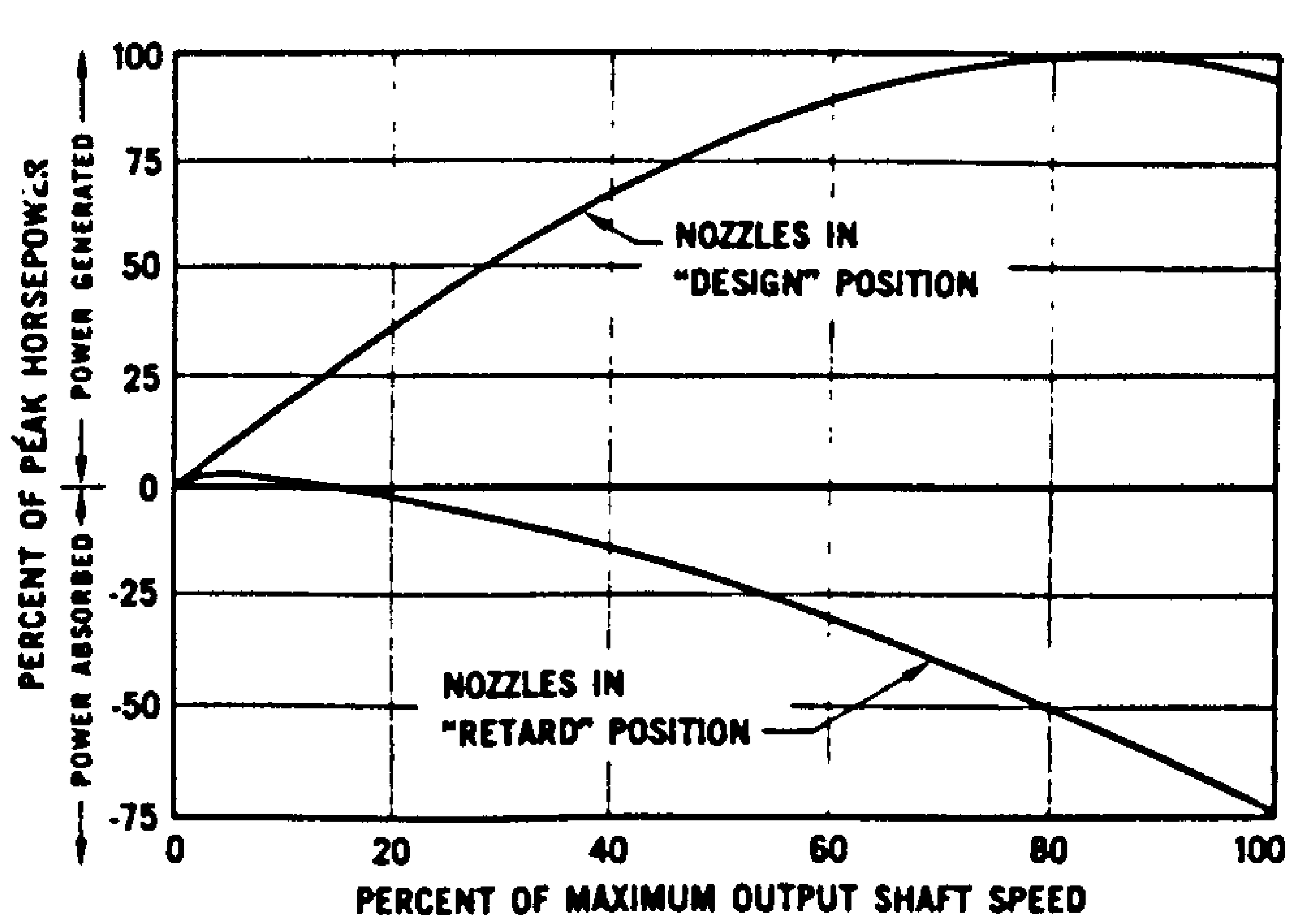



Fig. 9.7 Braking charateristics of a variable area turbine [45]

gas generator and power turbines at maximum speed and with the power turbine vanes in the design position, opening the vanes to the braking position causes the power turbine to absorb power which results in about 75 percent of the full power being dissipated. A limit on the amount of braking power developed is imposed by the limit on exhaust temperature, since a temperature rise occurs across the turbine during the braking process.

## 9.3 Mechanical Integrity

The hot environment in which variable area turbines are expected to operate has been of great concern to gas turbine designers and is one of the reasons why variable area turbines have not been widely used in gas turbines. It is expected that such turbines will first make their way into the less hotter areas of the expansion system and with increased confidence in their use aided by advanced materials and cooling technology, these turbines will eventually find their way into the high pressure end of the expansion process. Various tests have been carried out on variable area turbines of modest inlet temperatures and there have not been any reports of the system being mechanically unreliable. Reference 17 reports some problems encountered with maintaining acceptable bearing clearances which were easily solved. Some investigators have reported hysteresis when turbine area was set. These were not problems associated with the vanes but rather with the actuation mechanism; therefore, with careful design, satisfactory performance of the system can be obtained.

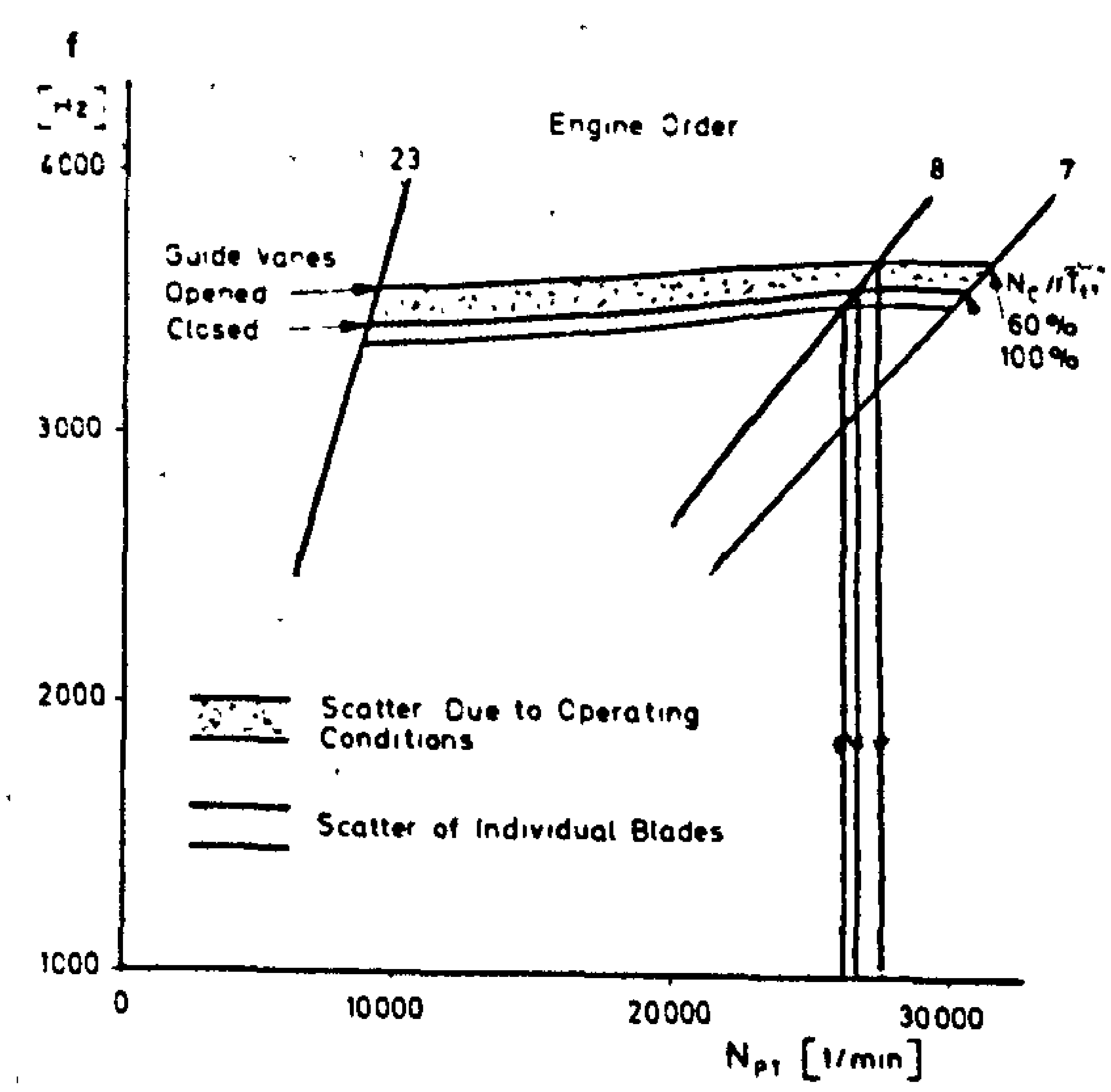



Fig. 9.8 Extension of resonance range caused by turbine area variability [17]

