

Cranfield Institute of Technology

School of Mechanical Engineering

PhD Thesis

Academic Year 1991-92

M.A.R. Nascimento

The Selective Bleed Variable Cycle Engine

Supervisor: Dr. P. Pilidis

July 1992

***This thesis is submitted in partial fulfilment of the requirements for
the Degree of Doctor of Philosophy.***

ABSTRACT

A new concept in aircraft propulsion is described in this work. In particular, variable jet engine is investigated for supersonic ASTOVL aircraft. This engine is a Selective Bleed Variable Cycle, twin shaft turbofan. At low flight speeds the engine operates as a medium bypass turbofan. At supersonic cruise it operates as low bypass turbofan without reheat.

The performance of the engine and its components is analyzed using a novel matching procedure. Off-design engine performance characteristics are explained, compressor running lines are shown, and variable geometry requirements are described.

The performance analysis shows that fuel savings are significant, thus reducing aircraft take-off weight. The major advantage of this engine is that all the components are used continuously for all operating modes, thus incurring low weight penalties. The benefits predicted by this preliminary investigation indicate that further research on this engine should be carried out.

The matching procedure developed and described shows itself to be very effective for two-shaft engines with a fixed geometry LP turbine. The method can also be extended to three shaft cases with or without a variable geometry LP turbine.

Extensive development of compressor and turbine design and performance software has been undertaken.

ACKNOWLEDGEMENTS

I would like to thank my supervisor, Dr P. Pilidis, for his invaluable advice, encouragement and patience, and K.W. Ramsden for his assistance during the course of my work and the preparation of this thesis. Thanks also to Dr. A. Mackenzie for his invaluable help.

I also wish to express my gratitude to all members of staff in the SME, specially Barbara Wilson for her patience and friendship. Special thanks are due to R.C. Adkins who thought of the name 'The Selective Bleed Variable Cycle Engine'.

Thanks to my parents and family Marina, Fabio and Felipe for their support and encouragement throughout the course of this work.

Finally, I would like to thank Escola Federal de Engenharia de Itajubá and CAPES (brazilian sponsors) for their financial support.

CONTENTS

ABSTRACT

ACKNOWLEDGEMENTS

CONTENTS

LIST OF FIGURES

NOTATION

CHAPTER 1 - INTRODUCTION	1
1.1- Aim of the Work	4
1.2- Bibliographic Review	5
CHAPTER 2 - The Selective Bleed Engine	
(Design and Off-design Performance)	10
2.1- Introduction	10
2.2- The Aircraft Mission	11
2.3- The Selective Bleed Engine	12
2.4- S.B.E. Design Point Optimization	14
2.4.1- Bypass Ratio	17
2.4.1.1- LP mode	17
2.4.1.2- HP mode	18
2.4.2- Ground Erosion	19
2.4.3- LP Compressor Pressure Ratio Optimization	19
2.5- Cycle Matching	19
2.6- Compressor Map Evaluation	20
2.7- Off-design Performance	21
2.8- Assessment of the Selective Bleed Engine	21
CHAPTER 3 - MATCHING PROCEDURE	24
3.1- Introduction	24
3.2- Williamson's Method	25
3.3- The Engine Matching Procedure	27
3.3.1- Part I: Turbine Matching	28
3.3.1.1- Matching of the HP Turbine	28

3.3.1.2- Matching of the LP Turbine	29
3.3.2- Part II: Compressor Matching	30
3.3.2.1- Determination of the Running Lines	30
3.3.2.2- Compressor Performance Analysis	32
CHAPTER 4 - COMPRESSOR DESIGN AND PERFORMANCE	34
4.1- Introduction	34
4.2- Modification of the Existing Computer Programmes	35
4.3- Design Point Objectives	37
4.4- Design Parameters Analysis	37
4.4.1- Inlet Axial Velocity	38
4.4.2- Inlet Mach Number	38
4.4.3- Hub-tip Ratio	38
4.4.4- Tip Speed	39
4.4.5- Stage Loading and Flow Coefficient	40
4.4.6- Degree of Reaction	41
4.4.7- Stage Temperature Rise	42
4.4.8- Diffusion Factor	42
4.4.9- Effect of Velocity Profile Distortions	44
(Work Done Factor)	
4.4.10- Radial Equilibrium Solution	45
4.4.11- Space/chord Ratio	46
4.4.12- Aspect Ratio	46
4.4.13- Tip Clearance	47
4.5- Design Problem and Solution	48
4.6- Overall Analysis of the Compressor Design	51
4.7- Compressor Off-design Performance	51
4.8- Off-design Performance Prediction	52
CHAPTER 5 - COMPRESSOR VARIABLE GEOMETRY	55
5.1- Introduction	55
5.2- Variable IGV and Stator Concepts	56
5.3- Procedure	57
5.3.1- LP Mode	59

5.3.1.1- LP Compressor	59
5.3.1.2- IP Compressor	65
5.3.2- HP Mode - LP compressor	69
CHAPTER 6 - TURBINE DESIGN AND PERFORMANCE	74
6.1- Introduction	74
6.2- Design Objectives	74
6.3- Computer Programmes	76
6.4- Implications of Design Selection Parameters	77
6.5- Design Assessment	83
6.6- Main Guidelines for Turbine Design Sequence	84
6.7- HP and LP Turbine -	86
6.7.1- HP Turbine	87
6.7.2- LP Turbine	89
6.8- Turbines Off-design Performance	91
6.9- HP Turbine Characteristics	94
6.10- LP Turbine Characteristics	94
CHAPTER 7 - SELECTIVE BLEED ENGINE SIMULATION	95
7.1- TURBOMATCH Preparation	95
7.2- Off-design Performance	97
7.3- Compressor Running Lines	98
CHAPTER 8 - ENGINEERING CONSIDERATIONS	101
8.1- Introduction	101
8.2- Other Design Aspects	101
8.2.1- Valves	101
8.2.2- Transition Phase	102
8.2.3- Sizing and Optimization	102
8.2.4- Mechanical Design	103
CHAPTER 9 - REMARKS AND CONCLUSIONS	104
9.1- Conclusions	104
9.2- Contributions	106
9.3- Further Work	107
REFERENCES	109

APPENDIX A	115
<i>AIRCRAFT MISSION PROFILE & THRUST REQUIREMENTS</i>	
APPENDIX B	122
<i>S.B.E. DESIGN AND OFF-DESIGN DATA</i>	
APPENDIX C	153
<i>EXAMPLE: THE SELECTIVE BLEED ENGINE</i>	
C.1- HP Turbine Matching	153
C.2- LP Turbine Matching	154
C.3- Compressor Matching - Running Lines	154
APPENDIX D	158
<i>COMPRESSOR DESIGN AND OFF-DESIGN SIMULATION</i>	
D.1- Introduction	158
D.2- Modification of CEMCOMP Efficiency Prediction	159
D.2.1- Blade Geometric Parameters	160
D.2.2- Blade Shape	160
D.2.3- Incidence Angle	162
D.2.4- Deviation Angle	162
D.2.5- Diffusion Factor	163
D.2.6- Equivalent Diffusion Factor	164
D.2.7- Profile Loss	164
D.2.8- Shock Loss	167
D.2.9- Secondary and Annulus Losses	169
D.2.10- Efficiency Prediction	170
D.3- Merger of ANNULUS and CEMCOMP	172
D.4- Modification of SCD15	172
D.5- COMPDES and BLADE Programme	174
D.5.1- Running COMPDES on the VAX Computer	175
D.5.2- Running BLADE on the VAX Computer	176
D.5.3- Running HOWELL on the VAX Computer	176
D.6- New Flow Diagram Calculation	177
APPENDIX E	197
<i>TURBINE DESIGN AND OFF-DESIGN SIMULATION</i>	

<i>E.1- Turbine Aerodynamic Design</i>	197
<i>E.1.1- TURBGEO Computer Programme</i>	197
<i>E.1.2- TURBDES Computer Programme</i>	198
<i>E.1.3- TURBOFF Computer Programme</i>	199
<i>E.2- Ainley and Mathieson's Method</i>	200
<i>E.2.1- Notation and Convention of signs</i>	201
<i>E.2.2- Outlet Gas Angle from a Blade Row</i>	201
<i>E.2.3- Profile Loss</i>	203
<i>E.2.4- Stalling Incidence</i>	204
<i>E.2.5- Secondary Loss</i>	204
<i>E.2.6- Tip Clearance Loss</i>	205
<i>E.2.7- Total Loss Coefficient</i>	206
<i>E.2.8- Stage Performance Prediction Procedure</i>	207
<i>E.2.9- Overall Stage Isentropic Efficiency</i>	212
APPENDIX F	232
COMPRESSOR AND TURBINE <u>INPUT</u> & <u>OUTPUT</u> DATA	
FIGURES	295

LIST OF FIGURES

- Figure 1.1 : S.F.C. of Jet Engines.
- Figure 1.2 : Internal Engine Hover Cruise Scheme.
- Figure 1.3 : The Selective Bleed Engine.
- Figure 2.1 : The Selective Bleed Engine (LP and HP modes).
- Figure 2.2 : TURBOMATCH LP mode Configuration.
- Figure 2.3 : TURBOMATCH HP mode Configuration.
- Figure 2.4 : SFC versus Sp. Thrust.
- Figure 2.5 : SFC versus Sp. Thrust.
- Figure 2.6 : F/R Thrust Ratio versus Pressure Ratio (LP mode).
- Figure 2.7 : SFC versus Sp. Thrust.
- Figure 2.8 : SFC versus Sp. Thrust.
- Figure 2.9 : Nozzle Exit Temperature versus Pressure Ratio (LP mode).
- Figure 2.10: SFC versus Pressure Ratio (LP mode).
- Figure 2.11: Compressor Operating Lines.
- Figure 2.12: Variation of Thrust with Mach Number.
- Figure 2.13: Change of Thrust with TET.
- Figure 2.14: Change of SFC with TET.
- Figure 2.15: Change of SFC with Thrust.
- Figure 2.16: Comparison of the Selective Bleed Engine to a Turbofan.
- Figure 2.17: Comparison of the Selective Bleed Engine to a Turbofan.
- Figure 3.1 : Crossplot of HP and LP Turbine Non-dimensional Mass Flows.
- Figure 3.2 : Compressor Design Point Characteristics.
- Figure 3.3 : Effect of TET for Both Modes.
- Figure 3.4 : Effect of TET and OPR.
- Figure 3.5 : Effect of TET & BPR.
- Figure 3.6 : Compressor Operating Lines.
- Figure 3.7 : Compressor Operating Lines.
- Figure 3.8 : Surge Line Displacement.
- Figure 4.1 : Compressor Stage Loading Chart.

- Figure 4.2 : Variation of Mean Work-done Factor with Number of Stages.*
- Figure 4.3 : Effect of Inlet Mach Number on the Loss Coefficient of Double-circular-arc Blading at Different Space/chord Ratios.*
- Figure 4.4 : LP Compressor.*
- Figure 4.5 : IP Compressor.*
- Figure 4.6 : HP Compressor.*
- Figure 5.1 : Guide Vane and Stator Adjustment Concepts.*
- Figure 5.2 : LP Compressor Fixed Geometry.*
- Figure 5.3 : LP Compressor Variable IGV.*
- Figure 5.4 : LP Compressor Variable IGV+1stator.*
- Figure 5.5 : LP Compressor Variable IGV+1stator+2stator.*
- Figure 5.6 : LP compressor Schedule of Variable Stators.*
- Figure 5.7 : IP Compressor Fixed Geometry.*
- Figure 5.8 : IP Compressor Variable IGV.*
- Figure 5.9 : G.E. Variable Cycle Engine.*
- Figure 5.10: IP Compressor Variable Last Stator by 40 degrees.*
- Figure 5.11: IP Compressor Variable All Stators by 10 degrees.*
- Figure 5.12: IP Compressor Variable All Stators by 20 degrees.*
- Figure 5.13: IP Compressor Variable All Stators by 30 degrees.*
- Figure 5.14: IP Compressor Variable All Stators by 40 degrees.*
- Figure 5.15: IP Compressor Variable All Stators by 30 degrees (only LP mode).*
- Figure 5.16: LP Compressor Variable All Stators by 10 degrees.*
- Figure 5.17: LP Compressor Variable IGV by 20 degrees+All Stators by 10 degrees.*
- Figure 5.18: LP Compressor Variable IGV by 30 degrees+All Stators by 10 degrees.*
- Figure 5.19: LP Compressor Variable IGV by 30 degrees+ 1 Stator by 20+ Remaining Stators by 10 degrees.*
- Figure 5.20: LP Compressor Variable IGV by 30 degrees+all Stators by 15 degrees.*
- Figure 5.21: LP Compressor Variable IGV by 35 degrees+all Stators by 15 degrees.*

Figure 5.22: LP Compressor Variable IGV by 35 degrees+all Stators by 15 degrees (only HP mode).

Figure 5.23: LP Compressor Variable Geometry.

Figure 5.24: IP Compressor Fixed and Variable Geometry.

Figure 6.1 : Combustion Temperature Rise versus Fuel/air Ratio.

Figure 6.2 : Efficiency versus Loading and Flow Coefficients.

Figure 6.3 : Variation of Turbine Static Density Ratio and Hub-Shroud Diameter Ratio with Design Total Ratio.

Figure 6.4 : 'Optimum' Space/chord Ratio.

Figure 6.5 : Annulus Flare Angle.

Figure 6.6 : Axial-Flow Turbine Stage Efficiency versus Loading and Flow Coefficients.

Figure 6.7a: Maximum Stage Isentropic Efficiencies versus Loading Coefficient and Reaction.

Figure 6.7b: Stage Outlet Swirl Angles at Maximum Efficiencies versus Loading Coefficient and Reaction.

Figure 6.7c: Optimum Nozzle Flow Angle versus Loading Coefficient and Reaction.

Figure 6.8 : HP and LP Turbine Annulus Geometry.

Figure 6.9 : Single-Stage HP Turbine.

Figure 6.10: Four-Stages LP Turbine.

Figure 7.1 : LP Compressor Variable IGV by 35 degrees+All Stators by 20 degrees (only HP mode).

Figure 7.2 : Change of Thrust with TET.

Figure 7.3 : Change of SFC with TET.

Figure 7.4 : Change of SFC with Thrust.

Figure 7.5 : LP Compressor Variable IGV+1stator+2stator.

Figure 7.6 : LP Compressor Variable IGV by 35 degrees + All Stators by 20 Degrees.

Figure 7.7 : HP Compressor Fixed Geometry.

Figure 7.8 : IP Compressor Variable All Stators by 30 degrees.

Figure 7.9 : IP Compressor Fixed Geometry.

NOTATION

A	- Area
AR	- Aspect Ratio
a/c	- Position of maximum camber
c_p	- Specific heat at constant pressure
c_{ax}	- Axial chord
c	- Chord
C_{DA}	- Annulus drag coefficient
C_{DS}	- Secondary drag coefficient
C_L	- Lift coefficient
CN	- Non-dimensional speed relative to design $(\frac{N/\sqrt{T}}{(N/\sqrt{T})_{des}})$
D_p	- Dynamic Pressure
D	- Diffusion factor
D_{eq}	- Equivalent diffusion factor
e	- Mean radius of curvature of the suction surface of the blade profile between throat and trailing edge
ETA	- Isentropic efficiency
f_M	- Mach correction factor
f_{RE}	- Reynolds correction factor
H	- Enthalpy
h	- Blade height
H_{te}	- Trailing-edge boundary layer shape factor
H_{tec}	- Corrected boundary layer shape factor
i	- Incidence
i_{2D}	- Minimum loss incidence angle
$(i_0)_{10}$	- Minimum loss incidence angle for a NACA-65 cascade of zero camber and 10% thickness/chord ratio
K_i	- Correction factor

K_s - Average sand roughness

k - Clearance

k_a - Shrouded clearance

m - Mass flow

m_c - Coefficient (Carter's rule)

$\frac{m\sqrt{T}}{P}$ - Non-dimensional mass flow $[\frac{kg/s \cdot K}{N/m^2}]$ (SI) in the graph

M - Mach number

n - Operating speed line

N - Rotational speed

$\frac{N}{\sqrt{T}}$ - Non-dimensional rotational speed

o - Width of the throat between blades

P - Total pressure

PR_k - Pressure ratio relative to maximum efficiency

PR_s - Surge pressure ratio

PR_c - Choke pressure ratio

r - Leading and trailing edge radii

Re - Reynolds number

Re_{cr} - Critical Reynolds number

s/c - Space/chord ratio

S - Surge improvement

s - Space (pitch)

T - Total temperature

t - Maximum thickness

t_e - Trailing edge thickness

U - Blade speed

V - Velocity

V_a - Axial velocity

V_{max} - Maximum suction-surface velocity

- V_θ - Whirl velocity
 W - Non-dimensional mass flow
 Y - Loss (turbine)
 Y_t - Total loss coefficient (turbine)
 Y_{TET} - Trailing edge loss multiplier
 Z - Surge margin parameter $Z = \frac{PR - PR_c}{PR_s - PR_c}$

GREEK LETTERS

- α - Air angle
 α_0 - Inlet air angle
 β - Blade angle
 β_4 - Exit stator blade angle
 γ - Specific heat ratio
 Δ - Increment
 $\Delta\eta$ - Efficiency drop
 δ - Deviation
 ε - Deflection
 ζ - Stagger angle
 η - Isentropic efficiency
 θ - Blade camber angle
 θ_F - Flare angle
 (θ_t/c) - Wake momentum thickness
 $(\theta_t/c)_c$ - Corrected wake momentum thickness
 λ - Supersonic turning angle
 μ - Absolute viscosity
 ρ - Density
 σ - Cascade solidity
 ϕ - Flow coefficient
 ϕ_r - Restagger angle
 ψ - Stage loading coefficient
 ω - Loss

SUBSCRIPT

<i>1</i>	- Inlet cascade
<i>2</i>	- Outlet cascade
<i>A</i>	- Annulus
<i>ad</i>	- Adiabatic
<i>crit</i>	- Critical
<i>F</i>	- Fixed geometry
<i>i</i>	- Annulus inlet plane (turbine)
<i>k</i>	- Clearance
<i>m</i>	- Mean
<i>max</i>	- Maximum
<i>n_o</i>	- Inlet annulus area
<i>n2</i>	- Exit annulus area
<i>o</i>	- Nozzle exit (turbine)
<i>Opt</i>	- Optimized
<i>P</i>	- Profile
<i>R</i>	- Rotor
<i>S</i>	- Stator
<i>SD</i>	- Secondary
<i>SL</i>	- Stall
<i>ST</i>	- Stage
<i>Set</i>	- Variable geometry
<i>SH</i>	- Shock
<i>t</i>	- Throat
<i>T</i>	- Total
<i>tipcl</i>	- Tip clearance

ABBREVIATION

ASTOVL - *Advanced short takeoff and landing*

AB - *Afterburner*

ALT - *Altitude*

BPR - *By-pass ratio*

DCA - *Double circular arc*

DP - *Design point*

HP - *High pressure*

HPM - *High pressure mode*

HTP - *High pressure turbine*

IP - *Intermediate pressure*

IPR - *Intermediate pressure compressor pressure ratio*

IGV - *Inlet guide vane*

LP - *Low pressure*

LPR - *Low pressure compressor pressure ratio*

LPM - *Low pressure mode*

LTP - *Low pressure turbine*

NGTE - *National Gas Turbine Establishment*

NGV - *Nozzle guide vane*

OPR - *Overall pressure ratio*

PR - *Pressure ratio*

RPM - *Rotation per minute*

SFC - *Specific fuel consumption*

SEP - *Specific excess power*

SLS - *Sea level static*

SL - *Sea level*

Sp - *Specific*

SBE - *Selective Bleed Engine*

TET - *Turbine entry temperature*

VCE - *Variable cycle engine*

V/STOL - *Vertical or short takeoff and landing*

VSVA - *Variable stator vane angle*

CHAPTER 1

Introduction

The design of an aircraft engine is a compromise between the various operational requirements of the mission it is expected to fulfil. When engines are required to operate over wide ranges of flight speed and altitudes this compromise becomes progressively more difficult because to be able to achieve all the required mission criteria, the engine designer has to accept certain penalties at each operating point. As the range of operating conditions increases, the penalties become progressively larger. To partly overcome these penalties, the engine flexibility can be increased, by employing the Variable Cycle Engine (VCE) concept. This requires hardware that will operate with radically different thermodynamic cycles at different operating points. The most common example of such an engine is the afterburning turbojet or turbofan. The afterburner provides large thrust increases for a short (with one exception) period of time at the expense of specific fuel consumption (SFC). When supersonic cruise at Mach 1.5 - 2 is considered, the afterburning engine becomes very expensive in terms of SFC. In this case a jet engine operating in the 'dry' mode becomes very attractive. This engine can be a turbojet or a low bypass ratio turbofan. When the aircraft is required to operate at subsonic speeds, the best option is a high bypass ratio turbofan because it gives good fuel economy.

Different power plants exhibit varying characteristics for different flight Mach Numbers. Figure 1.1 shows the variation of specific fuel consumption with this parameter. The main cause for this variation is the increase of the inlet

momentum drag with flight speed. This Figure shows that for different flight regimes one power plant may be better than another with respect to fuel consumption and other criteria, such as specific thrust. Thus it would be desirable to be operating different engines at different speeds. If the thermodynamic cycle of the engine can be varied, mainly by decreasing bypass ratio as flight speed increases, this aim of operating different engines can be achieved. This is the philosophy behind the variable cycle engine, which requires hardware that will operate with radically different thermodynamic cycles at different operating points.

The driving force behind the research into Variable Cycle Engines has been the desire to optimise an aircraft engine at two or more design points, perhaps at opposite extremes of the flight envelope. The fixed cycle engine can only be optimised at its one design point, all other running points being off-design. Modern military engines particularly are expected to operate over a wide flight envelope, in which an engine optimised for one design point may operate very inefficiently at another running point or, more seriously, prevent the aircraft from operating there at all. This means that cycle variation may be used not only to benefit performance parameters such as thrust and SFC, but also may be used to improve engine handling or response times.

There are several potential applications of VCEs in military and commercial missions. This work is particularly concerned with military application for supersonic Vertical or Short Take-Off and Landing (V/STOL) fighter aircraft.

Supersonic V/STOL fighter aircraft requires propulsion systems which produce thrust well in excess of aircraft weight, for powered lift and control, and which can be integrated into an aerodynamically efficient air frame. The powered lift requirements, plus the flight performance requirements of fighter missions, result in extensive compromises in the cycle and scheduling of a fixed cycle engine (Figure 1.2 shows some of the internal engine schemes for both hover and cruise). These compromises have resulted in high take-off-gross- weight and

relatively poor payload and range performance in many V/STOL aircraft design. Because of their inherent operational flexibility, VCEs are being studied for V/STOL aircraft. Some results indicate that some VCEs concepts offer cycle flexibility, reduced fuel consumption, improved performance and the ability to obtain takeoff thrust without associated penalties over the rest of the mission.

Despite the variety of variable cycle engine concepts for civil and military application this work is concerned with the study of a new one entitled Selective Bleed Variable Cycle Engine for Advanced Short Take-Off and Vertical Landing (ASTOVL) multi-mission fighter aircraft. It has been suggested by Adkins and Pilidis (1) and a preliminary study has been carried out by Gammon (2). Figure 1.3 shows a sketch of the Selective Bleed Variable Cycle Engine studied herein.

The Selective Bleed Engine comprises a two spool gas generators with two turbines (high and low pressure turbines), and three compressors (low, intermediate and high pressure compressors) with LP and IP compressors at the same shaft, in which it is possible to bleed air from either the LP compressor delivery or from the IP compressor delivery. At subsonic speed the air bleed is taken from at the back of the LP compressor and it can be directed either vertically downwards through a vectoring front nozzle or expanded, to give thrust in the conventional direction, through a nozzle at the rear of the aircraft. At the supersonic speed the air bleed from the LP compressor is shut and another air bleed is opened at the back of the IP compressor. It is expanded through a convergent-divergent nozzle, mounted on the top of the core rear nozzle. The engine operates in three principal modes:

- i- vertical flight*
- ii- loiter and subsonic cruise*
- iii- supersonic cruise and dash.*

The engine operates as a turbofan of medium bypass ratio at subsonic flight conditions and it operates as a low bypass turbofan at supersonic cruise. Full details of the Selective Bleed Engine operation are given in Chapter 2.

This work describes a technique that allows the assessment of the engine performance based on the operability of the components. The method described treats fixed geometry turbines, and it can be easily extended to include the treatment of variable geometry turbines.

One of the greatest challenges, during the design phase, is how to optimise the various cycles and then to match them to the capabilities of the components which make up the engine.

Finally, a general matching method procedure is developed as part of this work. The method described is based on thermodynamic relationships and component characteristics. An example is included to illustrate the procedure.

1.1- Aim of the Work

The aim of this work is, in particular, concerned with overall engine performance analysis of a new variable cycle engine concept in order to identify its performance benefits. This involves cycle modelling study, engine optimization, matching for subsonic and supersonic modes, design and off-design performance for the specified mission and thrust requirements. In addition, the study includes compressor design, variable geometry and performance, turbine design, engine overall performance, and the creation of a computational model which will be able to predict the performance map for axial flow turbines. One of the products has been the geneses of an outlined performance analysis procedure for variable cycle jet engines.

1.2- Bibliographic Review

There are very few works published, in the public domain, on variable cycle engines. The majority of work have been dealt with different design concepts and potential applications. There is very little previous work assessing the component performance and feasibility of such engines.

The review carried out herein intends to show the tendency and analysis accomplished so far of some VCEs concepts.

Armstrong and Higon (3) have studied some aspects of variable cycle propulsion systems. Their work offers a broad discussion of the incentives, design features, problems and prospects of variable cycle gas turbine powerplants, illustrated by a number of potential applications considered in studies at N.G.T.E.. The assessment of such discussion has been carried out on the increase of thrust capability, subsonic civil transports, military combat aircraft, fuel economy and noise reduction. They have found that the launching costs of new aeronautical projects have reached high levels in relation to available resources, even for collaborative ventures between major partners, and valuable benefits of the variable cycle need to be in prospect, together with a good level of confidence in the success of the proposed development. This confidence, according to them, depends largely upon the availability of a basis of practical experience of the design feature in question and for the future, the variable cycle will continue to face strong competition from the alternative solution.

Facey and Glaser (4) have worked in a programme entitled the "Variable Cycle Engine Selection Program", sponsored by the Naval Air Propulsion Test Centre, which has sought to determine the impact of VCE on advanced V/STOL multi-mission fighters. The studies involved a description, a summary of current results, and the conclusions drawn from the initial phase of the programme. The results from the simplified analysis procedures, used in the first phase of the

programme, have shown a potential reduction of approximately 10% of take-off-gross-weight which was sufficient to guarantee proceeding to the next phases of the programme. In addition, the most promising VCE concepts were the Detroit Diesel Allison Variable Geometry Turbofan and General Electric Modulating Bypass Turbofan engines. They also found that an aircraft, which performs several missions of diverse nature will benefit from the use of a variable cycle engine.

Willis (5) has carried out a review on the progress and current status of variable cycle engines for supersonic cruise aircraft. The studies for supersonic cruise aircraft research engines were designed to identify the most promising VCE concepts and define their advanced technology requirements. The review has also included the presently-favoured variable cycle engines, which are two P&W concepts derived from a duct-burning turbofan and two GE engines based on a mixed-flow turbofan. It has revealed that VCEs may be prohibitively complex, heavy and expensive unless significant design and technology advances are accomplished. His work has concluded that the final choice among the current VCE candidates will depend on application and installation factors as well as further engine study/design and technology efforts.

Boxer, Morris, and Foss (6) have assessed VCEs for supersonic transports under the NASA Supersonic Cruise Aircraft Research Program. Their work evaluated each of three proposed supersonic cruise aircraft research propulsion systems in terms of relative aircraft range for a fixed payload and take-off-gross-weight with a design cruise Mach number of 2.7. A comparison of these engines, in order to put the performance of these engines in perspective, and the former U.S. SST engine (GE4) was made with an idealized variable cycle engine whose performance at all operating points matches that of an optimized point-design cycle within specified limits. In addition, range comparisons were made with and without noise level constraints to determine the influence of noise upon cycle selection. They have come to the conclusion that critical areas, for instance

rotating engine components which provide the flexibility to accept wide variations in flow with acceptable efficiency and surge margins, require new or improved technology for each VCE.

Military aircraft which require to operate far from the engine design point may have significant improvement of engine performance by cycle changes allowing operation nearer the design point. Hatton and Swann (7) have carried out a examination of cycle changes. They have illustrated the effect of representative losses due to cycle and geometric changes and with the inclusion of these not all the cases examined indicated success for VCEs. Their work was restricted to conventional aircraft and engines of moderate bypass ratio designed at sea level static conditions. They have found that the throttled specific fuel consumption may only be improved by variables if the high pressure turbine capacity, the most difficult, is varied. Then, high forward speeds, good prospects exist for increase of thrust up to 10% by varying nozzle and mixer areas.

Payzer (8) has discussed the potential applications for VCEs as well as the payoff, constraints on these engines both from a totally installed aspect and the internal engine restrictions. The outcomes were that installation effects, mission sizing effects, component complexity and off-design operation, as well as weight, all place limitations there-on to the degree of flexibility that can be achieved in an installed engine. Furthermore, the next generation of engines for multi-mission use could very well be VCEs if their increased weight, complexity and cost are more than offset by reduced aircraft size, cost, more flexible practical operation and greatly reduced fuel usage.

Allan (9) has published a review on the General Electric Company Variable Cycle Engine Technology Experimental Program, under contract from the government, which began in 1974. This program was established to test VCE features designed to improve mixed mission (supersonic/subsonic) propulsion system installed performance. He has pointed out most of the concepts identified

in the early stage have been feasible from an engine cycle standpoint, but have suffered from severe mechanical complexity and high weight and cost implications which overcame the cycle advantages. Later on, after studies, GE found two of the most promising variable cycle concepts, which were the Single Bypass Variable Cycle Engine and Double Bypass Variable Cycle Engine. The Single Bypass VCE is the simpler of the concepts, and is basically a low bypass ratio turbofan engine with a variable mixing device in the turbine discharge section of the engine called a variable area bypass injector. The Double Bypass VCE extends this capability with the design objective to eliminate spillage drag at subsonic cruise reduced power and during the transonic climb and acceleration operating conditions. The studies, design, fabrication and testing of VCE's have resulted in the advancement of the technologies required for successful application of the components identified as critical to the operation of the single and double bypass VCE concepts.

A similar work, under General Electric Company, has been carried out by Brown (10). His work has assessed a selected VCE called Double Bypass Variable Cycle Engine, which was considered the most promising concept to meet mission range and noise goals. The selection was based on a study with engines installed in an engine company in-house airplane and mission analysis results. The double bypass VCE selected is a relatively simple VCE concept. It is basically a low bypass ratio, two shaft turbofan engine with variable cycle features that allows the setting of engine controls to achieve installed performance levels better than a conventional mixed-flow turbofan engine. It is designed to operate at supersonic cruise (Mach numbers that are within 2.2 to 2.7) with dry power settings, and at transonic climb with low temperature augmentation power settings. The assessment on the VCEs was for a fan pressure ratio between 3.7 and 4.5, an engine overall pressure ratio between 14 and 20, and a bypass ratio between 0.2 and 0.5. The results revealed that the double bypass VCE concept has the flexibility of matching a wide range of supersonic cruise aircraft characteristics, design Mach numbers, inlet

characteristics, and takeoff requirements. This VCE concept is a relatively simple propulsion system, but it was found that, it must be fully integrated with aircraft requirements to fully exploit its performance advantages.

CHAPTER 2

The Selective Bleed Engine **(Design and Off-design Performance)**

2.1- Introduction

The engine optimization is a very important phase of the design to select the design point parameters as well as to achieve a good off-design performance. Therefore, a right selection of design point parameters leads to a significant engine overall performance.

In this Chapter, the optimization and selection of the design point parameters, and matching of operating modes have been outlined for the Selective Bleed Engine. The preliminary off-design performance has been assessed by means of a comparison of the Selective Bleed Engine to a standard Turbofan. It has shown that SFC of the Selective Bleed Engine is significantly better than that of the standard Turbofan in all modes.

Finally, this Chapter has provided data for the design and off-design performance of the engine components, which, therefore, has a significant influence in the final engine overall performance assessment.

2.2- The Aircraft Mission

The Selective Bleed Variable Cycle Engine Figure 2.1 has been designed with an ASTOVL aircraft in mind. This aircraft is meant to take off from a very short runway, cruise economically at Mach 1.5 or 1.6, loiter at Mach 0.6, and land vertically.

Table 2.1 shows the thrust requirement for this type of engine (A full data presentation is in Table A.1, Appendix A), where for several operating points in the flight envelope, the specific excess thrust (S.E.T.), and the thrust are shown.

Point	Alt. (Km)	Mach No	S.E.T. (m/s)	Velocity (m/s)	Thrust (KN)
1	9	1.5	0	456	100
2	0	0.9	0	306	102
3	0	0.5	0	170	92
4	0	0.9	0	306	74
5	6	0.9	0	285	101
6	6	1.2	0	380	135
7	9	1.4	150	425	121
8	9	1.5	0	456	81
9	0	0.9	300	306	174
10	6	0.9	190	285	113
11	0	1.1	0	374	144
12	11	1.6	0	472	68
13	0	0.0	0	0	146
14 (Loiter)	9	0.65	0	197	13
15 (Range)	9	0.78	0	236	14

TABLE 2.1 ASTOVL Aircraft Thrust Requirements.

The particular mission that the aircraft was designed to fulfill has been simplified to the sequence of operating points shown in Table 2.2.

2.3- The Selective Bleed Engine

The Selective Bleed Engine is a jet engine designed to achieve three widely differing missions satisfactorily. These missions are good SFC at low flight Mach numbers, good thrust split, and supersonic cruise at Mach 1.5 with a dry engine. To achieve these goals the engine has two modes of operation, the LP (low pressure) mode and the HP (high pressure) mode. The nomenclature indicates the pressure of the bypass flow: in the LP mode the bypass flow is split from the core flow at LP compressor delivery, while in the HP mode from the IP compressor delivery.

Take-off	(pt 13)	1 min
Subsonic cruise	(range)	25 min
Loiter	(loiter)	60 min
Supersonic cruise	(pt 8)	10 min
Subsonic cruise	(range)	38 min
Vertical landing	(pt 13)	1 min

TABLE 2.2 Typical ASTOVL Aircraft Mission.

The Selective Bleed Engine is a two-spool engine (Figure 2.1). It has two compressors in the LP spool and one compressor in the HP spool. These compressors are the LP, IP and HP compressors respectively. Each spool is driven by its turbine; the HP turbine drives the HP compressor and the LP turbine drives the IP and the LP compressor. This configuration proved to be the best after trials with two other spool engines, and with two and three compressors. One of the main criteria for its selection was the good matching of the operating conditions in the two operating modes. The study has been carried out with turbines which are of a fixed geometry. Variable geometry turbines would

increase greatly the flexibility of the power plant but the author decided to reduce the technological risk by choosing fixed geometry turbines in the first instance. If further studies indicate the need, a variable geometry LP turbine may then be adopted.

As stated above the engine will operate in two modes, the LP and the HP mode (Figures 2.1a and 2.1b, respectively). For subsonic speeds the bypass and the core flow will be split immediately downstream of the LP compressor. This is the LP mode of operation with the LP valve open and the HP valve closed (Figure 2.1a). In these circumstances the engine will operate as a turbofan with a bypass ratio of 1, ie with a high mass flow, a low SFC and a low specific thrust; these characteristics being suited to subsonic flight conditions. The high bypass mass flow will also give a good thrust split between cold and hot (front and rear) streams, a pre-condition of V/STOL operations. In the LP mode the jets will be discharged through separate front and rear nozzles. One of the constraints imposed on the engine has been a jet temperature limit of about 1000 K during the V/STOL operating modes, because of ground erosion problems. For supersonic cruise, a cycle with high specific thrust will be required. This is achieved in the HP mode by reducing the bypass ratio to 0.4 and increasing the bypass flow pressure ratio. The split between bypass and core flows now takes place just downstream of the IP compressor, with the LP valve closed and the HP valve open (Figure 2.1b). The cold and hot flows are not mixed downstream of the bypass duct and the core, and they are then discharged through a separate nozzles. The engine is a turbofan with a low bypass ratio, a cycle suitable for supersonic cruise. The engine has an afterburner fitted to it for conditions, such as combat, where a thrust boost is required. Table 2.3 illustrates the main cycle characteristics of both the LP and the HP modes. The selection of design points was carried out using a procedure (described in the next sections) that simplifies the process of matching and optimising the two cycles.

2.4- S.B.E. Design Point Optimization

As mentioned in the previous section, the Selective Bleed Variable Cycle Engine was split into two modes of operation. Each one is considered as a single cycle engine and they have to have their on design point conditions separately. Therefore, the simulation and optimization have been carried out in each mode and the design point conditions for each mode were selected as follows:

Conditions	LP mode	HP mode
Altitude (km)	0	6
Mach Number	0	1.2
Turbine entry temperature (K)	1500	1800
Fan pressure ratio	3.0	6.2
Compressor pressure ratio	8.7	4.1
Bypass ratio	1.0	0.4
Mass flow (kg/s)	331	243
Thrust (kN)	173	134
Specific thrust (m/s)	524	552
SFC at design (kg/MNs)	16	31
Front/Rear thrust split	0.7	n/a

TABLE 2.3 Engine Cycle Details

i) LP mode - Sea Level Static (S.L.S.). This is the condition where the sea level static hover at the end of the sortie needs to be accomplished without reheat, due to the ground erosion and the possibility of hot gas reingestion.

ii) HP mode - Altitude 6km and Mach number of 1.2. This is the condition that suits best the engine matching at the supersonic speed. It is explained in section 3.3.1.1, Chapter 3.

The design point and off-design performance, and optimization of the Selective Bleed Engine were simulated with the TURBOMATCH scheme, a software system developed at Cranfield Institute of Technology which can simulate any type of open cycle gas turbine engine. The programme was originally written by Macmillan (11) and subsequently enhanced by Palmer (12).

The programme links a series of thermodynamic subroutines, each representing a different engine component such as a compressor or turbine. Each subroutine takes specific input data and results from previous subroutines to calculate output data. So that, the programme calculates the full range of engine performance parameters both for design point and, using simultaneous iteration techniques to calculate residual errors, for off-design performance.

Before optimization, it was necessary to draw the TURBOMATCH configurations for the LP and HP modes. These configurations can be seen in Figures 2.2 and 2.3 for LP and HP modes, respectively. With the help of these configurations, the required TURBOMATCH input data were prepared for each of the operating modes and they can be seen in Figures B.1 and B.2, Appendix B. During the preparation process of input files, some standard values were required and they were selected as follows:

<i>Inlet conditions -</i>	<i>ISA</i>
<i>Compressor Isentropic efficiency -</i>	<i>88%</i>
<i>Combustion total pressure loss -</i>	<i>3%</i>
<i>Combustion efficiency -</i>	<i>100%</i>
<i>Turbine isentropic efficiency -</i>	<i>90%</i>
<i>Pressure loss in unheated duct -</i>	<i>2%</i>
<i>Pressure loss in reheated duct -</i>	<i>3%</i>
<i>Reheat combustion efficiency -</i>	<i>90%</i>

This selection was sought from the most recently developed fighter aircraft engine and predictions as to the likely progress of specific technologies over the

next ten to fifteen years.

Following the preparation of TURBOMATCH input data, the rest of the parameters such as Bypass Ratio (BPR), Overall Pressure Ratio (OPR), and Turbine Entry Temperature (TET) were obtained in the optimization process. The optimization of them somehow depends on some of the constraints imposed on the engine, related to engine-aircraft integration, in particular:

- i) Runway erosion and hot-gas reingestion. Runway erosion may result from the hot, high velocity engine exhaust during powered-lift operation. The gases will also be reingested into the inlet, resulting in degraded engine performance.
- ii) Engine location. Engine location is related to the balance problem. It can be exemplified by transition to hover, which requires acceptable stability during airborne flight and a balance of thrust moments during jet-borne flight. Successful hover balance depends primarily on insuring that thrust centre coincides with the centre of gravity.
- iii) Matching conditions (see Chapter 3).

A series of design point calculations covering a suitable range of the remaining parameters have been carried out with TURBOMATCH by using its loop facility, as can be seen in Figures B.1 and B.2, Appendix B. Such calculations were made for the selected flight conditions of forward speed and altitude of each mode as stated earlier.

The following subsections have been introduced as the optimization procedure of the remaining parameters considering the limits imposed by the constraints and matching conditions.

2.4.1- Bypass Ratio

The bypass ratio of each mode was carried out independently as follows:

2.4.1.1- LP mode

Figures 2.4 and 2.5 are a typical variation of Specific Fuel Consumption (SFC) with specific thrust for TET of 1500K and 1750K, respectively, and with OPR of 14, 16, 20, 26, 30 (Figure 2.4), and 12, 14, 16, 20, 26 (Figure 2.5) and BPR of 1.0, 1.5, 2.0, and 2.5 as parameters.

Comparing both Figures, it can be observed that the values of SFC are lower in Figure 2.4 than those in Figure 2.5 at a constant OPR. In both Figures, the SFC reduces as OPR increases at a constant BPR without a significant change in specific thrust and at a constant OPR, it reduces as BPR increases at the expense of a significant reduction in specific thrust.

Due to the Selective Bleed Engine being a design for military aircraft with supersonic capability, a small BPR of 1.0 was selected in order to keep the frontal area down, to satisfy the compressor operating conditions and engine matching requirements and to meet the engine-aircraft integration, which requires BPR ranges from 0.6 to 1.4 (Reference 13). The other two values selected were OPR of 26 and TET of 1500K. The selected design parameters are represented by point A in Figure 2.4.