From the point of view of blade vibration, the investigation carried out in [17] on a single stage variable area power turbine discovered that because variable geometry can change the flow conditions at inlet to the turbine independent of gas generator speed, there is an extension in the scatter range of the speed at which resonance occurs, Fig. 9.8. Also, the point at which the nozzle guide vanes are pivoted is of importance since the axial clearances between the vanes and blades vary as the vanes are rotated with the possibility that the guide vane wake may excite resonance of the blades. It was also observed that interfering objects in the flow ahead of the vanes are potential exciters of vibration. Therefore, consideration should be given to the distribution of support struts and probes ahead of variable turbine vanes, the scatter in frequencies and speeds at which resonance occurs, and the spacing between variable vanes and the stage rotors during the design of a variable area turbine.

## 9.4 VIGV Thrust Modulation Capability

Variable geometry in a compressor is generally used to improve the performance of the inlet stages at off-design so that the useful incidence range can be increased to give a larger variation in mass flow. Its potential to control, engine airflow and hence, thrust, at constant speed has been realized and substantial performance benefits may be obtained when such a compressor or fan is used to improve engine response, more so of lifting engines.

There are mechanical and aerodynamic design considerations that are to be taken into account when compressor variable geometry is used to improve the response of an engine. On one hand, VIGVs could be used either alone or in combination with variable stators to provide flow control while on the other hand variable pitch could be used on the first stage rotors to do the same job. Each has advantages and disadvantages when compared with the other but in general, a vane assembly poses less difficulty as far as control is concerned.

To cover the thrust requirements of the V/STOL aircraft examined in the last chapter, the VIGV fan has to operate over its entire range of angle setting and not only does this affect the performance of the fan but also that of the inlet. For effective thrust control, the inlet flow should remain attached at all operating conditions so that loss of thrust due to total pressure loss and high fan vibratory stresses arising from the distorted flow are avoided. However, the possible thrust range that can be obtained without inlet separation occurring depends greatly on the separation criteria used [60] and even when the flow is separated, thrust can be further modulated with acceptable levels of fan blade stresses [43,60,61]. At the high thrust end, separation may be caused by the large mass flows encountered whereas high angles of attack are responsible for separation at low thrust. The VIGVs are themselves responsible for limiting the angle of attack capability of the inlet and fan blade stresses may be increased at some engine order (more so at transitional flight) not present when the vanes are not installed.

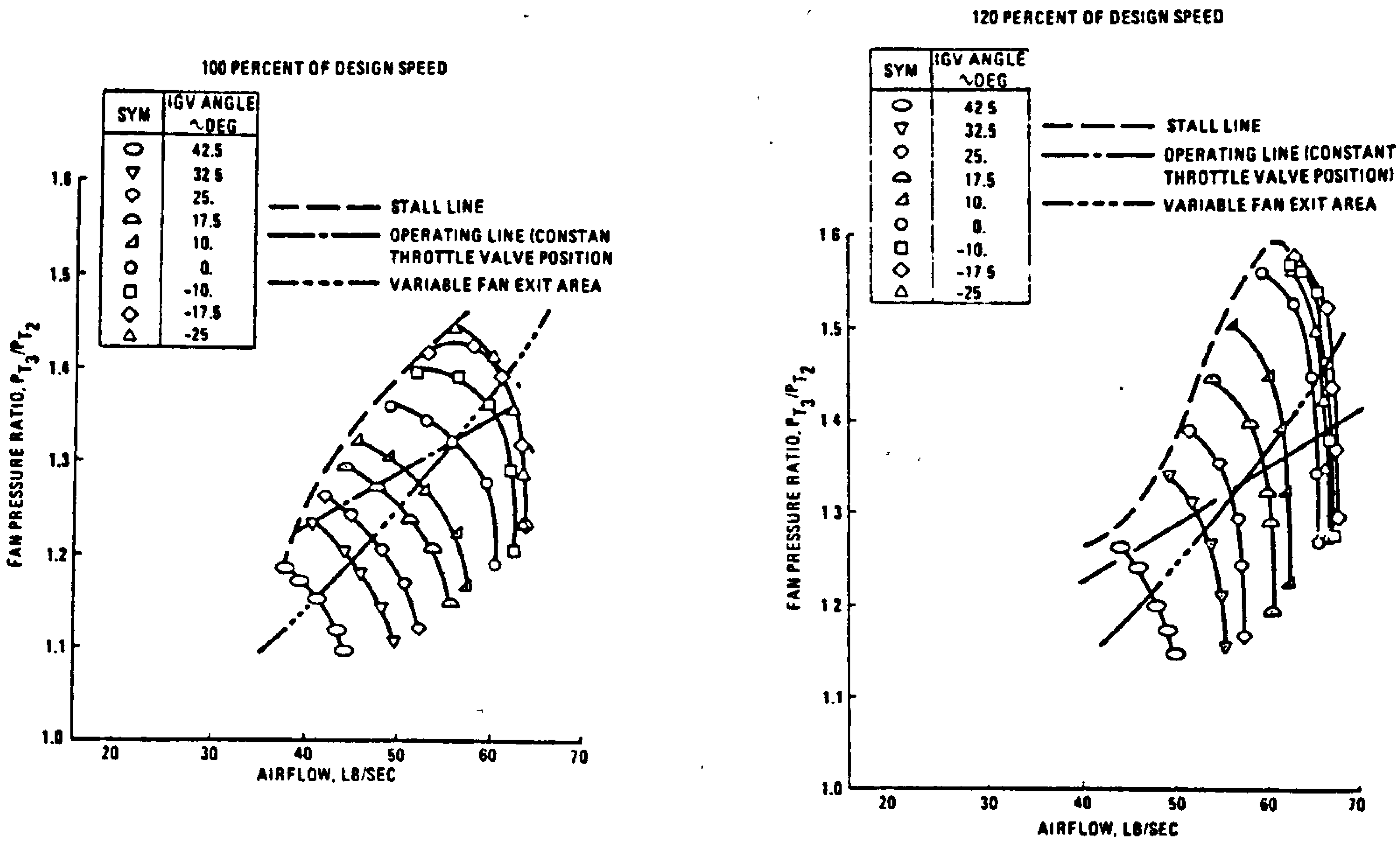



Fig. 9.9 Design consideration for rapid response engines [43]

Also, the thrust can be limited at the high end by choking and at the low end, by fan stall. The stall line may change with fan speed as can be seen in Fig. 9.9 and therefore, a knowledge of the complete fan performance is essential before selecting a fan speed schedule at which VIGV attenuation is used for modulating thrust. It may be necessary to use some other control such as nozzle area variation to increase the thrust range.

From the findings of the work undertaken in [16], variable pitch rotors are much more effective in controlling thrust at constant speed than do VIGVs. The characteristics of the variable pitch rotor and VIGV fans tested are reproduced in Figs 9.10 and 9.11 with their matching operating lines, from which it can be seen that variable pitch rotors can easily produce a large thrust range at constant speed while VIGV fans may need a considerable change in speed to meet the thrust requirements.

A requirement for a twin-engine V/STOL aircraft is that if an engine fails during vertical flight, the operating core must be able to drive both fans to produce the required thrust to accomplish vertical landing safely. This means that power has to be transferred from the operable core to the inoperable fan while high-flowing the fan on the

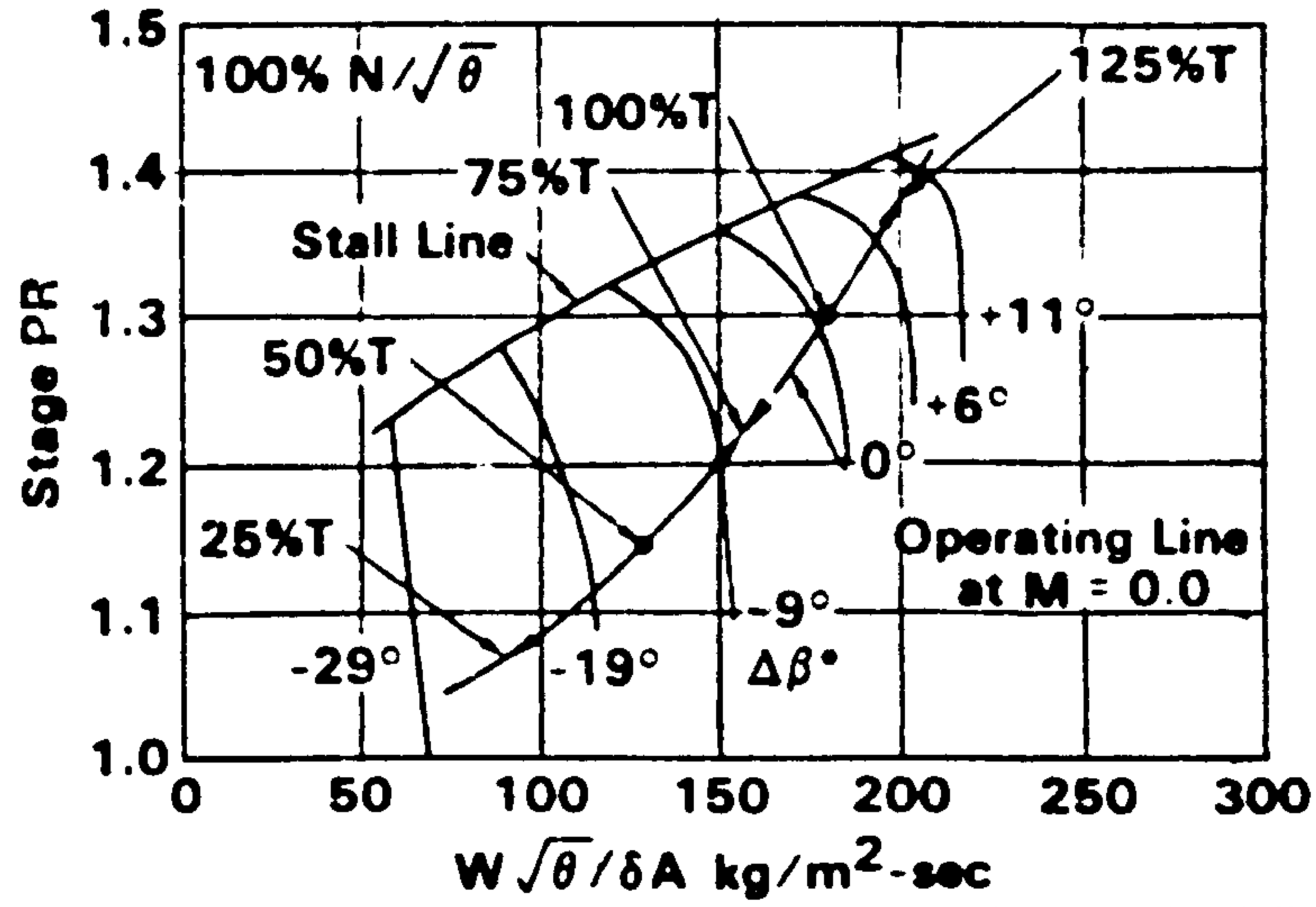



Fig. 9.10 Operating line to satisfy thrust requirements of a variable pitch rotor fan engine [16]

inoperable engine and low-flowing that on the operable engine to produce balanced thrust. This requirement determines the size of the gas generator.

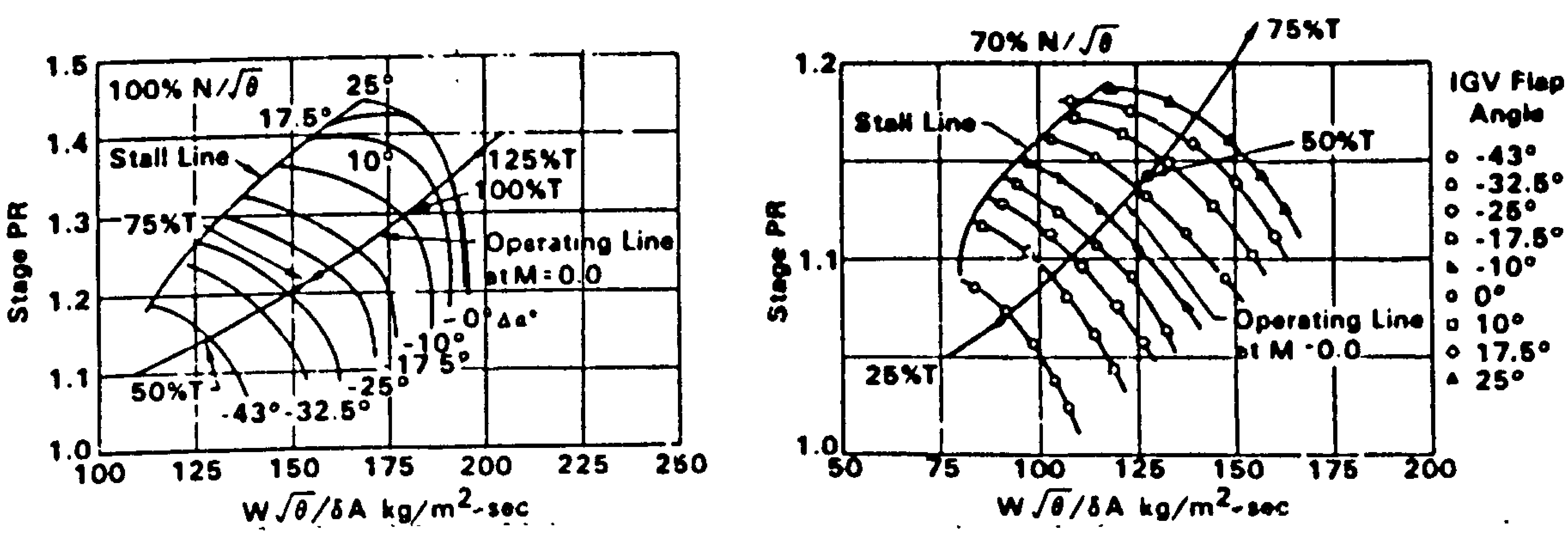



Fig. 9.11 Speed augmentation of variable inlet guide vane thrust modulation [16]

Most thrust of a high bypass turbofan is developed in the tip and if IGVs covering the entire annulus are used to modulate thrust then the core is desupercharged when the vanes are closed to reduce fan thrust. In meeting the one engine inoperative requirement, core performance is reduced as the vanes are low-flowed to give balanced thrust and as a result an even bigger core is required to produce the driving power for the fans.

If the vanes are positioned to cover the outer part of the annulus so that only fan tip performance is affected when thrust is modulated, then hub performance and hence, core supercharging, could be maintained while thrust decreases. A smaller core is then required to do the same job and the saving in engine weight due to a smaller core and lighter VIGVs could give substantial savings in cruise sfc. As reported in [2,46], part span VIGVs are an effective method of modulating thrust at constant shaft speed, and beneficial effects on the engine system can be obtained.

## 9.5 Recommendations for Future Work

Further work that can be carried out can be categorized into program development and cycle studies.

One area for program development is that of installed performance analysis. A comprehensive and complete model to arrive at the installed thrust was presented in Chapter 4. However, because of the lack of data on installation losses, the model used in the program is only good for trend studies and moreover, the model assumes an installation wherein the engines are buried in the fuselage as is typical of fighter aircraft. As a result, the installed performance "switch" requires a value for the ratio of maximum nozzle exit area to maximum fuselage area to turn it on. This switch should be made independent of engine installation. Also, the inlet capture area should be specified rather than calculated from the specified off-design condition.