A higher BPR reduces SFC and increases the inlet mass flow for a given design thrust requirement. However, the engine inlet frontal area would be larger, making engine-aircraft integration difficult, and would increase the gap between compressor running lines resulting in the engine operation becoming impossible even with compressor variable stator blades.

Another aspect to be considered, it is that of front/rear thrust ratio. It is

important for engine-aircraft integration. Figure 2.6 shows the variation of front/rear thrust ratio with LP compressor pressure ratios for BPR of 0.8, 1.0, and 1.5. It can be seen in Figure 2.6 that at constant pressure ratios, F/R thrust ratio increases significantly with increasing BPR. At a constant BPR, an increase in pressure ratio causes increase in F/R thrust ratio. An F/R thrust ratio between 0.67 and 1.5 (Reference 13) has been considered as a good range of its selection. The final value is a result of interaction of the aircraft and engine design. Lower values cause the engine to be located nearer the front of aircraft and higher values nearer the back.

It is important to select a suitable LP compressor pressure ratio, which also depends on the matching between the two modes. Thus, the author has selected a range of LP compressor pressure ratios, for BPR of 1.0, between 2.0 and 3.5.

2.4.1.2- HP Mode

As in section 2.4.1.1, Figures 2.7 and 2.8 are also a typical variation of Specific Fuel Consumption with specific thrust for TET of 1700K and 1800K, respectively, and with OPR of 20, 26, and 30, and BPR of 0.3, 0.4, 0.6, 0.8, and 1.0 as parameters.

The same analysis carried out in section 2.4.1.1, on the behaviour of SFC with BPR, OPR, and TET, is still applicable in this section. BPR of 0.4, OPR of 26, and TET of 1800K have been selected for this mode and they are represented by point **B** in Figure 2.8. Again the selection was made looking at the matching condition, low inlet mass flow and turbine blade cooling. The air cooling is bled at the exit of the HP compressor and goes to the turbine blades. An OPR over 26 implies high temperatures (A maximum of 900K is allowed for cooling) in the last stages of HP compressor HP mode, making the turbine blade cooling difficult.

2.4.2- Ground Erosion

As stated before, ground erosion is caused by high velocity and hot temperature at the engine exhaust during the takeoff and hover. Figure 2.9 shows the variation of rear nozzle exit temperature with LP compressure pressure ratio for BPR of 0.8, 1.0, and 1.5 at LP mode. It can be seen that nozzle exit temperature reduces with the increase of pressure ratio at a constant BPR and the same occurs with BPR at a constant pressure ratio.

To keep the ground erosion under control, it is suggested, in the public domain, that the nozzle exit temperature should be below 1000K. Therefore, the selected value for LP mode parameters had to meet this requirement also.

2.4.3- LP Compressor Pressure Ratio Optimization

SFC may be further optimized by increasing to some extent the LP compressure pressure ratio. Typical variations of SFC with LP compressor pressure ratio for BPR of 0.8, 1.0, 1.5 can be seen in Figure 2.10. At a constant BPR, the SFC does not change significantly with increasing pressure ratio, as it does with increasing BPR at a constant pressure ratio. Therefore, an increase in the pressure ratio at a constant BPR will not affect substantially the SFC.

2.5- Cycle Matching

The matching of LP and HP modes has been carried out in Chapter 3 with optimized parameters selected in the previous sections, which were:

LP mode

BPR=1.0

OPR=26

TET=1500K

HP mode

BPR=0.4

OPR=26

TET=1800K,

The method deals with fixed geometry turbines, which at the design point normally operate choked. It means that the non-dimensional mass flow of these turbines has reached its maximum value. Because the turbines are choked at the design point, the non-dimensional mass flow of each turbine must be the same in the two cycles being analyzed. This process of ensuring the equality of non-dimensional mass flow is called matching. Once this equality of turbine mass flows is achieved, the compressor operating lines of the compressors have to be generated and checked that they can be delivered by one compressor, with or without variable geometry. This compressor evaluation was executed in the following section for LP, IP and HP compressors.

2.6- Compressor Map Evaluation

One of the main features of the matching method procedure, in Chapter 3, is to allow the variation of compressor pressure ratios without losing the matching. Therefore, the LP compressor pressure ratio was chosen within the range 2.0 to 3.5 based on the analysis carried out in Figures 2.6, 2.9, and 2.10.

For each pressure ratio selected, the compressor operating lines of compressors have to be generated and checked whether they can be delivered by one compressor, with or without variable geometry. The running lines were obtained from TURBOMATCH output data by running it for each mode with the chosen compressor pressure ratios and optimized parameters, for the given mission profile (Table 2.1).

As a result of compressor performance analysis (section 3.3.2.1, Chapter 3), the compressor running lines for each compressor have been accomplished and they can be seen in Figure 2.11 for LP, IP, and HP compressors. The design pressure ratios of each mode were found to be for:

<u>LP mode</u>	<u>HP mode</u>
LP compressor PR=3.00	LP compressor PR=2.10
IP compressor PR=2.41	IP compressor PR=3.13
HP compressor PR=3.60	HP compressor PR=3.96

As can be seen in Figure 2.11, the running lines (LP and HP modes) in the LP and IP compressors are not aligned. Therefore, this requires compressors with variable stator blades to operate. However, the HP compressor does not present the same problem with its running lines.

2.7- Off-design Performance

Having determined the design point parameters, the off- design performance of the Selective Bleed Engine was simulated with the TURBOMATCH for the mission profile in Table 2.1 (full TURBOMATCH output data for LP and HP mode can be found in Figure B.3 and B.4, Appendix B). The analysis of the off-design performance was carried out for Thrust and SFC, with variation of TET and Flight Mach Number. Figure 2.12 shows the thrust produced by the engine in both modes as the Mach number changes. Reheat is available only in the HP mode of operation. Figures 2.13 and 2.14 show the change of thrust and SFC as throttle setting changes. In this cases throttle setting is represented by TET. It can be observed, as in all gas turbine engines, that there is a very large change of thrust with TET. Figure 2.14 shows the change of SFC with TET. The influence of TET on SFC, in both operating modes, is similar to any other jet engine. Figure 2.15 shows the change of SFC with thrust. Again the curves are similar to that of any other jet engine.

2.8- Assessment of the Selective Bleed Engine

To assess the performance of the Selective Bleed Engine, a simulation of an alternative power plant of traditional design, a reheated Turbofan with BPR=1.3, OPR=16 and TET=1800k, was carried out, using TURBOMATCH (The

reheated Turbofan used for comparison is not upgraded version of the Rolls Royce Pegasus engine, its performance has to be optimized and reassessed). This Turbofan has to carry out the same mission as the Selective Bleed Engine, so one of the main constraints is engine performance in the hovering mode. This constraint requires a Turbofan of a medium bypass ratio. The supersonic cruise condition is met by using the afterburner. As a result a Turbofan with the characteristics stated above was selected as a standard for comparison. As a consequence the main difference between the two engines manifests itself in the supersonic cruise mode where one engine is in the HP mode dry condition and the other is employing the afterburner. The standard Turbofan operates at a higher TET than the Selective Bleed Engine in the LP mode, but both engines operate at the same TET for high speed operation. A comparison of the performance of the two engines is shown in Figures 2.16 and 2.17. It can be seen that the Selective Bleed Engine LP mode can achieve higher levels of thrust than a conventional Turbofan (Figure 2.16). The LP mode can achieve significantly higher levels of thrust than the Turbofan in the dry mode up to Mach number 1, mainly because it has been designed for lower TET so there is a significant TET margin which can be usefully employed. The HP mode in reheat, due to the higher nozzle pressure ratio can achieve a higher level of thrust than the Turbofan in the reheated mode. The HP mode running dry is halfway between the dry and afterburning output of the Turbofan. Figure 2.17 shows the comparison of the SFC for the Turbofan and the Selective Bleed Engine. It can be seen that the SFC of the Selective Bleed Engine is significantly better than that of the Turbofan in all modes.

When analyzing the fuel consumption required to accomplish the mission outlined in Table 2.2, it was found that the Selective Bleed Engine saved 337 kg of fuel (As the mission fuel bill calculation indicates in Table 2.4), this being approximately 8.49 percent of the total fuel consumption. The largest difference in fuel consumption manifests itself in the supersonic cruise leg, which if increased would further increase the fuel savings.

Table 2.4 also shows a comparison of the Selective Bleed Engine to a standard Turbofan with a mission fuel bill greater than that of the Selective Bleed Engine. The number of mission phases with reheat for a standard Turbofan has been shown to be considerably larger than that of the Selective Bleed Engine.

The main disadvantage of the Selective Bleed Engine, is that due to the requirement of having to cruise at Mach 1.5 in the dry mode, it will require a larger compressor mass flow and therefore a larger compressor face area. Reheated engines achieve high levels of thrust because specific thrust is very high. A dry engine will have a lower specific thrust, so mass flow and hence frontal area are going to be higher. In this particular case the compressor face diameter of a standard Turbofan will be approximately 15 percent smaller than that of the Selective Bleed Engine. Naturally this larger compressor face is going to have an impact on total aircraft drag that needs to be quantified.

	Selective Bleed Engine		Standard Turbofan
Mission Fuel Bill [kg]	4440		4777
Mode	LP mode	HP mode	-----
Design point	S.L.S.	6km & M=1.2	S.L.S.
BPR	1.0	0.4	1.3
OPR	26	26	16
TET [K]	1500	1800	1800
m [kg/s]	331	243	248
Hot nozzle temperature [K]	883	-----	1299
Thrust split F/R	0.76	-----	0.58
Number of mission phase with reheat	1		8

Table 2.4 Engine Comparison Data

CHAPTER 3

Matching Method

3.1- Introduction

The performance analysis of the Selective Bleed Engine has been carried out in the previous Chapter. One of greatest problems faced in this work was how to perform the engine calculation using a computer programme which was developed for single cycle engines. The VCE combines two cycles within the same engine body. In fact, cycle matching is the most crucial matter in VCE because component design requirements depend on cycle data.

A survey of the literature was undertaken to establish how the problem had previously been dealt with and it was revealed that little research has been carried out in this direction. Most of the technology research is regarded as generic particularly in areas such as low weight, materials, pollutant emissions, performance and new design concepts.

The most recent study in cycle matching was carried out by Williamson (14). In the present Chapter his method is demonstrated and its advantages and disadvantages are assessed.

As a result of the difficulty of engine optimization encountered in Williamson's method, a new technique has been developed. The new method, presented here, enables the designer to match the cycles and to improve

compressor operating conditions without affecting engine optimization.

Pre-assessment of the design point of each mode on the compressor map was carried out to help the designer locate the running lines in the compressor operating range. The method described covers fixed geometry turbines. However, it can easily be extended to include the treatment of variable geometry turbines.

3.2- Williamson's Method

Williamson's method is based on making sure that the non-dimensional mass flow of each turbine is the same in the two operating modes. That is, the turbine is assumed to be choked at the design points for each mode. Hence the size of the engine is determined by turbine components alone. This then ensures that variable geometry is not necessary in the turbine but is automatically introduced for the compressors.

Mathematically the method can be written as follows:

for HP turbine

$$\left(\frac{m\sqrt{T}}{P}\right)_{LPM} = \left(\frac{m\sqrt{T}}{P}\right)_{HPM} \quad (3.1)$$

and for LP turbine

$$\left(\frac{m\sqrt{T}}{P}\right)_{LPM} = \left(\frac{m\sqrt{T}}{P}\right)_{HPM} \quad (3.2)$$

where m , T , and P are the inlet mass flow, total temperature and total pressure, respectively. LPM and HPM stand for LP and HP modes, respectively.

The systematic method is achieved by using the computer programme (TURBOMATCH) which allows the design point looping for each operating mode.

The matching procedure consists of a run at each mode whilst varying the

bypass ratio, compressor pressure ratios and turbine entry temperature until the turbine non-dimensional mass flows match. Figure 3.1 shows a set of design point loops for both LP and HP modes (Two shafts, two compressors and two turbines) which were carried out in "TURBOMATCH". Figure B.1 and B.2, in Appendix B, are examples of how the "TURBOMATCH" input file should be prepared to get the result for each mode. The set of curves in Figure 3.1 stands for a set of design point loops, so that the intersection point is the matching point in which each mode should be performed. In Figure 3.1 there are two curves which overlap and more intersection points may be chosen. However, the intersection point is only achieved by a trial and error procedure. Point A was chosen as an example to demonstrate the method. Once the intersection point is reached the compressor characteristics are obtained from the compressor operating conditions. If the gaps between design point conditions for both modes of the compressor operation are large, the VCE is not feasible even when both compressor and LP turbine have variable geometries. Figure 3.2 is an example. This shows the characteristics for both LP and HP compressors as a result of the matching condition of Figure 3.1. As can be seen in Figure 3.2, the LP compressor pressure ratio change from LP to HP mode makes the operation impossible.

If the compressor operating condition is not satisfied the process must be repeated. Otherwise, it is very time consuming and an improved operating condition may well not, in fact, be achieved.

The advantages of Williamson's method are twofold. Firstly, it is easy to manage when a computer program like "TURBOMATCH" is available and secondly, it can be used for different types of VCE.

The disadvantage lies with engine optimization because even if the compressor operating conditions are accomplished, it does not mean that the VCE is optimised as a whole in terms of SFC and specific thrust. The optimization is a function of by-pass ratio, TET and overall pressure ratio (OPR)

(fish/hook curves). Thus, an attempt to optimise the engine mismatches the matching conditions. To overcome the difficulties presented above, a new method was developed and it is demonstrated in the following section.

3.3- The Engine Matching Procedure

The new method initially treats the two or more cycles as independent engines and then assesses the operation of the components. The first stage of the method is to make an assessment of the major cycle parameters of the various (two in this case) cycles to be analyzed. From the point of view of cycle selection, high bypass ratios and low fan pressure ratios are suitable for high subsonic speed applications. As the flight speed increases however, the requirement is for lower bypass ratios and higher fan pressure ratios.

The method, which is described for a two-spool engine, can be readily extended to a three-spool layout. The first part of the method consists of matching the two turbines. The second part consists of matching the compressors.

The present method deals with fixed geometry turbines, which, at the design point normally operate choked. Therefore, the non-dimensional mass flow of these turbines is the maximum value. Because the turbines are choked at the design point, the non-dimensional mass flow of each turbine must be the same in the two cycles being analyzed. This process of ensuring the equality of non-dimensional mass flow is called matching (the same approach as Williamson). Once this equality of turbine mass flows is achieved, the compressor operating lines have to be generated and checked that they can be followed by a single compressor, with or without variable geometry.

3.3.1- PART I: Turbine Matching

As previously stated, the first part of the method consists of ensuring equality of non-dimensional mass flow of the turbines. This is the case because in this work only fixed geometry turbines are analyzed. When variable geometry turbines are envisaged, the procedure can be modified to allow for a certain pre-determined difference in non-dimensional mass flow. This will reflect the turbine variable geometry.

3.3.1.1- Matching of the HP Turbine

In the case of the two spool engine, the high pressure turbine will be matched first. This implies a choice of the main cycle parameters (OPR, TET and BPR) which is carried out from the stand point of the engine mission optimisation covered in Chapter 2 (fish/hook curves). On a graph of overall engine pressure ratio (vertical axis) and HP turbine inlet $\sqrt{T/P}$, the appropriate values are plotted for several turbine inlet temperatures at the two main flight conditions. This is, one appropriate to the subsonic and one to the supersonic mode. Figure 3.3 shows such a plot with the selected flight conditions of sea level static and flight at 6km altitude at a Mach number of 1.2, for curves of constant turbine entry temperature (TET). It is the variation of flight conditions that causes the separation of the two groups of curves. For Mach numbers greater than 1.2, it has been found that the separation of the two groups of curves becomes even larger due to the ram ratio effect. In consequence, the core mass flow ratio between the two modes of operation also is larger, requiring the engine to swallow more mass flow. As a result, in the LP compressor, the LP mode running line moves to the left and HP mode running line to the right increasing the gap between them. The same occurs in the subsequent compressor. This reduces possibility of accomplishing the compressor matching successfully even with variable geometry.

The overall pressure ratios of the two modes are selected which will yield the appropriate values of HP turbine $\sqrt{T/P}$ for each cycle. The non-dimensional mass flow of the HP turbine has to be the same for the two cycles. This will then define the ratio of mass flows for both cases, as follows:

$$\frac{m_{LPM}}{m_{HPM}} = \frac{(\frac{\sqrt{T}}{P})_{HPM}}{(\frac{\sqrt{T}}{P})_{LPM}} \quad (3.3)$$

derived from equation (3.1).

3.3.1.2- Matching of the LP Turbine

Once the HP Turbine has been matched, the next step is the matching of the LP Turbine. For the two modes of operation of the engine, for different values of TET and overall pressure ratio (Figure 3.4), a graph of HP Compressor pressure ratio vs LP Turbine inlet $\sqrt{T/P}$ can be produced from the work compatibility of the HP Shaft. Once the plots are produced by means of a graphical procedure, equality of the non-dimensional mass flows of the LP turbine can be equalised for the two cycles. This calculation also determines the pressure ratio of all the components. The procedure is as follows:- The HP compressor pressure ratio is selected arbitrarily for one of the modes. From the appropriate curve, the value of LP turbine $\sqrt{T/P}$ at LP turbine inlet can be found for this mode. The ratio of the mass flows is known from the matching of the HP Turbine and hence if $\sqrt{T/P}$ at LP turbine inlet is known for one mode, it can easily be calculated for the other mode using the following equation:

$$\left(\frac{\sqrt{T}}{P}\right)_{HPM} = \frac{m_{LPM}}{m_{HPM}} \left(\frac{\sqrt{T}}{P}\right)_{LPM} \quad (3.4)$$

derived from equation (3.2), if the LP mode is selected. The reverse procedure is now carried out for the other mode. The knowledge of LP Turbine inlet $\sqrt{T}/P$ determines the HP compressor pressure ratio for the other mode, by employing the appropriate curve. With this information, the pressure ratios of the compressors and the HP turbine can be calculated. Bypass ratio is known, so the LP turbine pressure ratio can be obtained.

3.3.2- PART II: Compressor Matching

The next part of the exercise is to ascertain that the design of the compressors can satisfy the requirements imposed by the turbines in the two cycles. This method of compressor matching also consists of two steps: the determination of the compressor running lines and the preliminary analysis of the compressors.

3.3.2.1- Determination of the Running Lines

The design point parameters for the two cycles have been calculated from the first part. It is possible now to carry out an approximate off-design performance analysis of the two cycles to obtain the approximate position of the compressor running lines. Figure 3.5 shows the results of such an analysis. All the running lines correspond to different engines, all having the same turbine non-dimensional mass flow.

An example of how to employ the method has been carried out for the Selective Bleed Engine in Appendix C. Figure 3.6 illustrates the first attempt at compressor matching for the Selective Bleed Engine after turbine matching. The selected parameters were:

<u>LP mode</u>	<u>HP mode</u>
LP compressor PR=3.00	LP compressor PR=2.6
IP compressor PR=2.41	IP compressor PR=2.52
HP compressor PR=3.60	HP compressor PR=3.96

As can be seen in Figure 3.6, both LP and IP compressors do not have running lines (LP and HP mode) close enough to allow the compressors to operate. Thus, further attempts in order to bring the LP and HP running lines closer, were required. The HP compressor was not necessary because LP and HP running lines are nearly aligned and therefore the HP compressor can operate.

The second attempt has kept the same compressor pressure ratio as that for the LP mode. For HP mode, the LP and IP compressor pressure ratios were changed, without changing HP compressor pressure ratio, as follows:

<u>LP mode</u>	<u>HP mode</u>
LP compressor PR=3.00	LP compressor PR=2.10
IP compressor PR=2.41	IP compressor PR=3.13
HP compressor PR=3.60	HP compressor PR=3.96

The product of LP and IP compressor pressure ratio is the same in both modes. In this case, it was not necessary to rematch the turbines again.

With the change of the LP compressor pressure ratio from 2.6 to 2.10, in the HP mode, the HP mode running line has moved further down for the LP compressor and further up for the IP compressor, as can be seen in Figure 3.7. Comparing Figures 3.6 and 3.7, a significant improvement is made in the LP and IP compressors. Therefore, the conditions given in Figure 3.7 were selected to carry out the compressor performance analysis.

Before moving into the next section the following points should be borne

CHAPTER 4

Compressor Design and Performance

4.1- Introduction

After cycle calculation, optimization, and analysis, the compressor design and off-design performance is carried out for LP, IP, and HP compressors. It is the general purpose of this Chapter to present a summary of the major aspects on the preliminary aerodynamics design of an axial-flow compressor, essentially for subsonic and transonic configuration with relative rotor tip Mach number up to 1.3, and to upgrade some computer programmes available in the SME directory. These computer programmes are very powerful tools, but naturally they are not so sophisticated as the design facilities available to the gas turbine manufacture. Nevertheless, the main aim is to establish that the compressors could be designed to perform as the cycle demanded.

Analyses and difficulties in the design imposed by aircraft constraints are assessed and some solutions are considered in order to minimize these problems.

Whilst the LP compressor has two transonic stages the IP and HP compressors are entirely subsonic. The estimated efficiency of the LP compressor is 2.5 per cent below the target efficiency. However, the other two compressors are above their target values.

For off-design conditions, stage stacking techniques for performance prediction using a computer programme have been used to obtain the overall performance maps.

4.2- Modification of the Existing Computer Programmes

The design of the three compressors required the use of several computer programmes. Three computer programmes were found in the SME directory, forming a complete package for the design of compressors. The first one, entitled ANNULUS calculates thermodynamics properties and physical dimensions at the inlet and outlet of each stage. The second, called CEMCOMP, calculates quantities related to the aerodynamics of the compressor, stage by stage at five radial sections. The third called SCD15 executes the blading of the compressor as well as the calculation of the blade stresses and vibration characteristics.

These computer programmes require short running time so that they enable the designer to concentrate more on the design optimisation and mechanical design rather than the mathematics behind it.

During the design process the following problems arose:

- 1- The programmes had to be used interactively with one another, by feeding the output of the one into the input of the other. The chance of errors were, therefore higher, especially as the number of stages increase.
- 2- The loss model used by CEMCOMP was insensitive to changes of, blade aspect ratio, space chord ratio, and small variation in hub-tip ratio. In addition, the shock model could not satisfactorily predict the shock losses.
- 3- A careful examination on profile loss, secondary losses, and annulus losses revealed that the calculation procedure was incorrect. Modifications

in mind to give good compressor matching:

1) BPR should not exceed 1.4 in the LP mode. The reason is that as the BPR increases, the LP mode running line moves to the left (Figure 3.5), increasing the gap between the two running lines (especially in LP and IP compressors).

2) A change in LP compressor pressure ratio moves the running lines up and down in the LP compressor and down and to the left and up and to the right for the IP compressor (Figures 3.6 and 3.7), respectively. Thus, a higher LP compressor pressure ratio in the LP mode than in the HP mode is advisable because it brings the LP and HP mode running lines closer. Fewer less iterations attempts are then required.

3) The variable stator blade compressor moves the surge line toward the bottom left hand corner in the compressor characteristics. Figure 3.8 shows the displacement of the surge line for different blade settings (0°, 10°, 20°, 30°, 40°). Therefore, the running lines (LP and HP modes) have to be placed taking into account what is stated above, otherwise one of running lines will be outside of the compressor characteristics.

It is very difficult to establish firm rules for compressor matching. The only way is to use a trial and error procedure. Following the general rules, only two or three attempts are necessary before the designer achieves a suitable location for the running lines. The other thing that may help is making use of imaginary compressor characteristics for one mode of operation. This facilitates more rapid selection of the appropriate parameters for the second mode of operation.

3.3.2.2- Compressor Performance Analysis

Having obtained the position of the running lines, the next step is to define the compressors that can operate in the required conditions. Therefore, both

design and off-design compressor analysis is required. The design point analysis will produce the compressor geometry, and the off-design analysis will produce the compressor characteristics. These have been carried out in Chapters 4 and 5 and it is anticipated that the necessary use of variable geometry will be extensive.

If the performance of the compressors is not satisfactory, the main cycle parameters are reviewed and the calculation repeated. Normally not more than two iterations are necessary. If the performance of the compressors is deemed to be satisfactory, the analysis of the engine can now be refined.

The method as described treats fixed geometry turbines as mentioned earlier. The inclusion of turbine variable geometry is very simple. The steps of turbine matching are modified to allow for differences in turbine non-dimensional mass flow, these differences reflecting the variability of turbine nozzle area.

were therefore necessary.

As a result after a considerable literature review, new loss models were introduced to the programmes. Full details of these can be seen in Appendix D. Basically, the modification were executed on:

- a) profile loss;
- b) shock loss;
- c) Reynolds number and roughness effects;
- d) secondary losses;
- e) annulus losses;

In addition, incidence and deviation correlations were introduced to calculate the blade angles in CEMCOMP and SCD15. Moreover, to avoid the frustrating interruption of manually feeding data between ANNULUS and CEMCOMP, ANNULUS was introduced into CEMCOMP as a subroutine.

Since the modifications to the programmes were extensive, it was decided to change the name of CEMCOMP to COMPDES and SCD15 to BLADE. The features of COMPDES and BLADE and the users' guides are in Appendix D.

The COMPDES programme uses the radial equilibrium approach to design the compressor. However, this is no longer the way compressors are designed since numerically based methods provide generally more accurate results.

Although simple radial equilibrium is no longer a major tool for final design of compressors, it can still be used to undertaken preliminary design and it is therefore, appropriate for this work. In addition, very high accuracy is not important in this work since the aim is to establish that the compressors could in principle, be designed to perform as the cycle demanded.

Finally, the modifications to the existing programmes had as an objective the qualitative analysis of the losses and corresponding assessment of efficiency

change.

The following sections deal with both the design and off-design performance of the three compressors with the help of the modified computer programmes.

4.3- Design Point Objectives

The design points for LP, IP and HP compressors are defined by the cycle simulation and analysis carried out in Figure 2.11 of Chapter 2. The controlling parameters are drawn from the TURBOMATCH simulation. The selected design point data can be seen in Table 4.1 for each compressor.

PARAMETER	COMPRESSOR		
	LP	IP	HP
Pressure Ratio	3.0	3.31	3.96
Target Efficiency (%)	88.0	88.0	88.0
Mass Flow (kg/s)	331.0	243.4	173.8
Inlet Total Pressure (atm)	1.0	2.35	7.36
Inlet Total Temperature (K)	288.0	406.8	580.3

Table 4.1 Design point parameters.

4.4- Design Parameters Analysis

Having modified computer programmes for the design point conditions, the analysis on aerodynamic parameters was carried out in order to select appropriate values to achieve the target efficiency.

The following sections provide ranges in which both aerodynamic and mechanical parameters can be selected without serious problems as well as the resource to make decisions during the design process.

4.4.1- Inlet Axial Velocity

The inlet axial velocity in a multistage axial-flow compressor depends on the application. For instance, inlet axial velocity for an industrial gas turbine is of the order of 150 m/s whereas for advanced aero engines can be in excess of 200 m/s. The latter higher values of axial velocity are used to provide the high flow rate per unit frontal area required for turbojet and turbofan engines.

On the other hand, exit axial velocity is usually fixed by the diffuser and combustor requirements. Table 4.2 shows the inlet axial velocity selected for LP, IP, and HP compressors.

4.4.2- Inlet Mach Number

For a subsonic compressor cascade at zero incidence, the inlet 'critical' and 'maximum' Mach numbers are in the range of 0.7 and 0.85. High Mach number narrows the effective operating incidence range and can lead to poor off-design performance. However, with the development of high speed profiles (thin blade sections of quite different shape) such as double circular arc and controlled diffusion airfoils, a stage can be designed to operate efficiently with a relative Mach number up to 1.2 over part of the blade height. This is called a transonic stage. With fans of large by-pass ratio however, the inlet relative Mach number at the rotor tip may be of the order of 1.5. The use of IGVs will slightly reduce the inlet relative Mach number.

4.4.3- Hub-tip Ratio

The inlet hub-tip ratio normally varies between 0.4 and 0.6 while for the outlet no more than 0.92 is used. Using very low hub-tip ratio increases the mismatch between compressor and turbine diameters. In addition, both mechanical and aerodynamic design of the first stage is more complex.

COMPRESSOR

PARAMETER	LP	IP	HP
Number of stages	5	7	7
Rotational Speed (rpm)	4450.0	4450.0	10000.
Inlet Axial Velocity (m/s)	200.0	190.0	220.0
Inlet Mach Number	0.58	0.46	0.44
Inlet Hub-Tip Ratio	0.43	0.70	0.74
Outlet Hub-Tip Ratio	0.71	0.83	0.91
Radial Equilibrium solution	Free Vortex	Free Vortex	Free Vortex
Inlet Stage Loading Coefficient (pitch)	0.43	0.35	0.34
Inlet Flow Coefficient (pitch)	0.76	0.72	0.65
Inlet Tip Speed (m/s)	367.0	308.0	388.1
Rel. Rotor Tip Inlet Mach Number	1.27	0.92	0.95
Annulus Geometry	Const. Mean Radius	Const. Hub Radius	Const. Tip Radius

Table 4.2 Selected Compressor Design Parameters.

For a fixed limited blade speed the rotational speed is a function of the hub-tip ratio. There are, therefore, a wide range of solutions and final choice will depend on the experience of the particular designer.

The chosen values for LP, IP, and HP compressors are shown in Table 4.2. Further discussion on the choice of appropriated hub-tip ratio is given in subsequent sections.

4.4.4- Tip Speed

The centrifugal stress in the rotor blade is determined by the tip speed, blade material, and hub-tip ratio. Once blade material and hub-tip ratio are

selected the centrifugal stress is a function only of tip speed. In fact, the first stage is the longest and the most highly stressed. The blades are subject to fluctuating gas bending stresses which add to the centrifugal stress and may cause fatigue failure.

Normally, in compressor design the tip speed is of the order of 350 m/s. Up to this value, stress problems are not usually critical in the sizing of the annulus. However, tip speeds around 450 m/s are commonly used in fans of high by-pass ratio turbofans with low hub-tip ratio. The selected values for LP, IP, and HP compressors are presented in Table 4.2.

4.4.5- Stage Loading and Flow Coefficient

The choice of a realistic stage loading is the single most important decision in the design process. The wrong choice may lead to untold problems later with little possibility of reaching the combination of efficiency, pressure ratio, mass flow, and operating range originally intended. However, fundamental to all of the aerodynamics design are the basic decisions of an aerodynamic nature. At blade mid-height a choice must be made for the local flow coefficient, $\phi = V_a/U$ and the stage loading coefficient $\psi = \Delta H/U^2$, where V_a is the axial velocity, U is the blade speed, and ΔH is the stagnation enthalpy rise.

The flow coefficient effectively determines the performance of the stage. The axial velocity used may be the local value or the mean value across the annulus. As ϕ is reduced at the off-design conditions, the incidence rises bringing changes in the blade operation. Most axial-flow compressors are designed with ϕ between 0.3 and 0.6. The lower end of this range is more common nowadays.

The relationship between stage loading and flow coefficient is plotted in Figure 4.1 (Reference 14). Typical design values are shown.

The values of stage loading and flow coefficient for LP, IP, and HP

compressors were selected in the range of 0.3 to 0.5 and 0.3 to 0.6, respectively. The corresponding range of static pressure rise coefficient $\Delta P/D_p$ is 0.6 to 0.5. Table 4.2 shows these values for each compressor at blade mean radius.

4.4.6 Degree of Reaction

The degree of reaction is a useful concept in compressor design. It relates to the ratio of static pressure rise in the rotor to that in the entire stage. It can also be defined in terms of enthalpy rise.

The degree of reaction can range from values 0% to 100%. The limiting values are of particular interest. For zero reaction, the rotor blades are of impulse type and all the static pressure rise occurs in the stator. For 100 per cent reaction, the stators are of impulse type. Therefore, 50 per cent reaction is attractive since both components of the stage share the diffusion equally.

Efficiency is not very sensitive to the precise degree of reaction. However, reaction will, in general, vary across the annulus, especially for stages of low hub-tip ratio. In practise the designer is more concerned with the levels of diffusion factor and Mach number rather than the degree of reaction.

In the COMPDES programme the degree of reaction is function of inlet swirl angle when the free vortex method is used. To set a particular degree of reaction with the free vortex solution, the user has to vary the inlet swirl angle. As a guide, increasing the inlet swirl reduces reaction.

In this work, zero inlet swirl angle has been selected for each stage of the LP compressor. For the IP and HP compressor however it was found more appropriate to keep the degree of reaction close to 50 per cent at mean blade height, after trials with other values. This can be seen in Table 4.3 a,b,c for LP, IP and HP compressors, respectively.

Zero exit swirl angle was chosen for the compressors since the flow at entry to the combustion chamber should be axial to avoid excessive diffuser losses.

4.4.7- Stage Temperature Rise

In different compressor designs, the stage temperature rise can vary widely, depending on the application. For instance, it may vary from 10K to 30K for subsonic stages and may be 45K or even higher in high-performance transonic stages. The temperature rise is not normally considered constant for each stage through the compressor. It is normal to design for lower temperature rise in the first and last stages. The first stage, for example, has a high Mach number at the tip of the rotor and the inlet flow may become distorted with significant variations in axial velocity across the compressor annulus. By reducing the temperature rise these problems can be partially alleviated. For the last stage, the outlet flow goes into the diffuser at entry of the combustion chamber. It would be desirable to have a purely axial velocity at exit. Therefore, the swirl should be kept as low as possible implying lower stage loading.

The choice of the number of stages is directly related to the temperature rise chosen for each stage and individual blade loading. As tip speed increases the temperature rise is higher for the same stage loading so that the number of stages is reduced. The temperature rise distribution through the compressor has been selected for LP, IP, and HP compressors and is shown in Table 4.3. The distribution starts with lower values for the first stage gradually increasing to the middle stage and progressively falling to the last stage.

4.4.8- Diffusion Factor

The blade loading limits are usually assessed by the level of diffusion factor (or equivalent diffusion ratio). Essentially, the diffusion factor defines the ratio of the peak velocity on the suction surface of the blade to that at the trailing

edge. Additionally, the flow turning is accounted for. Values of diffusion factor between 0.45 and 0.6 are usually taken as typical at the design point. The lower value relates to tip and the higher to the hub.

LP COMPRESSOR

PARAMETER	STAGE						
	1st	2nd	3rd	4th	5th	6th	7th
Aspect Ratio	3.0	3.0	2.8	2.5	2.3	-	-
Stage Temperature Raise (K)	23	25	28	27	26	-	-
Work Done Factor	0.98	0.95	0.92	0.90	0.88	-	-
Tip Clearance (%)	1	1	1	1	1	-	-
Stage Inlet Swirl angle Distribution (degree)	0.0	0.0	0.0	0.0	0.0	-	-
Degree of Reaction (pitch)	83	81	78	78	78	-	-
Tip Relative Mach Number (rotor)	1.27	1.18	1.10	1.03	0.97	-	-
Blade s/c (pitch)							
Rotor -	0.80	0.80	0.80	0.80	0.80	-	-
Stator -	0.70	0.70	0.70	0.70	0.70	-	-

Table 4.3(a) Selected Compressor Design Parameters.

Another method used for assessing the allowable diffusion is de Haller number. This is defined the ratio of the blade inlet to outlet velocity, V_2/V_1 , values of de Haller number are usually > 0.70 . This is equivalent to a static pressure rise (ΔP) not exceeding about 0.50 times the dynamic pressure (D_p) into a blade row. Because of the simplicity of de Haller number it is frequently used in preliminary design work but detailed blading uses the diffusion factor.

For the preliminary design, de Haller number static pressure rise coefficient, stage loading, and flow coefficient relationship are given in Figure 4.1. All these values have been selected, within their respective ranges for each compressor.

IP COMPRESSOR

PARAMETER	STAGE						
	1st	2nd	3rd	4th	5th	6th	7th
Aspect Ratio	3.5	3.2	3.0	2.9	2.9	2.8	2.8
Stage Temperature Rise (K)	23	25	26	27	28	26	26
Work Done Factor	0.98	0.95	0.92	0.90	0.90	0.88	0.88
Tip Clearance (%)	1	1	1	1	1	1	1
Stage Inlet Swirl angle Distribution (degree)	0.0	5.0	5.0	5.0	5.0	5.0	5.0
Degree of Reaction (pitch)	83	73	71	68	68	67	66
Tip Relative Mach Number (Rotor)	0.92	0.84	0.80	0.76	0.73	0.70	0.67
Blade s/c (pitch)							
Rotor -	0.80	0.80	0.80	0.80	0.80	0.80	0.80
Stator -	0.70	0.70	0.70	0.70	0.70	0.70	0.70

Table 4.3(b) Selected Compressor Design Parameters.**4.4.9- Effect of Velocity Profile Distortions****(Work Done Factor)**

For a given inlet mass flow through the blade row, the effect of the wall boundary layers is to increase the axial velocities across the mainstream portion of the annulus and to decrease the axial velocity near the end walls. Because the values of velocity over the major portion of the annulus are greater, the mass-averaged energy addition at the design flow is lower than the design, resulting in reduction of work capacity. Hence, the actual stage temperature rise is lower than that envisaged by the design values.

Due to the non-uniformity of the axial velocity profile there is a reduction in the effective work done in a blade row, the ratio between the actual work done and that achievable with uniform inlet conditions is known as work done factor. The work done factor varies with the stage in a compressor and typical values

(for each stage starting from the first) can be seen in Figure 4.2 (Reference 15).

HP COMPRESSOR

PARAMETER	STAGE						
	1st	2nd	3rd	4th	5th	6th	7th
Aspect Ratio	2.0	2.0	2.0	2.0	1.9	1.9	1.8
Stage Temperature Rise (k)	38	40	43	44	44	44	43
Work Done Factor	0.99	0.95	0.92	0.90	0.88	0.88	0.88
Tip Clearance (%)	1	1	1	1	1	1	1
Stage Inlet Swirl Angle Distribution (degree)	0.0	9.0	10.0	12.0	13.0	14.0	13.0
Degree of Reaction (%) (pitch)	82	71	69	66	65	64	66
Tip Relative Mach No (Rotor)	0.95	0.87	0.83	0.79	0.77	0.74	0.72
Blade s/c (pitch)							
Rotor -	0.80	0.80	0.80	0.80	0.80	0.80	0.80
Stator -	0.75	0.75	0.75	0.75	0.75	0.75	0.75

Table 4.3(c) Selected Compressor Design Parameters.

The chosen values for the present work were taken from Figure 4.2 for each compressor and they are shown in Table 4.3.

4.4.10- Radial Equilibrium Solution

The free vortex hub to casing design has been selected for all three compressors. The reason for choosing the free vortex is that it keeps the axial velocity more or less uniform in the radial direction. However, it gives substantial blade twist across from root to tip especially for low hub-tip ratio.

Alternative design solutions were tried but did not show a significant change in efficiency except for the LP compressor. However, for simplicity, the free vortex method was adopted throughout.

4.4.11- Space/chord Ratio

An appropriate choice of space/chord ratio for compressor blading at subsonic speed is extremely important. For example, some correlations allow for the effect of space/chord ratio on incidence and deviation angles. Figure 4.3 (Reference 16), displays the variation of total pressure loss coefficient with inlet Mach number for three different space/chord ratio in the transonic range. It can be seen in Figure 4.3 that between a Mach number of 1.0 and 1.26, space/chord ratio 1.0 offers lowest total pressure loss, and between Mach number 1.26 and 1.4, space/chord ratio of 0.7 offers lowest total pressure loss. Space/chord ratios in the range 0.7 to 0.8 are used at the rotor tip where the Mach number is higher. However, evidence suggests that near the design point, the efficiency tends to be slightly lower if the space/chord ratio is on the low side (and low aspect ratio), but the pressure rise and operating range are greater.

The selected values of space/chord ratio for LP, IP, and HP compressors are shown in Table 4.3. They were selected according to the Mach number variation along the blade height based on the values given in Figure 4.3.

4.4.12- Aspect Ratio

The choice of aspect ratio is related to the number of blades, stage length, effect on secondary losses and vibration. The use of low values leads to an increase the secondary and tip clearance losses. High values of aspect ratio lead to vibration problem.

The choice of a suitable value depends on the application. For fans and LP compressors the inlet values may be chosen between 4.0 and 4.5. However, new design trends indicate aspect ratios between 1.5 and 2.5 for most subsonic and transonic designs. This is because large chord blades are more effective in the endwall regions and which are crucial in determining both efficiency and the stall point.

The selected values of aspect ratio, for LP, IP, and HP compressors are presented in Table 4.3.

4.4.13- Tip Clearance

The effect of tip clearance on overall compressor performance is not straightforward simple. Lakshminarayana (17) in his experiments found that optimum tip clearance lies between 1.0 and 1.5 per cent of chord for rotors, compared with between 3.0 and 5.0 per cent for cascades. However, the optimum clearance for rotors is generally smaller than that which can be tolerated for mechanical reasons.

Some experiments have shown that not only does the pressure recovery rise and efficiency fall as the tip clearance is increased but the surge line moves a considerable way downwards a lower pressure ratio.

In a multistage axial-flow compressor the precise effect of tip clearance becomes confused because the performance of each blade row affects all those downstream. For instance, increased tip clearance yields extra blockage and extra loss in the front stages and the matching of the downstream stages is altered. In the case of stators it is relatively easy to provide a shroud seal at the hub.

Amongst all experimental work in the literature the following rule may be drawn for tip clearance: every 1 per cent increase in tip clearance results in about 2 per cent deterioration in efficiency. A good range to take tip clearance is between 0.5 and 1.5 per cent of chord at tip. The chosen values for LP, IP, and HP compressors can be seen in COMPDES input data files in Figures F.1, F.2, and F.3 (Appendix F), respectively.

4.5- Design Problem and Solution

The compressor design for the three compressors has been carried out based on the information available in the literature and section 4.4. Despite a wide range of solutions some difficulties arose during the design process due to aircraft constraints. The solutions taken and the difficulties met, for each compressor, are discussed as follows:

1- LP compressor

The LP compressor plays a prominent role in determining the frontal area of the engine. Because in general, the maximum area of the engine is controlled by the LP compressor tip diameter, it is important to keep this diameter as small as possible to reduce nacelle drag. Therefore, high flow capacity was required. This is also a factor in minimizing specific engine weight.

The first attempt to design for minimum inlet outer diameter was to set hub-tip ratio at 0.5, inlet axial velocity at 150 m/s and the inlet relative tip Mach number to 1.2. As a result, the inlet outer diameter reached values over 1.8m, which is very large for this application. For instance, the Rolls Royce Pegasus engine has an inlet outer diameter about 1.23m. Further attempts were carried out in order to reduce the outer diameter. Assessing all results within the aerodynamical and mechanical limits stated in previous sections, the following data were selected: inlet hub-tip ratio equal 0.43, inlet axial velocity equal 200m/s, and tip speed 367.2m/s.

As a result, the inlet tip diameter dropped to 1.59m and the inlet tip Mach number relative to the first and second rotor reached 1.27 and 1.18, respectively, characterizing transonic stages. Moreover, the stage loading and the flow coefficients at the hub were 0.83 and 1.2, respectively, for the first stage. These values were out of the ranges quoted in the previous sections. Nevertheless, the design can be accepted as reasonable if the pitch line values are within the

acceptable ranges. The pitch line values are shown Table 4.2.

The compressor isentropic efficiency achieved was 86 per cent. Real compressors of good aerodynamic design have an efficiency in the range of 86 to 92 per cent. However, the achieved efficiency is 2 per cent below the target value of 88 per cent. For the purpose of this work the efficiency achieved can be considered satisfactory. Any modification to improve the efficiency increased the outer diameter.

A constant tip radius design was chosen in the first place. Constant tip radius allows the designer to increase the aerodynamic loading for downstream stages. This means that the temperature raise per stage may be increased, consequently, reducing the number of stages. Unfortunately, it was not possible to retain constant tip radius due to swan neck problems encountered between IP and HP compressor. Constant tip radius increased the difference in levels between exit hub radius of the IP compressor and inlet hub radius of the HP compressor since the design of the IP compressor depends on the design of the LP compressor, which are on the same spool. On the other hand, constant hub radius would reduce this difference in diameter levels relieving the swan neck problem. However, the number of stages in both LP and IP compressors would be even larger and increase the aerodynamic problems. Therefore, constant mean radius was selected as an intermediate solution to the problem.

The remaining parameters were within their respective ranges and a full detailed printout from COMPDES is shown in Figure F.4 (Appendix F). The principal values are also shown in Tables 4.2 and 4.3.

2- IP compressor

The rotational speed of IP compressor, is determined since it is connected to the LP compressor ie 4450 rpm. This low rotational speed implies low blade speed and therefore, lower temperature rise is achievable in the later stages. As

a result the number of stages have been increased in the IP compressor to 7 for a pressure ratio about the same as in LP. High inlet hub-tip ratio was also required to keep the design within acceptable aerodynamic and mechanical limits.

A lower hub-tip ratio would indeed improve the swan neck between IP and HP compressors, as explained earlier, by reducing the hub radius. However, the number of stages would then increase even more because of the reduction of blade speed and therefore, temperature rise per stage.

A three spool design could have overcome this problem. In this case, each compressor would have been driven by a different turbine and at different more favourable rotational speeds. On the other hand, the matching procedure would obviously have been much more complicated.

The estimated compressor isentropic efficiency reached 89 per cent, which is greater than the target efficiency. The final values of the main parameters and dimensions are presented in Tables 4.2 and 4.3. The remaining values are given in Figure F.5 (Appendix F).

3- HP compressor

Once again the need for a swan neck influenced the design of the HP compressor. A high inlet hub-tip ratio was also require (Table 4.2) to reduce the difference in diameters between IP and HP compressor. Consequently, a high outlet hub-tip ratio and low rotational speed was necessary.

The design of the HP compressor is considered satisfactory for the purpose of this work. No further problems were found during the design process. The compressor isentropic efficiency was estimated to be of 88 per cent, which is the same as the target value.

A constant outer diameter has been selected to reduce the number of

stages and to improve the geometrical matching to the IP compressor, diffuser, and combustion chamber.

4.6- Overall Analyses of the Compressor Design

Table 4.4 summarizes the main parameters taken from Tables 4.1, 4.2, 4.3. The remainders can be seen in the COMPDES output data files in Figures F.4, F.5, F.6, Appendix F for LP, IP, and HP compressors, respectively.

Examination of Table 4.4 shows a relatively large number of stages required by the compressors. As stated, the design tools available to the engine manufacture are more advanced than those available in the open literature. With the modern techniques employed by engine manufactures the compressors would be smaller and shorter, probably a 3 stage LP compressor, a 4 stage IP compressor, and a 5 stage HP compressor. Similarly, diameters could also be reduced with more complex design tools.

The above compressor design matching has demonstrated the feasibility of the design, leaving room for improvement only with more sophisticated design tools.

4.7- Compressor Off-design Performance

Having obtained the design point performance of the compressors, the next step was to evaluate their off-design performance and to estimate how the actual compressor characteristics would fit within those predicted. It is well known that there are several different methods of predicting performance maps of compressors of known geometry. A survey of these has been carried out and the results are presented in the following section.

COMPRESSOR

PARAMETER	LP	IP	HP
<i>Pressure Ratio</i>	3.0	3.13	3.96
<i>RPM (LP mode)</i>	4450.0	4450.0	9500
<i>RPM (HP mode)</i>	4450.0	4450.0	10000
<i>Inlet Tip Diameter (m)</i>	1.59	1.32	0.74
<i>Outlet Tip Diameter (m)</i>	1.32	1.11	0.74
<i>Inlet Hub Diameter (m)</i>	0.69	0.92	0.55
<i>Outlet Hub Diameter (m)</i>	0.94	0.92	0.68
<i>Non-dimensional mass Flow (SI)</i>	0.0560	0.021	0.0057
<i>Length (m)</i>	1.28	0.75	0.46
<i>Efficiency (%)</i>	85.94	89.18	88.40

Table 4.4 *Compressor Design Parameters.***4.8- Off-design Performance Prediction**

The behaviour of the compressor operating at off-design condition is non-linear and depends on the performance of the individual stages in terms of both pressure ratio and temperature raise.

Most of the methods for predicting off-design performance are based on the in-house data bases. Some of few methods in the open literature are by Serovy (18), Wall (19) and Howell and Calvert (20). Some methods assess the performance on a pitch line radius, others use blade-to-blade analysis at several radii. The performance of individual stages, in all methods is evaluated in terms of blade speed and inlet non-dimensional flow rate.

It was not aim of this work to develop a computer programme to predict the off-design performance of axial flow compressors. The idea was to look for one which could give the required compressor characteristics with least running time. The reason is both LP and IP compressors require variable stator blades

and a programme with reduced running time may achieve the end point more quickly and with less cost. Precise accuracy was not vital.

A mean radius, stage stacking method to predict the complete performance characteristics of a multistage axial flow compressor, named HOWELL, was identified in the Turbomachinery Group of the School written by A.R. Howell of NGTE.

The programme had been evaluated by NGTE. Four compressors were selected for the evaluation, the number of stages varied from two to eight and the overall pressure ratio from 1.88 to 10.26. The test and predicted overall characteristics presented were generally in good agreement.

Analysis of the users' guide, written by Calvert (21), and the programme has revealed that it is very quick and easy to use. The programme provides a full output data with the inlet conditions, performance of each stage and overall compressor performance. The users' guide gives complete input file data requirements and how to run it.

COMPDES was prepared to provide an output file, which can be selected when COMPDES is running, and used as an input file to HOWELL to avoid lengthy manual procedures. The running sequence for HOWELL, COMPDES and BLADE computer programme are shown in Figure D.20, Appendix D.

The LP, IP, and HP compressor input data files for HOWELL, produced by COMPDES, can be seen in Figures F.7, F.8, and F.9 (Appendix F). These input files were in turn loaded into HOWELL and the output prepared in graphical form, as shown in Figures 4.4, 4.5, and 4.6 for LP, IP, HP compressor, respectively. The full HOWELL output data files are not detailed here, because of their size. These Figures represent typical axial-flow compressor performance maps with pressure ratio and isentropic efficiency (η) against the non-

dimensional mass flow $(\dot{m}\sqrt{T}/P)$ for given values of the non-dimensional speed $(N/\sqrt{T})$. Five rotational speed lines were chosen starting from 100 to 50 per cent of the design speed. For each rotational speed line there is the corresponding efficiency line. Moreover, the LP and HP mode running lines were placed on the characteristics in order to evaluate the surge line for part load performance and cycle change.

Both LP and HP mode running lines present a surge problem at low speed for the LP compressor (see Figure 4.4). In addition, the HP mode running line operates at low speed reducing the engine overall efficiency. For the IP compressor (Figure 4.5) the situation is significantly different. The HP mode running line is within the compressor operating range and operates at high efficiency. On the other hand, the LP mode running line is entirely out of the compressor operating range. The HP compressor (Figure 4.6) has both LP and HP mode running lines inside the compressor operating range at high efficiency.

As a result of the analysis of the compressor characteristics both LP and IP compressors required variable geometry. This was envisaged at the outset. The compressor variable geometry procedure and analysis is described in the following Chapter.

CHAPTER 5

Compressor Variable Geometry

5.1- Introduction

The demand for improved fuel economy, reduced first cost and reduced weight has led to the development of compressors incorporating variable stator vanes.

The improvement of overall compressor range of operation reduces to individual blade row performance improvement. The aerodynamic behaviour of a compressor of fixed geometry is optimised only at design conditions. At other speeds and flows stages are mismatched to the local flow conditions. Variable stators restore rotor incidences close to their design values, improving performance at off-design. In addition, a compressor having variable stator blades may be utilized to achieve rapid engine thrust modulation; a desirable feature for ASTOVL.

In this work, the two modes of operation produce two running lines in different positions on the compressor characteristics. Hence use of variable stators was strongly indicated for both LP and IP compressors. This not only to improves surge margin, but reduces mass flows when the engine shifts to the alternative operating mode.

The LP and IP compressor with variable IGV's and stators have been investigated using a computer programme. For the subsonic mode (LP mode),

the LP compressor performance, with optimal restaggers, was significantly improved in comparison with the original. The schedule of optimal restaggers is presented. The IP compressor performance map was achieved by setting IGV plus all stators at 30° . For the supersonic mode (HP mode), the LP compressor performance map was established by setting IGV's at 35° and all the stators at 15° . For the IP compressor there were no changes. These data cover a range of operating speed lines from 0.5 to 1.00 of the design speed. Therefore, the LP compressor requires an individual stage control system and the IP compressor one control system for all stages.

5.2- Variable IGV and Stator Concepts

There are three ways of making variable IGV's and stators variable compressor operation:

1- Solid vanes rotated about a radial or nearly radial axis (Figure 5.1a). These have the restriction that the flow direction can be optimized at only one radial location.

2- Guide vanes or stators with variable camber along the span. This gives additional gains in compressor off-design performance because the air direction entering at all radii can more nearly be optimized. One possible concept of doing this is hinging the blade along a radial axis or possibly a canted axis (Figure 5.1b).

3- Another is mechanical twist or distortion of the vanes (Figure 5.1c).

The first was chosen in this work, since variable stagger can more easily be incorporated using the existing computer programme and is easier to employ in practice. The alternatives would need a special computer programme to deal with variable camber. In addition, the mechanical complexity and method of control must be weighed against the aerodynamic improvements. Such

considerations could make some of the blade types impractical.

5.3- Procedure

The variable stator blade technique has a great potential for achieving significant surge margin improvement with less mechanical and computational complexity. Rotor incidence in the main flow is closer to design values, which are chosen to achieve the best performance for each stage. Thus, compressor performance at off-design points is improved.

The procedure adopted herein was the same as that employed by Lihua and oth. (22). This obtains optimized restagger (ϕ_r) of each stator along each operating speed line. The optimized restagger is selected to give the highest efficiency. The restaggers of stators were executed by decreasing stator by stator as follows:

$$\phi_{r0} > \phi_{r1} > \phi_{r2} > \phi_{rn} \quad (5.1)$$

where the subscripts 0 stand for IGV and $1, 2$ and n stator numbers, respectively.

To perform the variable stator calculation, a short running time computer programme was required in order to obtain the compressor map at each restagger value.

The HOWELL computer programme was again selected to determine the schedule for variable geometry stators, due to its short running time and simple input data file requirements.

The HOWELL programme calculates overall compressor performance for various air mass flows and compressor speed relationship. Individual stage performance is also calculated as already mentioned in the previous Chapter.

During the investigations of the adjustable stators, the deviation angle of

each blade row was assumed independent of stagger angle.

For the convenience of comparison, the change of surge margin and peak efficiency (Reference 22) were introduced and defined as follow:

1- Surge Margin

The improvement in surge limit was calculated on basis of the relevant surge point of variable and fixed geometry stators as follow:

$$S = \left[\frac{PR_{set} W_F}{PR_F W_{set}} - 1 \right] 100\% \quad (5.2)$$

where PR and W are the compressor pressure ratio and non-dimensional mass flow, respectively. The subscript set and F are the values corresponding to variable and fixed geometry, respectively.

2- Peak Efficiency

$$\Delta \eta_{max} = \eta_{max(set)} - \eta_{max(F)} \quad (5.3)$$

At each setting angle, the rotor blade air inlet angle α_0 for the second and subsequent stages were calculated from the following (Reference 21):

$$\alpha_0 = 0.6 + \beta_4 \frac{(0.90(\frac{a}{c})^2 + 0.002\beta_4)\theta\sqrt{(\frac{s}{c})}}{1 - 0.002\theta\sqrt{(\frac{s}{c})}} \quad (5.4)$$

where β_4 , a/c , s/c , θ are the exit blade angle, position of maximum camber, space/chord ratio, and camber angle of the previous stator row at the current reference radius. The position of maximum camber, a/c is taken as 0.5 (see Users' Guide).

The optimization was first carried out in the LP mode and then the HP mode for LP and IP compressors. For the HP compressor, optimization was not necessary since the running lines for both modes were within the compressor operating range as mentioned earlier.

The run was made with the inlet guide vanes to facilitate matching of the first stage to the downstream stage, although originally, the compressors were designed without IGV's. The off-design settings were selected to match the first stage to the downstream stages for all stages at all operating speeds. Hence, the optimized restagger of each stator was made stage by stage for each operating speed line. The operating speed lines covered were in the range 0.5 to 1.00 relative to the design non-dimensional speed ($N/\sqrt{T}$). The calculations were executed on the above as follows.

5.3.1- LP Mode

5.3.1.1- LP Compressor

Variable IGV

Eight different values of restaggers (ϕ_r) were selected for each operating speed line, as shown in Table 5.1 (n is relative to the design rotational speed and PR_k is the pressure ratio related to the maximum efficiency at a constant operating speed line). The optimized restagger was obtained by maximizing compressor efficiency by resetting the restagger of the IGV at each operating speed line. The optimal values of restaggering were then taken from each operating speed line (Table 5.1) and placed in Table 5.4. The outcome of optimal values for variable IGV only (Table 5.4), calculated by the computer programme, is shown in Figure 5.3. This is the compressor map with variable IGV's. To compare the surge line displacement, Figure 5.2 (compressor map for fixed geometry) was used.

Operating speed line $n=0.90$								
η_{max}	85.55	85.92	85.87	85.49	84.81	83.86	82.56	81.02
PR_k	2.52	2.47	2.41	2.35	2.18	2.22	2.14	2.14
Operating speed line $n=0.80$								
η_{max}	85.18	85.92	86.12	85.93	85.41	84.54	83.44	80.97
PR_k	2.09	2.06	2.03	1.99	1.94	1.92	1.91	1.91
Operating speed line $n=0.70$								
η_{max}	83.78	84.74	85.25	85.47	85.35	84.86	83.94	82.52
PR_k	1.76	1.76	1.75	1.73	1.71	1.67	1.64	1.60
Operating speed line $n=0.60$								
η_{max}	80.77	82.16	83.04	83.52	83.67	83.30	82.45	81.02
PR_k	1.48	1.48	1.47	1.46	1.45	1.43	1.41	1.41
Operating speed line $n=0.50$								
η_{max}	75.43	77.16	78.60	77.71	80.19	80.27	80.23	79.78
PR_k	1.33	1.33	1.33	1.33	1.33	1.32	1.31	1.30

Table 5.1 LP Compressor Variable IGV (LP mode).

Comparing Figures 5.3 to 5.2, when the restagger of the IGV changes from 0° to the optimal values (Table 5.4), the surge line shifts towards upper left improving the surge margin. The surge pressure ratios, at each operating speed line, reduced slightly, the non-dimensional mass flow falls significantly and the efficiency lines at the lower operating speed lines, increased considerably.

The phenomenon may be explained as follow: when the restagger increase but it is close to the optimal value, the incidence of both rotor and stator approaches the design values in the main stream. Meanwhile the mass flow is reduced which leads to a reduction in the loss in the flow passage and the profile loss. The overall performance of the compressor is obviously improved.

As is seen in Figure 5.3, the surge margin has improved significantly but

not sufficiently. Thus, to obtain a desirable surge margin, a variable first stator was required.

Variable First Stator

Lihua's work investigated whether the existence of optimized restagger downstream would affect the optimized restagger upstream. His result revealed little or no effect. Accordingly, the settings for both first and subsequent stators were employed.

Using the same procedure employed for variable IGV's, the maximum efficiencies and corresponding pressure ratios were calculated by the computer programme within the range of restaggers (Table 5.2) for each operating speed line with the respective optimal restagger of the variable IGV. The optimal restaggers were chosen (Table 5.2) in a progressive way close to the optimal ones, giving maximum efficiency. This assumption does not affect the performance significantly. The result can be seen in Figure 5.4.

Figure 5.4 represents the compressor map for optimal restagger of the variable IGV and the first stator (Table 5.4). Again, comparing Figures 5.4 and 5.2, the surge line has shifted towards the upper left. The surge margin improvement is a little better than the variable IGV case (Figure 5.3). The efficiency lines (Figure 5.4) increase significantly at lower operating speed lines.

The LP mode running line (Figure 5.4) is now within the compressor operating range, but it has not met the requirements for S.B.E. to operate. As is well-known, engine acceleration, the running line moves up toward the surge line. In this case, for low operating speeds, the engine would need more surge margin to operate. Hence, a variable second stator was required.

Operating speed line $n=0.90$					
IGV=12°	ϕ_r	6°	8°	10°	12°
	η_{max}	87.11	87.21	87.22	87.16
	PR_k	2.41	2.39	2.38	2.36
Operating speed line $n=0.80$					
IGV=16°	ϕ_r	8°	10°	12°	14°
	η_{max}	87.92	87.99	87.98	87.89
	PR_k	1.99	1.98	1.97	1.95
Operating speed line $n=0.70$					
IGV=20°	ϕ_r	10°	12°	14°	16°
	η_{max}	87.91	88.00	88.04	88.03
	PR_k	1.71	1.72	1.71	1.61
Operating speed line $n=0.60$					
IGV=25°	ϕ_r	14°	16°	18°	20°
	η_{max}	86.79	86.90	86.93	86.89
	PR_k	1.43	1.43	1.43	1.43
Operating speed line $n=0.50$					
IGV=30°	ϕ_r	16°	18°	20°	22°
	η_{max}	85.27	86.47	85.57	85.53
	PR_k	1.32	1.32	1.32	1.32

Table 5.2 LP Compressor Variable 1 Stator (LP mode)

Variable Second Stator

Table 5.3 represents the restagger optimization carried out using the same procedure as for the variable IGV and first stator. The range of restaggering varies from one operating line to another to suit the optimization. The optimal restaggers, at each operating speed line, are given in Table 5.4. The outcome of that is shown in Figure 5.5.

n	Stator Group	ϕ_r	PR_s	$S(\%)$	$\eta_{max}(\%)$	$\Delta\eta(\%)$	$W_{sl} \times 10^{-2}$ (SI)
0.50	Original	0°	1.35	-----	73.53	-----	1.85
	IGV	30°	1.35	3.93	80.27	6.74	1.78
	First	$30^\circ+20^\circ$	1.36	7.72	85.57	12.04	1.73
	Second	$30^\circ+20^\circ+18^\circ$	1.36	13.64	87.55	14.02	1.64
0.60	Original	0°	1.55	-----	78.67	-----	2.39
	IGV	25°	1.55	3.91	83.67	5.00	2.30
	First	$25^\circ+18^\circ$	1.55	9.33	86.93	8.36	2.20
	Second	$25^\circ+18^\circ+16^\circ$	1.55	13.80	87.70	9.03	2.10
0.70	Original	0°	1.82	-----	82.15	-----	3.08
	IGV	20°	1.82	4.40	85.47	3.32	2.95
	First	$20^\circ+14^\circ$	1.82	10.00	88.03	5.88	2.80
	Second	$20^\circ+14^\circ+12^\circ$	1.82	14.49	89.17	7.02	2.69
0.80	Original	0°	2.19	-----	83.70	-----	3.86
	IGV	16°	2.17	3.09	86.11	2.41	3.71
	First	$16^\circ+12^\circ$	2.17	7.43	87.99	4.29	3.56
	Second	$16^\circ+12^\circ+10^\circ$	2.17	14.85	89.00	5.30	3.33
0.90	Original	0°	2.69	-----	84.78	-----	4.66
	IGV	12°	2.62	0.86	85.95	1.17	4.50
	First	$12^\circ+9^\circ$	2.59	2.67	87.22	2.44	4.37
	Second	$12^\circ+9^\circ+8^\circ$	2.63	9.78	88.16	3.38	4.15
1.00	Original	0°	3.35	-----	84.66	-----	5.47

Table 5.5 LP Compressor - Main Parameters Obtained at Each Operating Speed line (LP mode).

5.3.1.2- IP COMPRESSOR

Variable IGV

The settings for IP compressor were evaluated, firstly, in the same way as the LP compressor. Table 5.6 shows the restaggers, maximum efficiency and pressure ratio for each operating speeds. The results of optimized restagger (Table 5.6), at each operating speed, are shown in Table 5.7.

The output produced by the computer programme for optimal restaggers is given in Figure 5.8. Comparing Figure 5.8 with Figure 5.7, (fixed geometry), it can be seen that surge margin has not been changed. On the other hand, the

intermediate compressor with a single stage. Because the variable stators are modulated to reduce the flow into the HP compressor, the same procedure was employed for S.B.E.. The last stator was set at 40° and the result is presented in Figure 5.10.

Operating speed line $n=0.90$				
ϕ_r	3°	4°	5°	10°
η_{max}	90.56	90.60	90.63	90.62
PR_k	2.62	2.61	2.60	2.56
Operating speed line $n=0.80$				
ϕ_r	8°	10°	12°	15°
η_{max}	90.49	90.53	90.53	90.48
PR_k	2.14	2.13	2.11	2.10
Operating speed line $n=0.70$				
ϕ_r	20°	22°	24°	26°
η_{max}	89.82	89.86	89.85	89.82
PR_k	1.84	1.84	1.83	1.82
Operating speed line $n=0.60$				
ϕ_r	25°	26°	28°	30°
η_{max}	88.95	88.97	88.97	88.94
PR_k	1.56	1.56	1.55	1.55
Operating speed line $n=0.50$				
ϕ_r	28°	30°	32°	34°
η_{max}	87.66	87.66	87.63	87.60
PR_k	1.35	1.35	1.34	1.33

Table 5.6 IP Compressor Variable IGV (LP mode)

It can be seen, even though the LP mode running line had been reached, the IP compressor characteristics was much deteriorated. In Figure 5.10 can be seen that for medium and low speed lines, the compressor is of significantly low efficiency. These low efficiencies are most likely attributable to a radial redistribution of flow, which results from the upstream stages operating at off-design incidence angles and therefore, with higher losses.

Operating speed line $n=0.90$					
IGV=12° + 1Stator=9°	ϕ_r	4°	6°	8°	10°
	η_{max}	88.01	88.14	88.16	88.08
	PR_k	2.38	2.38	2.36	2.35
Operating speed line $n=0.80$					
IGV=16° + 1Stator=12°	ϕ_r	6°	8°	10°	12°
	η_{max}	88.95	88.97	88.88	88.69
	PR_k	1.96	1.95	1.94	1.93
Operating speed line $n=0.70$					
IGV=20° + 1Stator=14°	ϕ_r	8°	10°	12°	14°
	η_{max}	89.31	89.28	89.17	88.98
	PR_k	1.70	1.70	1.69	1.68
Operating speed line $n=0.60$					
IGV=25° + 1Stator=18°	ϕ_r	8°	10°	12°	14°
	η_{max}	88.00	88.06	88.02	87.90
	PR_k	1.42	1.41	1.41	1.41
Operating speed line $n=0.50$					
IGV=30° + 1Stator=20°	ϕ_r	10°	12°	14°	16°
	η_{max}	87.76	87.84	87.83	87.73
	PR_k	1.32	1.32	1.32	1.32

Table 5.3 LP Compressor Variable 2nd Stator (LP mode)

As can be seen in Figure 5.5, there is a significant surge margin improvement, in particular between $n=0.75$ and $n=0.80$, where the main problem lies. The change of surge margin, at $n=0.75$, calculated by equation 5.2 is about 14%. This is a considerable improvement for the engine operation. It can be seen in Figure 5.5, it shows that there is a significant improvement in efficiency for both medium and low speed zones compared with the compressor fixed geometry (Figure 5.2). For instance, when $n=0.50$ the maximum efficiency is enhanced

from 73% to 87%.

Once the LP mode running line is within the compressor operating range, further variable stators were not required. To provide the same compressor map as Figure 5.5, the compressor stators have to be restaggered, mechanically, according to the schedule in Table 5.4. The mechanical and control system complexity were not a matter for analysis in this work.

In Figure 5.6 can be seen that the IGV, at medium and low speeds, requires a larger amount of restaggering than downstream stages. Table 5.5 illustrates all the parameters obtained at each operating speed line for the LP compressor.

Stage	IGV	First	Second
n	ϕ_{r0}	ϕ_{r1}	ϕ_{r2}
1.00	0°	0°	0°
0.90	12°	9°	8°
0.80	16°	12°	10°
0.70	20°	14°	12°
0.60	25°	18°	16°
0.50	30°	20°	18°

Table 5.4 LP Compressor - Schedule of Variable Stators (LP mode)

An assessment on Table 5.5 reveals that at $n=0.80$, the maximum surge margin reaches about 14.85% with a peak efficiency increase of 5.30%. The maximum efficiency achieves 89% at $n=0.80$. This is above the design value. For lower operating speeds, the LP mode running line has fallen to a higher efficiency zone, thus improving engine overall performance.

Stage	IGV
n	ϕ_{ro}
1.00	0°
0.90	14°
0.80	28°
0.70	35°
0.60	35°
0.50	35°

Table 5.7 IP Compressor - Schedule of Variable IGV (LP node).

Because the upstream stages do not match the design conditions, it was decided to set the IGV and all stators with the same restagger angle (Table 5.8), in order to match all stages for the new conditions, even though they did not match with the previous compressor design.

Stator	IGV	1st	2nd	3rd	4th	5th	6th	7th
Attempt	ϕ_r	ϕ_r	ϕ_r	ϕ_r	ϕ_r	ϕ_r	ϕ_r	ϕ_r
1st	10°	10°	10°	10°	10°	10°	10°	10°
2nd	20°	20°	20°	20°	20°	20°	20°	20°
3rd	30°	30°	30°	30°	30°	30°	30°	30°
4th	40°	40°	40°	40°	40°	40°	40°	40°

Table 5.8 IP Compressor Variable Stators (LP mode)

Restaggering of each stator blade by the same amount, as if they had been redesigned for another operating condition, has rematched them all. Therefore, the surge problem has been significantly reduced.

The first setting chosen was 10° and the outcome is shown in Figure 5.11. As is seen, the characteristics lines shifted down and to the left. The surge margin increased 15.42% at $n=0.80$. Similarly, the efficiency lines have shifted

downwards and to the left. In spite of this efficiency deterioration, it is still within a satisfactory range for the compressor operating conditions.

As more restaggers were set, such as 20° , 30° and 40° for all stators (Table 5.8), the characteristics and efficiency lines shifted to down and to the left as shown in Figures 5.12, 5.13 and 5.14 for 20° , 30° and 40° , respectively. For 40° the LP mode running line is outside the IP compressor map.

Table 5.9 summarizes the main parameters obtained so far for IP compressor (LP mode). It shows that the surge margin may be expanded up to 64.50% at $n=0.50$ and the efficiency dropped by up to 10.32% at $n=0.80$, whilst this is not satisfactory, it is still considered acceptable.

The compressor characteristics of Figures 5.13 (IGV plus all stators at 30°) was chosen for the IP compressor in the LP mode since the LP mode running line was within the compressor operating range. Figures 5.15 represents an enlargement of Figures 5.13 in order to better display the IP compressor map in the LP mode.

The main parameters for the IP compressor are given in Table 5.9 (IGV plus all stators at 30°). The maximum surge margin achieves 51.56% at $n=0.50$ with the peak efficiency reduced by 4.43%. The maximum efficiency is 87.48% at $n=1.00$.

5.3.2- HP MODE -LP Compressor

As was stated before, the LP and IP compressor are at the same shaft. This means that at each off-design condition they must be at the same rotational speed. For instance, in the HP mode the running line starts at $n=1.00$ on the IP compressor map and on the LP compressor map about $n=0.83$. Therefore, the S.B.E. can not operate at this condition even though the running lines are within the compressor operating range.

n	Stator Group	ϕ_r	PR_s	$S(\%)$	$\eta_{max}(\%)$	$\Delta\eta(\%)$	$W_{sl} \times 10^2$ (SI)
0.50	Original	0°	1.40	-----	85.00	-----	0.670
	IGV	35°	1.38	1.87	87.66	2.66	0.653
	All stators	10°	1.35	17.04	84.15	-0.85	0.552
	All stators	20°	1.31	35.11	82.72	-2.28	0.464
	All stators	30°	1.27	51.56	80.57	-4.43	0.401
	All stators	40°	1.21	64.50	76.98	-8.02	0.352
0.60	Original	0°	1.62	-----	86.83	-----	0.857
	IGV	35°	1.60	1.12	88.97	2.14	0.837
	All stators	10°	2.54	21.77	85.29	-1.54	0.669
	All stators	20°	1.47	33.15	83.27	-3.56	0.584
	All stators	30°	1.40	48.12	80.96	-5.87	0.500
	All stators	40°	1.32	60.15	77.75	-9.08	0.436
0.70	Original	0°	1.92	-----	88.38	-----	1.07
	IGV	35°	1.88	-0.21	89.85	1.47	1.05
	All stators	10°	1.78	14.15	86.76	-1.62	0.869
	All stators	20°	1.68	30.00	85.53	-2.85	0.720
	All stators	30°	1.58	44.11	82.94	-5.44	0.611
	All stators	40°	1.45	52.75	78.71	-9.67	0.529
0.80	Original	0°	2.30	-----	90.16	-----	1.34
	IGV	28°	2.27	0.95	90.53	0.37	1.31
	All stators	10°	2.10	15.42	89.19	-0.97	1.06
	All stators	20°	1.95	29.69	87.51	-2.65	0.876
	All stators	30°	1.80	42.48	84.40	-5.76	0.736
	All stators	40°	1.62	49.57	79.84	-10.32	0.631
0.90	Original	0°	2.80	-----	90.66	-----	1.63
	IGV	14°	2.77	0.15	90.64	-0.02	1.61
	All stators	10°	2.51	13.26	90.41	-0.25	1.29
	All stators	20°	2.29	26.96	88.44	-2.27	1.05
	All stators	30°	2.08	38.06	86.19	-4.47	0.877
	All stators	40°	1.84	43.58	80.50	-10.16	0.746
1.00	Original	0°	3.41	-----	90.00	-----	1.99
	All stators	10°	3.03	14.08	90.74	0.74	1.55
	All stators	20°	2.72	26.98	89.89	-0.11	1.25
	All stators	30°	2.43	37.67	87.48	-2.25	1.03
	All stators	40°	2.10	40.37	80.42	-9.58	0.873

Table 5.9 IP Compressor - Main Parameters Obtained at Each Operating Speed (LP mode).

In an attempt to overcome this problem, the same procedure as used for the IP compressor in the LP mode was used for the LP compressor in the HP mode. Table 5.10 presents the restaggers for each individual stator at six different settings. Table 5.11 presents all main parameters obtained at each operating speed line in the LP compressor for these different settings.

Stator	IGV	1st	2nd	3rd	4th	5th
Attempt	ϕ_r	ϕ_r	ϕ_r	ϕ_r	ϕ_r	ϕ_r
1st	10°	10°	10°	10°	10°	10°
2nd	20°	10°	10°	10°	10°	10°
3rd	30°	10°	10°	10°	10°	10°
4th	30°	20°	10°	10°	10°	10°
5th	30°	15°	15°	15°	15°	15°
6th	35°	15°	15°	15°	15°	15°

Table 5.10 LP Compressor Variable Stators (HP mode).

The first setting was 10° for all the IGV and the stators. The result is presented in Figure 5.16. The characteristics lines have shifted downwards and to the left. The maximum surge margin increased by 17.10% at $n=0.70$. The efficiency lines have dropped slightly. The maximum reduction of 1.34% occurred at $n=0.60$ (Table 5.11). Further settings were necessary since the target had not been accomplished.

The next setting was 20° for IGV and 10° for the remaining stators. The result is given in Figure 5.17. The maximum surge margin, in this case, is improved by 19.01% at $n=0.70$. The maximum peak efficiency has risen by 3.29% at $n=1.00$ (Table 5.11). It was noticed that when the IGV setting was changed from 10° to 20°, the efficiency improved because better matching for the first stage was reached corresponding to reduced losses.

Further attempts were carried out with the IGV and first stage settings in

order to bring the characteristics lines of the LP compressor to the condition in which the HP mode could work. With IGV at 30° and all remaining stators at 10° , the characteristics lines displaced downwards and to the left. Correspondingly, the efficiency lines rose slightly as is be seen in Figure 5.18.

Figure 5.19 shows the results for IGV at 30° , the first stator at 20° and all remaining stators at 10° . Figure 5.20 represents the results for IGV at 30° and all remaining stators at 15° . In both Figures, there has been a displacement of the surge line downwards and to the left as well as an efficiency improvement. However, the problem still remained.

The final solution was achieved (Figure 5.21) when the IGV was set to 35° and all remaining stators at 15° . At this condition, the maximum surge margin position risen 31.19% at $n=0.70$ and maximum peak efficiency reached 4.07% at $n=0.50$ (Table 5.11). Figure 5.22 shows the compressor characteristic when the IP compressor is operating in the LP mode. Figures 5.23 and 5.24 show the final compressor characteristics for LP and IP compressors in the LP and HP modes, respectively. Thus, the LP compressor requires an individual stage control system and the IP compressor one control system for all stages.

n	Stator Group	ϕ_r	PR _s	S(%)	$\eta_{\max}$ (%)	$\Delta\eta$ (%)	W _{SL} x10 ⁻² (SI)
0.50	Original	0°	1.35	-----	73.53	-----	1.85
	All stators	10°	1.32	16.70	72.24	-1.29	1.55
	IGV + All St.	20°+10°	1.32	18.22	75.25	1.72	1.53
	IGV + All St.	30°+10°	1.32	19.79	77.55	4.02	1.51
	IGV+1St.+R' St.	30°+20°+10°	1.33	21.50	80.72	7.19	1.50
	IGV + All St.	30°+15°	1.31	28.22	76.60	3.07	1.40
	IGV + All St.	35°+15°	1.31	29.15	77.60	4.07	1.39
0.60	Original	0°	1.55	-----	78.67	-----	2.39
	All stators	10°	1.50	16.81	77.33	-1.34	1.98
	IGV + All St.	20°+10°	1.50	18.61	80.56	1.89	1.95
	IGV + All St.	30°+10°	1.50	20.46	82.72	4.05	1.92
	IGV+1St.+R' St.	30°+20°+10°	1.50	21.73	85.21	6.54	1.90
	IGV + All St.	30°+15°	1.48	28.93	81.06	2.39	1.77
	IGV + All St.	35°+15°	1.41	29.14	82.04	3.37	1.76
0.70	Original	0°	1.82	-----	82.15	-----	3.08
	All stators	10°	1.73	17.10	80.97	-1.18	2.50
	IGV + All St.	20°+10°	1.73	19.01	83.39	1.24	2.46
	IGV + All St.	30°+10°	1.73	21.48	84.92	2.83	2.41
	IGV+1St.+R' St.	30°+20°+10°	1.74	24.77	87.22	5.07	2.36
	IGV + All St.	30°+15°	1.70	30.17	83.76	1.61	2.21
	IGV + All St.	35°+15°	1.69	31.19	84.23	2.08	2.18
0.80	Original	0°	2.19	-----	83.70	-----	3.86
	All stators	10°	2.05	15.80	84.08	0.38	3.12
	IGV + All St.	20°+10°	2.05	18.08	86.19	2.49	3.06
	IGV + All St.	30°+10°	2.03	20.47	86.86	3.16	2.97
	IGV+1St.+R' St.	30°+20°+10°	2.04	24.84	88.55	4.85	2.88
	IGV + All St.	30°+15°	1.99	28.47	86.17	2.47	2.73
	IGV + All St.	35°+15°	1.98	30.70	86.44	2.74	2.67
0.90	Original	0°	2.69	-----	84.78	-----	4.66
	All stators	10°	2.49	12.33	85.97	1.19	3.84
	IGV + All St.	20°+10°	2.47	15.96	87.51	2.73	3.69
	IGV + All St.	30°+10°	2.40	14.53	87.56	2.78	3.63
	IGV+1St.+R' St.	30°+20°+10°	2.41	19.28	88.96	4.18	3.50
	IGV + All St.	30°+15°	2.36	23.14	87.37	2.59	3.32
	IGV + All St.	35°+15°	2.33	22.68	87.35	2.57	3.29
1.00	Original	0°	3.35	-----	84.66	-----	5.47
	All stators	10°	3.06	10.00	87.25	2.59	4.54
	IGV + All St.	20°+10°	2.98	10.58	87.95	3.29	4.40
	IGV + All St.	30°+10°	2.85	9.75	87.38	2.72	4.24
	IGV+1St.+R' St.	30°+20°+10°	2.81	10.82	88.29	3.63	4.14
	IGV + All St.	30°+15°	2.78	14.05	87.95	3.29	3.98
	IGV + All St.	35°+15°	2.75	15.13	87.59	2.93	3.90

Table 5.11 LP Compressor - Main Parameters Obtained at Each Operating Speed (HP mode). (* Remaining)

CHAPTER 6

Turbine Design and Performance

6.1- Introduction

The design and off-design performance of the HP and LP turbines has been executed by a computer programme package. An existing programme has been modified and another two, developed by the author. The design and off-design performance have been assessed on the basis of a mean-diameter treatment.

The design of the turbines has yielded a single stage HP turbine and a 4 stage LP turbine. The HP turbine design isentropic efficiency, predicted by the TURBDES computer programme, was 88%, 2% below the target efficiency of 90%. The LP turbine design isentropic efficiency, predicted by the same computer programme, was 90%, the same as the target value.

The off-design performance calculation has been successfully accomplished by the TURBOFF computer programme and the HP and LP turbine characteristics are demonstrated.

6.2- Design Objectives

The design objectives are given by the cycle specifications. These usually include the turbine entry temperature and pressure, turbine outlet pressure, turbine power required (and/or the gas mass flow) the shaft speed, and the fuel-

air ratio.

To achieve these requirement, degrees of freedom are:

- 1- Number of stages,*
- 2- Work split (multistage turbine),*
- 3- Annulus area,*
- 4- Annulus shape (constant tip diameter, constant mean diameter or constant hub diameter),*
- 5- Stage reaction,*
- 6- Type of vortex flow,*
- 7- Radial distribution of work and flow,*
- 8- NGV and rotor space/chord ratio and aspect ratio and*
- 9- NGV and rotor profile design.*

Further degrees of freedom are constrained for mechanical integrity reasons, such as:

- 1- Blade centrifugal stress,*
- 2- Disc centrifugal stress,*
- 3- Maximum diameter for installation,*
- 4- NGV and blade cooling and*
- 5- Overall weight.*

For the present design, Table 6.1 gives the turbine inlet parameters, taken from the TURBOMATCH output, for both HP and LP turbines. The fuel-air ratios for both turbines were found from Figure 6.1 (Reference 15). The value of fuel-air ratio for the HP turbine was obtained by interpolation. The input data for the graph in Figure 6.1 are: combustion inlet temperature and combustion temperature rise.

PARAMETERS	TURBINE	
	HP	LP
INLET TOTAL PRESSURE (N/m ²)	28.27x10 ⁵	10.54x10 ⁵
INLET TOTAL TEMPERATURE (K)	1800	1210.76
INLET MASS FLOW (kg/s)	152.73	160.20
FUEL-AIR RATIO	0.0280	0.0207
PRESSURE RATIO	2.328	4.365
TEMPERATURE DROP (K)	277.00	327.35
ROTATIONAL SPEED (RPM)	10000	4450
ISENT. EFFICIENCY (%)	90	90

Table 6.1 Design Objectives

6.3- Computer Programmes

Despite the existence of sophisticated computer programmes, preliminary pitch line design calculations remain an important starting point.

It is the aim of this work to undertake only preliminary turbine aerodynamic design and simple blade stress analysis. NGV and blade cooling considerations were not considered in detail despite the effect on turbine and therefore, engine overall efficiency. A fully detailed turbine design involves many interactions and design experience. The ultimate prediction will be more accurate when additional losses introduced by the cooling system are taken into account (Reference 15). In this work the aim was to obtain approximate turbine characteristics to assess the engine overall performance.