The model for the variable area mixer comprises that of a variable area convergent nozzle and a fixed area mixer, and since the calculations for the bypass stream are done before that of the main stream, a static pressure has to be specified to which the flow in the bypass stream expands. The MIXFUL brick which does the calculations for the fixed area mixer should be modified to include the variable area mixer. This may require including area as a possible variable.

A wider range of component performance maps should be incorporated so that the effects of different variable geometry designs on engine performance can be considered. The importance of this was noted in the previous section when the performance of V/STOL cycles was discussed where it was mentioned that variable pitch rotors could give better response rates than do VIGVs. Also, more representative installation maps should be included with the user having the option to input his own maps.

Going back to the part-span VIGVs, the pressure of the airflow entering the bypass duct will be different from that entering the core since fan hub performance differs from that of the tip. The program should be modified to take care of this.

The graphics routine should be extended to produce "carpet" plots for design point performance analysis. This, it is hoped, will speed up the cycle selection process.

Variable cycles should be studied in more depth. As reported in the previous chapter, the variable bypass ratio cycle has great potential for improving both subsonic and supersonic performances of an aircraft gas turbine. This cycle should be investigated further to find out what the limitations are on the range of bypass ratios that can be attained. The scheduling of various areas should also be looked at at various points in a flight envelope with a view to optimizing performance.

Other types of variable cycle obtained principally by flow switching and augmentation should also be investigated with the view of identifying certain types of cycle which give substantial performance gains for certain duties.

# CHAPTER 10

## CONCLUSIONS

Several gas turbine cycles incorporating one or more variable geometry components were studied to find out if such cycles have any potential to improve the off-design performance of gas turbine engines. The performances of the study engines were compared with those of conventional fixed geometry engines and the results obtained led to the following conclusions.

The off-design performance of gas turbines can be significantly improved by the use of variable geometry in one or more components. The improvement may occur as a saving in fuel consumption, or an increase in output, or both, brought about by improvement in component performance and/or cycle parameters. Rapid engine response may also be the result of the improvement.

The variable area turbine can be used to control the operating line on a compressor such that a compressor pressure ratio or maximum cycle temperature can be maintained at the design value or at a higher level than that obtained with conventional fixed geometry engines, resulting in lower fuel consumptions at a given output.

The regenerative shaft power cycle benefits more from operation at higher temperatures than does the simple cycle which may give a worse performance compared with the conventional cycle, whereas operation at higher pressure ratios is beneficial to the simple cycle and not the regenerative cycle.

Variable geometry in compressors and turbines can be used to improve engine response dramatically. It is possible to modulate output at constant gas generator speed by using variable inlet guide vanes to control the flow into a compressor. It may be necessary to use other power modulating devices to augment VIGV power attenuation in certain parts of the power spectrum either to preclude a higher fuel burn due to low compressor efficiencies or to further modulate power when the vanes can no longer be rotated any further.

The installed performance of an aircraft engine can be improved by using both a variable area turbine and a variable area propelling nozzle to hold compressor design point fixed over a wide range of operating conditions and power settings which results in reduced inlet and aft-end losses to give higher thrusts with an accompanying reduction in specific fuel consumption levels. A turbojet engine can benefit more from this mode of control as the installed fuel burn obtained at subsonic speeds can be quite comparable to that of a turbofan while at the same time maintaining its good supersonic characteristics.

Cycle variability has the potential to improve the off-design performance of a gas turbine engine as engine cycle can be tailored to suit a current duty. By varying bypass ratio, the thrust of a turbofan engine can be increased when an engine limit is reached by the use of variable geometry to increase one or more performance variables while keeping the limiting variable or parameter fixed. With afterburning engines, there is scope for a large saving in fuel consumption over that of conventional engines in the afterburning thrust range. The advent of such engines will not only revolutionize gas turbine technology but also give users increased confidence in the operation of their fleet.

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# APPENDIX A

## The Balancing Technique

An iteration is carried out using the multi-variable Newton-Raphson method to solve the governing differential equations. During an off-design run, guesses are made for some variables in order to effect the calculations in a brick. Also, the various matching constraints as defined by the compatibility equations are to be met. The independent variables are perturbed and matching errors are generated which are all reduced to zero when the engine is balanced.

When an off-design point is given, a pass is made around the engine and the matching errors caused by a change of operating conditions are generated as base errors. Thereafter, each independent variable is perturbed and the changes in the errors caused by a change in each variable are noted during each loop. The total change in the ith error after all the variables have been perturbed can be taken as the sum of the individual change in that error caused by a change in the individual variables, viz.,

$$dE_i = \sum_{j=1}^{j_{max}} \frac{\partial E_{ij}}{\partial V_j} dV_j \qquad (A.1)$$

where,

$E$ = Error
$V$ = Variable
$i$ = $i_{th}$ error
$j$ = $j_{th}$ variable
$j_{max}$ = Maximum number of variables
$E_{ij}$ = Change in $E_i$ caused by a change in $V_j$

On the assumption that the change in a variable is small, one can write

$$dE = E - E_b \qquad (A.2)$$
$$dV = V - V_b \qquad (A.3)$$
$$\frac{\partial E}{\partial V} = \frac{dE}{dV} \qquad (A.4)$$

with the suffix b representing the base value. On substituting these expressions into (A.1) and remembering that the errors must reduce to zero at the balanced point, we have,

$$E_i - E_{bi} = -E_{bi} = \sum_{j=1}^{j_{max}} \frac{\Delta E_{ij}}{\Delta V_j} dV_j \qquad (A.5)$$

A matrix loaded with the $\Delta E/\Delta V$s is inverted to solve (A.5) for $dV_j$ which in turn gives the new values of the variables using the expression

$$V_j = V_{jb} + dV_j \qquad (A.6)$$

If the errors were indeed linear functions of the variables, then a balanced point would be obtained with the new values of the variables. However, this is not so and therefore the process is repeated a number of times until convergence eventually occurs within the given tolerance. The solution thus obtained gives the correct values of the variables which satisfy the various matching constraints.

# APPENDIX B1

Brick Name: INTAKE

Purpose: To calculate free stream static and total conditions at a given altitude and flight Mach number, using the International Standard Atmosphere, with allowance for deviation from standard static temperature. Also, to calculate conditions at exit from the intake, using a value of Pressure Recovery which is either stated explicitly, or calculated from the USAF Standard Pressure Recovery-Flight Mach number relations, or from maps which give pressure recovery as a function of Mach number. Both internal and external installed thrust loss can also be calculated from pressure recovery functions and inlet drag maps, respectively.

Station Vectors:
1. Free Stream
2. Intake Outlet

Essential Station Vector Data: Inlet Mass Flow (lb/s or kg/s)

Brick Data:
1. Altitude (ft or m)
2. ISA Deviation (K)
3. Flight Mach Number
4. Either,
   (a) Explicit Pressure Recovery (in the range 0 to 1.0),
   or (b) the value "-1.", indicating that the USAF standard value is required,
   or (c) a value (in the range -2. to -6.) indicating which pressure recovery map is used.
5. A value (in the range 1 to 4) indicating the number of the drag map for installed analysis. This is optional.
6. Either,
   (a) Map number of intake characteristic
   or (b) absolute value of non-dimensional mass flow, $\frac{W_B\sqrt{T_B}}{A_C P_B}$

Engine Vector Results: Momentum Drag (lbf or N)

Other Quantities
Calculated:

1. Rest of Inlet Station Vector
2. Outlet Station Vector
3. Explicit Pressure Recovery (if option 4(b) or 4(c) of Brick Data applies)
4. Scaling Factor if BD(6) > 0. (design point only)

Possible Variables: Nil

Errors Generated: Nil

# APPENDIX B2

**Brick Name:** COMPRE

**Purpose:** To calculate the outlet conditions from a compressor (or fan) and the compressor work, given the inlet conditions, design point pressure ratio and isentropic efficiency, and off-design values of the relative non-dimensional speed and of distance from the surge line.