To carry out the design two computer programmes were used. The first called TURBGEO written by the author, (a pitch line method) provides the rapid determination of design velocity diagrams for turbine stages. It also facilitates the

selection of the number of turbine stages and the selection of optimum work split among the stages for any given application. In addition, TURBGEO calculates annulus geometry as well as all mean aerodynamic parameters for preliminary assessment. Moreover, output data for the second programme called TURBDES is given directly.

The second, called TURBDES (formerly called CEMTURB) written by Borthwick (23) and improved by Dener (24) is a free vortex design programme approach. This allows for three-dimensional effects. It also estimates design point efficiency and undertakes a simple blade stress analysis.

The turbine off-design performance was carried out by TURBOFF, written by the author. It also is a pitch line axial-flow turbine off-design performance prediction method. More details about TURBOFF are given in section 6.8 and Appendix E.

Further information about TURBGEO and TURBDES can also be found in Appendix E. An explanation is given to how to make the input files as well as the sequence in which each one has to be run.

The selection of the input and the assessment of the output parameters for the design procedure were relied on the information given below (basically References 15, 25 & 26). A laborious description of the design parameter selections is not included, however, the output is discussed.

6.4- Implications of Design Selection Parameters

Blade Loading Coefficient ψ

Blade loading Coefficient is defined as:

$$\Psi = \frac{\Delta H}{U^2} = \frac{\bar{c}_p \Delta T}{U^2} \quad (6.1)$$

where

ΔH =enthalpy drop,

$\bar{c}_p$ =mean specific heat at constant pressure,

ΔT =total temperature and

U =blade speed.

In general, turbines are designed to achieve efficiencies over 90%. Smith (27) has concluded that the turbine efficiency depends primarily on the shape of the velocity triangles. Smith's efficiency correlation (Figure 6.2) shows that high efficiency is obtained when stage loadings and to a lesser extent flow coefficients are low. All of the test turbine efficiency values shown in this chart are corrected to exclude tip clearance losses. Figure 6.2 indicates that the most efficient turbines are designed for ψ between 1.0 and 2.5. If a low value of ψ is used then more stages are required for a given overall turbine output. For an industrial gas turbine, a low value of ψ is adopted since size and weight are not so important but low SFC is vital. Jet engines, however, must be of low weight and small frontal area. Accordingly, higher values of ψ are adopted.

All the test data collected by Smith (Figure 6.2) were taken from 69 different Rolls Royce cold flow model turbines, all having approximately 50% reaction at mean diameter.

Flow coefficient ϕ

Flow coefficient is defined as a ratio V_a/U where V_a is the axial velocity and U is the blade speed. Figure 6.2 also indicates the range of values of ϕ for best turbine efficiency lies between 0.6 and 0.8. For industrial and aircraft gas turbines, ϕ follows the same trends as ψ stated above.

The required mean blade speed U_m may be calculated by the following equation knowing the enthalpy drop and blade loading coefficient (Reference 25).

$$U_m = \sqrt{\left(\frac{\Delta H}{\psi}\right)} \quad (6.2)$$

where

$$\Delta H = \bar{c}_p \Delta T \quad (6.3)$$

Degree of Reaction

Degree of reaction represents the fraction of the stage expansion which occurs in the rotor. Normally the turbines are designed for reaction of 50% at blade mid height. For the last stage, exit swirl angle less than 20 degrees is necessary to minimise losses in the jet pipe and propelling nozzle. In this circumstance, it might be necessary to use a degree of reaction somewhat less than 50% if a high value of ψ is also required.

Negative value of reaction must be avoided since this would produce expansion in the nozzle followed by recompression in the rotor increasing the losses.

Number of stages

The number of stages is determined primarily from considerations of performance at the design point. Other factors have also to be considered such as: engine cost and weight, engine geometry and the number of blade rows needing to be cooled.

The calculation of the number of stages depends on the choice of work split between turbine stages. Performance cycle data merely stipulates the overall

work required from the turbine on each shaft. The initial choice of work split is based on experience. Table 6.2 gives some directions on how to split the work according to the number of stages and Table 6.3 gives the ranges of stage loading for different types of turbine as well as range of exit Mach number.

Number of stages	Work split %		
	Stage 1	Stage 2	Stage 3
2	60	40	-
3	43	34	23

Table 6.2 Work Split per stage

Type of turbine	Range of exit Mach number	Range of average stage loading
Transonic HP turbine	1.0 - 1.2	2.0 - 2.4
Subsonic, high Mach number HP and IP turbines	0.7 - 1.0	1.5 - 2.0
Subsonic, low Mach number IP and LP turbines	< 0.7	1.5 - 2.5

Table 6.3 Exit Mach Number and Stage Loading

There are some points to be allowed for during the work split selection, such as:

- 1- if each stage falls on the Smith chart,
- 2- if the stages are well matched aerodynamically, in particular with good accelerations being achieved over all blade rows,
- 3- if the exit swirl and Mach number are excessive for the following turbine or exhaust system.
- 4- if the first stage in two-stage HP turbine is so loaded-up that it

becomes transonic.

The choice of work split also affects the cooling requirements. If a work split of 60% and 40% is assumed for a two stage HP turbine instead of 50% and 50%, the extra work on the first stage results in both rotor blades being cooler probably to the extent that the second blade may not need to be cooled.

From the mechanical point of view, the work splits have to be selected in order to provide enough camber and stiffness in the blading.

Hub tip ratio or hub-shroud ratio

Correct choice of hub-tip ratio in the first and the last stage is important. Good practice is to select a hub-tip ratio for the high-pressure first stage not above 0.875, to maintain low tip clearance annulus losses. The hub-tip ratio of the last (low-pressure) stage should not be below 0.6, for stress considerations.

Figure 6.3 (Reference 25) is an alternative method for selecting hub-tip ratio. It is the result of a calculation of hub-tip ratio as a function of design-point pressure ratio.

Space-chord ratio (s/c)

Optimum values of space-chord ratio with respect to profile losses only are given in Figure 6.4 (Reference 15). This is for a conventional T6 profile having a thickness-chord ratio of 0.2 and a trailing edge thickness-pitch ratio of 0.02. The optimum value of the space-chord ratio in the NGV and rotor blade rows can also be found using Zweifel's criterion. Zweifel's formula at blade mid height is given by:

$$C_L = 2(s/C_{ax}) \cos^2 \alpha_2 (\tan \alpha_1 + \tan \alpha_2) \quad (6.4)$$

where station 1 and 2 are inlet and outlet of the blade row, s is the pitch and c_{ax} is axial chord, then:

$$C_{L\ opt} = \frac{0.8}{\sin^2 \alpha_2} \quad (6.5)$$

Once the space-chord is determined the following should be checked:

- 1- that the pitch (s) is not so small that the blades can not be attached safely to the turbine disc rim.
- 2- that there is sufficient chord (c) for blade cooling.
- 3- that the number of the blades is not more than 100.

The number of the blades with common multiples should be avoided in order to minimise the probability of vibration. The usual practice is to adopt an even number for the nozzle blades and a prime number for the rotor blades.

Aspect ratio (h/c)

Low values of aspect ratio are likely to increase secondary loss for the nozzle row and secondary and tip clearance loss for the rotor row. High values of aspect ratio will, conversely, increase the probability of vibration problems and reduce heat transfer. Vibration problems caused by high aspect ratio blading can be significantly reduced using tip shrouds. Typical limits for aspect ratio are $1 < AR < 5.5$, where AR for the NGV is often lower than that for rotor, because of cooling requirements

Exit swirl angle

The swirl angle should be kept below 20 degrees to minimise losses in the jet pipe and propelling nozzle.

6.5- Design AssessmentMach number relative to the rotor blade

A check should be made at the rotor root where the Mach number is highest and should not exceed about 0.75, because additional losses may be incurred by formation of shock waves in the rotor blade passages.

Negative value of degree of reaction

A check should be made at the root. Higher values of swirl angle at blade mid height would increase the degree of reaction at the root.

Pressure ratio across the convergent nozzle

This must not be above the critical value otherwise the nozzle will be choked. However, values slightly above the critical value can be accepted if the rotor inlet Mach number proves to be satisfactory.

Annulus flare angle

Figure 6.5 shows the annulus flare angle θ and it is usually limited as follows:

$$(\theta_F / 2)_{tip} < 25 \text{ degree}$$

$$(\theta_F / 2)_{root} < 20 \text{ degree}$$

$$\text{included angle } \theta_F < 30 \text{ degree.}$$

Higher values of flare angle than these will involve a risk of flow separation from the inner wall where the reaction is low and therefore, the acceleration is not large. If annulus walls are not straight, sharp corners should be avoided. Annulus flare angle also depends on the magnitude of axial chords and axial space between the blade rows.

Stage outlet Mach number

This should be in the range given in Table 6.3 to avoid high friction losses in the jet pipe.

Nozzle or blade exit angle

Nozzle or blade exit angle should not exceed 70 degrees. Large angles lead to long profiles which are difficult to cool and worsen the effect of manufacturing tolerances on throat area.

Velocity ratio

Velocity ratio across NGV tip section and rotor root section should be:

$V_2/V_1 > 1.2$ for first NGV and

$V_2/V_1 > 1.1$ for subsequent blade rows.

6.6- Main Guidelines for Turbine Design Sequence

The guidelines are summarized as follows:

1- The high-pressure turbine should be a single-stage if possible by specifying the blade loading coefficient of up to 1.8. However, the blade loading coefficient in a low-pressure stage, for turbine overall pressure ratio above 8:1, is limited by the value at the hub. Perhaps a better rule

for preliminary design would then be to have a limiting value of hub blade loading coefficient up to 2.0 for the high-pressure turbine up to 4.0 for the low-pressure turbine.

2- The use of 50 per cent reaction at the mean diameter in the high-pressure stages gives best efficiency. The corresponding values can be found in Figures 6.6 and 6.7. These curves should be used in the initial design phase when the number of stages and therefore, the blade loading coefficient has been selected.

Note:

a)- Figures 6.6a to c give optimum values of flow coefficient and stage exit flow angle, for a selected values of blade loading coefficient and reaction. The full lines, which are of total-to-total efficiency, are suitable for all stages (except the last) of multistage turbines. The dotted lines, for total-to-static efficiency across the blading, are suitable for single-stage turbines without an exit diffuser. Finally, the chain dotted lines are suitable for a single-stage turbine followed by a diffuser with 50 per cent pressure recovery.

b)- Figure 6.7a to c represent the cross correlations obtained from Figure 6.6. Figure 6.7a shows peak isentropic efficiency plotted against blade loading coefficient for different reactions. Figure 6.7b and c are for the effects of blade loading coefficient on rotor outlet swirl and optimum nozzle angle.

3- The hub-tip ratios (Figure 6.3) should be specified between a maximum of 0.875 and minimum of 0.6 according to pressure ratio. However, if the pressure ratio is below 3 the flow path configuration and inlet and outlet hub-tip ratio may be chosen freely.

Note:

a)- As stated before, Figure 6.3 helps to select the hub-tip ratios as a

function of pressure ratio.

b)- It is advisable to start the design with the low-pressure turbine at a higher mean diameter and a consequently high hub-tip ratio. High-pressure turbine design follows. The transition duct is placed between turbines.

6.7- HP and LP Turbine Design

The constraints stated in the previous sections are but not a unique choice. They are the ranges in which preliminary turbine design is undertaken.

The turbine designs herein were based on the following assumptions:

- 1- Constant axial velocity for TURBGEO.*
- 2- About 50% reaction at blade mid height.*
- 3- Free vortex flow distribution.*
- 4- HP turbine inlet axial Mach number of about 0.3.*
- 5- Straight sided annulus walls.*
- 6- Shrouded blades are selected, for HP and LP turbines, since they feature inter-locking shrouds which both stiffen the blades and help to damp vibration. On the other hand, unshrouded blades are more likely to vibrate and therefore, the chords and adjacent gaps are larger than those corresponding shrouded equivalents. Nevertheless, unshrouded blades are also popular.*

The HP and LP turbine design has been carried out with the help of the TURBGEO and TURBDES computer programmes. They were used for both aerodynamic and mechanical design. First, TURBGEO was used to select the number of stages and a suitable annulus geometry. When an acceptable compromise was reached, TURBDES was then applied to refine losses, aerodynamic parameters and stress analysis. The blade stress calculation

performed by TURBDES is a simple approximate method adequate for preliminary design.

The assessment and analysis of HP and LP turbine design are carried out in the following sections.

6.7.1- HP Turbine

After many trials, a single stage HP turbine was selected, even though a two stage would be more appropriated from an aerodynamic point view. The choice of a single stage was made on the basis of the fact that a two stage turbine would increase the length, weight and cost of the engine. However, a single stage turbine would generate higher losses in the swan-neck duct between HP and LP turbine due to high exit swirl angle resulting from a high blade loading coefficient.

The selected design parameters are given in Table 6.4. They have been chosen within the ranges stated previously so that the mechanical conditions may to a large extent, be satisfied. The input data file for TURBGEO is given in Figure F.10, Appendix F. A constant hub radius design has been selected taking into account the LP turbine geometry after estimating the location of the combustion chamber. Finally, the main design output parameters, such as Mach numbers, deflection, hub-tip ratio, etc, are given in Table 6.5. Full TURBGEO and TURBDES output data can be found in Figures F.11 and F.12, Appendix F, respectively.

As a result of design calculation the stage isentropic efficiency (η) was found as follows:

In Smith's efficiency correlation (Figure 6.2), with the design values of $\psi=1.83$ and $\phi=0.57$, the efficiency was found to be $\eta=91\%$. Allowing for, say, -3%

tip clearance it became 88%. The efficiency, predicted by TURBDES, was the same. The efficiency predicted did not reach the target efficiency of 90%. It could be improved by using a sophisticated computational programme and carrying out further interactions. Furthermore, a more detailed assessment of the mechanical design may well change, for instance, the number of blades, aspect ratios, space-chord ratios, and annulus geometry. In consequence, these change would affect the efficiency.

HP TURBINE	
PARAMETER (at blade mid height)	STAGE
Inlet axial velocity (m/s)	240.0
Temperature drop (K)	277.
Flow coefficient ϕ	0.57
Blade loading coefficient ψ	1.83
Inlet air angle (degree)	0.0
Inlet total pressure $\times 10^5$ (N/m ²)	28.27
Inlet total temperature (K)	1800.
Design type	CHD
Nozzle/rotor aspect ratio	1.2/2.0
Nozzle/rotor space-chord ratio	1.0/0.95
Outlet stage tip radius (m)	0.4453
Blade mean speed (m/s)	419.14
Annulus flare angle (degree)	19.68
Stage length (cm)	12.04

Table 6.4 Design Parameters

Note: CHD = Constant Hub Diameter

HP TURBINE	
PARAMETER (at blade mid height)	STAGE
Nozzle/rotor deflections (degree)	66.8/100.7
Nozzle/rotor Abs. exit gas angle (degree)	66.8/37.
Relative inlet gas angle (degree)	32.98
Relative exit gas angle (degree)	67.70
Exit swirl angle (degrees)	37.0
Nozzle/rotor accelerations V_{out}/V_{in}	2.58/2.22
Rotor inlet Mach number (root)	0.37
Nozzle exit Mach number	0.78
Exit axial Mach number	0.41
Root/tip reaction	44/60
Inlet hub/tip ratio	0.8750
Outlet hub/tip ratio	0.7975
Stage isentropic efficiency %	88
Overall isentropic efficiency %	88

Table 6.5 Turbine Design Assessment

6.7.2- LP Turbine

A four stage LP turbine with constant mean radius has been chosen. The input data file for TURBGEO is given in Figure F.13, Appendix F. The number of stages could not be reduced from 4 to 3 for the following reasons: Firstly, because in order to make the LP compressor operates efficiently, their tip speeds, especially in the first stages, have to be limited. As a consequence, the LP shaft speed became low compared to the HP shaft. Thus, the LP turbine has more stages than the HP turbine. To reduce the number of stages of LP turbine

would be necessary to push up the blade mid height radius. This would increase the losses in the swan-neck duct between the two turbines and the length of the engine. Secondly, a limit on exit tip radius of 0.65 m was imposed by the by-pass duct in the HP mode and the low exit tip radius of HP the turbine.

The design mean parameters selected for the LP turbine are given in Table 6.6. Table 6.7 shows the LP turbine assessment. For the last stage, the exit swirl angle is below 20 degrees which ensures low losses in the jet pipe. Full TURBGEO and TURBDES output data are given in Figures F.14 and F.15, Appendix F, respectively.

LP turbine isentropic efficiency was found in the same way as for HP turbine, namely using both TURBDES and Smith's correlation. For all stages, the predictions from both were close enough that only one method need be used.

stage 1	$\psi=1.60$
	$\phi=0.86$
	$\eta=90\%$
stage 2	$\psi=1.53$
	$\phi=0.86$
	$\eta=89\%$
stage 3	$\psi=1.50$
	$\phi=0.86$
	$\eta=90\%$
stage 4	$\psi=1.46$
	$\phi=0.86$
	$\eta=91\%$

The overall isentropic efficiency by TURBDES was 90%. This is the target

efficiency.

After the HP and LP turbine design had been carried out, a final geometry involving both turbine annulus and swan-neck duct was drawn in order to check the overall dimensions. The drawing is given in Figure 6.8.

The swan-neck duct is a form of inter-turbine connecting duct which provides to some extent an outward step change in radius and diffusion of the main-stream flow. Excessive diffusion of gas flow within turbomachinery blading should be avoided. For this reason, the swan-neck duct with slight diffusion (with increasing flow area) was placed between the HP and LP turbines in order to slow-down the main-stream flow.

The next step was to estimated the turbine off-design performance using the TURBOFF programme.

6.8- Turbines Off-design Performance

The off-design performance prediction of HP and LP turbines have been carried out using TURBOFF. As mentioned before, it was written by the author and the calculations were based on Ainley and Mathieson's method (28) involving correlations of cascade data. Some new correlations of cascade data obtained by Dunhan and Came (29) have been introduced to replace some of those originally used by Ainley and Mathieson. In addition, TURBOFF was especially designed to handle multistage turbines.

Basically, the performance of a turbine, as calculated by TURBOFF, is assumed to be the same as that of a turbine of constant section blading, the geometry of which is identical with that at the mean section of the real turbine. The assumption of "mean section" calculations is sufficiently accurate for the purpose of this preliminary design assessment.

LP TURBINE				
PARAMETER (at blade mid height)	STAGE			
	1st	2nd	3rd	4th
<i>Inlet axial velocity (m/s)</i>	205	205	205	205
<i>Temperature drop (K)</i>	85.00	82.00	81.00	79.35
<i>Flow coefficient ϕ</i>	0.86	0.86	0.86	0.86
<i>Blade loading coefficient ψ</i>	1.60	1.53	1.50	1.46
<i>Inlet air angle (degree)</i>	0.0	21.0	21.0	21.0
<i>Inlet total pressure $\times 10^5$ (N/m²)</i>	10.54	7.73	5.62	4.01
<i>Inlet total temperature (K)</i>	1210.76	1125.76	1043.76	962.76
<i>Design type</i>	CMD	CMD	CMD	CMD
<i>Nozzle/rotor aspect ratio</i>	2.00/ 2.29	2.38/ 2.99	3.07/ 3.57	3.20/ 3.60
<i>Nozzle/rotor space-chord ratio</i>	1.10/ 0.95	1.11/ 1.10	1.11/ 1.00	1.12/ 1.00
<i>Outlet stage tip radius (m)</i>	0.5677	0.5826	0.6030	0.6311
<i>Blade mean speed (m/s)</i>	239	239	239	239
<i>Annulus flare angle (degree)</i>	14.23	14.02	14.82	19.78
<i>Stage Length (cm)</i>	12.58	16.10	20.02	17.59

Table 6.6 Design Parameters

Note: CMD = Constant Mean Diameter

In Appendix E, the method due to Ainley and Mathieson method and TURBOFF are presented in detail. The input files for the HP and LP turbines (Figures F.16 and F.17, respectively) and the output file for the HP turbine (Figure F.18) are in Appendix F. However, the output file for the LP turbine has not been included due to its length.

LP TURBINE				
PARAMETER (at blade mid height)	STAGE			
	1st	2nd	3rd	4th
Nozzle/rotor deflections (degrees)	56.1/ 74.9	75.2/ 70.4	74.8/ 68.5	74.7/ 67.7
Nozzle/rotor Abs. exit gas angle (degrees)	56.2/ 21.0	54.5/ 21.0	53.7/ 21.0	53.7/ 19.0
Relative inlet gas angle (degrees)	17.84	13.33	11.36	11.7
Relative exit gas angle (degrees)	57.14	57.06	57.10	56.66
Exit swirl angle (degrees)	21.00	21.00	21.00	19.00
Nozzle/rotor accelerations V_{out}/V_{in}	1.79/ 1.76	1.61/ 1.80	1.58/ 1.81	1.57/ 1.77
Nozzle exit Mach number	0.55	0.55	0.56	0.58
Rotor inlet Mach number (root)	0.32	0.33	0.34	0.35
Exit axial Mach number	0.34	0.35	0.36	0.36
Root/tip reaction	42/ 60	44/ 65	41/ 68	31/ 69
Inlet hub/tip ratio	0.8500	0.8119	0.7654	0.7055
Outlet hub/tip ratio	0.8119	0.7654	0.7055	0.6297
Stage isentropic efficiency %	90	89	90	91
Overall isentropic efficiency %	90			

Table 6.7 Turbine Design Assessment

The turbine off-design maps for both HP and LP turbines obtained with TURBOFF are presented in the following sections.

6.9- HP Turbine Characteristics

The characteristics of the HP turbine are shown in Figure 6.9. This shows non-dimensional mass flow ($m\sqrt{T}/P$) and efficiency versus pressure ratio for various values of design non-dimensional speed ($N/\sqrt{T}$). The design point conditions are included to compare the efficiency predicted by TURBOFF to that by TURBDES. The efficiency, predicted by TURBOFF, has been found to be 88%. This is the same as that predicted by TURBDES.

The form of the characteristic lines presented in Figure 6.9 is for choked nozzles. As can be seen, the constant speed lines merge into a single horizontal line. The reason for this is that the choking conditions do not depend on the rotational speed.

6.10- LP Turbine Characteristics

Figure 6.10 is the LP turbine characteristic. As with the HP turbine, the non-dimensional mass flow ($m\sqrt{T}/P$) and efficiency are plotted against pressure ratio for various values of design non-dimensional speed ($N/\sqrt{T}$). The design point conditions are also shown in Figure 6.10 in order to compare the efficiency predicted by TURBOFF with that for TURBDES. The efficiency, predicted by TURBOFF, has been found to be 90%. This is the same as that predicted by TURBDES. Unlike Figure 6.9, Figure 6.10 has characteristic for choked rotor passages. This means that, the choking condition depends on the rotational speed.

CHAPTER 7

Selective Bleed Engine Simulation

7.1- TURBOMATCH Preparation

This section explains the second simulation of the Selective Bleed Engine; employing, in TURBOMATCH, the characteristics obtained in Chapter 4, 5, 6. This allows the simulation of the Selective Bleed Engine with the estimated characteristics in order to assess its overall performance and to ascertain the compressors can perform the duty demanded. The replacement of the characteristics was carried out separately i.e. first the compressor characteristics and then turbine characteristics.

A TURBOMATCH input file was prepared with the same design inlet conditions as in Chapter 2 for each mode except for the compressors BRICK data and compressor characteristics. The replacement of compressor characteristics for each mode followed the instructions given in Reference 30. In each compressor BRICK data Z (surge margin parameter), CN (non-dimensional speed relative to design), PR (pressure ratio) and ETA (isentropic efficiency) were replaced by the appropriate design values as follows:

for LP mode

LP Compressor	IP Compressor	HP Compressor
Z=0.869	Z=0.986	Z=0.866
PR=3.0	PR=2.41	PR=3.60
CN=0.99	CN=0.99	CN=0.96
ETA=0.86	ETA=0.875	ETA=0.875

for HP mode

LP Compressor	IP Compressor	HP Compressor
Z=0.687	Z=0.95	Z=0.896
PR=2.1	PR=3.31	PR=3.96
CN=0.95	CN=0.95	CN=0.99
ETA=0.875	ETA=0.89	ETA=0.875

All above values were found manually in the estimated compressor characteristics with the design pressure ratios and non-dimensional mass flows calculated in the first simulation (Chapter 2).

The first run of TURBOMATCH with the predicted compressor maps failed. It was then decided to change the LP compressor characteristics in the HP mode because the HP mode design point CN in the estimated LP compressor characteristics was found to be about 0.91 and in the estimated IP compressor characteristics about 0.97. As is known, the LP and IP compressor are on the same shaft and therefore, the gap between the two CNs had to be reduced by varying further the LP compressor stator blades. The LP compressor stator blades were restaggered as follows: IGV 35° and all stators 20°. The result of it is shown in Figure 7.1. With the change, the design point CN in the LP compressor characteristics shifts from 0.91 to 0.95, which is much closer to that of the IP compressor. To make CN of the IP compressor the same as the LP compressor, a multiplier (0.97/0.95) was applied to the IP compressor. With the

new estimated LP compressor characteristics (Figure 7.1) in TURBOMATCH, the second run it was also unsuccessful.

The problem was in the IP compressor characteristics. For some reason TURBOMATCH did not give scale factors for pressure ratio, mass flow and efficiency equal to unity; would indicate that the maps were not being scaled. What could be happening in TURBOMATCH is that because of the shape of the characteristic of the IP compressor near the surge line, the TURBOMATCH interpolation procedure of the compressor characteristic was not working as desired and therefore, giving scale factors different from unity. Some tests carried out on the TURBOMATCH ARITHY brick data indicated that it was not locating the same CN in IP compressor characteristics as that in LP compressor in the second simulation. Because time was running out at that point, it was decided to replace the IP compressor characteristics by the TURBOMATCH compressor characteristics and to carry out the performance analysis on the LP and HP compressors only.

The second step was to replace the TURBOMATCH turbine characteristics by the estimated turbine characteristics obtained in Chapter 6. However, this was not possible because TURBOMATCH can only handle turbines with choked rotor. As is known, the HP turbine has choked nozzle and the LP turbine choked rotor. Although the LP turbine characteristics could be used by TURBOMATCH, it was decided not to replace the TURBOMATCH turbine characteristics by the estimated characteristics and carry on the off-design performance only with the predicted compressor characteristics, this being the most critical component.

7.2- Off-design Performance

The off-design performance of the Selective Bleed Engine was simulated with TURBOMATCH as in Chapter 2, with the estimated compressor characteristics as stated in the previous section. The off-design performance analysis was carried out for Thrust and SFC, with variation of TET and Flight

Mach Number. Figures 7.2 and 7.3 show the change of thrust and SFC as throttle setting changes for the first (Chapter 2) and second simulation. In these cases throttle setting is represented by TET. It can be observed, as in all gas turbine engines, that there is a very large change of thrust with TET. The HP mode in the second simulation achieves higher levels of thrust than that originally predicted and the TET operating range is reduced slightly for both modes. Figure 7.3 shows the change of SFC with TET. The influence of TET on SFC, in both operating modes, is similar to any other jet engine. The SFC is slightly greater for LP mode in the second than in the first simulation and slightly lower for HP mode in the second than in first simulation. Figure 7.4 shows the change of SFC with thrust. Again the curves are similar to that in Figure 7.3.

7.3- Compressor Running Lines

The steady state running lines derived from the second engine off-design performance simulation for the mission profile in Table 2.2 (Chapter 2), are shown in Figure 7.5 (LP mode), Figures 7.6 (HP mode) for the LP compressor, Figure 7.7 (both modes) for the HP compressor and Figure 7.8 (LP mode), Figure 7.9 (HP mode) for the IP compressor, in order to see how the compressor running lines fit within the estimated characteristics.

Comparing the LP compressor running lines (Figure 7.5) in the LP mode, there is no significant change in the LP compressor characteristics. However, the running line in the second simulation has more surge margin for CN between 0.98 and 0.80 than in the first simulation. This is because the estimated compressor efficiency in the second simulation is higher than that of the compressor characteristics in TURBOMATCH. Keeping the same engine main conditions but increasing the compressor efficiencies will result in lower off-design pressure ratios, because of the modifications in engine matching.

In Figure 7.6, it can be seen that there is no significant change in the trajectory of the running line in the second simulation. The surge margin is almost

the same in both simulations. Looking at the Figures 7.7, the LP mode running line does not change significantly, while for the HP mode running line changes considerably. The LP mode running line in the second simulation changes slope with the surge margin falling as speed falls. The design point surge margin remains almost the same as in the first simulation. The reason for that could be the same as that explained in Figure 7.5 for LP mode in the second simulation.

Figure 7.8 shows the IP compressor LP mode running lines. Comparing the running lines, the second simulation running line is shorter and it has a different trajectory. This could be because the LP compressor rotational speed is not compatible with that of the IP compressor and therefore, the LP compressor forces the IP compressor pressure ratio and non-dimensional mass flow towards the surge. To overcome this problem the variable LP compressor vanes could be used to control the LP compressor rotational speed.

Figure 7.9 shows the HP mode running lines for the IP compressor. In both simulations, the trajectory of HP mode running lines have the same direction. However, HP mode running line in the second simulation is much shorter than that in the second simulation.

The fuel consumption required to accomplish the mission profile outlined in Table 2.2 (Chapter 2) for the second simulation was calculated. It was found that the Selective Bleed Engine requires 4001kg of fuel, this being approximately 10 per cent less than that of the first simulation. The reduction in fuel consumption is caused by the higher efficiency of the compressors at off-design, when compared to TURBOMATCH. The supersonic cruise (Mach No=1.5 & Alt=9km), which is the most important phase of the mission, is accomplished without reheat and the SFC drops from 33.15 [mg/N sec] (first simulation) to 31.44 [mg/N sec] (second simulation).

A more accurate analysis of the running lines in both modes and of fuel consumption should be made. Before this is possible the analysis of several

problems, such as scale factors, interpolation procedures and TURBOMATCH ARITHY brick data has to be carried out.

CHAPTER 8

Engineering Considerations

8.1- Introduction

The aim of this investigation is to assess the performance of the Selective Bleed Engine. This preliminary investigation shows the subject warrants further investigation. This investigation should focus on the refining of the performance assessment and should now be broadened to include the engineering and the construction considerations. In this Chapter some key topics are outlined. These topics are valves, transition phase, sizing and optimisation.

8.2- Other Design Aspects

There are several other engine features that require very careful consideration because they are critical to the satisfactory operation of the Selective Bleed Engine.

8.2.1- Valves

Two sets of valves will be required by the engine. One will shut off the LP mode bypass duct (or vectoring nozzles) and the other will shut off the HP mode bypass duct. These valves will have to be light and effective. For the HP bypass duct, a system such as a vane valve could be employed. It would be lightweight and would require the same technology employed in variable stator design. For

the LP bypass duct the solution does not appear to be so straightforward because of the relatively large area that has to be blocked. A possibility is again a vane valve that would block the cold nozzle inlet.

8.2.2- Transition Phase

The transition from LP to HP modes of operation will require special attention to the transient performance of the engine. When this transition takes place, the shaft RPMs may have to change, and this change in RPM will be one of the factors that will determine the time required for the transition. The valves will have to be coordinated very carefully. As one of them is opening, the other one is closing. This change of area at the compressor outlet will have to be coordinated in such a way that there will be an adequate safety margin (the compressor outlet area is not too small) and that the right level of compressor pressure ratio will be retained (the compressor outlet area is not too large), thus retaining engine performance.

8.2.3- Sizing and Optimisation

The engine has been sized for the HP mode of operation, and some compromises, such as accepting a low TET and a degree of oversizing, had to be made in the LP mode. Because the LP compressor is designed with the conditions in the LP mode, the HP mode is also oversized. Similarly, the number of stages, and the overall diameter of the engines could be reduced if the latest turbomachinery design codes, which are only available to the major aero-engine manufacturers, could be employed. The tools available at Cranfield are powerful enough to demonstrate the feasibility of the concept, but cannot compete with the most modern design procedures available only to engine manufacturers. Further optimisation, that could yield some improvement in engine performance and characteristics is still necessary. It is also necessary to carry out an analysis of variable geometry in the LP turbine to assess the potential impact on overall engine performance.

A variable geometry turbine can be used to improve engine performance and its advantages and disadvantages depend on overall engine objective and applications. For instance, some applications show that LP turbine variable geometry is effective in increasing the LP compressor surge margin at all rotor speeds. On the other hand, it has shown an additional engine complexity and manufacturing cost.

Therefore, based on the information in the open literature, it is likely that the overall engine and compressor performance for the Selective Bleed Engine will improve by employing a variable LP Turbine.

The analysis of such an engine, the Selective Bleed Engine, has been carried out. However, the transition from one mode to the other, a very important engine handling feature, and the transition performance will not be investigated here.

8.2.4- Mechanical Design

The mechanical design is a very important phase which needs to be assessed due to the complexity of the variable cycle concepts. Some of the performance gained may have to be compromised because of mechanical considerations.

CHAPTER 9

Remarks and Conclusions

9.1- Conclusions

The study of a variable cycle engine has been carried out. This variable cycle engine operates at subsonic speeds as a medium bypass turbofan and at supersonic speeds and combat conditions as a low bypass ratio turbofan.

The main topics investigated have been the overall performance of the engine, the matching method and the behaviour of the compressors. The overall performance of the engine is predicted to be significantly better than that of a conventional reheated turbofan. One of the products of this work is a design method for variable cycle jet engines. Some of the penalties, such as additional weight and handling problems have still to be assessed.

A cycle matching method has been developed. It has been presented as a mixed numerical-graphical method, but it can be easily adapted to a fully numerical method for digital computers.

The main advantage of this method is that it starts with engine parameters which have been selected from cycle optimisation procedures. Thus it is known from the beginning of the study that the cycles on which the engine is going to operate have been optimised. The procedure is a check of the capability of the components to satisfy the requirements of the two cycles.

The method is based on thermodynamic relations and the characteristics of the various components. This method has been employed for the analysis of two other variable cycle engines the Tandem Fan (References 31) and the Variable Cycle Turbojet Turbofan (Reference 32) and it has proved to be a very powerful analytical tool.

The method as described treats fixed geometry turbines. The inclusion of turbine variable geometry is very simple. The steps of turbine matching can be easily modified to allow for differences in turbine non-dimensional mass flow, these differences reflecting the variability of turbine nozzle area.

The variable compressor geometry is widely used to improve off-design performance of highly loaded multistage axial flow compressor with high pressure ratio. This is the first time such a technique has been used in a low pressure ratio multistage axial flow compressor to displace the surge line in order to allow sufficient surge margin under all operating conditions. The results obtained indicate that compressor variable geometry, a well mastered technology, is essential for satisfactory engine operation. The use of variable compressor geometry has expanded the surge margin up to 65 per cent at non-dimensional rotational speed relative to design equal to 0.50 (Table 5.9, Chapter 5).

Turbine variable geometry, currently a high-risk technology, can be avoided although it will improve the flexibility of engine operation, and will improve the matching of the two cycles. The full potential of LP turbine variable geometry needs to be assessed.

The study described in this work indicates that the Selective Bleed Engine appears to yield important benefits in performance. For instance, the Selective Bleed Engine saves 337kg of fuel compared to the standard Turbofan to accomplish the mission outlined in Table 2.2 (Chapter 2). This is approximately 9 per cent of total fuel consumption. In the second simulation (Chapter 7), the Selective Bleed Engine saves up to 776kg of fuel. This is approximately 18 per

cent of total fuel consumption calculated in the first simulation (Chapter 2). However, the main advantage of the Selective Bleed Engine is at Mach number 1.5 (supersonic cruise) where it is approximately 50 per cent more efficient than standard Turbofan (Figure 2.17, Chapter 2).

The modifications carried out in the existing computer programmes allow better assessment of the aerodynamic parameters and faster running time during the component design. Therefore, more time may be spent on overall engine performance analysis. The turbine off-design performance prediction computer programme (TURBOFF) is simple and can quickly predict the turbine characteristics. A comparison of TURBOFF results with experimental data is necessary in order to verify its empirical correlations in test cases for the programme.

The TURBOMATCH computer programme is indeed a very powerful tool to simulate engine design point and off-design performance. However, the scale factors which correct the default compressor and turbine characteristics, still do so when replaced by predicted characteristics. This affects the accuracy of the engine off-design performance simulation. A modification in TURBOMATCH in order to deactivate (when desirable) the scale factors could be done and it would be very useful for further work.

Some of the difficulties anticipated need scrutiny, but the performance gains justify further analysis of the concept.

9.2- Contributions

The main contribution of this work was the assessment of the new variable cycle jet engine concept. This has shown that the Selective Bleed Engine is a feasible design offering a significant benefit in performance.

An original contribution was the development of the matching method

procedure to match variable cycle jet engines, which can be extended to three shaft jet engines and with or without variable nozzle LP turbines.

The successful use of variable compressor geometry to displace the surge line in low pressure ratio compressor was another significant contribution, especially for variable cycle engines.

Finally, equally important, was the development of an axial-flow turbine off-design performance programme and the modifications and update of the existing computer programmes.

This work is the genesis of a preliminary performance analysis procedure for variable cycle jet engines.

9.3- Further Work

The present work shows that the Selective Bleed Engine is a feasible design. A more detailed assessment should encompass the following work:

- 1- Analysis of the transient performance and the transition from one mode to the other.*
- 2- Assessment of engine overall performance with LP turbine variable geometry.*
- 3- Further development of the matching method for three-shaft engines and variable nozzle LP turbines.*
- 4- Mechanical design of components and assessment of the Selective Bleed Engine.*
- 5- Development of a computer programme specially designed to simulate*

the design point and off-design performance analysis of the Selective Bleed Engine.

REFERENCES

- 1 *PILIDIS, P.
ADKINS, R.C.* *Alternative engine system for ASTOVL aircraft. Cranfield Institute of Technology, May 1975.*
- 2 *GAMMON, N.* *Supersonic STOVL fighter engine design project, selective bleed cycle study, HP turbine, intake. M.Sc. Thesis, Cranfield Institute of Technology, 1988.*
- 3 *ARMSTRONG, F.W.
HIGTON, D.R.* *Some aspects of variable cycle propulsion systems. AGARD CP 205, September 1976.*
- 4 *FACEY, J.R.
GLASER, F.C.* *Variable cycle engines for V/STOL fighters. AGARD CP 205, September 1976.*
- 5 *WILLS, E.* *Variable cycle engines for supersonic cruise aircraft. AGARD CP 205, September 1976.*
- 6 *BOXER, E.
MORRIS, S.J.
FOSS, W.E.* *Assessment of variable cycle engines for supersonic transports. AGARD CP 205, September 1976.*
- 7 *HATTON, N.G.
SWANN, B.* *Use of engine variables to improve military performance. AGARD CP 205, September 1976.*
- 8 *PAYZER, R.J.* *Variable cycle engine applications and constraints. AGARD CP 205, September 1976.*

- 9 ALLAN, R.D. *General electric company variable cycle engine technology demonstrator programs.*
AIAA No 78-1047, July 1978.

- 10 BROWN, R.H. *Integration of a variable cycle engine concept in a supersonic cruise aircraft.*
AIAA No 78-1049, July 1978

- 11 MACMILLAN, W.L. *Development of a modular type computer programme for the calculation of gas turbine off-design performance.*
Ph.D. Thesis, Cranfield Institute of Technology, September 1974.

- 12 PALMER, J.R. *The TURBOMATCH scheme for gas turbine performance calculations - Users' guide.*
Cranfield Institute of technology, October 1983.

- ✓ 13 BUCKNELL, R.L. *STOVL Engine/Airframe Integration.*
AIAA No 87-1711, June 1987.

- 14 WILLIAMSON, C. M. *Supersonic STOVL fighter engine design project - Tandem Fan cycle study, LP compressor and rear nozzle design.*
M.Sc. Thesis, Cranfield Institute of Technology, September 1988.

- 15 COHEN, H.
 ROGERS, G. F. C.
 SARAVANAMUTTOO, H.I.H. *Gas turbine theory.*
Longman, Scientific and Technical, third edition, 1987.

- 16 HELMANN, W.
 STARKEN, H.
 WEYER, H. *Cascade wind tunnel tests on blades designed for transonic compressors.*
AGARD CP 34, 1968.

- 17 LAKSHMINARAYANA, B. *Methods of predicting the tip clearance effect in axial flow turbomachinery.*
Trans ASME, Journal of Basic Engineering Vol.92, 467-82, 1970.

- 18 SEROVY, G. K. *Compressor and turbine performance prediction system development - lessons from thirty years of history.*
AGARD Lecture Series 83, ch 3, 1976.

- 19 WALL, R. A. *Axial flow compressor performance prediction.*
AGARD Lecture Series 83, ch 4, 1976.

- / 20 HOWELL, A. R.
CALVERT, W. J. *A new stage stacking technique for axial flow compressor performance prediction.*
Trans ASME Journal of Engineering for Power, vol.100, 698-703, 1978.

- X 21 CALVERT, W. J. *Users' guide to the Howell multistage compressor performance prediction program.*
N.G.T.E. M77023, December 1977.

- 22 LIHUA, W.
FUSHENG, H.
FENG, L.
YANLI, W. *Investigation and experiment on optimizing angles of variable stator of axial flow compressor.*
ASME 85-IGT-32, September 1985.

- 23 BORTHWICK, D. *The influence of vortex flows on turbine aerodynamic design.*
M.Sc. Thesis, Cranfield Institute of Technology, October 1975.

- 24 DENER, C. *European fighter aircraft engine design - HP compressor and HP turbine.*
M.Sc. Thesis, Cranfield Institute of Technology, September 1985.