**Station Vectors:**

1. Compressor (or fan) Inlet
2. Compressor (or fan) Outlet

**Essential Station Vector Data:**

1. Inlet Fuel-Air Ratio
2. Inlet Mass Flow (lb/s or kg/s)
3. Inlet Total Pressure (atm)
4. Inlet Total Temperature (K)

**Brick Data:**

1. $Z = \dfrac{R - R_{choke}}{R_{surge} - R_{choke}}$ R = Pressure Ratio

   If this value is given the design value "-1.", a default value of 0.85 is invoked.
2. PCN = rotational speed N, as expressed as a decimal fraction of a standard value. If this is given the design value "-1.", a default value of 1.0 is invoked.
3. Design Pressure Ratio $R_{des}$
4. Design Isentropic Efficiency $\eta_{is}$ (in the range 0. to 1.)
5. Error Selection ( =1., if error is required, otherwise, = 0.)
6. Compressor Map Number.
7. Design IGV Angle, $\zeta$ (if variable geometry map is used; otherwise, -100.)
8. PCNLIM = Limit on rotational shaft speed, $N_{max}/N_{des}$.
9. Surge Margin Limit, if required (maximum Z).
10. Switch ( "= 1." if compressor pressure ratio is fixed).
11. Switch for mode of operation when surge margin limit is reached, ( "=1." for conventional control, or "2." if Surge Margin is fixed).
12. Switch ( "=1." if mode of operation

of variable geometry of fan or compressor affects operation of the downstream variable fan or compressor; otherwise, "-1.").

Engine Vector Result: COMWK = Compressor Work (HP or W)

Other Quantities Calculated:
1. Outlet Station Vector
2. Scaling Factors (design point only)

Possible Variables:
1. Z (see BD(1) above)
2. PCN (see BD(2) above)
3. $\zeta$ (see BD(7) above)

Possible Errors Generated:
1. $\frac{W_{outlet} - W_{inlet}}{W_{inlet}}$ (if BD(5) = 1.)
2. $\frac{PR - PR_{des}}{PR_{des}}$ (only if BD(10) =1.)

# APPENDIX B3

**Brick Name:** TURBIN

**Purpose:** To calculate the outlet conditions from a compressor or power turbine with fixed or variable geometry, given the inlet conditions. The relevant design point thermodynamic variables are required to effect operation of this brick.

**Station Vectors:**

1. Turbine Inlet
2. Turbine Outlet

**Essential Station Vector Data:**

1. Inlet Fuel-Air Ratio
2. Inlet Mass Flow (lb/s or kg/s)
3. Inlet Total Pressure (atm)
4. Inlet Total Temperature (K)

**Brick Data:**

1. Auxiliary Work AUXWK (compressor turbine) or power output POWER required (power turbine) (HP or W)
2. (a) Relative non-dimensional inlet mass flow TF (in range 0. to 1.). If this is given the design value "-1.", a default value of 0.8 is invoked (if switch is set "-1." in BD(10)),

   or (b) Relative work function $\Delta H/T$ (in range 0. to 1.). If this is given the design value "-1.", a default value of 0.7 is invoked if the turbine area is fixed, or 0.4 if variable (if switch is set "=1." in BD(10))
3. (a) Relative non-dimensional speed CN (in range 0. to 1.). If this is given the design value "-1.", a default value of 0.6 is invoked (if switch is set "=-1." in BD(10)),

   or (b) Relative Power/Speed ratio $\Delta H/N^2$ (in range 0. to 1.). If this is given the design value "=-1.", a value of 0.5 is invoked (if switch is set "=1." in BD(10)).
4. Design Isentropic Efficiency, $\eta_{is}$ (in range 0. to 1.)
5. Relative rotational speed PCN (in range 0. to 1.) (for power turbine only; for compressor turbine, the value "-1." is used).

6. Compressor number (from "1." at low pressure end) (for compressor turbine only; for power turbine, the value "0." is used).
7. Turbine map number.
8. Power law index, n, in the relation AUXWK or POWER $\propto$ PCN$^n$. If n = "-1.", AUXWK or POWER is assumed constant.
9. Compressor work COMWK for compressor (for compressor turbine; for power turbine the value "-1." is used).
10. Switch ("=1." if variable area turbine map is used; otherwise, "-1.").
11. Design Turbine Area Ratio $A/A_{min}$ if BD(10) =1.
12. Design Exit Mach Number (LP Turbine only) if BD(10) = 1.
13. Limiting value of exit Mach number (LP Turbine only) if BD(10) = 1.
14. Switch ("=1." if driven compressor(s) use variable geometry to control flow when turbine reverts to fixed mode of operation; otherwise, "-1." for speed control).
15. Switch ("=1." if variable area turbine operation affects operation of variable area turbine immediately upstream)

Engine Vector Results: Nil

Other Quantities Calculated:

1. Outlet Station Vector
2. Scaling Factors (design point only)

Possible Variables:

1. TF (see BD(2) above)
2. $\Delta H/T$ (see BD(2) above)
3. PCN or POWER (Power turbine only - see BD(5) and BD(1) above).
4. $A/A_{min}$ (see BD(11) above)

Possible Errors Generated:

1. $$\frac{TF_{calc} - TF_{map}}{TF_{calc}}$$
2. $$\frac{H/T_{calc} - H/T_{map}}{H/T_{calc}}$$

## APPENDIX B4

**Brick Name:** MIXFUL

**Purpose:** To calculate the outlet conditions resulting from the mixing of two flows (at constant final area) with given inlet conditions and with full allowance for total pressure change resulting from momentum balance. If one inlet flow is much smaller than the other (e.g. cooling air bleedback), the simpler MIXEES may be used instead.

**Station Vectors:**

1. Inlet No. 1
2. Inlet No. 2
3. Outlet

**Essential Station Vector Data:**

1. Fuel-Air Ratio
2. Mass Flow (lb/s or kg/s)
3. Static Pressure (atm)
4. Static Temperature (K)
5. Velocity (ft/s or m/s)

**Brick Data:**

1. Number of Compressor (fan) providing stream number 2 (i.e. bypass stream)
2. Switch ("=1." if BD(3) = Mach Number; "-1." if BD(3) = Static Pressure)
3. Mach Number or Static Pressure of Stream number 1 (according to value of BD(2))
4. Mach Number Limit in stream 2, if necessary

**Engine Vector Results:**

1. Fan Pressure Ratio Multiplier
2. Main Compressor Pressure Ratio Multiplier (Note: These multipliers are to be applied manually to the previously specified values of these pressure ratios if the latter are such that mixing is impossible)

**Other Quantities Calculated:** Outlet Station Vector

**Possible Variables:** Nil

**Errors Generated:** $\dfrac{p_1 - p_2}{p_1}$ where p = static pressure

## APPENDIX B5

**Brick Name:** PREMAS

**Purpose:** To calculate the outlet conditions from a component such as a splitter, bleed, auxiliary door, bypass duct or jet pipe, given the absolute and/or relative changes of mass flow and/or total pressure, according to the equations.

$$W_{out1} = \lambda_w \times W_{in} - \Delta W$$
$$P_{out1} = \lambda_p \times P_{in} - \Delta P$$
$$T_{out1} = T_{in}$$

and also (optionally)

$$W_{out2} = W_{in} - W_{out1}$$
$$P_{out2} = P_{out1}$$
$$T_{out2} = T_{in}$$

**Station Vectors:**

1. Inlet
2. Outlet No. 1
3. Outlet No. 2 (optional: for use with a splitter or bleed. Write as "0" if not used).

**Essential Station Vector Data:**

1. Inlet Fuel-Air Ratio
2. Inlet Mass Flow (lb/s or kg/s)
3. Inlet Total Pressure (atm)
4. Inlet Total Temperature (K)

**Brick Data:**

1. $\lambda_w$
2. $\Delta W$ as defined under "Purpose" above
3. $\lambda_p$
4. $\Delta P$
5. Switch ("=1." for auxiliary door; otherwise "-1.")
6. Mach Number at Outlet No. 1, if area is to be calculated