- 25 WILSON, D. G. *New guideline for the preliminary design and performance prediction of axial flow turbines.*
Prpc Inst Mech Engrs, vol 201, No A4, 1987.

- 26 HORLOCK, J. H. *Axial flow turbine, fluid mechanics and thermodynamics.*
Robert E. Krieger Publishing Company, 1973.

- 27 SMITH, S. F. *A simple correction of turbine efficiency. Jour of Roy Aero Soc, vol. 63, July 1965.*

- 28 AINLEY, D. G.
MATHIESON, C. R. *A method of performance estimation for axial-flow turbine. N.G.T.E. R 2974, December, 1951.*

- 29 DUNHAN, J.
CAME, P. M. *Improvement to the Ainley Mathieson method of turbine performance prediction. Transactions of the ASME, July 1970, pp 252-256.*

- 30 HOUT, F. V. D. *Gas turbine performance simulation improvement to the TURBOMATCH scheme. M.Sc. Thesis, Cranfield Institute of technology, September 1991.*

- 31 KONTOS, D. *SST engine design, engine cycle study. M.Sc. Thesis, Cranfield Institute of Technology, 1991*

- 32 MAHMOOD, S. *SST engine design, Tandem fan engine. M.Sc. Thesis, Cranfield Institute of Technology, 1991*

- 33 LOCKET, R. D. *Supersonic variable by-pass engine. M.Sc. Thesis, Cranfield Institute of technology, 1976.*

- 34 ANDREWS, S. J. *Tests related to the effect of profile shape and camber line on compressor cascade performance. NGTE Report No. 60, October 1949.*

- 35 WRIGHT, L. C. *Blade selection for a modern axial flow compressor. In Fluid Mechanics, Acoustics and Design of Turbomachinery, Part II. NASA SP 304, 1970.*

- 36 OLDHAM, R. K. *Some design data for double circular arc compressor blading. NGTE Note No. NT 589, November 1965.*

- 37 LIEBLEIN, S. *Incidence and deviation-angle correlations for compressor cascades. Trans. ASME Journal of Basic Engineering, vol. 82, p. 575-87, 1960.*

- 38 LIEBLEIN, S. *Experimental flow in 2D cascades. Chapter VI of The Aerodynamic Design of Axial Flow Compressor. Reprinted NASA SP 36 in 1965, 1956.*

- 39 LIEBLEIN, S.
 et al. *Diffusion factor for estimating losses and limiting blade loadings in axial-flow-compressor blade elements. NACA RM E53D01, 1953.*

- 40 KOCH, C. C.
 SMITH, L. H. *Loss sources and magnitudes in axial-flow compressors. Trans. ASME Journal of Engineering for Power, p.411-24, July 1976.*

- 41 MILLS, A. F.
 XU HANG *On the skin friction coefficient for a full rough flat plate. ASME Journal of Fluids Engineering. Vol. 105, p. 364-65, September 1983.*

- 42 LIEBLEIN, S. *Loss and stall analysis of compressor cascade. Trans. ASME, Series D, Journal of Basic Engineering, Vol. 81, p.387, 1959.*

- 43 HIRSCH, CH.
 DENTON, J. D. *Through flow calculations in axial turbomachinaries. AGARD AR 175, October 1981.*

- 44 MEMBERS OF THE
 COMPR. AND TURBINE
 RES. DIV. (Revised) *Aerodynamic design of axial flow compressors. NASA SP 36, 1965.*

- 45 RAIT, J. W. *Powerplant for a VTOL airliner lift engine - fan and compressor design. M.Sc. Thesis, Cranfield Institute of Technology, 1970.*

- 46 MACKENZIE, A. B. *Axial compressor design lecture notes. Cranfield Institute of Technology, 1990-91.*

- 47 BARTHWICK, D. *The influence of vortex flow on turbine aerodynamic design.*
M.Sc. Thesis, Cranfield Institute of technology, 1975.
- 48 PALMER, J.
RAMSDEN, K. *Compressible flow tables for engineers.*
Macmillan Education, 1987.

APPENDIX A

This Appendix consists of aircraft mission profile and thrust requirements taken from Gammon's thesis (2).

Aircraft Mission Profile and Thrust Requirements

The air superiority design mission comprises:

- 1- Warm-up, taxi (5 minutes at ground idle).
- 2- STO (200m roll, dry concrete).
- 3- Minimum fuel climb to best cruise altitude and Mach number (BCAM).
- 4- Cruise at BCAM to point 200 nm from take off.
- 5- Instantaneous transfer to 9km and loiter for 60 minutes at best loiter speed.
- 6- Accelerate at maximum power to Mach 1.5 and best cruise altitude and hold this Mach number to a point 100nm from the loiter area.
- 7- Supersonic combat (fire 2 AMRAAM then one 360 degree, 3g sustained turn at $M=1.5$, $Alt=9km$).
- 8- Subsonic combat (two 360 degree, 8g sustained turns at $M=0.9$, sea level, fire AMRAAM, 2 ASRAAM, expend ammunition).
- 9- Minimum fuel climb to BCAM.
- 10- Return to base (300nm from subsonic combat) at BCAM.
- 11- One minute hover out of ground effect, vertical landing with 5 percent internal fuel remaining at entry hover.

The offensive support fall out mission comprises:

- 1- Warm-up, taxi (5 minutes at ground idle), take off climb to 100m.
- 2- Outbound cruise at 100m, best cruise Mach number.
- 3- Penetration for 60nm at 100m, $M=0.8$.
- 4- Drop 6 anti-tank cluster weapons.
- 5- Combat (five 360 degree turns at 8g, sea level, $M=0.9$).
- 6- Egress for 60nm at 100m, $M=0.8$.
- 7- Return cruise at 100m, best cruise Mach number.
- 8- One minute hover out of ground effect, vertical landing with 5 percent internal fuel remaining at entry hover.

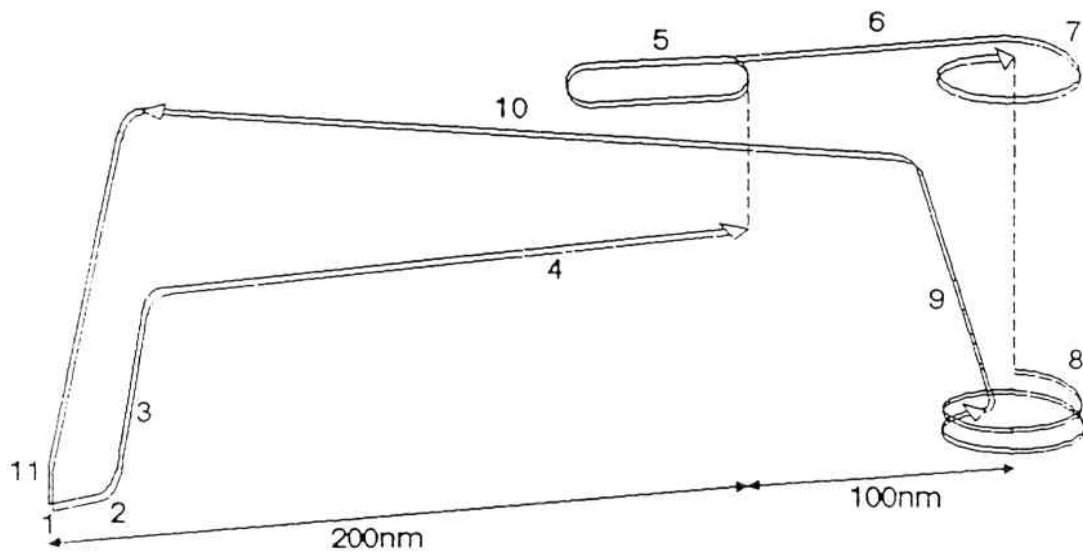
The following mass definitions should be borne in mind whilst the mission is being studied:

- i) Design basic mass empty shall include any fixed store carriers, all lubricating oil, anti-icing and refrigerating liquids, unusable fuel, avionics equipment and chaff and flare dispensers.
- ii) Design take-off mass is designed as basic mass empty, plus full internal fuel, any water injection fluid, crew (90kg), role-fit equipment such as jettisonable stores pylons, launchers and ejector release units, 4 AMRAAM, 2 ASRAAM, gun, ammunition, chaff and flares.
- iii) Combat mass is defined as the design take-off mass less 40 percent of usable fuel, any injection water consumed during take-off and 2 AMRAAM.
- iv) Design landing mass is defined as the design take-off mass less 80 percent of usable fuel and any injection water consumed during take-off.

The following performance requirements must be met by the aircraft at combat mass under ISA conditions:

- 1- *Sustained turn rates of :*
 16 deg s⁻¹ at sea level, M=0.5, max dry power
 11 deg s⁻¹ at sea level, M=0.9, max dry power
 12 deg s⁻¹ at 6000m, M=0.9, max dry power
 8 deg s⁻¹ at 6000m, M=1.2, max dry power.
- 2- *Attained (maximum instantaneous) turn rate of 22 deg s⁻¹ at 3000m, M=0.55, max power.*
- 3- *Specific excess power of :*
 150 m/s at 9000m, M=1.4, max power
 300 m/s at sea level, M=0.9, max power
 190 m/s at 9000m, M=0.9, max power.
- 4- *Mach numbers of : M=1.1 at sea level, M=1.6 at 11000m.*

Figure A.1 shows the air superiority design mission selected. In Figure A.2, the point performance thrust requirements are shown graphicly. Finally, Table A.1 shows the required engine point performance data.



1. Warm – up, taxi (5 minutes at ground idle)
2. STO (600ft roll, dry concrete)
3. Minimum fuel climb to best cruise altitude and Mach number (BCAM)
4. Cruise at BCAM to a point 200nm from take – off
5. Instantaneous transfer to 9km and loiter for 60 mins at best loiter speed
6. Accelerate at maximum power to $M = 1.5$ and best cruise altitude (BCA) and hold to a point 100nm from loiter area
7. Supersonic combat (fire 2 AMRAAM then one 360 degree, 3g sustained turn @ $M = 1.5$, 9km)
8. Subsonic combat (Two 360 degree, 8g sustained turns @ $M = 0.9$, SL, fire 2 AMRAAM, 2ASRAAM, expend ammunition)
9. Minimum fuel climb to BCAM
10. Return to base (300nm from subsonic combat) at BCAM
11. One minute hover out – of – ground effect, vertical landing with 5% internal fuel remaining at entry to hover

Figure A.1 Air Superiority Design Mission.

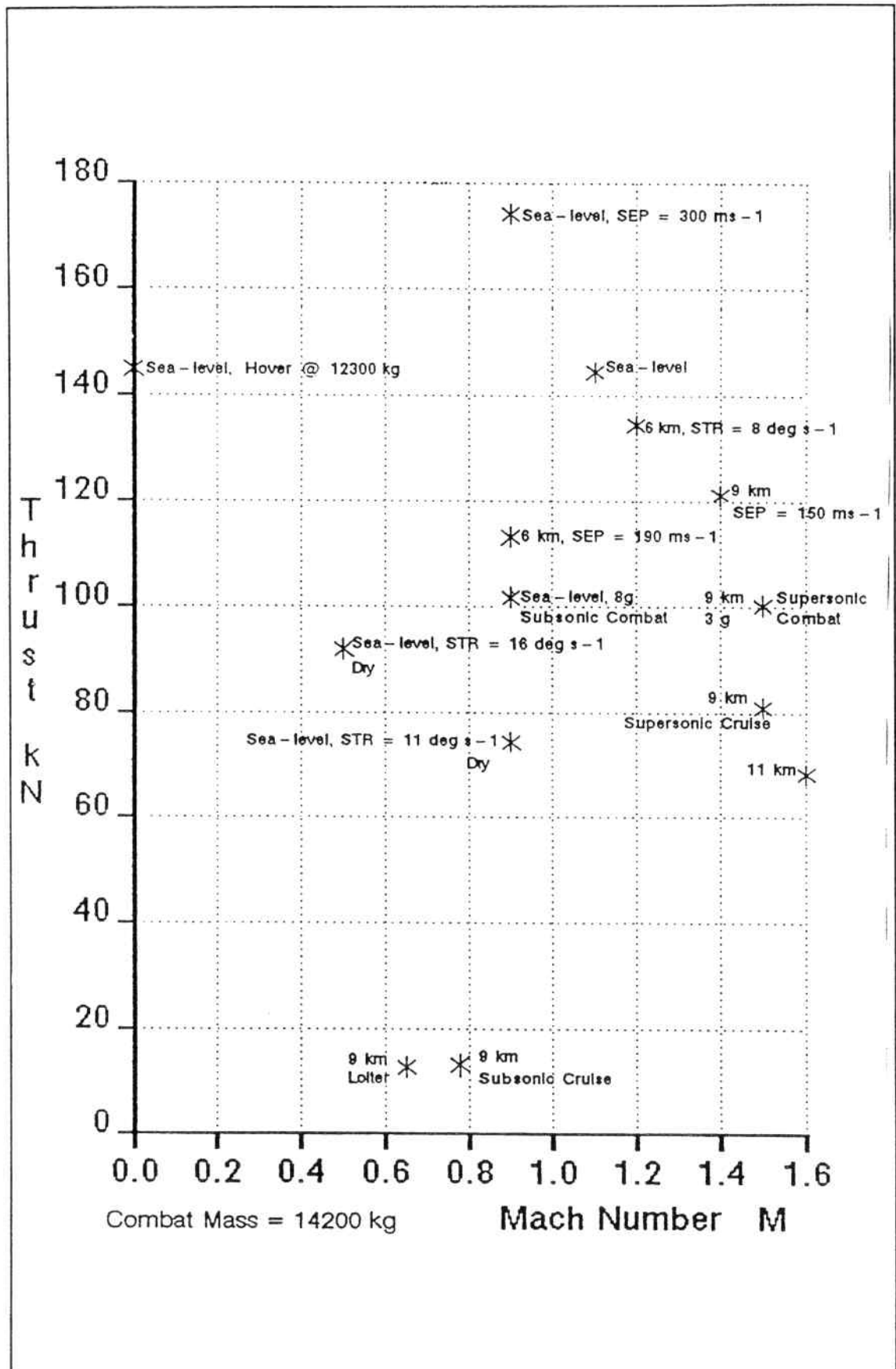


Figure A.2 Point Performance Thrust Requirements.

ALTITUDE (km)	MACH NUMBER	g LOADING	SPECIFIC EXCESS POWER (ms^{-1})	SUSTAINED TURN RATE (deg s^{-1})	q (kPa)	VELOCITY (ms^{-1})	C_L	C_{D_z}	C_{D_i}	C_D	D (kN)	REQUIRED THRUST (kN)
9.0	1.50	3.00	0.0	0.0	48.4	455.7	0.178	0.0335	0.0091	0.0427	100.2	100.2
0.0	0.90	8.00	0.0	0.0	57.5	306.3	0.399	0.0133	0.0233	0.0366	102.0	102.0
0.0	0.50	4.95	0.0	16.0	17.7	170.1	0.799	0.0138	0.0934	0.1071	92.1	92.1
0.0	0.90	6.08	0.0	11.0	57.5	306.3	0.303	0.0133	0.0134	0.0268	74.5	74.5
6.0	0.90	6.16	0.0	12.0	26.7	284.8	0.660	0.0141	0.0637	0.0778	100.9	100.9
6.0	1.20	5.50	0.0	8.0	47.5	379.7	0.331	0.0351	0.0232	0.0583	134.4	134.4
9.0	1.40	1.00	150.0	0.0	42.2	425.3	0.068	0.0341	0.0012	0.0354	72.3	121.3
9.0	1.50	1.00	0.0	0.0	48.4	455.7	0.059	0.0335	0.0010	0.0346	81.1	81.1
0.0	0.90	1.00	300.0	0.0	57.5	306.3	0.050	0.0133	0.0004	0.0137	38.1	174.3
6.0	0.90	1.00	190.0	0.0	26.7	284.8	0.107	0.0141	0.0017	0.0158	20.5	113.2
0.0	1.10	1.00	0.0	0.0	85.8	374.3	0.033	0.0345	0.0002	0.0347	144.5	144.5
11.0	1.6	1.00	0.0	0.0	40.5	472.1	0.071	0.0332	0.0016	0.0348	68.4	68.4

Table A.1 Required Engine Point Performance Data.

MODE	MACH NUMBER	q (kPa)	VELOCITY (ms ⁻¹)	C _L	C _{D_z}	C _{D_i}	C _D	D (kN)	REQUIRED THRUST (kN)
LOITER	0.65	9.1	197.4	0.316	0.0145	0.0146	0.0291	12.8	12.8
RANGE	0.78	13.0	236.1	0.221	0.0142	0.0071	0.0213	13.4	13.4

Table A.1 Required Engine Point Performance Data.

APPENDIX B

S. B. E. Design and Off-design Data

This Appendix consists of TURBOMATCH input and output files for LP and HP mode. These files are for engine optimization and design and off-design performance.

Figures B.1 and B.2 represent the TURBOMATCH input files for LP and HP mode created to carry out the engine optimization and illustrate the use of Loop required during the optimization process.

Figures B.3 and B.4 are the TURBOMATCH output files for LP and HP mode with the design and off-design parameters as well as the engine input data at the begin.

Figure B.1 TURBOMATCH Input Data File for LP mode.

```

-----
*** SELECTIVE BLEED ENGINE - LP MODE ***
*** OPTIMIZATION ***
-----
////

DP,SI,KE,FP
-1
-1
INTAKE S1-2 D1-4 R100
COMPRES2-3 D5-10 R101 V5 V6
PREMAS S3,17,4 D11-14 V11
DUCTER S17-18 D15-18 R103
NOZCON S18,19,1 D19 R104
ARITHY D200-204
COMPRES4-5 D20-25 R102 V20
COMPRES5,6 D26-31 R109 V26 V27
ARITHY D220-224
PREMAS S6,20,7 D32-35
COMPRES7,8 D36-41 R105 V36
ARITHY D210-216
ARITHY D230-236
PREMAS S8,21,9 D42-45
PREMAS S9,22,10 D46-49
BURNER S10-11 D50-52 R106
TURBIN S11-12 D53-60,111 V54
MIXEES S12,22,13
TURBIN S13-14 D61-68,110 V62
DUCTER S14-15 D69-72 R107
NOZCON S15,16,1 D73 R108
ARITHY D261-267
ARITHY D271-277
ARITHY D281-287
ARITHY D301-307
ARITHY D311-317
ARITHY D321-327
ARITHY D340-344
OUTPUT D7,11,22,400
OUTPUT D328,288
PLOTIT D328,288,7,11,22,400
PERFOR S1,0,0 D74-77,104,100,103,108,0,106,0,0,107
CODEND

```

DATA ITEM	LP MODE
DESIGN POINT AT S.L.S.	
1 0.0	Altitude
2 0.0	INTAKE ISA Deviation
3 0.0	Mach Number
4 -1.0	Pressure Recovery (default)
5 -1.0	Z Surge margin (default)
6 -1.0	PCN Corrected RPM (default)
1001 7 1 1 2.0	COMPRES Pressure Ratio PR1 (loop)
8 0.88	Isentropic Efficiency
9 0.0	Error switch (off)
10 1.0	Map Number
1002 11 2 2 0.7 0.75	Mass flow factor(BY-PASS FLOW)
12 0.0	PREMAS Flow loss
13 1.0	Pressure factor
14 0.0	Pressure loss
15 1.00	Reheat switch(off)
16 0.03	DUCTOR Pressure factor
17 0.9	Combustion efficiency
18 1000000.0	Dummy limiting fuel flow
19 -1.0	NOZCON Switch(Fixed exit area)
20 -1.0	Z Surge margin (default)
21 -1.0	PCN Corrected RPM (default)
1003 22 3 2 9. 10.0	COMPRES Pressure Ratio PR2 (loop)
23 0.88	Isentropic Efficiency
24 0.0	Error switch (off)
25 1.0	Map Number

```

26 -1.0      !      Z Surge margin (default)
27 -1.0      !      PCN Corrected RPM (default)
28 1.2       ! COMPRE Pressure Ratio PR3
29 0.89      !      Isentropic Efficiency
30 0.0       !      Error switch (off)
31 1.0       !      Map Number

32 0.0       !      Mass flow factor(Surge Control)
33 0.0       ! PREMAS Flow loss
34 1.0       !      Pressure factor
35 0.0       !      Pressure loss

36 -1.0      !      Z Surge margin (default)
37 -1.0      !      PCN Corrected RPM (default)
38 3.0       ! COMPRE Pressure Ratio PR3
39 0.89      !      Isentropic Efficiency
40 1.0       !      Error switch (off)
41 1.0       !      Map Number

42 0.05      !      Mass flow factor(Reaction Control)
43 0.0       ! PREMAS Flow loss
44 1.0       !      Pressure factor
45 0.0       !      Pressure loss

46 0.10      !      Mass flow factor(HPT Cooling Air)
47 0.0       ! PREMAS Flow loss
48 1.0       !      Pressure factor
49 0.0       !      Pressure loss

50 0.03      !      Pressure loss factor
51 1.0       ! BURNER Combustion efficiency
52 -1.0      !      Fuel flow not specified

53 25000.0   !      AUXWORK
54 -1.0      !      TF Mass flow(default)
55 -1.0      !      CN Corrected RPM (default)
56 0.90      ! TURBIN Isentropic efficiency
57 -1.0      !      PCN-Compressor Turbine (default)
58 2.0       !      Compressor number
59 5.0       !      Map number
60 -1.0      !      Power law index (default)

61 0.0       !      AUXWORK
62 -1.0      !      TF Mass flow (default)
63 -1.0      !      CN Corrected RPM (default)
64 0.90      ! TURBIN Isentropic efficiency
65 -1.0      !      PCN-Compressor Turbine (default)
66 1.0       !      Compressor number
67 5.0       !      Map number
68 -1.0      !      Power law index (default)

69 1.0       !      Reheat switch (off)
70 0.03      ! DUCTER Pressure loss factor
71 0.9       !      Combustion efficiency
72 1000000.  !      Dummy fuel flow limit

73 -1.0      ! NOZCON switch (fixed throat area)

74 -1.0      !      Power turbine (none)
75 -1.0      ! PERFOR Propeller efficiency
76 0.0       !      Scaling index switch (off)
77 147600.0  !      Design Thrust

200 5        !
201 -1       !
202 21       ! ARITHY PCN(COMPRES1)=PCN(COMPRES2)
203 -1      !
204 6        !

210 1        !
211 -1       !
212 110      !
213 -1       ! ARITHY W(TOTAL)=W(compres1)+W(compres2)
214 101      !
215 -1       !
216 102      !

220 5        !
221 -1       !
222 37       ! ARITHY PCN(COMPRES3)=PCN(COMPRES4)
223 -1      !
224 27       !

230 1        !

```

```

231 -1      !
232 111     !
233 -1      ! ARITHY W(TOTAL)=W(compre3)+W(compre4)
234 105     !
235 -1      !
236 109     !

261 6.      !
262 -1.     !
263 268.    !
264 11.     !
265 6.      ! ARITHY BD(268)=sqrt(T(11))
266 -1.     !
267 269.    !
269 0.5     !

271 3.      !
272 -1.     !
273 278.    !
274 -1.     ! ARITHY BD(278)=m(11)*sqrt(T(11))
275 268.    !
276 11.     !
277 2.      !

281 4.      !
282 -1.     !
283 288.    !
284 -1.     ! ARITHY BD(288)=m(11)*sqrt(T(11))/P(11)
285 278.    !
286 11.     !
287 4.      !

301 6.      !
302 -1.     !
303 308.    !
304 13.     ! ARITHY BD(308)=sqrt(T(13))
305 6.      !
306 -1.     !
307 309.    !
309 0.5     !

311 3.      !
312 -1.     !
313 318.    !
314 -1.     ! ARITHY BD(318)=m(13)*sqrt(T(13))
315 308.    !
316 13.     !
317 2.      !

321 4.      !
322 -1.     !
323 328.    !
324 -1.     ! ARITHY BD(328)=m(13)*sqrt(T(13))/P(13)
325 318.    !
326 13.     !
327 4.      !

340 5.      !
341 -1.     !
342 400.    ! ARITHY COPY D400=SV(11,6) Temperature(TET)
343 11.     !
344 6.      !

-1
1 2 331.    !Dummy INTAKE mass flow
1004 11 6 4 3 1500. 1600. 1800. !Turbine entry temperature
-1
-3

```

Figure B.2 TURBOMATCH Input Data File for HP mode.

```

*** SELECTIVE BLEED ENGINE - HP MODE ***
*** OPTIMIZATION ***
-----
DP,SI,KE,FP
-1
-1
INTAKE S1-2 D1-4 R100
COMPRES2-3 D5-10 R101 V5 V6
ARITHY D200-204
COMPRES3-4 D11-16 R102 V11
PREMAS S4,17,5 D17-20 V17
DUCTER S17-18 D21-24 R103
NOZDIV S18,19,1 D25-26 R104
ARITHY D210-216
COMPRES5,6 D27-32 R109 V27 V28
ARITHY D220-224
PREMAS S6,20,7 D33-36
COMPRES7,8 D37-42 R105 V37
ARITHY D230-236
PREMAS S8,21,9 D43-46
PREMAS S9,22,10 D47-50
BURNER S10-11 D51-53 R106
TURBIN S11-12 D54-61,111 V55
MIXEES S12,22,13
TURBIN S13-14 D62-69,110 V63
DUCTER S14-15 D70-73 R107
NOZDIV S15,16,1 D74-75 R108
ARITHY D261-267
ARITHY D271-277
ARITHY D281-287
ARITHY D301-307
ARITHY D311-317
ARITHY D321-327
ARITHY D340-344
OUTPUT D7,11,22,400
OUTPUT D328,288
PLOTIT D328,288,7,11,22,400
PERFOR S1,0,0 D76-79,104,100,103,108,0,106,0,0,107
CODEND
-----
**** DATA ITEM *** HP MODE ****
**** DESIGN POINT AT 6000m, Mach=1.2 ****
-----
1 6000.0 ! Altitude
2 0.0 ! INTAKE ISA Deviation
3 1.2 ! Mach Number
4 -1.0 ! Pressure Recovery (default)

5 -1.0 ! Z Surge margin (default)
6 -1.0 ! PCN Corrected RPM (default)
1001 7 1 3 2. 3. 4. ! COMPRES Pressure Ratio PR1 (loop)
8 0.88 ! Isentropic Efficiency
9 0.0 ! Error switch (off)
10 1.0 ! Map Number

11 -1.0 ! Z Surge margin (default)
12 -1.0 ! PCN Corrected RPM (default)
13 2.0 ! COMPRES Pressure Ratio PR2
14 0.88 ! Isentropic Efficiency
15 0.0 ! Error switch (off)
16 1.0 ! Map Number

1003 17 2 3 .1 .2 .3 ! Mass flow factor
18 0.0 ! PREMAS Flow loss
19 1.0 ! Pressure factor
20 0.0 ! Pressure loss

21 1.00 ! Reheat switch(off)
22 0.03 ! DUCTOR Pressure factor
23 0.9 ! Combustion efficiency
24 1000000.0 ! Dummy limiting fuel flow

25 -1.0 ! NOZDIV Switch(Fixed exit area)
26 -1.0 ! Throat Area

27 -1.0 ! Z Surge margin (default)
28 -1.0 ! PCN Corrected RPM (default)
1002 28 2 3 6. 4. 3. ! COMPRES Pressure Ratio PR2

```


30	0.89	!	Isentropic Efficiency
31	0.0	!	Error switch (off)
32	1.0	!	Map Number
33	0.0	!	Mass flow factor(Surge Control)
34	0.0	! PREMAS	Flow loss
35	1.0	!	Pressure factor
36	0.0	!	Pressure loss
37	-1.0	!	Z Surge margin (default)
38	-1.0	!	PCN Corrected RPM (default)
39	3.3	! COMPRE	Pressure Ratio PR3
40	0.885	!	Isentropic Efficiency
41	0.0	!	Error switch (off)
42	1.0	!	Map Number
43	0.05	!	Mass flow factor (Reaction Control)
44	0.0	! PREMAS	Flow loss
45	1.0	!	Pressure factor
46	0.0	!	Pressure loss
47	0.10	!	Mass flow factor(HPT Cooling Air)
48	0.0	! PREMAS	Flow loss
49	1.0	!	Pressure factor
50	0.0	!	Pressure loss
51	0.03	!	Pressure loss factor
52	1.0	! BURNER	Combustion efficiency
53	-1.0	!	Fuel flow not specified
54	25000.0	!	AUXWORK
55	-1.0	!	TF Mass flow (default)
56	-1.0	!	CN Corrected RPM (default)
57	0.90	! TURBIN	Isentropic efficiency
58	-1.0	!	PCN-Compressor Turbine (default)
59	2.0	!	Compressor number
60	5.0	!	Map number
61	-1.0	!	Power law index (default)
62	0.0	!	AUXWORK
63	-1.0	!	TF Mass flow (default)
64	-1.0	!	CN Corrected RPM (default)
65	0.90	! TURBIN	Isentropic efficiency
66	-1.0	!	PCN-Compressor Turbine (default)
67	1.0	!	Compressor number
68	5.0	!	Map number
69	-1.0	!	Power law index (default)
70	1.0	!	Reheat switch (off)
71	0.03	! DUCTER	Pressure loss factor
72	0.9	!	Combustion efficiency
73	1000000.	!	Dummy fuel flow limit
74	-1.0	! NOZDIV	switch (fixed throat area)
75	-1.0	!	Throat Area
76	-1.0	!	Power turbine (none)
77	-1.0	! PERFOR	Propeller efficiency
78	0.0	!	Scaling index switch (off)
79	134400.0	!	Design Thrust
200	5	!	
201	-1	!	
202	12	! ARITHY	PCN(COMPRES1)=PCN(COMPRES2)
203	-1	!	
204	6	!	
210	1	!	
211	-1	!	
212	110	!	
213	-1	! ARITHY	W(TOTAL)=W(compres1)+W(compres2)
214	101	!	
215	-1	!	
216	102	!	
220	5	!	
221	-1	!	
222	38	! ARITHY	PCN(COMPRES3)=PCN(COMPRES4)
223	-1	!	
224	28	!	
230	1	!	
231	-1	!	
232	111	!	

```

233 -1      ! ARITHY  W(TOTAL)=W(compre3)+W(compre4)
234 105     !
235 -1      !
236 109     !

261 6.      !
262 -1.     !
263 268.    !
264 11.     !
265 6.      ! ARITHY  BD(268)=sqrt(T(11))
266 -1.     !
267 269.    !
268 0.5     !

271 3.      !
272 -1.     !
273 278.    !
274 -1.     ! ARITHY  BD(278)=m(11)*sqrt(T(11))
275 268.    !
276 11.     !
277 2.      !

281 4.      !
282 -1.     !
283 288.    !
284 -1.     ! ARITHY  BD(288)=m(11)*sqrt(T(11))/P(11)
285 268.    !
286 11.     !
287 4.      !

301 6.      !
302 -1.     !
303 308.    !
304 13.     ! ARITHY  BD(308)=sqrt(T(13))
305 6.      !
306 -1.     !
307 309.    !
308 0.5     !

311 3.      !
312 -1.     !
313 318.    !
314 -1.     ! ARITHY  BD(318)=m(13)*sqrt(T(13))
315 308.    !
316 13.     !
317 2.      !

321 4.      !
322 -1.     !
323 328.    !
324 -1.     ! ARITHY  BD(328)=m(13)*sqrt(T(13))/P(13)
325 308.    !
326 13.     !
327 4.      !

340 5.      !
341 -1.     !
342 400.    ! ARITHY COPY D400=SV(11,6) Temperature(TET)
343 11.     !
344 6.      !

```

```

-1
1 2 201.    !Dummy INTAKE mass flow
1004 8 6 4 3 1500 1600 1700. !Turbine entry temperature
-1
-3

```

Figure B.3 TURBOMATCH Output Data for LP mode.

TURBOMATCH SCHEME-VAX VERSION (30/6/88)

Input *Program* follows

```

*** SELECTIVE BLEED ENGINE - LP MODE ***
*** PERFORMANCE ***

```

OD,SI,KE,FP

-1

-1

INTAKE	S1-2	D1-4	R100		
COMPRES	S2-3	D5-10	R101	V5	V6
PREMAS	S3,17,4	D11-14		V11	
DUCTER	S17-18	D15-18	R103		
NOZCON	S18,19,1	D19	R104		
ARITHY		D200-204			
COMPRES	S4-5	D20-25	R102	V20	
COMPRES	S5,6	D26-31	R109	V26	V27
ARITHY		D220-224			
PREMAS	S6,20,7	D32-35			
COMPRES	S7,8	D36-41	R105	V36	
ARITHY		D210-216			
ARITHY		D230-236			
PREMAS	S8,21,9	D42-45			
PREMAS	S9,22,10	D46-49			
BURNER	S10-11	D50-52	R106		
TURBIN	S11-12	D53-60,111		V54	
MIXEES	S12,22,13				
TURBIN	S13-14	D61-68,110		V62	
DUCTER	S14-15	D69-72	R107		
NOZCON	S15,16,1	D73	R108		
PERFOR	S1,0,0	D74-77,104,100,103,108,0,106,0,0,107			
CODEND					

```

**** DATA ITEM      *** LP MODE ****
DESIGN POINT AT SEA LEVEL STATIC
RP1=3.0, RP2=2.41, PR3=3.6, BY PASS RATIO=1.0 and TET=1500K

```

1	0.0	!	Altitude
2	0.0	!	INTAKE ISA Deviation
3	0.0	!	Mach Number
4	-1.0	!	Pressure Recovery (default)
5	-1.0	!	Z Surge margin (default)
6	-1.0	!	PCN Corrected RPM (default)
7	3.0	!	COMPRES Pressure Ratio PR1
8	0.88	!	Isentropic Efficiency
9	0.0	!	Error switch (off)
10	1.0	!	Map Number
11	0.5	!	Mass flow factor
12	0.0	!	PREMAS Flow loss
13	1.0	!	Pressure factor
14	0.0	!	Pressure loss
15	1.00	!	Reheat switch(off)
16	0.03	!	DUCTOR Pressure factor
17	0.9	!	Combustion efficiency
18	1000000.0	!	Dummy limiting fuel flow
19	-1.0	!	NOZCON Switch(Fixed exit area)
20	-1.0	!	Z Surge margin (default)
21	-1.0	!	PCN Corrected RPM (default)
22	2.41	!	COMPRES Pressure Ratio PR2
23	0.88	!	Isentropic Efficiency
24	1.0	!	Error switch (on)
25	1.0	!	Map Number
26	-1.0	!	Z Surge margin (default)
27	-1.0	!	PCN Corrected RPM (default)
28	1.2	!	COMPRES Pressure Ratio PR3
29	0.89	!	Isentropic Efficiency
30	1.0	!	Error switch (on)
31	1.0	!	Map Number

```

32  0.0      !      Mass flow factor(Surge Control)
33  0.0      ! PREMAS Flow loss
34  1.0      !      Pressure factor
35  0.0      !      Pressure loss

36 -1.0      !      Z Surge margin (default)
37 -1.0      !      PCN Corrected RPM (default)
38  3.0      ! COMPRE Pressure Ratio PR3
39  0.89     !      Isentropic Efficiency
40  1.0      !      Error switch (on)
41  1.0      !      Map Number

42  0.05     !      Mass flow factor(Reaction Control)
43  0.0      ! PREMAS Flow loss
44  1.0      !      Pressure factor
45  0.0      !      Pressure loss

46  0.10     !      Mass flow factor(HPT Cooling Air)
47  0.0      ! PREMAS Flow loss
48  1.0      !      Pressure factor
49  0.0      !      Pressure loss

50  0.03     !      Pressure loss factor
51  1.0      ! BURNER Combustion efficiency
52 -1.0      !      Fuel flow not specified

53 25000.0   !      AUXWORK
54 -1.0      !      TF Mass flow(default)
55 -1.0      !      CN Corrected RPM (default)
56  0.90     ! TURBIN Isentropic efficiency
57 -1.0      !      PCN-Compressor Turbine(default)
58  2.0      !      Compressor number
59  5.0      !      Map number
60 -1.0      !      Power law index(default)

61  0.0      !      AUXWORK
62 -1.0      !      TF Mass flow(default)
63 -1.0      !      CN Corrected RPM (default)
64  0.90     ! TURBIN Isentropic efficiency
65 -1.0      !      PCN-Compressor Turbine(default)
66  1.0      !      Compressor number
67  5.0      !      Map number
68 -1.0      !      Power law index(default)

69  1.0      !      Reheat switch (off)
70  0.03     ! DUCTER Pressure loss factor
71  0.9      !      Combustion efficiency
72 1000000.  !      Dummy fuel flow limit

73 -1.0      ! NOZCON switch (fixed throat area)

74 -1.0      !      Power turbine (none)
75 -1.0      ! PERFOR Propeller efficiency
76  0.0      !      Scaling index switch (off)
77 147600.0  !      Design Thrust

200  5       !
201 -1       !
202  21      ! ARITHY PCN(COMPRES1)=PCN(COMPRES2)
203 -1       !
204  6       !

210  1       !
211 -1       !
212  110     !
213 -1       ! ARITHY W(TOTAL)=W(compre1)+W(compre2)
214  101     !
215 -1       !
216  102     !

220  5       !
221 -1       !
222  37      ! ARITHY PCN(COMPRES3)=PCN(COMPRES4)
223 -1       !
224  27      !

230  1       !
231 -1       !
232  111     !
233 -1       ! ARITHY W(TOTAL)=W(compre3)+W(compre4)
234  105     !
235 -1       !
236  109     !

```

```

-1
1 2 331.      !Dummy INTAKE mass flow
11 6 1500.    !Turbine entry temperature
-1
!

```

The Units for this Run are as follows:-
 Temperature = K Pressure = Atmospheres Length = metres
 Area = sq metres Mass Flow = kg/sec Velocity = metres/sec
 Force = Newtons s.f.c.(Thrust) = mg/N sec s.f.c.(Power) = mug/J
 Sp. Thrust = N/kg/sec Power = Watts
 ***** DESIGN POINT ENGINE CALCULATIONS *****

```

***** AMBIENT AND INLET PARAMETERS *****
Alt. = 0.0      I.S.A. Dev. = 0.000      Mach No. = 0.00
Etar = 1.0000      Momentum Drag = 0.00
***** COMPRESSOR 1 PARAMETERS *****
PRSF = 0.49020E+01      ETASF = 0.10107E+01      WASF = 0.5E+00E+00
Z = 0.85000      PR = 3.000      ETA = 0.88000
PCN = 1.0000      CN = 1.00000      COMWK = 0.40141E+08
***** DUCT/AFTER BURNING 1 PARAMETERS *****
ETA = 0.9000      DLP = 0.0900      WFB = 0.0000
***** CONVERGENT NOZZLE 1 PARAMETERS *****
Area = 0.2803      Exit Velocity = 369.86      Gross Thrust = 75056.73
Nozzle Coeff. = 0.97531E+00
***** COMPRESSOR 2 PARAMETERS *****
PRSF = 0.34559E+01      ETASF = 0.10107E+01      WASF = 0.11133E+00
Z = 0.85000      PR = 2.410      ETA = 0.88000
PCN = 1.0000      CN = 1.00000      COMWK = 0.22007E+08
***** COMPRESSOR 3 PARAMETERS *****
PRSF = 0.49020E+00      ETASF = 0.10222E+01      WASF = 0.52553E-01
Z = 0.85000      PR = 1.200      ETA = 0.89000
PCN = 1.0000      CN = 1.00000      COMWK = 0.55718E+07
***** COMPRESSOR 4 PARAMETERS *****
PRSF = 0.49020E+01      ETASF = 0.10222E+01      WASF = 0.45146E-01
Z = 0.85000      PR = 3.000      ETA = 0.89000
PCN = 1.0000      CN = 1.00000      COMWK = 0.35898E+08
***** COMBUSTION CHAMBER PARAMETERS *****
ETASF = 0.10000E+01
ETA = 1.00000      DLP = 0.7808      WFB = 2.9775
***** TURBINE 1 PARAMETERS *****
CNSF = 0.11000E+03      ETASF = 0.10056E+01      TFSF = 0.19442E+01
DHSF = 0.59211E+04      ETA = 0.90000      CN = 2.840
TF = 430.916      AUXWK = 0.25000E+05
***** TURBINE 2 PARAMETERS *****
CNSF = 0.98828E+02      ETASF = 0.10056E+01      TFSF = 0.81479E+00
DHSF = 0.92820E+04      ETA = 0.90000      CN = 2.840
TF = 430.916      AUXWK = 0.00000E+00
***** DUCT/AFTER BURNING 2 PARAMETERS *****
ETA = 0.9000      DLP = 0.0725      WFB = 0.0000
***** CONVERGENT NOZZLE 2 PARAMETERS *****
Area = 0.4994      Exit Velocity = 539.90      Gross Thrust = 98414.05
Nozzle Coeff. = 0.97949E+00

```

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 1.0000

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	331.000	1.00000	1.00000	288.15	288.15	0.0	*****
2	0.00000	331.000	*****	1.00000	*****	288.15	*****	*****
3	0.00000	331.000	*****	3.00000	*****	408.48	*****	*****
4	0.00000	165.500	*****	3.00000	*****	408.48	*****	*****
5	0.00000	165.500	*****	7.23000	*****	538.20	*****	*****
6	0.00000	165.500	*****	8.67600	*****	569.47	*****	*****
7	0.00000	165.500	*****	8.67600	*****	569.47	*****	*****
8	0.00000	165.500	*****	26.02800	*****	789.18	*****	*****
9	0.00000	157.225	*****	26.02800	*****	789.18	*****	*****
10	0.00000	141.503	*****	26.02800	*****	789.18	*****	*****
11	0.02104	144.480	*****	25.24716	*****	1500.00	*****	*****
12	0.02104	144.480	*****	10.54031	*****	1253.50	*****	*****
13	0.01894	160.202	*****	10.54031	*****	1210.70	*****	*****
14	0.01894	160.202	*****	2.41527	*****	883.41	*****	*****
15	0.01894	160.202	*****	2.34282	*****	883.41	*****	*****
16	0.01894	160.202	1.27060	2.34282	755.05	883.41	539.9	0.4994

17	0.00000	165.500	*****	3.00000	*****	408.48	*****	*****
18	0.00000	165.500	*****	2.91000	*****	408.48	*****	*****
19	0.00000	165.500	1.54066	2.91000	340.75	408.48	369.9	0.2803
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	8.275	*****	26.02800	*****	789.18	*****	*****
22	0.00000	15.723	*****	26.02800	*****	789.18	*****	*****

Gross Thrust = 173470.77
Momentum Drag = 0.00
Net Thrust = 173470.77
Fuel Flow = 2.9775
s.f.c. = 17.16410
Sp. Thrust = 524.081

-1 ! LP MODE DESIGN POINT CONDITION
11 6 1390 !
-1 !