**Engine Vector Results:** Nil

**Other Quantities Calculated:** Outlet Station Vector(s)

**Possible Variable:** $\lambda_w$ (for bypass splitter of turbofan)

**Errors Generated:** Nil

## APPENDIX B6

| | |
|---|---|
| Brick Name: | NOZCON |
| Purpose: | To calculate the outlet conditions and gross thrust, if required, from a convergent nozzle, given the inlet conditions and (optionally) exit area. |
| Station Vectors: | 1. Nozzle Inlet<br>2. Nozzle Outlet<br>3. Ambient or Specified |
| Essential Station Vector Data: | 1. Inlet Fuel-Air Ratio<br>2. Inlet Mass Flow (lb/s or kg/s)<br>3. Inlet Total Pressure (atm)<br>4. Inlet Total Temperature (K)<br>5. Ambient or Specified Static Pressure (atm) |
| Brick Data: | 1. Switch( "=1." if exit area "floats", i.e., if this area is calculated afresh for each point; or "-1." if the exit area is fixed).<br>2. Number of Variable Area Turbine to which nozzle is linked, if nozzle is in bypass stream of a Variable Area Mixer. |
| Engine Vector Result: | Gross Thrust, if a propelling nozzle (lbf or N) |
| Other Quantities Calculated: | 1. Outlet Station Vector<br>2. Nozzle Thrust Coefficient |
| Possible Variable: | Static Pressure |
| Error Generated: | $\frac{P_{req} - P}{P}$<br>where P = calculated value of inlet total pressure and $P_{req}$ = value of P required to pass specified mass flow. (Note: this error is calculated only if the exit area is fixed, i.e., if BD(1) = -1.) |

# APPENDIX B7

**Brick Name:** NOZDIV

**Purpose:** To calculate the outlet conditions and gross thrust from a convergent - divergent nozzle, given the inlet conditions and (optionally) throat area.

**Station Vectors:**
1. Nozzle Inlet
2. Nozzle Outlet
3. Ambient

**Essential Station Vector Data:**
1. Inlet Fuel-Air Ratio
2. Inlet Mass Flow (lb/s or kg/s)
3. Inlet Total Pressure (atm)
4. Inlet Total Temperature (K)
5. Ambient Static Pressure (atm)

**Brick Data:**
1. Switch("=1." if throat and exit areas "float", i.e., if these areas are calculated afresh for each point, or "=-1." if these areas are fixed, or "=-2." if throat area is fixed and exit area floats)
2. Throat Area (if not previously calculated, otherwise "=-1.")

**Engine Vector Result:** Gross Thrust (lbf or N)

**Other Quantities Calculated:**
1. Outlet Station Vector
2. Nozzle Thrust Coefficient

**Possible Variables:** Nil

**Errors Generated:**

$$\frac{P_{req} - P}{P}$$

where P = calculated value of inlet total pressure, and Preq = value of P required to pass specified mass flow. (Note: this error is calculated only if throat area is fixed, i.e. if BD(1) = -1.)

# APPENDIX B8

**Brick Name:** DUCTER

**Purpose:** To calculate the outlet conditions from a duct, given the inlet conditions and relative total pressure loss; also, if called for, to calculate the reheat fuel flow, given the outlet total temperature and combustion efficiency.

**Station Vectors:**

1. Duct Inlet
2. Duct Outlet

**Essential Station Vector Data:**

1. Inlet Fuel-Air Ratio
2. Inlet Mass Flow (lb/s or kg/s)
3. Inlet Total Pressure (atm)
4. Inlet Total Temperature (K)
5. Outlet Total Temperature (K) (if reheat is specified by BD(1)).

**Brick Data:**

1. Switch ("=0." if no reheating at all; "=1." if reheating is required later; "=2." if reheating is required now and burner is separated from engine; "3." if reheating is required now and burner is attached to engine)
2. $P/P_{in}$ = Total Pressure Loss/Inlet Total Pressure
3. Combustion Efficiency (in range 0. to 1.) (if BD(1) > "0.")
4. Limiting value of Fuel Flow (or 1000000. if not needed)
5. Duct Exit Mach Number (optional and for a duct preceding a variable area nozzle) (Note: this determines duct exit area and hence, the maximum value of area to which the downstream nozzle exit area can expand).

**Engine Vector Results:** Fuel Flow (lb/s or kg/s) (Note: this must always be allowed for, even if BD(1) = "0.")

**Other Quantities Calculated:** Outlet Station Vector

**Possible Variables:** Nil

**Errors Generated:** Nil

# APPENDIX C1

## Change of Variable Procedure

The program incorporates a facility that will allow the user to change any of the prescribed variables at any time during a run. The data is given as part of the off-design input data sandwiched by two minus fours (-4s). A variable can be added to or deleted from a codeword. The procedure is,

```
-4
I  VW  D1  D2  VN  VO
.
.
.
-4
```

where,

- I = Component or codeword number in the codeword hierarchy
- VW = "1" or "2" for first or second variable (i.e. V or W)
- D1 = Station Vector number or "-1" (Brick Data)
- D2 = Station Vector or Brick Data item number
- VW = New value of SV or BD item associated with the changed variable
- VO = New value of SV or BD item previously associated with the changed variable

The last four data values are all optional

Examples.

(a) 2 1 -1 6 .8 1

This means that the first variable in the second codeword has now been changed to brick data item 6 with a new value of .8 while the brick data or station vector item previously connected to the this variable now takes a value of unity.

(b) 6 2

This simply means that the second variable associated with the sixth codeword has been deleted.

# APPENDIX C2

## Parameter Scheduling

It is possible to schedule a brick data or station vector item as a function of one or two parameters for the functional relationship

$$P1 = P1(P2,P3) \tag{C2.1}$$

These parameters can be a single brick data or station vector item or a combination of these. The codeword descriptor letter "A" makes this possible. The sequence is akin to that used in the "ARITHY" brick [34] and some or all of the following data are to be given where necessary.

1. Operation code for dependent parameter,P1 ("-1" = no operation, "1" = add, "2" = subtract, "3" = multiply, "4" = divide)
2. Operation code for first independent parameter, P2
3. Operation code for second independent parameter, P3
4. Station Vector number or "-1"
5. Station Vector or Brick Data item number in numerator
6. Station Vector number or "-1"
7. Station Vector or Brick Data item number in denominator
8. Station Vector number or "-1" (if a constant is given)
9. Station Vector item number or value of constant to be multiplied by denominator

For Dependent Parameter

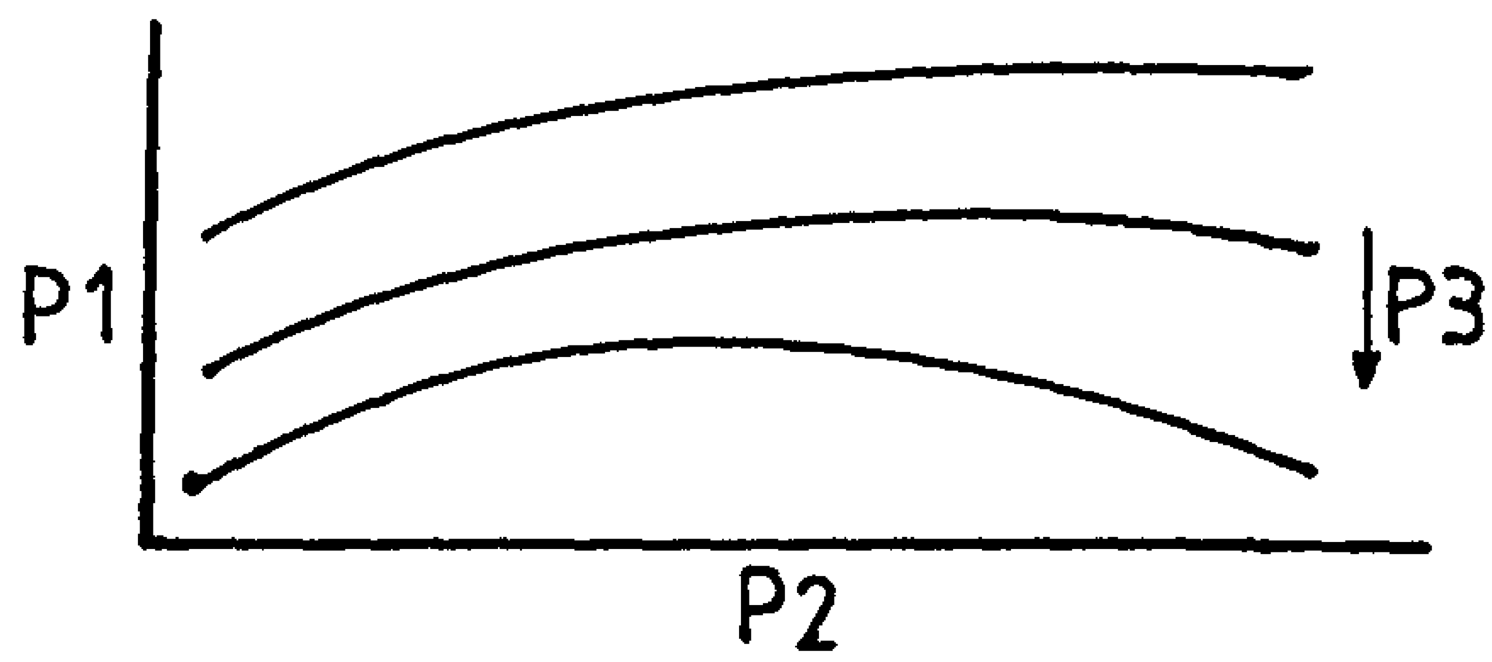
10.
11.
12.
13.
14.
15.