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 3 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

Alt. = 0.0 I.S.A. Dev. = 0.000 Mach No. = 0.00
Etar = 1.0000 Momentum Drag = 0.00

***** COMPRESSOR 1 PARAMETERS *****

Z = 0.85540 PR = 2.799 ETA = 0.88995
PCN = 0.9362 CN = 0.93615 COMWK = 0.33786E+08

***** DUCT/AFTER BURNING 1 PARAMETERS *****

ETA = 0.9000 DLP = 0.0639 WFE = 0.0000

***** CONVERGENT NOZZLE 1 PARAMETERS *****

Area = 0.2803 Exit Velocity = 365.33 Gross Thrust = 68178.06
Nozzle Coeff. = 0.97708E+00

***** COMPRESSOR 2 PARAMETERS *****

Z = 0.83075 PR = 2.258 ETA = 0.89957
PCN = 0.9362 CN = 0.94779 COMWK = 0.17192E+08

***** COMPRESSOR 3 PARAMETERS *****

Z = 0.83969 PR = 1.195 ETA = 0.89738
PCN = 0.9683 CN = 0.99236 COMWK = 0.44188E+07

***** COMPRESSOR 4 PARAMETERS *****

Z = 0.83428 PR = 2.940 ETA = 0.90031
PCN = 0.9683 CN = 0.99306 COMWK = 0.31948E+08

***** COMBUSTION CHAMBER PARAMETERS *****

ETA = 1.00000 DLP = 0.6874 WFE = 2.3590

***** TURBINE 1 PARAMETERS *****

TF = 433.057 ETA = 0.89896 CN = 2.762
AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****

TF = 432.785 ETA = 0.89893 CN = 2.764
AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****

ETA = 0.9000 DLP = 0.0620 WFE = 0.0000

***** CONVERGENT NOZZLE 2 PARAMETERS *****

Area = 0.4994 Exit Velocity = 518.77 Gross Thrust = 76508.90
Nozzle Coeff. = 0.98001E+00

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 1.0000

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	303.921	1.00000	1.00000	288.15	288.15	0.0	*****
2	0.00000	303.921	*****	1.00000	*****	288.15	*****	*****
3	0.00000	303.921	*****	2.79854	*****	398.51	*****	*****
4	0.00000	147.665	*****	2.79854	*****	398.51	*****	*****
5	0.00000	147.576	*****	6.31951	*****	512.49	*****	*****

6	0.00000	147.576	*****	7.55217	*****	541.44	*****	*****
7	0.00000	147.580	*****	7.55217	*****	541.44	*****	*****
8	0.00000	147.580	*****	22.20571	*****	745.45	*****	*****
9	0.00000	140.201	*****	22.20571	*****	745.45	*****	*****
10	0.00000	126.181	*****	22.20571	*****	745.45	*****	*****
11	0.01870	128.540	*****	21.51826	*****	1390.00	*****	*****
12	0.01870	128.540	*****	8.99099	*****	1158.75	*****	*****
13	0.01683	142.560	*****	8.99099	*****	1120.44	*****	*****
14	0.01683	142.560	*****	2.05953	*****	813.43	*****	*****
15	0.01683	142.560	*****	1.99757	*****	813.43	*****	*****
16	0.01683	142.560	1.07966	1.99757	692.76	813.43	518.8	0.4994
17	0.00000	156.266	*****	2.79854	*****	398.51	*****	*****
18	0.00000	156.266	*****	2.71462	*****	398.51	*****	*****
19	0.00000	156.266	1.43649	2.71462	332.37	398.51	365.3	0.2803
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	7.379	*****	22.20571	*****	745.45	*****	*****
22	0.00000	14.020	*****	22.20571	*****	745.45	*****	*****

Gross Thrust = 144686.97
Momentum Drag = 0.00
Net Thrust = 144686.97
Fuel Flow = 2.3590
s.f.c. = 16.30432
Sp. Thrust = 476.067

3 0.5 ! Mach number
-1 !
11 6 1320. ! TET
-1 !

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 3 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

Alt. = 0.0 I.S.A. Dev. = 0.000 Mach No. = 0.50
Etar = 1.0000 Momentum Drag = 52805.76

***** COMPRESSOR 1 PARAMETERS *****

Z = 0.85747 PR = 2.536 ETA = 0.89955
PCN = 0.8702 CN = 0.84921 COMWK = 0.31931E+08

***** DUCT/AFTER BURNING 1 PARAMETERS *****

ETA = 0.9000 DLP = 0.0902 WFB = 0.0000

***** CONVERGENT NOZZLE 1 PARAMETERS *****

Area = 0.2803 Exit Velocity = 368.12 Gross Thrust = 75294.78
Nozzle Coeff. = 0.97520E+00

***** COMPRESSOR 2 PARAMETERS *****

Z = 0.80730 PR = 2.067 ETA = 0.90853
PCN = 0.8702 CN = 0.87433 COMWK = 0.14802E+08

***** COMPRESSOR 3 PARAMETERS *****

Z = 0.83152 PR = 1.188 ETA = 0.90398
PCN = 0.9458 CN = 0.97593 COMWK = 0.40688E+07

***** COMPRESSOR 4 PARAMETERS *****

Z = 0.80768 PR = 2.830 ETA = 0.89834
PCN = 0.9458 CN = 0.97768 COMWK = 0.29437E+08

***** COMBUSTION CHAMBER PARAMETERS *****

ETA = 1.0000 DLP = 0.6745 WFB = 2.0878

***** TURBINE 1 PARAMETERS *****

TF = 436.348 ETA = 0.89711 CN = 2.635
AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****

TF = 435.928 ETA = 0.89756 CN = 2.637
AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****

ETA = 0.9000 DLP = 0.0588 WFB = 0.0000

***** CONVERGENT NOZZLE 2 PARAMETERS *****

Area = 0.4994 Exit Velocity = 505.38 Gross Thrust = 69571.20
Nozzle Coeff. = 0.98025E+00

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 1.0000

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	310.303	1.00000	1.18630	288.15	302.58	170.2	*****
2	0.00000	310.303	*****	1.18630	*****	302.58	*****	*****
3	0.00000	310.303	*****	3.00865	*****	404.65	*****	*****
4	0.00000	143.675	*****	3.00865	*****	404.65	*****	*****
5	0.00000	143.800	*****	6.21972	*****	505.51	*****	*****
6	0.00000	143.800	*****	7.38779	*****	532.91	*****	*****
7	0.00000	143.665	*****	7.38779	*****	532.91	*****	*****
8	0.00000	143.665	*****	20.90548	*****	726.61	*****	*****
9	0.00000	136.482	*****	20.90548	*****	726.61	*****	*****
10	0.00000	122.834	*****	20.90548	*****	726.61	*****	*****
11	0.01700	124.922	*****	20.23100	*****	1320.00	*****	*****
12	0.01700	124.922	*****	8.44118	*****	1098.30	*****	*****
13	0.01530	138.570	*****	8.44118	*****	1063.64	*****	*****
14	0.01530	138.570	*****	1.94399	*****	770.68	*****	*****
15	0.01530	138.570	*****	1.88523	*****	770.68	*****	*****
16	0.01530	138.570	1.01824	1.88523	654.82	770.68	505.4	0.4994
17	0.00000	166.682	*****	3.00865	*****	404.65	*****	*****
18	0.00000	166.682	*****	2.91848	*****	404.65	*****	*****
19	0.00000	166.682	1.54421	2.91848	337.52	404.65	368.1	0.2803
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	7.183	*****	20.90548	*****	726.61	*****	*****
22	0.00000	13.648	*****	20.90548	*****	726.61	*****	*****

Gross Thrust = 144865.98
Momentum Drag = 52805.76
Net Thrust = 92060.23
Fuel Flow = 2.0878
s.f.c. = 22.67900
Sp. Thrust = 296.679

1 9000.0 ! Altitude
3 0.65 ! Mach number
-1 !
11 6 925. ! TET
-1 !

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 7 Loops *****

***** AMBIENT AND INLET PARAMETERS *****
Alt. = 9000.0 I.S.A. Dev. = 0.000 Mach No. = 0.65
Etar = 1.0000 Momentum Drag = 16962.78

***** COMPRESSOR 1 PARAMETERS *****
Z = 0.83906 PR = 1.995 ETA = 0.89605
PCN = 0.6182 CN = 0.66488 COMWK = 0.52324E+07

***** DUCT/AFTER BURNING 1 PARAMETERS *****
ETA = 0.9000 DLP = 0.0241 WFB = 0.0000

***** CONVERGENT NOZZLE 1 PARAMETERS *****
Area = 0.2803 Exit Velocity = 322.20 Gross Thrust = 19118.32
Nozzle Coeff. = 0.97761E+00

***** COMPRESSOR 2 PARAMETERS *****
Z = 0.72153 PR = 1.682 ETA = 0.89590
PCN = 0.6182 CN = 0.70980 COMWK = 0.19459E+07

***** COMPRESSOR 3 PARAMETERS *****
Z = 0.82802 PR = 1.174 ETA = 0.91309
PCN = 0.7702 CN = 0.93509 COMWK = 0.65794E+06

***** COMPRESSOR 4 PARAMETERS *****
Z = 0.76805 PR = 2.621 ETA = 0.89704
PCN = 0.7702 CN = 0.93819 COMWK = 0.47555E+07

***** COMBUSTION CHAMBER PARAMETERS *****
ETA = 0.99979 DLP = 0.1429 WFB = 0.3202

***** TURBINE 1 PARAMETERS *****

TF = 444.759 ETA = 0.87041 CN = 2.236
 AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****

TF = 440.755 ETA = 0.87741 CN = 2.243
 AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****

ETA = 0.9000 DLP = 0.0110 WFB = 0.0000

***** CONVERGENT NOZZLE 2 PARAMETERS *****

Area = 0.4994 Exit Velocity = 311.14 Gross Thrust = 10194.16
 Nozzle Coeff. = 0.97683E+00

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 1.0000

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	85.864	0.30334	0.40303	229.65	249.13	197.6	*****
2	0.00000	85.864	*****	0.40303	*****	249.13	*****	*****
3	0.00000	85.864	*****	0.80405	*****	309.68	*****	*****
4	0.00000	34.964	*****	0.80405	*****	309.68	*****	*****
5	0.00000	34.970	*****	1.35249	*****	365.19	*****	*****
6	0.00000	34.970	*****	1.58749	*****	383.61	*****	*****
7	0.00000	34.969	*****	1.58749	*****	383.61	*****	*****
8	0.00000	34.969	*****	4.16080	*****	517.03	*****	*****
9	0.00000	33.221	*****	4.16080	*****	517.03	*****	*****
10	0.00000	29.899	*****	4.16080	*****	517.03	*****	*****
11	0.01071	30.219	*****	4.01793	*****	925.00	*****	*****
12	0.01071	30.219	*****	1.68710	*****	765.84	*****	*****
13	0.00964	33.541	*****	1.68710	*****	741.70	*****	*****
14	0.00964	33.541	*****	0.43021	*****	542.61	*****	*****
15	0.00964	33.541	*****	0.41926	*****	542.61	*****	*****
16	0.00964	33.541	0.30334	0.41926	496.16	542.61	311.1	0.4994
17	0.00000	50.896	*****	0.80405	*****	309.68	*****	*****
18	0.00000	50.896	*****	0.77996	*****	309.68	*****	*****
19	0.00000	50.896	0.41201	0.77996	258.13	309.68	322.2	0.2803
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	1.748	*****	4.16080	*****	517.03	*****	*****
22	0.00000	3.322	*****	4.16080	*****	517.03	*****	*****

Gross Thrust = 29312.48
 Momentum Drag = 16962.78
 Net Thrust = 12349.70
 Fuel Flow = 0.3202
 s.f.c. = 25.92726
 Sp. Thrust = 143.828

3 0.78 ! Mach number
 -1 !
 11 6 950. ! TET
 -1 !

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 1 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

Alt. = 9000.0 I.S.A. Dev. = 0.000 Mach No. = 0.78
 Etar = 1.0000 Momentum Drag = 23033.19

***** COMPRESSOR 1 PARAMETERS *****

Z = 0.84206 PR = 2.034 ETA = 0.8932
 PCN = 0.6401 CN = 0.67686 COMWK = 0.03057E+07

***** DUCT/AFTER BURNING 1 PARAMETERS *****

ETA = 0.9000 DLP = 0.0276 WFB = 0.0000

***** CONVERGENT NOZZLE 1 PARAMETERS *****

Area = 0.2803 Exit Velocity = 328.61 Gross Thrust = 23126.67
 Nozzle Coeff. = 0.97475E+00

***** COMPRESSOR 2 PARAMETERS *****

Z = 0.73881 PR = 1.716 ETA = 0.89955
 PCN = 0.6401 CN = 0.72053 COMWK = 0.23905E+07

***** COMPRESSOR 3 PARAMETERS *****
 Z = 0.82481 PR = 1.172 ETA = 0.91304
 PCN = 0.7849 CN = 0.93173 COMWK = 0.77742E+06

***** COMPRESSOR 4 PARAMETERS *****
 Z = 0.76171 PR = 2.598 ETA = 0.89608
 PCN = 0.7849 CN = 0.93508 COMWK = 0.56159E+07

***** COMBUSTION CHAMBER PARAMETERS *****
 ETA = 0.99986 DLP = 0.1671 WFB = 0.3707

***** TURBINE 1 PARAMETERS *****
 TF = 443.775 ETA = 0.87384 CN = 2.284
 AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****
 TF = 441.573 ETA = 0.87781 CN = 2.291
 AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****
 ETA = 0.9000 DLP = 0.0132 WFB = 0.0000

***** CONVERGENT NOZZLE 2 PARAMETERS *****
 Area = 0.4994 Exit Velocity = 352.19 Gross Thrust = 13180.85
 Nozzle Coeff. = 0.97809E+00

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 1.0000

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	97.160	0.30334	0.45354	229.65	257.70	237.1	*****
2	0.00000	97.160	*****	0.45354	*****	257.70	*****	*****
3	0.00000	97.160	*****	0.92257	*****	322.37	*****	*****
4	0.00000	39.802	*****	0.92257	*****	322.37	*****	*****
5	0.00000	39.886	*****	1.58331	*****	381.97	*****	*****
6	0.00000	39.886	*****	1.85563	*****	401.23	*****	*****
7	0.00000	39.887	*****	1.85563	*****	401.23	*****	*****
8	0.00000	39.887	*****	4.82153	*****	538.71	*****	*****
9	0.00000	37.893	*****	4.82153	*****	538.71	*****	*****
10	0.00000	34.103	*****	4.82153	*****	538.71	*****	*****
11	0.01087	34.474	*****	4.65440	*****	950.00	*****	*****
12	0.01087	34.474	*****	1.94904	*****	785.64	*****	*****
13	0.00978	38.263	*****	1.94904	*****	762.13	*****	*****
14	0.00978	38.263	*****	0.47089	*****	551.48	*****	*****
15	0.00978	38.263	*****	0.45766	*****	551.48	*****	*****
16	0.00978	38.263	0.30334	0.45766	492.31	551.48	352.2	0.4994
17	0.00000	57.209	*****	0.92257	*****	322.37	*****	*****
18	0.00000	57.209	*****	0.89498	*****	322.37	*****	*****
19	0.00000	57.209	0.47238	0.89498	268.54	322.37	328.6	0.2803
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	1.994	*****	4.82153	*****	538.71	*****	*****
22	0.00000	3.789	*****	4.82153	*****	538.71	*****	*****

Gross Thrust = 36307.52
 Momentum Drag = 23033.19
 Net Thrust = 13274.33
 Fuel Flow = 0.3707
 s.f.c. = 27.92340
 Sp. Thrust = 136.623

 1 0.0 ! Altitude
 3 0.9 ! Mach number
 -1 !
 11 6 1420. ! TET
 -1 !

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 7 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

Alt. = 0.0 I.S.A. Dev. = 0.000 Mach No. = 0.90
 Etar = 1.0000 Momentum Drag = 124965.08

***** COMPRESSOR 1 PARAMETERS *****
 Z = 0.85584 PR = 2.475 ETA = 0.90144
 PCN = 0.8934 CN = 0.82873 COMWK = 0.44947E+08

***** DUCT/AFTER BURNING 1 PARAMETERS *****
 ETA = 0.9000 DLP = 0.1253 WFB = 0.0000

***** CONVERGENT NOZZLE 1 PARAMETERS *****
 Area = 0.2803 Exit Velocity = 385.41 Gross Thrust = 114751.34
 Nozzle Coeff. = 0.96227E+00

***** COMPRESSOR 2 PARAMETERS *****
 Z = 0.79904 PR = 2.024 ETA = 0.90659
 PCN = 0.8934 CN = 0.85717 COMWK = 0.10400E+08

***** COMPRESSOR 3 PARAMETERS *****
 Z = 0.83131 PR = 1.186 ETA = 0.90496
 PCN = 0.9815 CN = 0.97106 COMWK = 0.56925E+07

***** COMPRESSOR 4 PARAMETERS *****
 Z = 0.80217 PR = 2.803 ETA = 0.89796
 PCN = 0.9815 CN = 0.97326 COMWK = 0.41100E+08

***** COMBUSTION CHAMBER PARAMETERS *****
 ETA = 1.00000 DLP = 0.9129 WFB = 2.9645

***** TURBINE 1 PARAMETERS *****
 TF = 436.904 ETA = 0.89545 CN = 2.606
 AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****
 TF = 436.553 ETA = 0.89597 CN = 2.605
 AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****
 ETA = 0.9000 DLP = 0.0794 WFB = 0.0000

***** CONVERGENT NOZZLE 2 PARAMETERS *****
 Area = 0.4994 Exit Velocity = 526.56 Gross Thrust = 112511.34
 Nozzle Coeff. = 0.97760E+00

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 1.0000

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	407.963	1.00000	1.69166	288.15	334.65	306.3	*****
2	0.00000	407.963	*****	1.69166	*****	334.65	*****	*****
3	0.00000	407.963	*****	4.18653	*****	443.71	*****	*****
4	0.00000	186.696	*****	4.18653	*****	443.71	*****	*****
5	0.00000	186.711	*****	8.47407	*****	549.93	*****	*****
6	0.00000	186.711	*****	10.05167	*****	579.13	*****	*****
7	0.00000	186.713	*****	10.05167	*****	579.13	*****	*****
8	0.00000	186.713	*****	28.17850	*****	784.64	*****	*****
9	0.00000	177.377	*****	28.17850	*****	784.64	*****	*****
10	0.00000	159.640	*****	28.17850	*****	784.64	*****	*****
11	0.01857	162.604	*****	27.26557	*****	1420.00	*****	*****
12	0.01857	162.604	*****	11.40235	*****	1185.58	*****	*****
13	0.01671	180.342	*****	11.40235	*****	1148.35	*****	*****
14	0.01671	180.342	*****	2.65296	*****	838.68	*****	*****
15	0.01671	180.342	*****	2.57359	*****	838.68	*****	*****
16	0.01671	180.342	1.38885	2.57359	715.03	838.68	526.6	0.4994
17	0.00000	221.357	*****	4.18653	*****	443.71	*****	*****
18	0.00000	221.357	*****	4.06121	*****	443.71	*****	*****
19	0.00000	221.357	2.14974	4.06121	370.44	443.71	385.4	0.2803
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	9.336	*****	28.17850	*****	784.64	*****	*****
22	0.00000	17.738	*****	28.17850	*****	784.64	*****	*****

Gross Thrust = 227262.67
 Momentum Drag = 124965.08
 Net Thrust = 102297.59
 Fuel Flow = 2.9645
 s.f.c. = 28.97886
 Sp. Thrust = 250.752

```

*****
1  0.0  !  Altitude
3  0.9  !  Mach number
-1      !
11 6 1315. !  TET
-1      !

```

```

*****

```

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 1 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

Alt. = 0.0 I.S.A. Dev. = 0.000 Mach No. = 0.90
 Etar = 1.0000 Momentum Drag = 111955.12

***** COMPRESSOR 1 PARAMETERS *****

Z = 0.85112 PR = 2.271 ETA = 0.90256
 PCN = 0.8170 CN = 0.75785 COMWK = 0.35957E+08

***** DUCT/AFTER BURNING 1 PARAMETERS *****

ETA = 0.9000 DLP = 0.1150 WFB = 0.0000

***** CONVERGENT NOZZLE 1 PARAMETERS *****

Area = 0.2803 Exit Velocity = 380.37 Gross Thrust = 103304.55
 Nozzle Coeff. = 0.96653E+00

***** COMPRESSOR 2 PARAMETERS *****

Z = 0.77463 PR = 1.879 ETA = 0.90983
 PCN = 0.8170 CN = 0.79429 COMWK = 0.14955E+08

***** COMPRESSOR 3 PARAMETERS *****

Z = 0.82733 PR = 1.179 ETA = 0.90895
 PCN = 0.9397 CN = 0.95248 COMWK = 0.44479E+07

***** COMPRESSOR 4 PARAMETERS *****

Z = 0.78117 PR = 2.702 ETA = 0.89664
 PCN = 0.9397 CN = 0.95546 COMWK = 0.32086E+08

***** COMBUSTION CHAMBER PARAMETERS *****

ETA = 1.00000 DLP = 0.7692 WFB = 2.2442

***** TURBINE 1 PARAMETERS *****

TF = 439.786 ETA = 0.88691 CN = 2.478
 AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****

TF = 438.978 ETA = 0.88814 CN = 2.476
 AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****

ETA = 0.9000 DLP = 0.0653 WFB = 0.0000

***** CONVERGENT NOZZLE 2 PARAMETERS *****

Area = 0.4994 Exit Velocity = 506.98 Gross Thrust = 83085.14
 Nozzle Coeff. = 0.98000E+00

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 1.0000

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	365.490	1.00000	1.69166	288.15	334.85	306.3	*****
2	0.00000	365.490	*****	1.69166	*****	334.85	*****	*****
3	0.00000	365.490	*****	3.84213	*****	432.13	*****	*****
4	0.00000	158.998	*****	3.84213	*****	432.13	*****	*****
5	0.00000	159.386	*****	7.21888	*****	523.88	*****	*****
6	0.00000	159.386	*****	8.51284	*****	550.80	*****	*****
7	0.00000	159.408	*****	8.51284	*****	550.80	*****	*****
8	0.00000	159.408	*****	22.99820	*****	740.40	*****	*****
9	0.00000	151.437	*****	22.99820	*****	740.40	*****	*****
10	0.00000	136.294	*****	22.99820	*****	740.40	*****	*****
11	0.01647	138.538	*****	22.22903	*****	1315.00	*****	*****
12	0.01647	138.538	*****	9.28952	*****	1096.75	*****	*****
13	0.01482	153.682	*****	9.28952	*****	1063.44	*****	*****
14	0.01482	153.682	*****	2.16592	*****	775.58	*****	*****

15	0.01482	153.682	*****	2.10064	*****	775.58	*****	*****
16	0.01482	153.682	1.13300	2.10064	659.04	775.58	507.0	0.4994
17	0.00000	205.867	*****	3.84213	*****	432.13	*****	*****
18	0.00000	205.867	*****	3.72710	*****	432.13	*****	*****
19	0.00000	205.867	1.97237	3.72710	360.67	432.13	380.4	0.2803
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	7.970	*****	22.99820	*****	740.40	*****	*****
22	0.00000	15.144	*****	22.99820	*****	740.40	*****	*****

Gross Thrust = 186389.70
 Momentum Drag = 111955.12
 Net Thrust = 74434.59
 Fuel Flow = 2.2442
 s.f.c. = 30.15049
 Sp. Thrust = 203.657

Figure B.4 TURBMATCH Output Data for HP mode.

TURBOMATCH SCHEME-VAX VERSION (30/6/88)

Input "Program" follows

*** SELECTIVE BLEED ENGINE - HP MODE ***

*** PERFORMANCE ***

```

-----
OD,SI,KE,FP
-1
-1
INTAKE S1-2      D1-4      R100
COMPRES2-3      D5-10     R101    V5  V6
ARITHY      D200-204
COMPRES3-4      D11-16     R102    V11
PREMAS S4,17,5   D17-20     V17
DUCTER S17-18    D21-24     R103
NOZDIV S18,19,1  D25-26     R104
ARITHY      D210-216
COMPRES5,6      D27-32     R109    V27  V28
ARITHY      D220-224
PREMAS S6,20,7   D33-36
COMPRES7,8      D37-42     R105    V37
ARITHY      D230-236
PREMAS S8,21,9   D43-46
PREMAS S9,22,10 D47-50
BURNER S10-11    D51-53     R106
TURBIN S11-12    D54-55,111 V55
MIXEES S12,22,13
TURBIN S13-14    D62-69,110 V63
DUCTER S14-15    D70-73     R107
NOZDIV S15,16,1  D74-75     R108
PERFOR S1,0,0    D76-79,104,100,103,108,0,106,0,0,107
CODEND

```

**** DATA ITEM *** HP MODE - ****

DESIGN POINT AT 6000m, Mach=1.2, THRUST=134.4 kN
 RP1=2.1, RP2=3.13, PR3=3.96, BY PASS RETIO=0.4 and TET=1800K.

```

-----
1  6000.0      !      Altitude
2  0.0         ! INTAKE ISA Deviation
3  1.2         !      Mach Number
4  -1.0        !      Pressure Recovery (default)

5  -1.0        !      Z Surge margin (default)
6  -1.0        !      PCN Corrected RPM (default)
7  2.1         ! COMPRES Pressure Ratio PR1
8  0.88        !      Isentropic Efficiency
9  0.0         !      Error switch (off)
10 1.0         !      Map Number

11 -1.0        !      Z Surge margin (default)
12 -1.0        !      PCN Corrected RPM (default)
13 3.13        ! COMPRES Pressure Ratio PR2
14 0.88        !      Isentropic Efficiency
15 1.0         !      Error switch (on)
16 1.0         !      Map Number

17 0.2857      !      Mass flow factor
18 0.0         ! PREMAS Flow loss
19 1.0         !      Pressure factor
20 0.0         !      Pressure loss

21 1.00        !      Reheat switch(off)
22 0.03        ! DUCTOR Pressure factor
23 0.9         !      Combustion efficiency
24 1000000.0   !      Dummy limiting fuel flow

25 -1.0        ! NOZDIV Switch(Fixed exit area)

```

26	-1.0	!	Throat Area
27	-1.0	!	Z Surge margin (default)
28	-1.0	!	PCN Corrected RPM (default)
29	1.2	!	COMPRES Pressure Ratio PR3
30	0.89	!	Isentropic Efficiency
31	1.0	!	Error switch (on)
32	1.0	!	Map Number
33	0.0	!	Mass flow factor(Surge Control)
34	0.0	!	PREMAS Flow loss
35	1.0	!	Pressure factor
36	0.0	!	Pressure loss
37	-1.0	!	Z Surge margin (default)
38	-1.0	!	PCN Corrected RPM (default)
39	3.3	!	COMPRES Pressure Ratio PR3
40	0.885	!	Isentropic Efficiency
41	1.0	!	Error switch (on)
42	1.0	!	Map Number
43	0.05	!	Mass flow factor(Reaction Control)
44	0.0	!	PREMAS Flow loss
45	1.0	!	Pressure factor
46	0.0	!	Pressure loss
47	0.10	!	Mass flow factor(HPT Cooling Air)
48	0.0	!	PREMAS Flow loss
49	1.0	!	Pressure factor
50	0.0	!	Pressure loss
51	0.03	!	Pressure loss factor
52	1.0	!	BURNER Combustion efficiency
53	-1.0	!	Fuel flow not specified
54	25000.0	!	AUXWORK
55	-1.0	!	TF Mass flow(default)
56	-1.0	!	CN Corrected RPM(default)
57	0.90	!	TURBIN Isentropic efficiency
58	-1.0	!	PCN-Compressor Turbine(default)
59	2.0	!	Compressor number
60	5.0	!	Map number
61	-1.0	!	Power law index(default)
62	0.0	!	AUXWORK
63	-1.0	!	TF Mass flow(default)
64	-1.0	!	CN Corrected RPM (default)
65	0.90	!	TURBIN Isentropic efficiency
66	-1.0	!	PCN-Compressor Turbine(default)
67	1.0	!	Compressor number
68	5.0	!	Map number
69	-1.0	!	Power law index(default)
70	1.0	!	Reheat switch (off)
71	0.03	!	DUCTER Pressure loss factor
72	0.9	!	Combustion efficiency
73	1000000.	!	Dummy fuel flow limit
74	-1.0	!	NOZDIV switch (fixed throat area)
75	-1.0	!	Throat Area
76	-1.0	!	Power turbine (none)
77	-1.0	!	PERFOR Propeller efficiency
78	1.0	!	Scaling index switch (on)
79	134400.0	!	Design Thrust
200	5	!	
201	-1	!	
202	12	!	ARITHY PCN(COMPRES1)=PCN(COMPRES2)
203	-1	!	
204	6	!	
210	1	!	
211	-1	!	
212	110	!	
213	-1	!	ARITHY W(TOTAL)=W(compre1)+W(compre2)
214	101	!	
215	-1	!	
216	102	!	
220	5	!	
221	-1	!	
222	38	!	ARITHY PCN(COMPRES3)=PCN(COMPRES4)
223	-1	!	

```

224 28      !
230 1        !
231 -1       !
232 111      !
233 -1       ! ARITHY W(TOTAL)=W(compre3)+W(compre4)
234 105      !
235 -1       !
236 109      !

```

```

-1
1 2 243.4    !Dummy INTAKE mass flow
11 6 1800.   !Turbine entry temperature
-1

```

The Units for this Run are as follows:-

Temperature = K Pressure = Atmospheres Length = metres
Area = sq metres Mass Flow = kg/sec Velocity = metres/sec
Force = Newtons s.f.c.(Thrust) = mg/N sec s.f.c.(Power) = mug/J
Sp. Thrust = N/kg/sec Power = Watts

***** DESIGN POINT ENGINE CALCULATIONS *****

***** AMBIENT AND INLET PARAMETERS *****

Alt. = 6000.0 I.S.A. Dev. = 0.000 Mach No. = 1.20
Etar = 0.9915 Momentum Drag = 92460.17

***** COMPRESSOR 1 PARAMETERS *****

PRSF = 0.26961E+01 ETASF = 0.10107E+01 WASF = 0.36534E+00
Z = 0.85000 PR = 2.100 ETA = 0.88000
PCN = 1.0000 CN = 1.00000 COMWK = 0.11058E+08

***** COMPRESSOR 2 PARAMETERS *****

PRSF = 0.52206E+01 ETASF = 0.10107E+01 WASF = 0.21655E+00
Z = 0.85000 PR = 3.130 ETA = 0.88000
PCN = 1.0000 CN = 1.00000 COMWK = 0.43440E+08

***** DUCT/AFTER BURNING 1 PARAMETERS *****

ETA = 0.9000 DLP = 0.2209 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****

Throat Area = 0.0572 Throat Velocity = 440.29 Throat Mach No. = 1.0000
Exit Area = 0.1410 Exit Velocity = 797.43 Exit Mach No. = 2.4277
Nozzle Coeff. = 0.9700 Gross Thrust = 53789.33

***** COMPRESSOR 3 PARAMETERS *****

PRSF = 0.49020E+00 ETASF = 0.10222E+01 WASF = 0.56198E-01
Z = 0.85000 PR = 1.200 ETA = 0.89000
PCN = 1.0000 CN = 1.00000 COMWK = 0.60827E+07

***** COMPRESSOR 4 PARAMETERS *****

PRSF = 0.56373E+01 ETASF = 0.10165E+01 WASF = 0.46344E-01
Z = 0.85000 PR = 3.300 ETA = 0.88500
PCN = 1.0000 CN = 1.00000 COMWK = 0.46681E+08

***** COMBUSTION CHAMBER PARAMETERS *****

ETASF = 0.10000E+01
ETA = 1.00000 DLP = 0.8746 WFB = 4.2938

***** TURBINE 1 PARAMETERS *****

CNSF = 0.12050E+03 ETASF = 0.10056E+01 TFSF = 0.16779E+01
DHSF = 0.57654E+04
TF = 430.916 ETA = 0.90000 CN = 2.84
AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****

CNSF = 0.10870E+03 ETASF = 0.10056E+01 TFSF = 0.81705E+00
DHSF = 0.75279E+04
TF = 430.916 ETA = 0.90000 CN = 2.84
AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****

ETA = 0.9000 DLP = 0.1161 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 2 PARAMETERS *****

Throat Area = 0.3806 Throat Velocity = 615.4 Throat Mach No. = 1.0000
Exit Area = 0.6683 Exit Velocity = 1044.2 Exit Mach No. = 2.0168
Nozzle Coeff. = 0.9790 Gross Thrust = *****

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 0.9986

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	243.068	0.46557	1.12964	243.15	321.07	379.9	*****
2	0.00000	243.068	*****	1.11999	*****	321.07	*****	*****
3	0.00000	243.068	*****	2.35199	*****	406.80	*****	*****
4	0.00000	243.068	*****	7.36172	*****	580.32	*****	*****
5	0.00000	173.624	*****	7.36172	*****	580.32	*****	*****
6	0.00000	173.624	*****	8.83407	*****	613.65	*****	*****
7	0.00000	173.624	*****	8.83407	*****	613.65	*****	*****
8	0.00000	173.624	*****	29.15242	*****	671.75	*****	*****
9	0.00000	164.943	*****	29.15242	*****	671.75	*****	*****
10	0.00000	148.448	*****	29.15242	*****	671.75	*****	*****
11	0.02889	152.736	*****	28.27785	*****	1800.00	*****	*****
12	0.02889	152.736	*****	12.14659	*****	1523.25	*****	*****
13	0.02600	169.230	*****	12.14659	*****	1464.70	*****	*****
14	0.02600	169.230	*****	3.86947	*****	1158.96	*****	*****
15	0.02600	169.230	*****	3.75339	*****	1158.96	*****	*****
16	0.02600	169.230	0.46557	3.75339	691.17	1158.96	1044.3	0.6674
17	0.00000	69.445	*****	7.36172	*****	580.32	*****	*****
18	0.00000	69.445	*****	7.14087	*****	580.32	*****	*****
19	0.00000	69.445	0.46557	7.14087	244.30	580.32	797.4	0.1408
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	8.681	*****	29.15242	*****	671.75	*****	*****
22	0.00000	16.494	*****	29.15242	*****	671.75	*****	*****

Gross Thrust = 226734.17
Momentum Drag = 92334.18
Net Thrust = 134400.00
Fuel Flow = 4.2880
s.f.c. = 31.90460
Sp. Thrust = 552.931

1 9000. : Altitude
3 1.5 : Mach number
-1 :
11 6 1720. : TET
-1 :

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 1 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

Alt. = 9000.0 I.S.A. Dev. = 0.000 Mach No. = 1.50
Eta = 0.9706 Momentum Drag = 94844.3

***** COMPRESSOR 1 PARAMETERS *****

Z = 0.88627 PR = 2.024 ETA = 0.87408
PCN = 0.9532 CN = 0.93560 DIMWK = 0.17772E+08

***** COMPRESSOR 2 PARAMETERS *****

Z = 0.83735 PR = 2.890 ETA = 0.89847
PCN = 0.9532 CN = 0.94055 DIMWK = 0.34405E+08

***** DUCT/AFTER BURNING 1 PARAMETERS *****

ETA = 0.9000 DLP = 0.1899 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****

Throat Area = 0.0572 Throat Velocity = 439.45 Throat Mach No. = 1.0000
Exit Area = 0.1410 Exit Velocity = 792.40 Exit Mach No. = 2.4279
Nozzle Coeff. = 0.9645 Gross Thrust = 47192.49

***** COMPRESSOR 3 PARAMETERS *****

Z = 0.83658 PR = 1.193 ETA = 0.89982
PCN = 0.9869 CN = 0.98882 DIMWK = 0.49581E+07

***** COMPRESSOR 4 PARAMETERS *****

Z = 0.82990 PR = 3.208 ETA = 0.89580

PCN = 0.9869 CN = 0.59995 COMWK = 0.39704E+08
 ***** COMBUSTION CHAMBER PARAMETERS *****
 ETA = 1.00000 DLP = 0.7547 WFB = 3.3670
 ***** TURBINE 1 PARAMETERS *****
 TF = 433.042 ETA = 0.89909 CN = 2.769
 AUXWK = 0.25000E+05
 ***** TURBINE 2 PARAMETERS *****
 TF = 432.780 ETA = 0.89904 CN = 2.771
 AUXWK = 0.00000E+00
 ***** DUCT/AFTER BURNING 2 PARAMETERS *****
 ETA = 0.9000 DLP = 0.1968 WFB = 0.0000
 ***** CONVERGENT/DIVERGENT NOZZLE 2 PARAMETERS *****
 Throat Area = 0.3806 Throat Velocity = 601.63 Throat Mach No. = 1.0000
 Exit Area = 0.6683 Exit Velocity = 1008.68 Exit Mach No. = 2.0211
 Nozzle Coeff. = 0.9748 Gross Thrust = *****

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 0.9986

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	207.757	0.30834	1.11433	229.65	333.23	455.9	*****
2	0.00000	207.757	*****	1.08154	*****	333.23	*****	*****
3	0.00000	208.224	*****	2.18886	*****	417.78	*****	*****
4	0.00000	208.224	*****	6.32670	*****	578.10	*****	*****
5	0.00000	148.391	*****	6.32670	*****	578.10	*****	*****
6	0.00000	148.391	*****	7.54885	*****	609.90	*****	*****
7	0.00000	148.271	*****	7.54885	*****	609.90	*****	*****
8	0.00000	148.271	*****	24.21682	*****	856.91	*****	*****
9	0.00000	140.857	*****	24.21682	*****	856.91	*****	*****
10	0.00000	126.772	*****	24.21682	*****	856.91	*****	*****
11	0.02652	130.134	*****	23.46210	*****	1720.00	*****	*****
12	0.02652	130.134	*****	10.04317	*****	1452.32	*****	*****
13	0.02387	144.220	*****	10.04317	*****	1398.42	*****	*****
14	0.02387	144.220	*****	3.21356	*****	1104.99	*****	*****
15	0.02387	144.220	*****	3.11677	*****	1104.99	*****	*****
16	0.02387	144.220	0.36907	3.11677	640.31	1104.99	1008.7	0.6674
17	0.00000	59.819	*****	6.32670	*****	578.10	*****	*****
18	0.00000	59.819	*****	6.13675	*****	578.10	*****	*****
19	0.00000	59.819	0.40111	6.13675	264.88	578.10	792.4	0.1408
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	7.414	*****	24.21682	*****	856.91	*****	*****
22	0.00000	14.086	*****	24.21682	*****	856.91	*****	*****

Gross Thrust = 194735.95
 Momentum Drag = 94715.12
 Net Thrust = 100020.82
 Fuel Flow = 3.3624
 s.f.c. = 33.61720
 Sp. Thrust = 481.431

 1 9000.0 ! Altitude
 3 1.5 ! Mach number
 -1 !
 11 6 1606. ! TET
 -1 !