Same as 4 - 9 but for first independent parameter

16.
17.
18.
19.
20.
21.

Same as 4 - 9 but for second independent parameter

An example will clarify this.




If for example the functional relation in (C2.1) can be represented graphically as shown above where the various parameters are obtained from a combination of brick data and station vector item, say for example,

P1 = BD(9)
P2 = BD(12)/(SV(10,5) SV(10,3))
and P3 = BD(79)xSV(4,2)

then, items 1 to 21 take the following values:-

1. -1
2. 4 Operation codes
3. 3
4. -1
5. 9
6. -1
7. -1 Not Data for P1
8. -1 Required
9. -1
10. -1
11. 12
12. 10 Data for P2
13. 5
14. 10
15. 3
16. -1
17. 79
18. 4 Data for P2
19. 2
20. -1
21. -1

The operation code is that of the first operand in a parameter term and the remaining variables in the term (at most two) are treated as a single entity. If there are two variables after the operand, then the first must be a station vector item of which the square root is taken and this will be multiplied by the second variable if it is a station vector item or divided by the square root of the second variable if this is a constant. Therefore, if the third variable in P2, that is, SV(10,3), were replaced by a constant taking the value 100, say, then items 14 and 15 would have the values,

14. -1

15. $1/100^2$

that is, the inverse of the square of a constant should be input and not the constant itself. This procedure is adopted to accomodate terms like non-dimensional fuel flow and speed, referred or not.

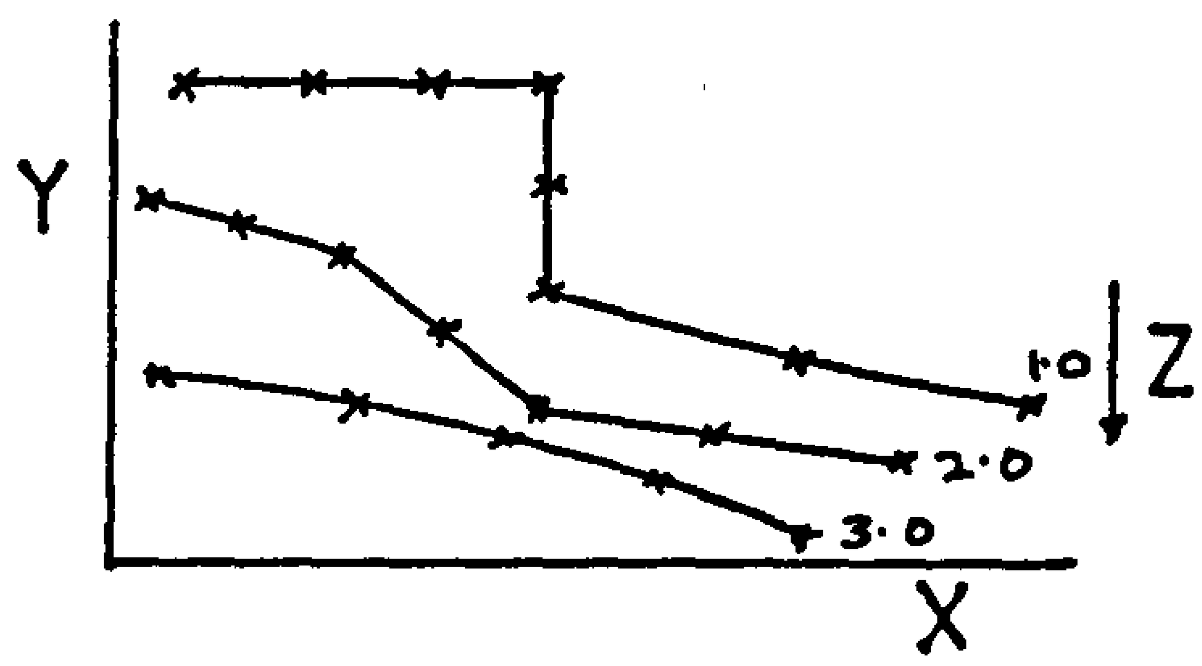
# APPENDIX C3

## Scheduled Data Input Procedure

An interpolation is carried out whenever it is required to extract data from a data set represented graphically. A parabolic or cubic interpolation may be performed but the vast majority are of the parabolic type. All of the input data are to be given with a certain format and some input procedures are more flexible than others. The general procedure is as follows:-

Read,

| | | | |
|---|---|---|---|
| 1. | I | Codeword number | |
| 2. | NL | Number of lines | |
| 3. | NP,ND | Number of points, Number of discontinuities | For each line |
| 4. | NDV | Data set point numbers at which discontinuities occur, if ND > 0 | For each line |
| 5. | (Xs,Ys),Z | Data set values and line value | For each line |



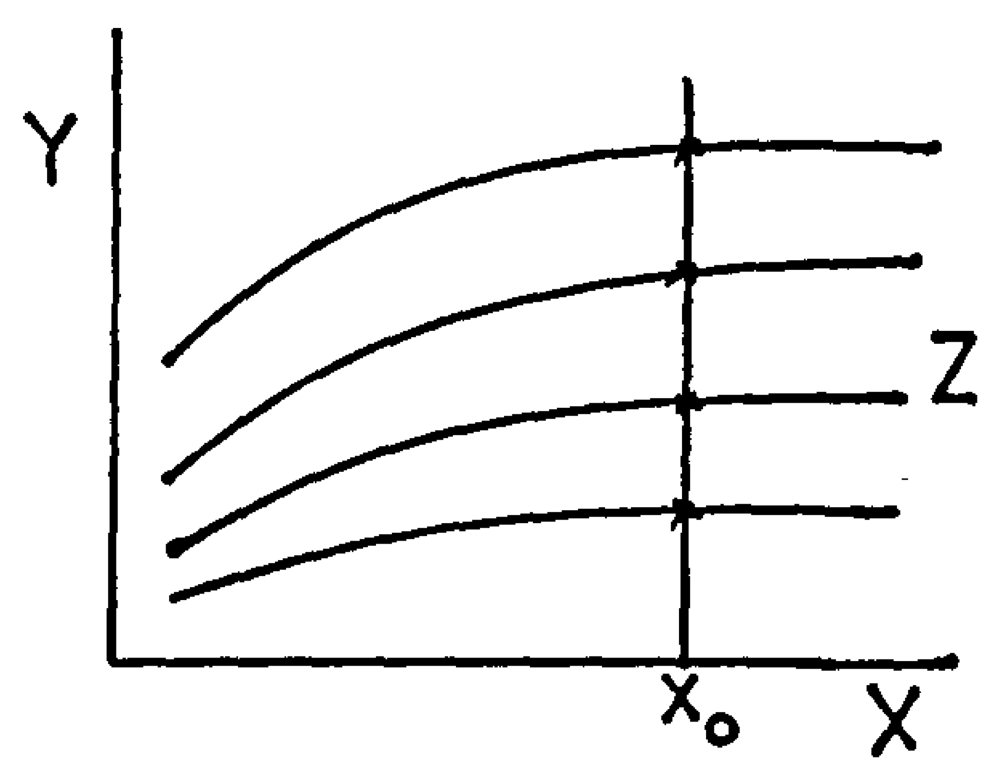

As an example, if we have a map as that shown above giving y as a function of x and z and which is associated with the second codeword, then the information is input as,