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 1 Loops *****

***** AMBIENT AND INLET PARAMETERS *****
 Alt. = 9000.0 I.S.A. Dev. = 0.000 Mach No. = 1.50
 Etar = 0.9706 Momentum Drag = 85502.95
 ***** COMPRESSOR 1 PARAMETERS *****
 Z = 0.91806 PR = 1.952 ETA = 0.86832
 PCN = 0.8937 CN = 0.8725 COMWK = 0.15217E+08
 ***** COMPRESSOR 2 PARAMETERS *****
 Z = 0.83682 PR = 2.707 ETA = 0.90725
 PCN = 0.8937 CN = 0.86038 COMWK = 0.28218E+08
 ***** DUCT/AFTER BURNING 1 PARAMETERS *****

ETA = 0.9000 DLP = 0.1704 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****

Throat Area = 0.0572 Throat Velocity = 432.57 Throat Mach No. = 1.0000
Exit Area = 0.1410 Exit Velocity = 779.84 Exit Mach No. = 2.4280
Nozzle Coeff. = 0.9679 Gross Thrust = 42186.61

***** COMPRESSOR 3 PARAMETERS *****

Z = 0.83418 PR = 1.186 ETA = 0.90500
PCN = 0.9515 CN = 0.96870 COMWK = 0.41298E+07

***** COMPRESSOR 4 PARAMETERS *****

Z = 0.80653 PR = 3.076 ETA = 0.89456
PCN = 0.9515 CN = 0.97072 COMWK = 0.33045E+08

***** COMBUSTION CHAMBER PARAMETERS *****

ETA = 1.00000 DLP = 0.6745 WFB = 2.6948

***** TURBINE 1 PARAMETERS *****

TF = 435.242 ETA = 0.89835 CN = 2.687
AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****

TF = 434.623 ETA = 0.89806 CN = 2.690
AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****

ETA = 0.9000 DLP = 0.0837 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 2 PARAMETERS *****

Throat Area = 0.3806 Throat Velocity = 580.74 Throat Mach No. = 1.0000
Exit Area = 0.6683 Exit Velocity = 971.94 Exit Mach No. = 2.0256
Nozzle Coeff. = 0.9775 Gross Thrust = *****

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 0.9986

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	187.295	0.30334	1.11433	229.65	333.23	455.9	*****
2	0.00000	187.295	*****	1.08154	*****	333.23	*****	*****
3	0.00000	187.435	*****	2.11092	*****	413.56	*****	*****
4	0.00000	187.435	*****	5.71434	*****	559.93	*****	*****
5	0.00000	132.951	*****	5.71434	*****	559.93	*****	*****
6	0.00000	132.951	*****	6.77743	*****	589.62	*****	*****
7	0.00000	132.910	*****	6.77743	*****	589.62	*****	*****
8	0.00000	132.910	*****	20.84653	*****	820.41	*****	*****
9	0.00000	126.264	*****	20.84653	*****	820.41	*****	*****
10	0.00000	113.638	*****	20.84653	*****	820.41	*****	*****
11	0.02368	116.329	*****	20.17205	*****	1606.00	*****	*****
12	0.02368	116.329	*****	8.63997	*****	1353.07	*****	*****
13	0.02131	128.956	*****	8.63997	*****	1304.45	*****	*****
14	0.02131	128.956	*****	2.76053	*****	1026.91	*****	*****
15	0.02131	128.956	*****	2.67681	*****	1026.91	*****	*****
16	0.02131	128.956	0.33176	2.67681	588.52	1026.91	971.9	0.6674
17	0.00000	54.709	*****	5.71434	*****	559.93	*****	*****
18	0.00000	54.709	*****	5.54396	*****	559.93	*****	*****
19	0.00000	54.709	0.36183	5.54396	256.51	559.93	779.8	0.1408
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	6.645	*****	20.84653	*****	820.41	*****	*****
22	0.00000	12.626	*****	20.84653	*****	820.41	*****	*****

Gross Thrust = 166566.94
Momentum Drag = 85386.44
Net Thrust = 81180.49
Fuel Flow = 2.6911
s.f.c. = 33.15009
Sp. Thrust = 433.437

1 11000. ! Altitude
3 1.6 ! Mach number
-1 !
11 6 1590. ! TET
-1 !

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 1 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

Alt. = 11000.0 I.S.A. Dev. = 0.000 Mach No. = 1.60
 Etar = 0.9624 Momentum Drag = 75961.74

***** COMPRESSOR 1 PARAMETERS *****

Z = 0.91563 PR = 1.955 ETA = 0.86923
 PCN = 0.8899 CN = 0.88067 COMWK = 0.12861E+08

***** COMPRESSOR 2 PARAMETERS *****

Z = 0.83736 PR = 2.718 ETA = 0.90679
 PCN = 0.8899 CN = 0.88964 COMWK = 0.23925E+08

***** DUCT/AFTER BURNING 1 PARAMETERS *****

ETA = 0.9000 DLP = 0.1457 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****

Throat Area = 0.0572 Throat Velocity = 429.57 Throat Mach No. = 1.0000
 Exit Area = 0.1410 Exit Velocity = 774.36 Exit Mach No. = 2.4280
 Nozzle Coeff. = 0.9639 Gross Thrust = 36325.79

***** COMPRESSOR 3 PARAMETERS *****

Z = 0.83531 PR = 1.186 ETA = 0.90450
 PCN = 0.9442 CN = 0.96808 COMWK = 0.34831E+07

***** COMPRESSOR 4 PARAMETERS *****

Z = 0.80735 PR = 3.075 ETA = 0.89491
 PCN = 0.9442 CN = 0.97003 COMWK = 0.27854E+08

***** COMBUSTION CHAMBER PARAMETERS *****

ETA = 1.00000 DLP = 0.5724 WFB = 2.2617

***** TURBINE 1 PARAMETERS *****

TF = 434.934 ETA = 0.89821 CN = 2.689
 AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****

TF = 434.663 ETA = 0.89812 CN = 2.692
 AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****

ETA = 0.9000 DLP = 0.0711 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 2 PARAMETERS *****

Throat Area = 0.3806 Throat Velocity = 577.77 Throat Mach No. = 1.0000
 Exit Area = 0.6683 Exit Velocity = 966.75 Exit Mach No. = 2.0262
 Nozzle Coeff. = 0.9749 Gross Thrust = *****

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 0.9986

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	160.608	0.22330	0.94970	216.65	327.86	472.3	*****
2	0.00000	160.608	*****	0.91396	*****	327.86	*****	*****
3	0.00000	160.607	*****	1.78679	*****	407.08	*****	*****
4	0.00000	160.607	*****	4.85613	*****	552.10	*****	*****
5	0.00000	113.627	*****	4.85613	*****	552.10	*****	*****
6	0.00000	113.627	*****	5.75979	*****	581.45	*****	*****
7	0.00000	113.628	*****	5.75979	*****	581.45	*****	*****
8	0.00000	113.628	*****	17.71420	*****	809.47	*****	*****
9	0.00000	107.947	*****	17.71420	*****	809.47	*****	*****
10	0.00000	97.152	*****	17.71420	*****	809.47	*****	*****
11	0.02345	99.431	*****	17.14179	*****	1590.00	*****	*****
12	0.02345	99.431	*****	7.36565	*****	1340.08	*****	*****
13	0.02111	110.225	*****	7.36565	*****	1291.64	*****	*****
14	0.02111	110.225	*****	2.35079	*****	1016.08	*****	*****
15	0.02111	110.225	*****	2.27972	*****	1016.08	*****	*****
16	0.02111	110.225	0.28169	2.27972	581.53	1016.08	966.8	0.6674
17	0.00000	46.974	*****	4.85613	*****	552.10	*****	*****
18	0.00000	46.974	*****	4.71039	*****	552.10	*****	*****
19	0.00000	46.974	0.30846	4.71039	252.90	552.10	774.4	0.1408
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	5.681	*****	17.71420	*****	809.47	*****	*****
22	0.00000	10.795	*****	17.71420	*****	809.47	*****	*****

Gross Thrust = 144112.45
 Momentum Drag = 75858.23
 Net Thrust = 68254.22
 Fuel Flow = 2.2786
 s.f.c. = 33.38453
 Sp. Thrust = 424.973

```

1 6000.      ! Altitude
3 0.9        ! Mach number
-1           !
11 6 1685.   ! TET
-1           !

```

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 10 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

```

Alt. = 6000.0      I.S.A. Dev. = 0.000      Mach No. = 0.90
Eta_r = 1.0000     Momentum Drag = 52540.00

```

***** COMPRESSOR 1 PARAMETERS *****

```

Z = 0.84503      PR = 2.124      ETA = 0.87370
PCN = 0.9619     CN = 1.01272    COMWK = 0.14744E+08

```

***** COMPRESSOR 2 PARAMETERS *****

```

Z = 0.85419      PR = 3.186      ETA = 0.87112
PCN = 0.9619     CN = 1.00979    COMWK = 0.30740E+08

```

***** DUCT/AFTER BURNING 1 PARAMETERS *****

```

ETA = 0.9000     DLP = 0.1599    WFB = 0.0000

```

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****

```

Throat Area = 0.0572      Throat Velocity = 421.91      Throat Mach No. = 1.0000
Exit Area = 0.1410       Exit Velocity = 760.41      Exit Mach No. = 2.4281
Nozzle Coeff. = 0.9730    Gross Thrust = 37070.83

```

***** COMPRESSOR 3 PARAMETERS *****

```

Z = 0.85472      PR = 1.203      ETA = 0.88476
PCN = 0.9615     CN = 1.00380    COMWK = 0.43168E+07

```

***** COMPRESSOR 4 PARAMETERS *****

```

Z = 0.85449      PR = 3.327      ETA = 0.88046
PCN = 0.9615     CN = 1.00298    COMWK = 0.34480E+08

```

***** COMBUSTION CHAMBER PARAMETERS *****

```

ETA = 1.00000     DLP = 0.6384    WFB = 3.0163

```

***** TURBINE 1 PARAMETERS *****

```

TF = 431.377      ETA = 0.89975      CN = 2.824
AUXWK = 0.25000E+05

```

***** TURBINE 2 PARAMETERS *****

```

TF = 431.279      ETA = 0.89981      CN = 2.828
AUXWK = 0.00000E+00

```

***** DUCT/AFTER BURNING 2 PARAMETERS *****

```

ETA = 0.9000     DLP = 0.0847    WFB = 0.0000

```

***** CONVERGENT/DIVERGENT NOZZLE 2 PARAMETERS *****

```

Throat Area = 0.3806      Throat Velocity = 594.59      Throat Mach No. = 1.0000
Exit Area = 0.6683       Exit Velocity = 996.46      Exit Mach No. = 2.0223
Nozzle Coeff. = 0.9723    Gross Thrust = *****

```

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 0.9986

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	184.163	0.46557	0.78773	249.15	289.63	284.9	*****
2	0.00000	184.163	*****	0.78773	*****	289.63	*****	*****
3	0.00000	184.113	*****	1.67291	*****	369.11	*****	*****
4	0.00000	184.113	*****	5.32972	*****	532.40	*****	*****
5	0.00000	131.776	*****	5.32972	*****	532.40	*****	*****
6	0.00000	131.776	*****	6.41041	*****	563.89	*****	*****
7	0.00000	131.808	*****	6.41041	*****	563.89	*****	*****
8	0.00000	131.808	*****	21.32791	*****	807.75	*****	*****
9	0.00000	125.217	*****	21.32791	*****	807.75	*****	*****
10	0.00000	112.696	*****	21.32791	*****	807.75	*****	*****
11	0.02673	115.708	*****	20.68954	*****	1685.00	*****	*****
12	0.02673	115.708	*****	8.88267	*****	1422.83	*****	*****
13	0.02406	128.230	*****	8.88267	*****	1367.32	*****	*****
14	0.02406	128.230	*****	2.83371	*****	1078.63	*****	*****
15	0.02406	128.230	*****	2.74900	*****	1078.63	*****	*****
16	0.02406	128.230	0.34086	2.74900	623.33	1078.63	996.5	0.6674
17	0.00000	52.488	*****	5.32972	*****	532.40	*****	*****
18	0.00000	52.488	*****	5.16985	*****	532.40	*****	*****
19	0.00000	52.488	0.33839	5.16985	243.84	532.40	760.4	0.1408

20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	6.590	*****	21.32791	*****	807.75	*****	*****
22	0.00000	12.522	*****	21.32791	*****	807.75	*****	*****

Gross Thrust = 152824.08
Momentum Drag = 52468.40
Net Thrust = 100355.68
Fuel Flow = 3.0122
s.f.c. = 30.01502
Sp. Thrust = 544.928

1 6000. ! Altitude
3 0.9 ! Mach number
-1 !
11 6 1775. ! TET
-1 !

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 1 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

Alt. = 6000.0 I.S.A. Dev. = 0.000 Mach No. = 0.90
Etar = 1.0000 Momentum Drag = 55002.45

***** COMPRESSOR 1 PARAMETERS *****

Z = 0.82913 PR = 2.165 ETA = 0.85793
PCN = 0.9873 CN = 1.03946 COMWK = 0.16164E+08

***** COMPRESSOR 2 PARAMETERS *****

Z = 0.85940 PR = 3.299 ETA = 0.85428
PCN = 0.9873 CN = 1.03124 COMWK = 0.34356E+08

***** DUCT/AFTER BURNING 1 PARAMETERS *****

ETA = 0.9000 DLP = 0.1687 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****

Throat Area = 0.0572 Throat Velocity = 427.50 Throat Mach No. = 1.0000
Exit Area = 0.1410 Exit Velocity = 770.59 Exit Mach No. = 2.4280
Nozzle Coeff. = 0.9726 Gross Thrust = 39482.93

***** COMPRESSOR 3 PARAMETERS *****

Z = 0.86241 PR = 1.207 ETA = 0.87636
PCN = 0.9803 CN = 1.00991 COMWK = 0.47905E+07

***** COMPRESSOR 4 PARAMETERS *****

Z = 0.86653 PR = 3.387 ETA = 0.87015
PCN = 0.9803 CN = 1.00832 COMWK = 0.38243E+08

***** COMBUSTION CHAMBER PARAMETERS *****

ETA = 1.00000 DLP = 0.6736 WFB = 3.4336

***** TURBINE 1 PARAMETERS *****

TF = 431.626 ETA = 0.89976 CN = 2.824
AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****

TF = 431.292 ETA = 0.89979 CN = 2.826
AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****

ETA = 0.9000 DLP = 0.0918 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 2 PARAMETERS *****

Throat Area = 0.3806 Throat Velocity = 610.70 Throat Mach No. = 1.0000
Exit Area = 0.6683 Exit Velocity = 1024.83 Exit Mach No. = 2.0187
Nozzle Coeff. = 0.9765 Gross Thrust = *****

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 0.9986

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	192.794	0.46557	0.78773	249.15	289.63	284.9	*****
2	0.00000	192.794	*****	0.78773	*****	289.63	*****	*****
3	0.00000	192.930	*****	1.70523	*****	372.85	*****	*****
4	0.00000	192.930	*****	5.62614	*****	546.74	*****	*****
5	0.00000	138.183	*****	5.62614	*****	546.74	*****	*****
6	0.00000	138.183	*****	6.79231	*****	579.96	*****	*****

7	0.00000	138.170	*****	6.79231	*****	579.96	*****	*****
8	0.00000	138.170	*****	23.00552	*****	836.69	*****	*****
9	0.00000	131.262	*****	23.00552	*****	836.69	*****	*****
10	0.00000	118.136	*****	23.00552	*****	836.69	*****	*****
11	0.02903	121.565	*****	22.33189	*****	1775.00	*****	*****
12	0.02903	121.565	*****	9.57693	*****	1501.29	*****	*****
13	0.02612	134.691	*****	9.57693	*****	1441.69	*****	*****
14	0.02612	134.691	*****	3.04952	*****	1140.10	*****	*****
15	0.02612	134.691	*****	2.95773	*****	1140.10	*****	*****
16	0.02612	134.691	0.37129	2.95773	664.74	1140.10	1024.8	0.6674
17	0.00000	54.670	*****	5.62614	*****	546.74	*****	*****
18	0.00000	54.670	*****	5.45740	*****	546.74	*****	*****
19	0.00000	54.670	0.35723	5.45740	250.43	546.74	770.6	0.1408
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	6.909	*****	23.00552	*****	836.69	*****	*****
22	0.00000	13.126	*****	23.00552	*****	836.69	*****	*****

Gross Thrust = 167850.80
Momentum Drag = 54927.50
Net Thrust = 112923.31
Fuel Flow = 3.4289
s.f.c. = 30.36498
Sp. Thrust = 585.719

1 0.0 ! Altitude
3 0.9 ! Mach number
-1 !
11 6 1778. ! TET
-1 !

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 2 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

Alt. = 0.0 I.S.A. Dev. = 0.000 Mach No. = 0.90
Etar = 1.0000 Momentum Drag = 103386.45

***** COMPRESSOR 1 PARAMETERS *****

Z = 0.87291 PR = 2.052 ETA = 0.87598
PCN = 0.9791 CN = 0.95877 COMWK = 0.29536E+08

***** COMPRESSOR 2 PARAMETERS *****

Z = 0.84242 PR = 2.977 ETA = 0.89166
PCN = 0.9791 CN = 0.96198 COMWK = 0.58457E+08

***** DUCT/AFTER BURNING 1 PARAMETERS *****

ETA = 0.9000 DLP = 0.3101 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****

Throat Area = 0.0572 Throat Velocity = 443.68 Throat Mach No. = 1.0000
Exit Area = 0.1410 Exit Velocity = 800.12 Exit Mach No. = 2.4278
Nozzle Coeff. = 0.9724 Gross Thrust = 70466.30

***** COMPRESSOR 3 PARAMETERS *****

Z = 0.84162 PR = 1.195 ETA = 0.89639
PCN = 0.9996 CN = 0.99182 COMWK = 0.83057E+07

***** COMPRESSOR 4 PARAMETERS *****

Z = 0.83747 PR = 3.238 ETA = 0.89354
PCN = 0.9996 CN = 0.99265 COMWK = 0.66416E+08

***** COMBUSTION CHAMBER PARAMETERS *****

ETA = 1.00000 DLP = 1.2277 WFB = 5.7585

***** TURBINE 1 PARAMETERS *****

TF = 432.065 ETA = 0.89940 CN = 2.798
AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****

TF = 432.044 ETA = 0.89939 CN = 2.797
AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****

ETA = 0.9000 DLP = 0.1598 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 2 PARAMETERS *****

Throat Area = 0.3806 Throat Velocity = 612.04 Throat Mach No. = 1.0000
Exit Area = 0.6683 Exit Velocity = 1027.03 Exit Mach No. = 2.0188

Nozzle Coeff. = 0.9615 Gross Thrust = *****

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 0.9986

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	337.057	1.00000	1.69166	288.15	334.85	306.3	*****
2	0.00000	337.057	*****	1.69166	*****	334.85	*****	*****
3	0.00000	337.195	*****	3.47134	*****	421.44	*****	*****
4	0.00000	337.195	*****	10.33505	*****	589.41	*****	*****
5	0.00000	240.379	*****	10.33505	*****	589.41	*****	*****
6	0.00000	240.379	*****	12.35378	*****	622.21	*****	*****
7	0.00000	240.368	*****	12.35378	*****	622.21	*****	*****
8	0.00000	240.368	*****	40.00554	*****	876.20	*****	*****
9	0.00000	228.350	*****	40.00554	*****	876.20	*****	*****
10	0.00000	205.515	*****	40.00554	*****	876.20	*****	*****
11	0.02798	211.265	*****	38.77783	*****	1778.00	*****	*****
12	0.02798	211.265	*****	16.66507	*****	1504.15	*****	*****
13	0.02518	234.100	*****	16.66507	*****	1447.54	*****	*****
14	0.02518	234.100	*****	5.31586	*****	1145.03	*****	*****
15	0.02518	234.100	*****	5.15611	*****	1145.03	*****	*****
16	0.02518	234.100	0.64649	5.15611	667.36	1145.03	1027.0	0.6474
17	0.00000	96.741	*****	10.33505	*****	589.41	*****	*****
18	0.00000	96.741	*****	10.02499	*****	589.41	*****	*****
19	0.00000	96.741	0.65677	10.02499	270.11	589.41	800.1	0.1408
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	12.018	*****	40.00554	*****	876.20	*****	*****
22	0.00000	22.835	*****	40.00554	*****	876.20	*****	*****

Gross Thrust = 277628.53
 Momentum Drag = 103245.57
 Net Thrust = 174382.98
 Fuel Flow = 5.7507
 s.f.c. = 32.97732
 Sp. Thrust = 517.369

 1 0.0 ! Altitude
 3 1.1 ! Mach number
 -1 !
 11 6 1675. ! TET
 -1 !

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 3 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

Alt. = 0.0 I.S.A. Dev. = 0.000 Mach No. = 1.10
 Eta = 0.9966 Momentum Drag = 128844.70

***** COMPRESSOR 1 PARAMETERS *****

Z = 0.92511 PR = 1.925 ETA = 0.84451
 PCN = 0.9048 CN = 0.85705 COMWK = 0.29356E+08

***** COMPRESSOR 2 PARAMETERS *****

Z = 0.83165 PR = 2.642 ETA = 0.91083
 PCN = 0.9048 CN = 0.86800 COMWK = 0.53515E+08

***** DUCT/AFTER BURNING 1 PARAMETERS *****

ETA = 0.9000 DLP = 0.3243 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****

Throat Area = 0.0572 Throat Velocity = 444.86 Throat Mach No. = 1.0000
 Exit Area = 0.1410 Exit Velocity = 802.28 Exit Mach No. = 2.4178
 Nozzle Coeff. = 0.9729 Gross Thrust = 74472.16

***** COMPRESSOR 3 PARAMETERS *****

Z = 0.83032 PR = 1.183 ETA = 0.90092
 PCN = 0.9718 CN = 0.96168 COMWK = 0.78437E+07

***** COMPRESSOR 4 PARAMETERS *****

Z = 0.79946 PR = 3.035 ETA = 0.89423
 PCN = 0.9718 CN = 0.96430 COMWK = 0.62843E+08

***** COMBUSTION CHAMBER PARAMETERS *****

ETA = 1.00000 DLP = 1.2673 WFB = 5.1630

***** TURBINE 1 PARAMETERS *****

TF = 435.631 ETA = 0.89804 CN = 2.664
AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****
TF = 435.253 ETA = 0.89778 CN = 2.663
AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****
ETA = 0.9000 DLP = 0.1564 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 2 PARAMETERS *****
Throat Area = 0.3806 Throat Velocity = 594.22 Throat Mach No. = 1.0000
Exit Area = 0.6683 Exit Velocity = 995.55 Exit Mach No. = 2.0230
Nozzle Coeff. = 0.9583 Gross Thrust = *****

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 0.9986

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	343.682	1.00000	2.13583	288.15	357.85	374.4	*****
2	0.00000	343.682	*****	2.12867	*****	357.85	*****	*****
3	0.00000	343.682	*****	4.09730	*****	442.04	*****	*****
4	0.00000	343.682	*****	10.82595	*****	592.58	*****	*****
5	0.00000	243.017	*****	10.82595	*****	592.58	*****	*****
6	0.00000	243.017	*****	12.80593	*****	623.21	*****	*****
7	0.00000	242.734	*****	12.80593	*****	623.21	*****	*****
8	0.00000	242.734	*****	38.86014	*****	861.55	*****	*****
9	0.00000	230.597	*****	38.86014	*****	861.55	*****	*****
10	0.00000	207.538	*****	38.86014	*****	861.55	*****	*****
11	0.02484	212.693	*****	37.59280	*****	1675.00	*****	*****
12	0.02484	212.693	*****	16.15419	*****	1414.14	*****	*****
13	0.02236	235.753	*****	16.15419	*****	1363.82	*****	*****
14	0.02236	235.753	*****	5.17913	*****	1076.82	*****	*****
15	0.02236	235.753	*****	5.02274	*****	1076.82	*****	*****
16	0.02236	235.753	0.12504	5.02274	621.12	1076.82	995.6	0.6674
17	0.00000	100.994	*****	10.82595	*****	592.58	*****	*****
18	0.00000	100.994	*****	10.50161	*****	592.58	*****	*****
19	0.00000	100.994	0.12753	10.50161	271.58	592.58	802.3	0.1408
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	12.137	*****	38.86014	*****	861.55	*****	*****
22	0.00000	23.060	*****	38.86014	*****	861.55	*****	*****

Gross Thrust = 273944.41
Momentum Drag = 128669.12
Net Thrust = 145275.30
Fuel Flow = 5.1559
s.f.c. = 35.49080
Sp. Thrust = 422.703

1 9000. ! Altitude
3 1.4 ! Mach number
70 2.0 ! Reheat switch (on)
74 1.0 ! Nozzle area (open)
9 -1.0 ! Compressor error switch (on)
-1 !
11 6 1750. ! TET
15 6 1550. ! Nozzle inlet temperature
-1 !

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 3 Loops *****

***** AMBIENT AND INLET PARAMETERS *****
Alt. = 9000.0 I.S.A. Dev. = 0.000 Mach No. = 1.40
Eta_r = 0.9782 Momentum Drag = 72489.92

***** COMPRESSOR 1 PARAMETERS *****
Z = 0.90581 PR = 1.951 ETA = 0.87371
PCN = 0.8831 CN = 0.38469 COMWK = 0.13186E+08

***** COMPRESSOR 2 PARAMETERS *****
Z = 0.82304 PR = 1.702 ETA = 0.91000
PCN = 0.8831 CN = 0.39432 COMWK = 0.24465E+08

***** DUCT/AFTER BURNING 1 PARAMETERS *****
ETA = 0.9000 DLP = 0.1491 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****

Throat Area = 0.0572 Throat Velocity = 423.67 Throat Mach No. = 1.0000
 Exit Area = 0.1410 Exit Velocity = 763.60 Exit Mach No. = 2.4281
 Nozzle Coeff. = 0.9700 Gross Thrust = 36373.60

***** COMPRESSOR 3 PARAMETERS *****

Z = 0.86010 PR = 1.202 ETA = 0.88440
 PCN = 0.9610 CN = 0.99913 COMWK = 0.40010E+07

***** COMPRESSOR 4 PARAMETERS *****

Z = 0.85834 PR = 3.316 ETA = 0.88058
 PCN = 0.9610 CN = 0.99842 COMWK = 0.31953E+08

***** COMBUSTION CHAMBER PARAMETERS *****

ETA = 1.00000 DLP = 0.5871 WFB = 3.0003

***** TURBINE 1 PARAMETERS *****

TF = 436.562 ETA = 0.89429 CN = 2.544
 AUXWK = 0.25000E+05

***** TURBINE 2 PARAMETERS *****

TF = 424.836 ETA = 0.89613 CN = 2.539
 AUXWK = 0.00000E+00

***** DUCT/AFTER BURNING 2 PARAMETERS *****

ETA = 0.9000 DLP = 0.0687 WFB = 1.5891

***** CONVERGENT/DIVERGENT NOZZLE 2 PARAMETERS *****

Throat Area = 0.3722 Throat Velocity = 708.25 Throat Mach No. = 1.0000
 Exit Area = 0.7744 Exit Velocity = 1273.67 Exit Mach No. = 2.1777
 Nozzle Coeff. = 0.9750 Gross Thrust = *****

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 0.9986

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	170.132	0.30334	0.96593	229.65	319.91	425.5	*****
2	0.00000	170.132	*****	0.94490	*****	319.91	*****	*****
3	0.00000	170.236	*****	1.84390	*****	396.66	*****	*****
4	0.00000	170.236	*****	4.98238	*****	536.88	*****	*****
5	0.00000	121.455	*****	4.98238	*****	536.88	*****	*****
6	0.00000	121.455	*****	5.98924	*****	568.51	*****	*****
7	0.00000	121.573	*****	5.98924	*****	568.51	*****	*****
8	0.00000	121.573	*****	19.86298	*****	813.25	*****	*****
9	0.00000	115.495	*****	19.86298	*****	813.25	*****	*****
10	0.00000	103.945	*****	19.86298	*****	813.25	*****	*****
11	0.02882	106.942	*****	19.27588	*****	1750.00	*****	*****
12	0.02882	106.942	*****	8.50403	*****	1489.55	*****	*****
13	0.02594	118.491	*****	8.50403	*****	1428.94	*****	*****
14	0.02594	118.491	*****	3.24775	*****	1173.74	*****	*****
15	0.04036	120.078	*****	3.17908	*****	1550.00	*****	*****
16	0.04036	120.078	0.30334	3.17908	904.95	1550.00	1273.7	0.7733
17	0.00000	48.798	*****	4.98238	*****	536.88	*****	*****
18	0.00000	48.798	*****	4.83332	*****	536.88	*****	*****
19	0.00000	48.798	0.31594	4.83332	245.90	536.88	763.6	0.1408
20	0.00000	0.000	*****	0.00000	*****	0.00	*****	*****
21	0.00000	6.079	*****	19.86298	*****	813.25	*****	*****
22	0.00000	11.549	*****	19.86298	*****	813.25	*****	*****

Gross Thrust = 185443.14
 Momentum Drag = 72391.14
 Net Thrust = 113052.00
 Fuel Flow = 4.5832
 s.f.c. = 40.54064
 Sp. Thrust = 664.496

 -3

APPENDIX C

Example: The Selective Bleed Engine

To illustrate the technique with an example, the Selective Bleed Engine has been chosen as a case study. The procedure is shown with the final results obtained after several loops of engine optimisation and component matching. The intermediate iterations to obtain the final result have been omitted. From the engine design point analysis, (this is an assessment of engine design point performance sensitivity to TET, pressure ratio and bypass ratio) it has been decided that from Chapter 2, the engines to be matched are:

a) LP mode with a bypass ratio of 1.0, pressure ratio of 26, TET of 1500K and with the design point at sea level static conditions.

b) HP mode with a bypass ratio of 0.4, pressure ratio of 26, TET of 1800K and with the design point at an altitude of 6km and a Mach number of 1.2.

Thus, the design point analysis has yielded two satisfactory engine cycles. The following exercise is carried out to assess the capability of the components to perform the duties demanded by the two cycles.

C.1- HP Turbine Matching

The appropriate plot of overall pressure ratio vs HP turbine inlet $\sqrt{T/P}$ is carried out for both modes. The pressure ratio selected is 26 for both LP and HP

modes. This will yield the appropriate ratio of mass flows at a value of 0.948 (equation 3.3, Chapter 3) to ensure equality of the HP turbine non-dimensional mass flow (Figure C.1).

C.2- LP Turbine Matching

The appropriate plots of HP compressor pressure ratio vs $\sqrt{T/P}$ at the LP turbine inlet are produced for both modes for the overall pressure ratios and TETs of section C.1 (Figure C.2). The overall pressure ratio of the HP compressor (LP mode) is now selected ($PR=3.6$) and this will yield the value of $\sqrt{T/P}$ at LP turbine inlet of 3.44. From the first step the ratio of turbine mass flows is known to be 0.984, thus the value of $\sqrt{T/P}$ at LP turbine inlet (HP mode) is calculated (equation 3.4, Chapter 3). From the appropriate curve of Figure C.2, the appropriate HP compressor pressure ratio is calculated.

After the process the HP and LP turbine non-dimensional mass flows were found to be 2.21×10^{-3} and 4.85×10^{-3} , respectively.

C.3- Compressor matching - Running lines

The result of sections C.1 and C2 is the design point for the two cycles with the matched HP and LP turbines. This information was employed in TURBOMATCH to generate the compressor running lines (Figure C.3).

The procedure on how to obtain the appropriate running lines for each compressor has been presented in Chapter 3, section 3.3.2.2. Figure C.3 is the selected conditions for compressor performance analysis carried out in Chapters 4 and 5 and it is exactly the same as Figures 2.11 and 3.7 in Chapters 2 and 3, respectively.

S.B.E. Part I(a) HPT Matching

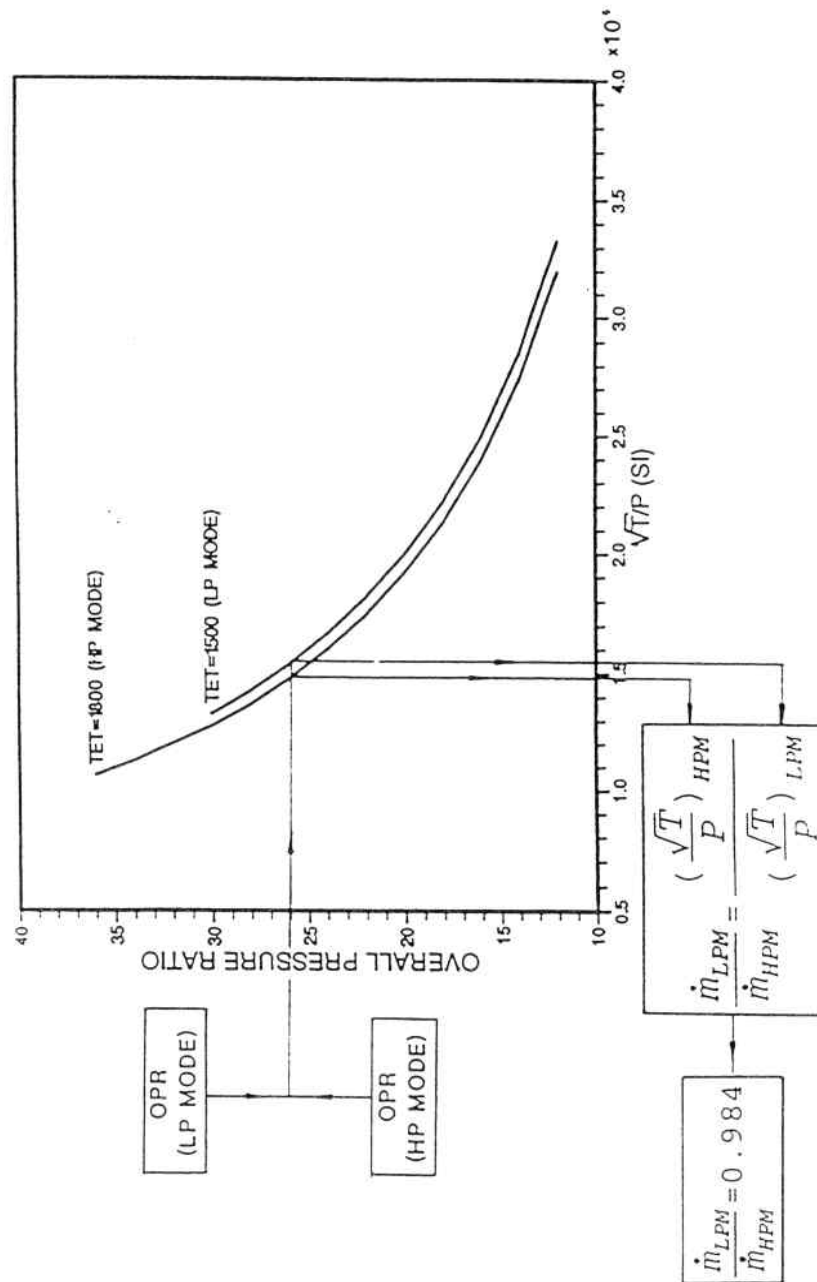


Figure C.1 HP Turbines

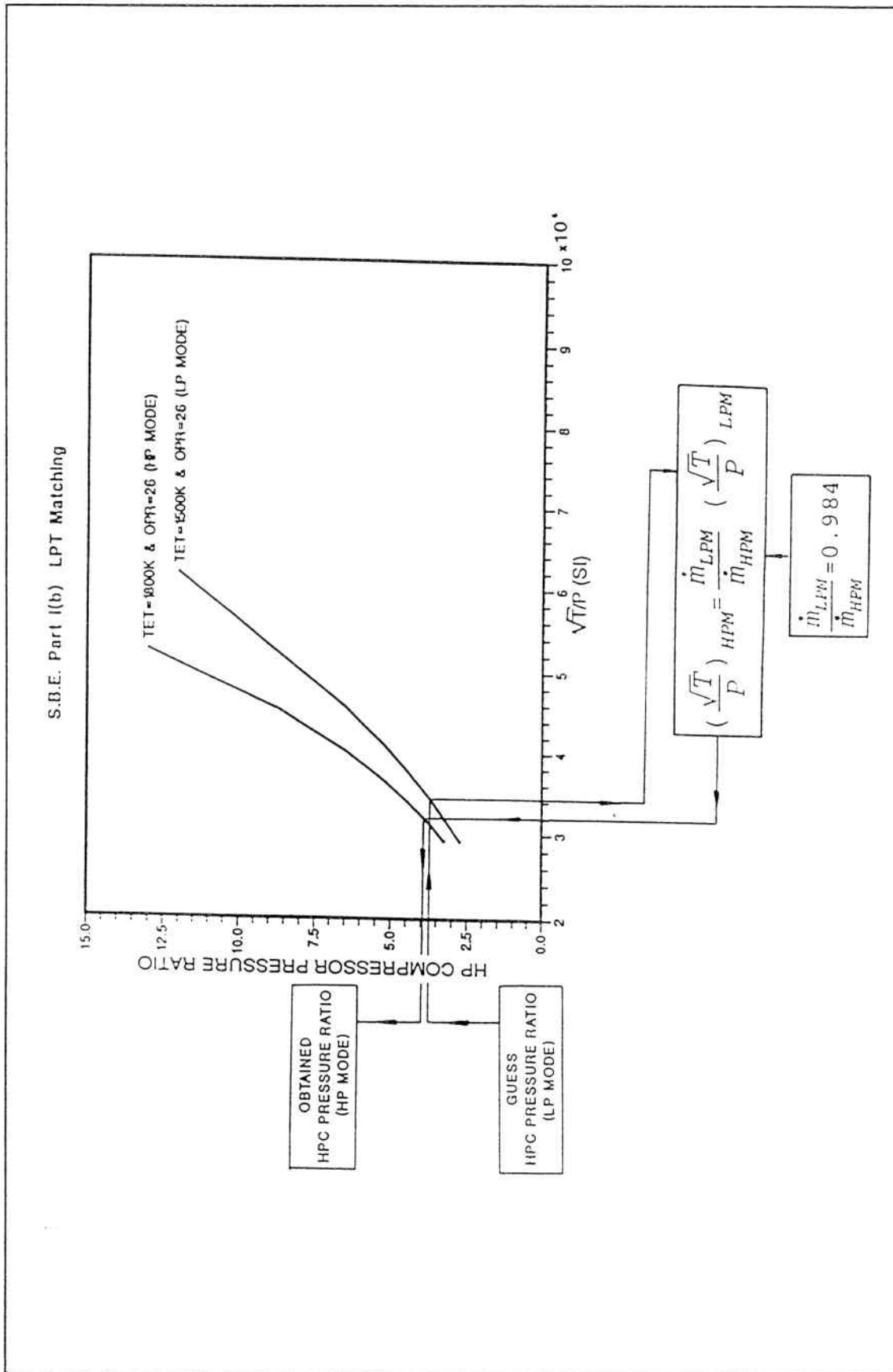


Figure C.2 LP turbines.

Matching Example

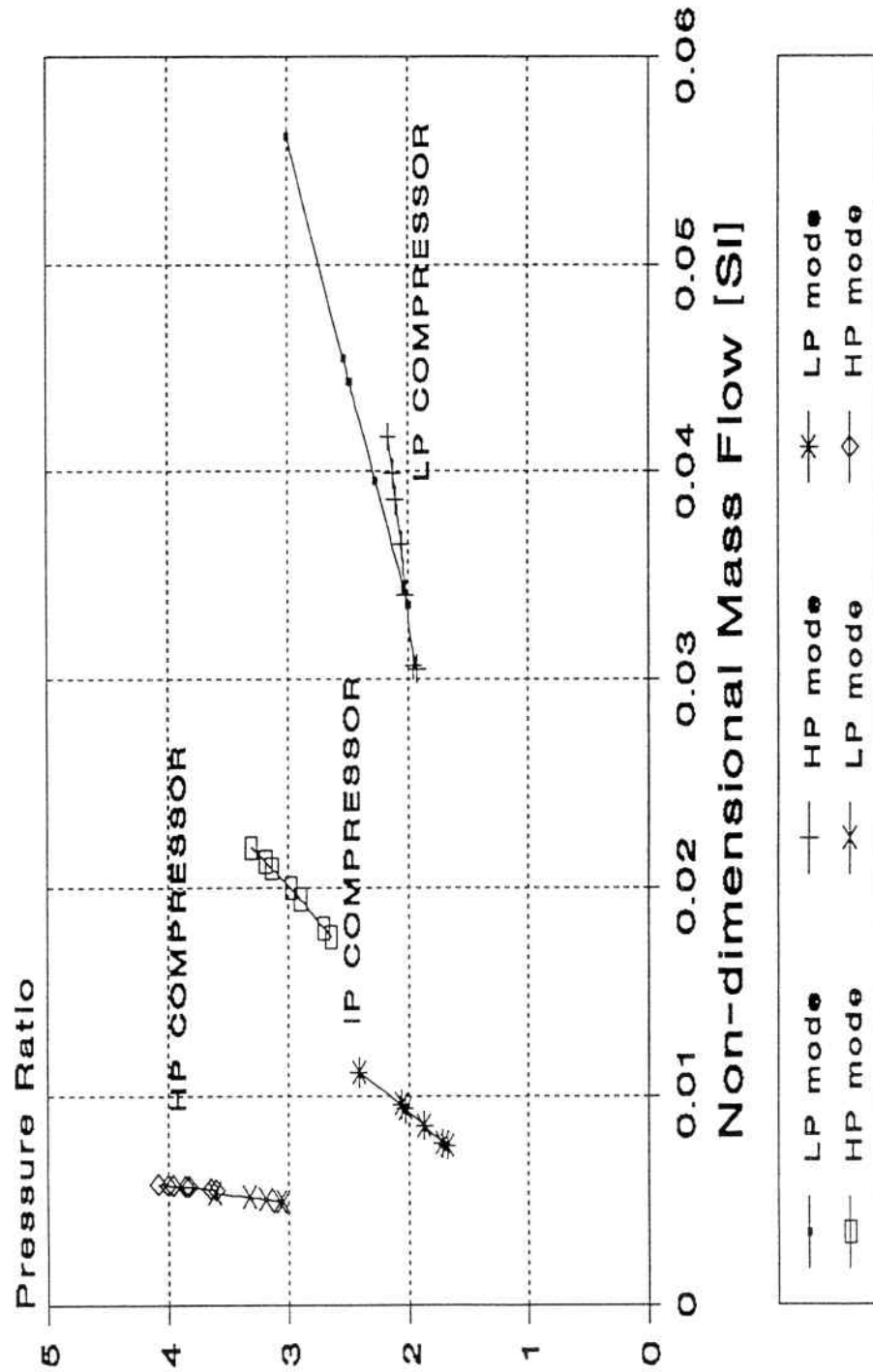


Figure C.3 Compressor Operating Lines.

APPENDIX D

Compressor Design and Off-design Simulation

D.1- Introduction

When the design and off-design performance of a compressor are required, correlation and correlation formulae are necessary in order to predict the losses and therefore, the efficiency.

A very large number of correlations (and correlation formulae) can be found in the literature.

Many of these have common points; for example, the use of a diffusion factor in estimating the minimum profile losses. On the other hand, large differences appear when it comes to estimating the effects of the various geometrical or physical variables on overall losses and deviation. This is a consequence of the complexity of the flow structure.

A survey has been carried out (in this Appendix) on some topics, to improve qualitatively the loss calculation for the existing programme. In particular:

- 1- diffusion factor,*
- 2- profile loss,*
- 3- shock loss,*
- 4- secondary losses,*

5- Reynolds number and roughness effects for design point conditions.