1. 2 codeword
2. 3
3. 5,0 (Z=3) — For first curve
5. (x1,y1),.........,(x5,y5), 3.0 — For first curve
3. 7,2 — For second curve
4. 3,5 — For second curve
5. (x1,y1),.........,(x7,y7), 2.0 — For second curve
3. 8,2 — For third curve
4. 4,6 — For third curve
5. (x1,y1),.........,(x8,y8), 1.0 — For third curve

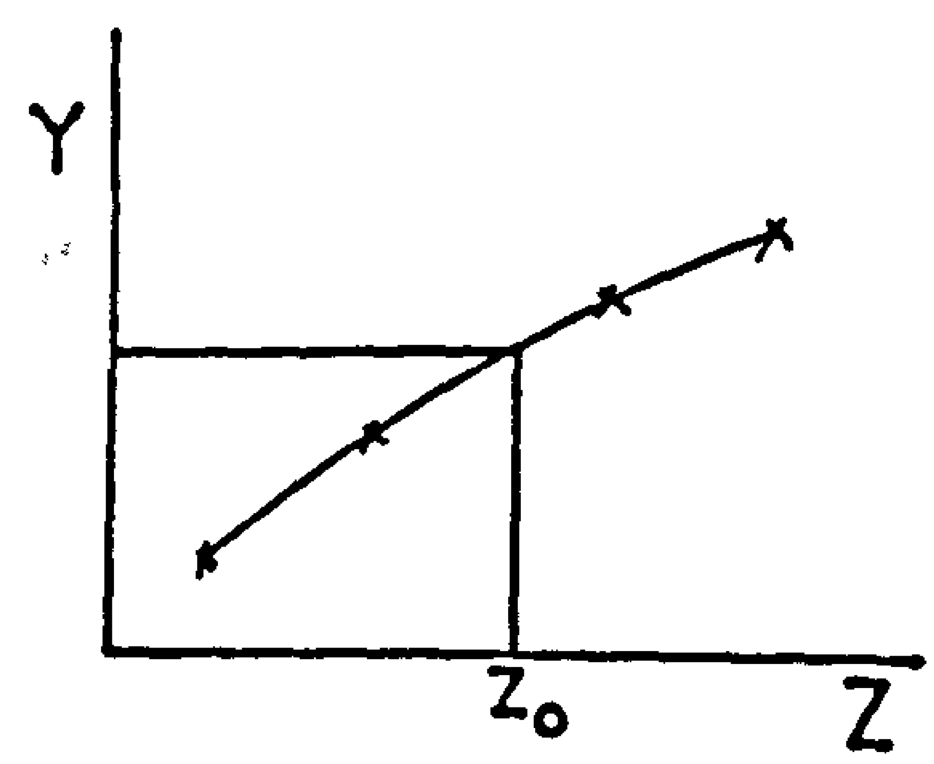
The first curve is always taken as the one at the lowermost level at the left, and the last, the topmost. In the above example, unlike curve 1, curves 2 and 3 have got points of discontinuity. These points of discontinuity are important as a parabolic interpolation uses three adjacent points, therefore, if a point of discontinuity happens to be the center point, then the interpolated value will be far off the actual value. For curve 2, the third and fifth points are points of discontinuity and are given in the input data and likewise, points 4 and 6 are to be input for curve 3. A maximum of five discontinuity points can be given and there should be at least one point between any two points of discontinuity. If there is a step in the function as shown in curve 3, the ordinate of the center point is the value of the function at the corresponding abscissa value; therefore, a step function will always have two identical adjacent data set points. If there is only one curve in the map, then "Z" takes a value of unity.

When cubic splines are used to perform the interpolation, a double interpolation is used to obtain the interpolated value if the function contains two variables. The procedure is as shown below.

At the given value of $x_o$, an interpolation is carried out to find the value of the function at each "Z" value. Another interpolation yields the value of the function at the given value of Z, that is, $z_o$.




Step 1




Step 2

# APPENDIX D

## The Graphics Routine

There is provision for presenting the off-design performance of an engine graphically. Any two of twenty possible parameters can be plotted and up to a maximum of six different data can be compared, with each plot having at most nine data set points. The data are either corrected to standard day or presented relative to design. The parameters that can be plotted are:-

1. Compressor corrected speed
2. Compressor corrected airflow
3. Compressor pressure ratio
4. Turbine inlet temperature
5. Turbine area ratio, $A/A_{min}$
6. Turbine work function, $\Delta H/T$
7. Turbine pressure ratio
8. LP turbine exit Mach number
9. Nozzle throat area
10. Nozzle exit area
11. Uninstalled specific thrust/power
12. Uninstalled sfc
13. Uninstalled thrust/power(eqv.)
14. Installed specific thrust/power
15. Installed sfc
16. Installed thrust/power(eqv.)
17. Nozzle pressure ratio
18. Compressor physical speed
19. Compressor IGV(stator) angle
20. Flight Mach number

It is recommended that thrust/power and Mach number be plotted on the abscissa and if these two are being plotted against each other, Mach number should be plotted on the abscissa.

As was mentioned earlier, up to six different performance curves can be compared. Before the program is run, the name of the file into which the data for plotting are to be written is assigned to a particular unit number as the program will prompt the user for a file input number if he desires to create a plot data file. When two or more performance plots are being compared, the data for plot 1 must be connected to unit 10 with subsequent engine data being connected to units 11,12, etc., as is necessary.

Although the program has got the capability to evaluate the performances of more than one engine in one run, separate runs must be performed for each program block if the plot data files are not in sequence, that is, 11.12,13, etc., say. At the end of the program run, the user is prompted to say if he now wishes to perform the graphics session. If he does not, the data are saved in the file or files that were connected to the respective unit numbers, otherwise, the plots are done.

The program will attempt to read the plotting codes from the last input data file, however, if there is an error in reading the first data or if an end-of-file is encountered, it assumes that the session will be done interactively and will prompt the user for the necessary information. The user is prompted for,

1. The number of data files
2. Data file names
3. Plotting device type (1. - HPOSTA4L; 2. - GLPS40A4; or 3. "blank space" denoting that choice will be made later on if necessary)
4. Orientation of paper and number of graph per page
5. Heading title for all plots (if any)
6. Is grid desired ("Y" or "N")
7. Figure number if any (read as a character)
8. Abscissa and Ordinate codes (from menu)
9. Either component number (1,2, or 3) or "Y" or "N" when sfc is plotted against thrust or power Optional For each curve

The first six points pertain to the desired plotting procedure while the last three pertain to the information to be plotted on the current graph. All the graphs on the same page have the same figure number in the sequence a,b,c, etc., if there are more than one graph on the sheet. The numbers corresponding to the parameters in the list of menu are given in 8 and if either of the parameters involve a nozzle, turbine, or compressor, then the position of this component is indicated in 9 with "1" representing the low pressure component and "3", the high pressure or core in the case of a nozzle. If sfc is being plotted against thrust, either uninstalled or installed, the user will have to say if he wishes to compare the uninstalled and installed values. The interactive version is quite straight forward and a new user is advised to use this first in order to familiarize himself with the procedure.

When the "plot" data are read from a file, the plots will be done without a graphical display appearing on the screen, otherwise, the graph will be displayed on the screen and the user will be prompted to say if he wishes to save the plot so that a hard copy can be obtained. If the data for the points on the plot are not in increasing or decreasing order, a smooth curve will not be drawn through the points. The plotting session can be restarted at any time during the run to change the settings such as orientation, number of graphs per page, etc., by inputting a "-1" for the figure number (item 7 in the above list). The program then restarts to read starting at item 3 in the above list.