It was found that the objective of all correlation is to determine the losses from a set of parameters related to the blade row geometry and the flow properties. This set of parameters is small and sufficiently representative to describe all of the main effects. This will obviously require a certain amount of empiricism but the actual trends and ways for improvement rely on a better understanding of the basic flow phenomena.

As a result, correlations have been drawn and introduced in CEMCOMP, for the loss calculation procedure and the ANNULUS programme has become a subroutine of CEMCOMP. A DCA profile with low losses was chosen for blade geometry since a transonic compressor was required.

In addition, new subroutines which substitute the manual feeding of the correlations for incidence and deviation angles have been developed for the SCD15 programme as well as stator blade calculations. Due to modifications in CEMCOMP and SCD15 programmes, they have been renamed COMPDES and BLADE, respectively. As a result, a new running sequence of these computer programmes has been created.

D.2- Modification of CEMCOMP Efficiency Prediction

CEMCOMP is a computer programme for compressor aerodynamic design, which is available in the S.M.E. directory of the Cranfield VAX computer. It was developed by Dener (24) from a programme called COMPDESN which was originally written by Locket (33). CEMCOMP calculates stage air angles, velocities, diffusion parameters and losses at 5 radial sections from blade hub to tip. In addition, stage and overall efficiencies, and pressure ratios are estimated. The simple radial equilibrium approach is used by CEMCOMP to solve the algebraic equations to determine the axial velocity distribution for a prescribed distribution of whirl velocity or blade outlet flow direction. CEMCOMP includes

choice from 6 radial equilibrium solutions, namely:

- Free Vortex,*
- Forced Vortex,*
- Constant Reaction,*
- Exponential,*
- Graded Work,*
- Infinite Aspect Ratio.*

However, a close examination in CEMCOMP revealed that the losses could not satisfactorily be predicted. A small change in profile loss, shock loss, secondary loss and efficiency prediction was required in order to achieve reasonable standard of loss assessment.

The changes in CEMCOMP carried out are described in the following sections. Some simplifications were introduced as appropriate to preliminary design.

D.2.1- Blade Geometric Parameters

To carry out the analysis on the losses, it is necessary to introduce a cascade notation in order to understand the correlations that will actually be used.

Figure D.1 shows a cross-section of three blades forming a part of a typical cascade (Reference 15), which also includes details of various angles, lengths and velocities associated with cascade experiments.

D.2.2- Blade Shape

There are many different ways to specify the blades of an axial compressor. The most common way is to base the design on families of profile such as the **C-series**, **NACA-65** series and double-circular-arc (**DCA**). The latter has great advantages at transonic Mach numbers.

At low Mach numbers the effect of blade shape for **C-series**, **NACA-65** and **DCA** seems to be very small. However, at high Mach numbers the superiority of the **DCA** blade becomes apparent with a very much wider operating range and no rise in minimum loss. Further information about blade profile can be found in Reference 34. Reference 35 recommended the use of **NACA-65** blades up to inlet Mach number of 0.78 and **DCA** in the range of inlet Mach number 0.78 to 1.2.

The need to design subsonic and transonic axial flow compressors, in this work, has led to the selection of the **DCA** blade profile, for both rotor and stator blade despite very little amount of data being available for stator blades, to achieve the aerodynamics requirement. The **DCA** blade has the additional advantage of simplicity of design and analysis. Figure D.2 (Reference 36), shows the essential features of a **DCA**. To calculate the camber line and surface radii, the following equations are given:

$$R_1 = \frac{c}{2 \sin \frac{\phi_D}{2}} \quad (\text{D.1})$$

$$R_2 = \frac{R_1(2R_1 + t) (1 - \cos \phi_D) + \frac{t^2}{4} - r^2}{2(R_1 - R_1 \cos \phi_D + \frac{t}{2} - r)} \quad (\text{D.2})$$

$$R_3 = \frac{R_1(2R_1 - t) (1 - \cos \phi_D) + \frac{t^2}{4} - r^2}{2(R_1 - R_1 \cos \phi_D - \frac{t}{2} + r)} \quad (\text{D.3})$$

where R_1 is radius of camber line, R_2 and R_3 are radii of suction and pressure surface, respectively. r is the leading and trailing edge radii and they are taken as 6% of maximum blade thickness with a minimum radius of 0.1225 mm, c is the chord, t is the thickness, θ is camber angle and ϕ is given by:

$$\sin\phi_D = \left(1 - 2\frac{r}{c}\right)\sin\frac{\theta}{2} \quad (\text{D.4})$$

D.2.3- Incidence Angle

ROTOR

The correlation of the minimum loss incidence angle (References 36, 37) is:

$$i_{2D} = K_i(i_0)_{10} + n\theta \quad (\text{D.5})$$

where $(i_0)_{10}$ is the minimum loss incidence angle for a **NACA-65** cascade of zero camber and 10 per cent thickness/chord ratio. Figure D.3 shows how $(i_0)_{10}$ and the slop factor n as a function of inlet air angle for a range of blade space/chord ratios. K_i is a correction factor which is applied to $(i_0)_{10}$ to allow for a different blade thickness and/or profile. The value of K_i for different thickness/chord ratios is given in Figure D.4. $(i_c - i_{2D})$ is the difference between the minimum loss incidence angle for a cascade and the measured value from rotor test. It is plotted against inlet Mach number, in Figure D.5, for rotor blade sections at the hub, mean and tip.

STATOR

For the stator the minimum loss incidence angle was assumed as a constant 5° (based on the analysis and suggestion given by Reference 36) from hub to tip.

D.2.4- Deviation Angle

The correlation which is still most often used, but with local modification and adaptations for deviation, is known as Carter's rule. It is expressed as:

$$\delta = m_c \theta \sqrt{\frac{s}{c}} \quad (\text{D.6})$$

where m_c varies with blade stagger angle (ζ) as shown in Figure D.6. To facilitate the computational design calculation the following equation is given:

$$m_c = 0.2162 + 0.0008595\zeta + 0.00002786\zeta^2 \quad (\text{D.7})$$

It has been found by Reference 36 that Carter's rule can be applied to DCA while the alternative deviation rule (Reference 37, 38), is more complex than Carter's method, and yet it predicts the true minimum loss deviation with much the same accuracy.

D.2.5- Diffusion Factor

The diffusion factor establishes a relevant loading parameter which described the low speed relation between blade loading and losses at optimum flow condition.

The diffusion factor given by Lieblein (39) has been rearranged in simplified form and expressed as:

$$D = \frac{V_{\max} - V_2}{V_1} \approx 1 - \frac{V_2}{V_1} + \frac{\Delta V_\theta s}{2V_1 c} \quad (\text{D.8})$$

where

$V_{\max}$ is maximum suction-surface velocity,

V_1 and V_2 are the inlet and outlet relative velocity to the blade element.

ΔV_θ is the change in whirl velocity and

s/c is the space-chord ratio.

The model is based on low speed measurements of NACA 65 - blades.

D.2.6- Equivalent Diffusion Factor

Equation (D.8) can be regarded as a simple one. For design purposes the inlet and outlet flow data are prescribed and the maximum surface velocity and the wake parameters are generally unknown. Hence, Lieblein formulated the following correlation between equivalent diffusion factor cascade inlet and outlet flow conditions:

$$D_{eq} = \frac{\cos\beta_2}{\cos\beta_1} [1.12 + 0.61 \frac{\cos^2\beta_1}{\sigma} (\tan\beta_1 - \tan\beta_2)] \quad (D.9)$$

where β_1 and β_2 are the inlet and outlet air angles, respectively, and σ is the cascade solidity.

D.2.7- Profile Loss

The profile loss of rotor and stator for two-dimensional incompressible cascade flow is given by:

$$\omega_p = \frac{2\sigma}{\cos\beta_2} \left(\frac{\theta_t}{c}\right)_c \left(\frac{\cos\beta_1}{\cos\beta_2}\right)^2 \{ [1 - \left(\frac{\theta_t}{c}\right)_c \frac{\sigma H_{tec}}{\cos\beta_2}]^{-3} \left(\frac{3H_{tec}-1}{2H_{tec}}\right)^{-1} \} \quad (D.10)$$

where

H_{tec} is the corrected boundary layer shape factor,

$(\theta_t/c)_c$ is the corrected wake momentum thickness and

The momentum thickness is given by Koch and Smith's data (40) as:

$$\left(\frac{\theta_t}{c}\right)_c = 0.00138 e^{1.1127 D_{eq}} + 0.0025 \quad (D.11)$$

and trailing-edge boundary layer shape factor as:

$$H_{te} = 1.26 + 0.795(D_{eq} - 1)^{1.681} \quad (D.12)$$

In the Koch and Smith correlations the values of (θ/c) and H_{te} given by equations (D.11) and (D.12) are for nominal conditions of

- 1- a Mach number below 0.05,
- 2- a streamtube (annulus) height ratio of 1.0,
- 3- a nominal value of Reynolds number of 10^6 , and
- 4- hydraulically smooth blades.

Figures D.7, D.8, D.9, and D.10 present the correction factor for conditions other than nominal. Figure D.7 is to correct the effect of the inlet Mach number on calculated trailing-edge thickness and form factor. Figure D.8 is to correct the effect of streamtube height variation on calculated trailing-edge form factor. Figure D.9 is to correct the effect of streamtube height variation on calculated trailing-edge momentum thickness and Figure D.10 is to correct the effect of Reynolds number and surface finish on calculated trailing-edge momentum thickness.

Therefore, The correction on trailing-edge momentum thickness is given by:

$$\left(\frac{\theta}{C}\right)_c = \left(\frac{\theta}{C}\right)_{M=0} \left(\frac{\theta}{\theta}\right)_{\frac{h_1}{h_2}=1} \left(\frac{\theta}{\theta}\right)_{Re=10^6} \quad (D.13)$$

and trailing-edge boundary layer shape factor by:

$$H_{tec} = H_{te} \left(\frac{H}{H}\right)_{M=0} \left(\frac{H}{H}\right)_{\frac{h_1}{h_2}=1} \quad (D.14)$$

The Reynolds number is calculated by:

$$Re = \frac{V_1 c \rho}{\mu} \quad (D.15)$$

where V_1 is the relative inlet velocity, c is the blade chord at blade section, ρ is the density, and μ is the absolute viscosity, which can be calculated by Sutherland's relation as follows:

$$\mu = \frac{145.8187 \cdot 10^{-8} T^{1.5}}{110.35 + T} \quad (D.16)$$

where T is the total inlet temperature.

In order to programme design calculations, the corrections for the influence of Reynolds number and roughness effect were based on the following formulations.

For the Reynolds number effect, given by Koch and Smith, it is:

$$F_{Re} = \left(\frac{Re}{Re_{cr}} \right)^n \quad (D.17)$$

where for laminar flow, $Re \leq Re_{cr} = 2.0 \cdot 10^5$, $n = -0.5$ and for turbulent flow, $Re > Re_{cr}$, $n = -0.166$.

For the roughness effect, given by Mill and Xu Hang (41), it is:

$$F_{ro} = \frac{(2.635 - 0.618 \log_e(\frac{k_s}{c}))^{-2.57}}{C_{Dref}} \quad (D.18)$$

where c is the chord, $C_{Dref} = 0.006$ and k_s is the average sand roughness in microns. It varies between 10^5 to 10^4 microns as average range. As suggested by Koch and Smith, the two corrections are combined such that $\left(\frac{\theta}{\theta} \right)_{Re=10^5}$ is given

by the largest value of F_{Re} and F_{ro} .

The value of k_s is placed the input file. If the value is unknown, it can certainly be selected within the range given above. For LP compressor is recommended a value near 10^{-4} microns and for IP and HP compressors a value near 10^{-5} microns. Re_{cr} and C_{Dref} are the reference values of Reynolds number and friction factor, respectively, given by Liebein (42).

D.2.8- Shock Loss

A flow model for shock loss has been selected (Reference 37) to estimate the shock loss contribution at the design condition for an inlet relative Mach number over 1.0. The selected flow model for shock loss estimation appeared to satisfy the requirements here.

The method is based on a very simplified shock model, with an oblique shock in front of the cascade with negligible losses, and normal shock in the blade passage which is assumed to be normal to the mean streamline. The Mach number immediately upstream of the shock on the suction surface is calculated as a function of the supersonic expansion from the inlet direction to the profile tangent at the shock position. The Mach number immediately upstream of the shock at the leading edge of the adjacent profile is equal to the inlet Mach number.

Normally, the calculation of shock loss coefficient is based on the Prandtl-Meyer flow approximation. This mathematical calculation may be avoided by using the graph presented in Figure D.11 (Reference 36).

Considering the geometry in Figure D.16, the supersonic turn λ can be calculated as follows:

$$\lambda = \alpha_1 - \zeta + \xi_c \quad (\text{D.19})$$

where

$$\sin \xi_c = \left(\frac{2x}{c'} - 1 \right) \sin \left(\frac{\phi_s}{2} \right) \quad (\text{D.20})$$

$$c' = c - 2r$$

$$\tan \left(\frac{\phi_s}{4} \right) = \left(\frac{1 - \cos \frac{\theta}{2}}{\sin \frac{\theta}{2}} \right) + \left(\frac{t - 2r}{c'} \right) \quad (\text{D.21})$$

and

$$\frac{2x}{c'} = 1 + \left(\sin 2\Sigma \cot \frac{\theta}{2} \right) - \cos 2\Sigma \quad (\text{D.22})$$

The angle Σ is given by:

$$\tan \Sigma = \frac{\sin \beta_1 \sin \frac{\theta}{2}}{\frac{c'}{s} + \cos \beta_1 \sin \frac{\theta}{2}} \quad (\text{D.23})$$

Having calculated the angle of turn, the normal shock loss can be found in Figure D.11 for the inlet relative Mach number.

A flow model does not exist for the inlet relative Mach number less than unity. However, between inlet relative Mach numbers 0.8 and 1.0, there is still some shock loss contribution. To deal with such cases Reference 24 formulated a reasonable interpolation between zero shock loss and the value given for the unity Mach number. The equation given by Reference 24 is:

$$\omega_{sh} = 0.035(M_1 - 0.82) \quad (D.24)$$

where M_1 is the blade inlet Mach number.

D.2.9- Secondary and Annulus Losses

An analysis on secondary and annulus losses in CEMCOMP has revealed that the drag coefficients have been calculated wrongly. The programme assumes that the two coefficients are zero at all positions other than the root and tip. This approximation is not true once the annulus and secondary drag coefficients are calculated for the condition outside the endwall region. For this exercise the calculation is carried out at pitch.

The new calculation executed by CEMCOMP follows the following equations for rotor and stator (Reference 15, 43).

Annulus loss

$$\omega_A = C_{DA} \left(\frac{c}{s} \right) \left(\frac{\cos^2 \alpha_1}{\cos^3 \alpha_m} \right) \quad (D.25)$$

where C_{DA} is the annulus drag coefficient and is given by:

$$C_{DA} = 0.020 \left(\frac{s}{h} \right) \quad (D.26)$$

and c , s and h are the space, pitch and blade height, respectively. α_m is:

$$\alpha_m = \tan^{-1} \left[\frac{1}{2} (\tan \alpha_1 + \tan \alpha_2) \right] \quad (D.27)$$

Secondary loss

$$\omega_{SD} = C_{DS} \left(\frac{c}{s} \right) \left(\frac{\cos^2 \alpha_1}{\cos^3 \alpha_m} \right) \quad (D.28)$$

where C_{DS} is the secondary drag coefficient and is given by:

$$C_{DS} = 0.0018 C_L^2 \quad (D.29)$$

and

$$C_L = 2 \left(\frac{s}{c} \right) (\tan \alpha_1 - \tan \alpha_2) \cos \alpha_m \quad (D.30)$$

where α_m is calculated by equation (D.27).

The secondary loss is of major importance and is significantly influenced by tip clearance which should be kept as small as possible.

To allow for the tip clearance it was decided to keep the same correlation as in the CEMCOMP once it is appropriated to the proposal of this work (see Reference 24).

D.2.10- Efficiency Prediction

The adiabatic temperature-rise efficiency of the flow along a stream surface (Reference 44), for rotor and stator, is given by definition as follows:

$$\eta_{ad,ST} = \frac{\left[\left(\frac{P_2}{P_1} \right)_R \left(\frac{P_2}{P_1} \right)_S \right]^{\frac{\gamma-1}{\gamma}} - 1}{\left(\frac{T_2}{T_1} \right)_R - 1} \quad (D.31)$$

From the developments of Reference 44 and for simplicity in the efficiency-estimation procedure the equation (D.31) becomes for the stage:

$$\eta_{ad,ST} = \frac{\left[\left(\frac{P_2'}{P_1} \right)_R \left(\frac{T_2}{T_1} \right)_R^{\frac{\gamma-1}{\gamma}} \left(\frac{P_2'}{P_1} \right)_S^{\frac{\gamma-1}{\gamma}} - 1 \right]}{\left(\frac{T_2}{T_1} \right)_R - 1} \quad (D.32)$$

and for the rotor:

$$\eta_{ad,R} = \frac{\left[\left(\frac{P_2'}{P_1} \right)_R \left(\frac{T_2}{T_1} \right)_R^{\frac{\gamma-1}{\gamma}} - 1 \right]}{\left(\frac{T_2}{T_1} \right)_R - 1} \quad (D.33)$$

where

$$\frac{P_2'}{P_1} = 1 - \omega_T \left\{ 1 - \left[\frac{1}{1 + \frac{\gamma-1}{2} (M_1)^2} \right]^{\frac{\gamma}{\gamma-1}} \right\} \quad (D.34)$$

As seen, four quantities are required for the determination of the blade element efficiency across the rotor or stage:

- 1- the rotor absolute total temperature T ,
- 2- the relative inlet Mach number M_1 ,
- 3- the relative total-pressure ratio (P_2'/P_1') , and
- 4- the relative total-pressure-loss coefficient ω_T , which is given by:

$$\omega_T = \omega_P + \omega_{SH} + \omega_A + \omega_{SD} \quad (D.35)$$

where

- ω_P is profile loss,
- ω_{SH} is shock loss,
- ω_A is annulus loss, and
- ω_{SD} is secondary loss .

To allow for the tip clearance effect, the overall adiabatic efficiency is given as follows:

$$\eta_{ad,o} = \eta_{ad,ST} - \Delta\eta_{tipcl} \quad (D.36)$$

D.3- Merger of ANNULUS and CEMCOMP

The traditional computer compressor design method was carried out based on the flow chart in Figure D.13. ANNULUS and CEMCOMP calculate the aerodynamics design, and SCD15 and SCD12 the mechanical design.

ANNULUS calculates compressor annulus dimensions, temperatures and pressures at the outlet from each stage of a compressor of constant mean, tip diameter or hub diameter design.

For the aerodynamic design, the input for CEMCOMP came from output of ANNULUS, which had to be made by hand. The chance of mistakes was enormous once the calculation was an iterative procedure. In order to avoid it, the ANNULUS programme became a subroutine of CEMCOMP, called ANNULUS. Thus, after the change a significant reduction in running time, during the design procedure, was achieved.

D.4- Modification of SCD15

The computer programme SCD15 was originally written by Rait (45). The programme calculates rotor blade angles and profile geometry for double circular arc blades based on the method from Reference 36, described previously. It also performs basic blade stressing calculations and estimates the first and second flap and first torsional natural frequencies.

To calculate the rotor blade angles and profile geometry for double circular arc blades, The SCD15 needed the correlations for incidence and deviation as

input data. These correlations had to be found in Figures D.3, D.4, D.5 and D.6 by hand reducing the accuracy and increasing the probability of mistakes. In addition, it could not perform stator blade angles and blade profile geometry, which are the requirements for compressor variable geometry.

Modifications were carried out on the aerodynamic design, but mechanical design is beyond the scope of this work. The following steps describe the modification in SCD15.

1- Conversion of Figures D.3, D.4, and D.5 into subroutines ZCIA10 and SFN, CFN, and MLIA, respectively. For Figure D.6 the equation (D.7) was assumed. Thus, the calculation is now done automatically according to the flow chart in Figure D.14 for the rotor blade row. The equation for stagger calculation, as approach, was suggested by Mackenzie (46). It is given by:

$$\zeta = \text{atan}(\tan \alpha_m - 0.15) \quad (\text{D.37})$$

where

$$\alpha_m = \frac{1}{2}(\tan \alpha_1 + \tan \alpha_2)$$

α_1 and α_2 are relative inlet and outlet air angles, respectively.

2- Introduction of stator blade calculation in accordance with section D.2.3. The flow chart in Figure D.15 shows the simplified sequence in which the programme carries out the stator blade calculation.

3- The thickness/chord ratio assumed (Reference 36) the typical values from hub to tip section as follows:

for rotor 10%, 8%, 7%, 5%, 3% and

for stator 9%, 7.5%, 6%, 4.5%, 3% .

These values are in DATA command for automatic calculation. However,

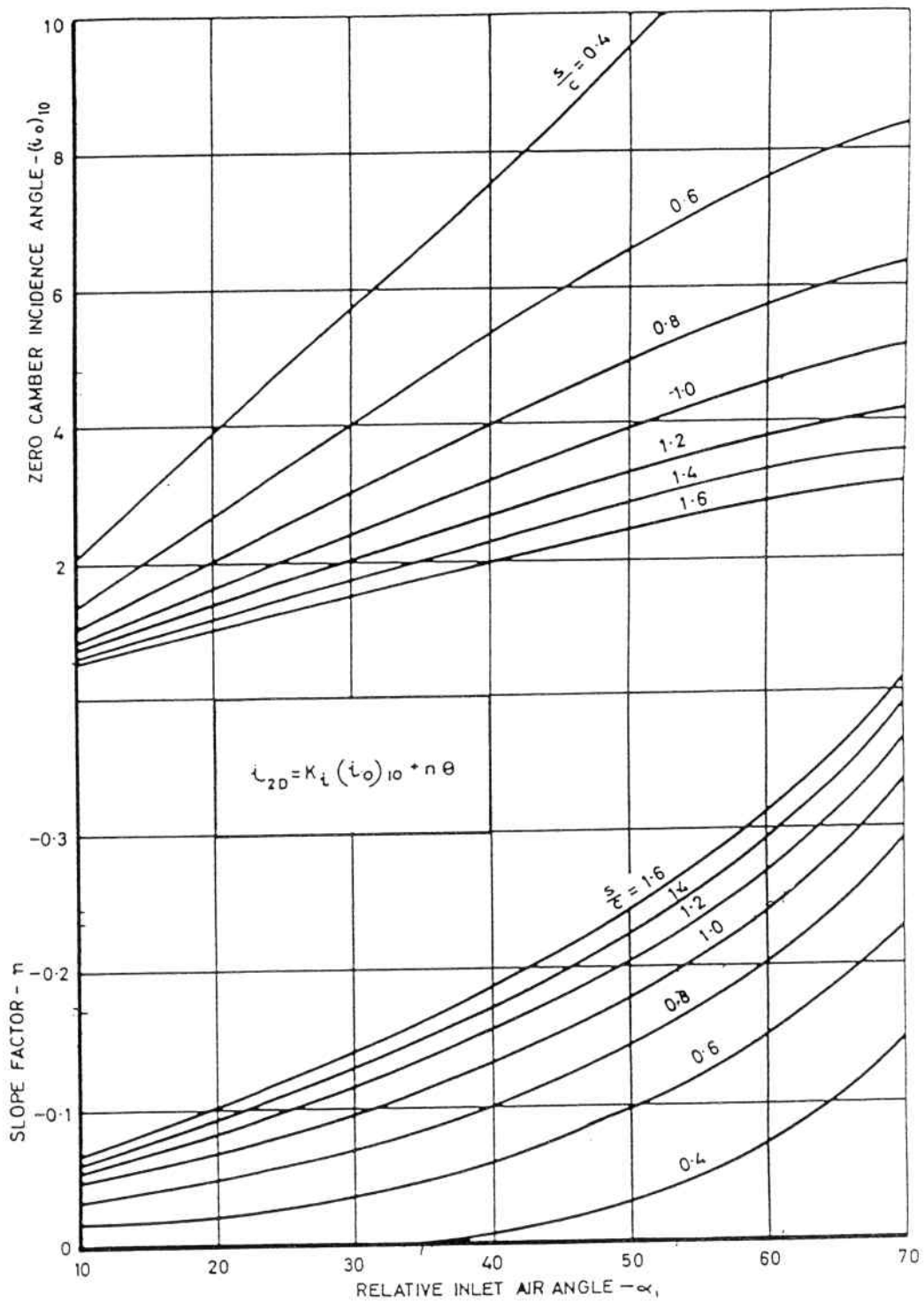


Figure D.3 Minimum Loss Incidence Angle.

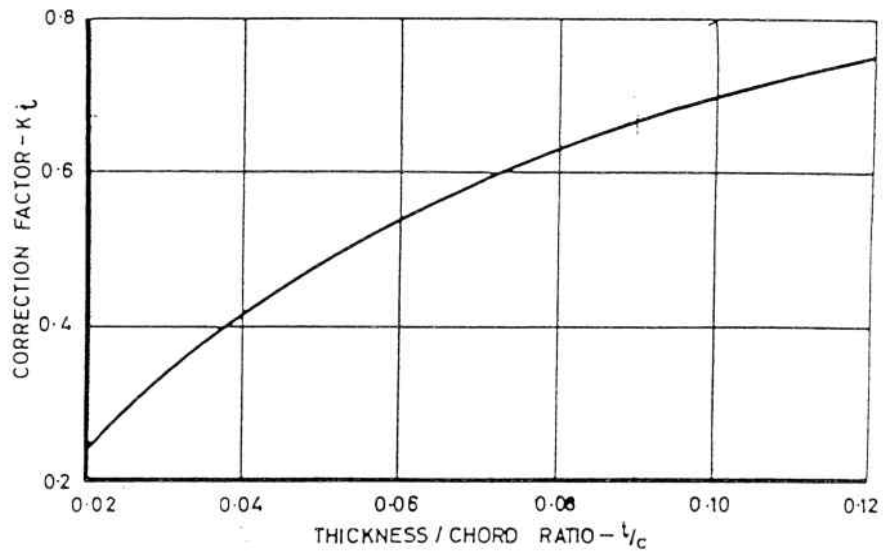


Figure D.4 Thickness Correction for zero camber reference incidence angle.

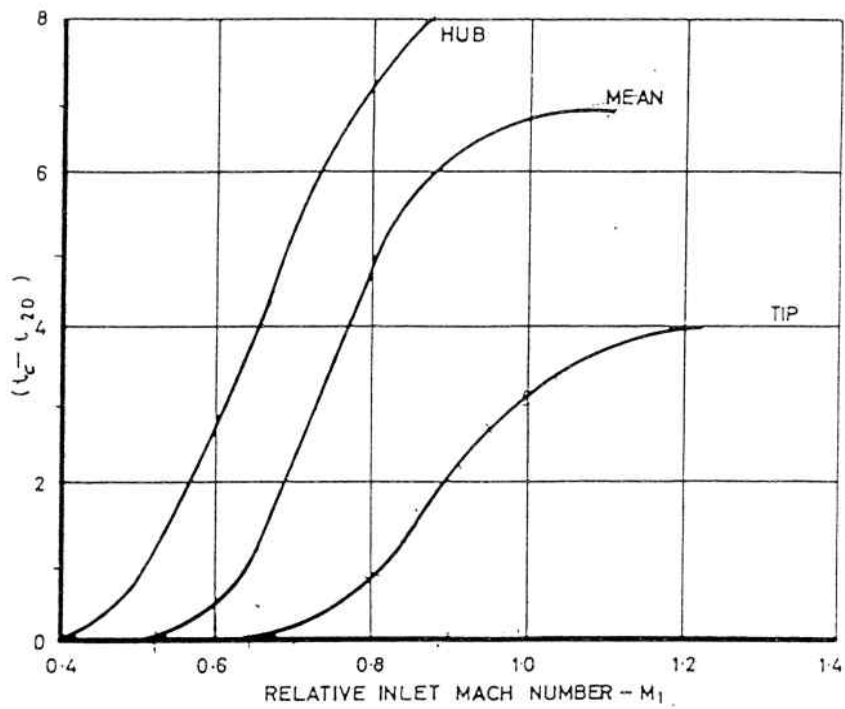


Figure D.5 Minimum Loss Incidence Angle.

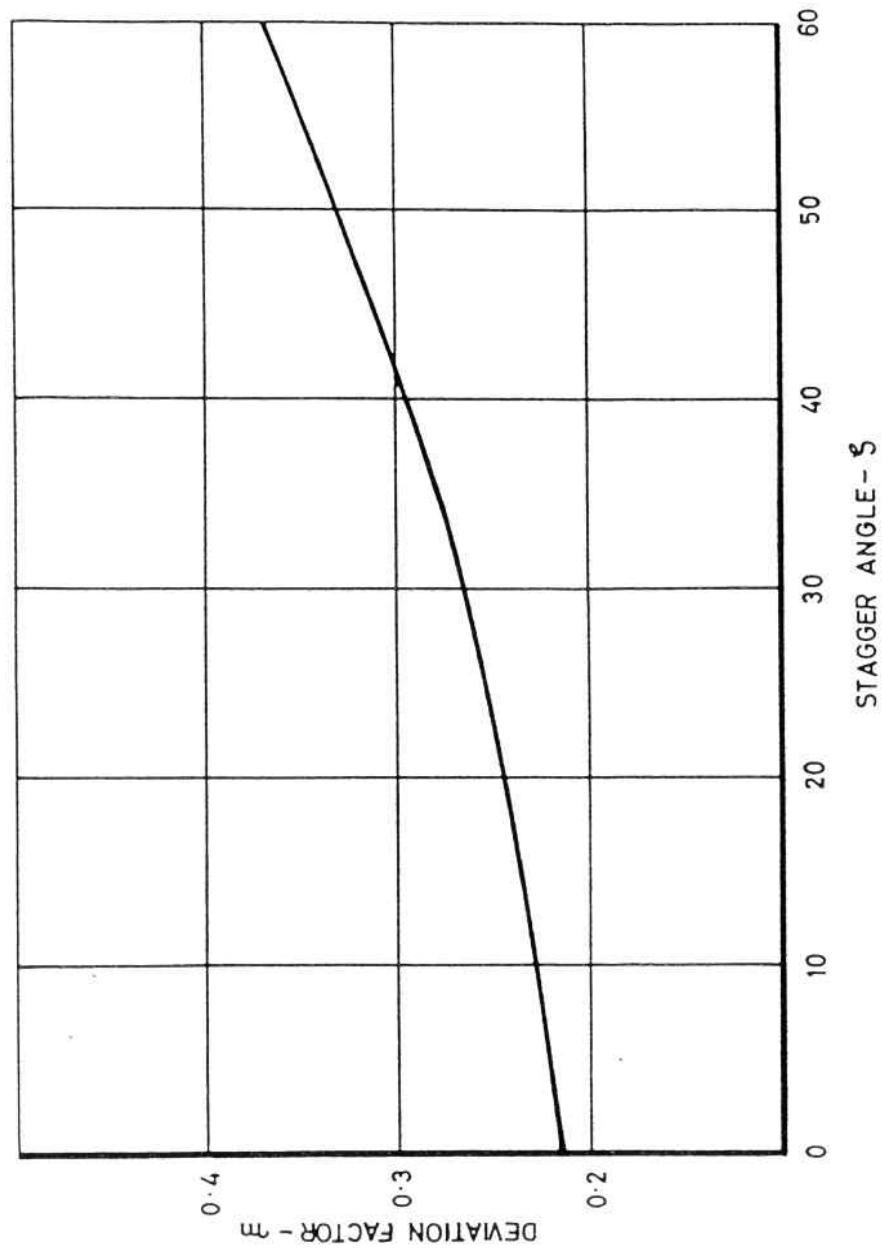


Figure D.6 Minimum Loss Deviation Angle.

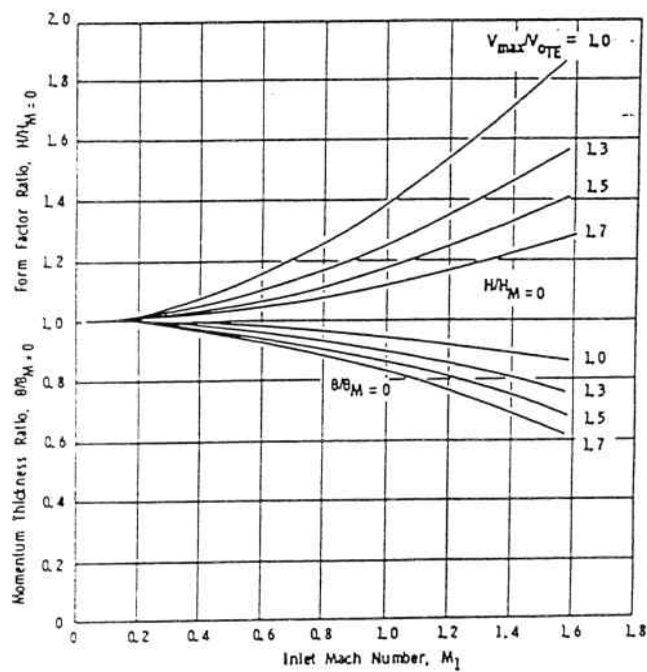


Figure D.7 Effect of Inlet Mach Number on Calculated Trailing-Edge Momentum Thickness and Form Factor.

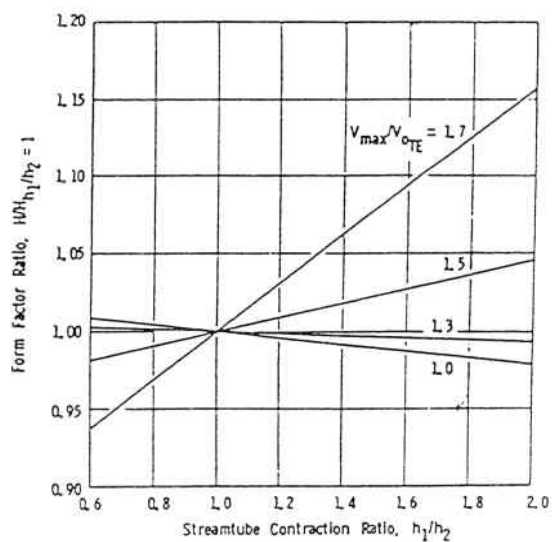


Figure D.8 Effect of Streamtube Height Variation on Calculated Trailing-edge.

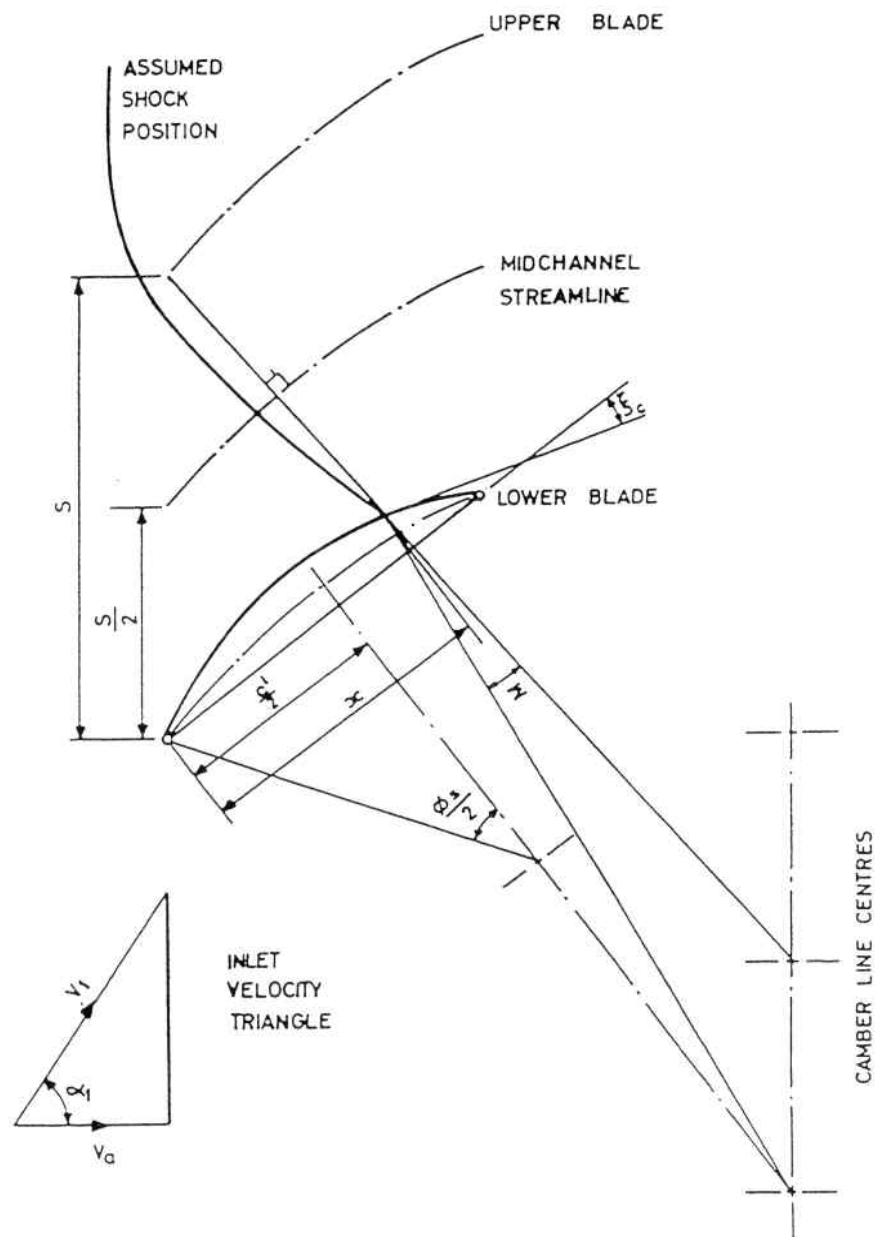
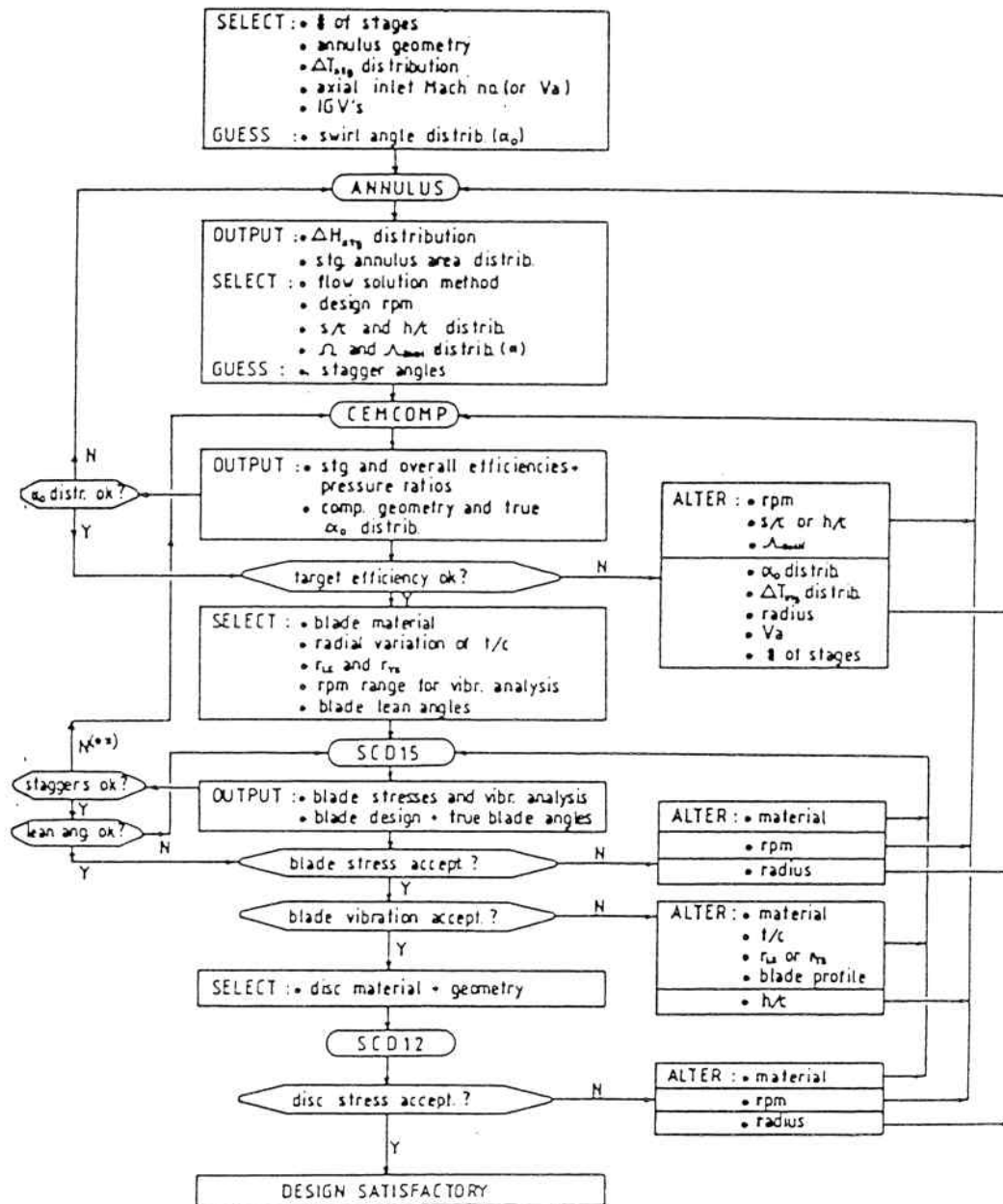


Figure D.12 Model Used to Locate Normal Shock.

FLOW CHART OF COMPUTER BASED COMPRESSOR DESIGN METHOD



(*) specify : • α_0 distrib. for free vortex • ind. aspect ratio solution

• Ω_{max} distrib. for other solutions

(**) affects geometry calcs only

Figure D.13 Flow Chart of Computer Package.

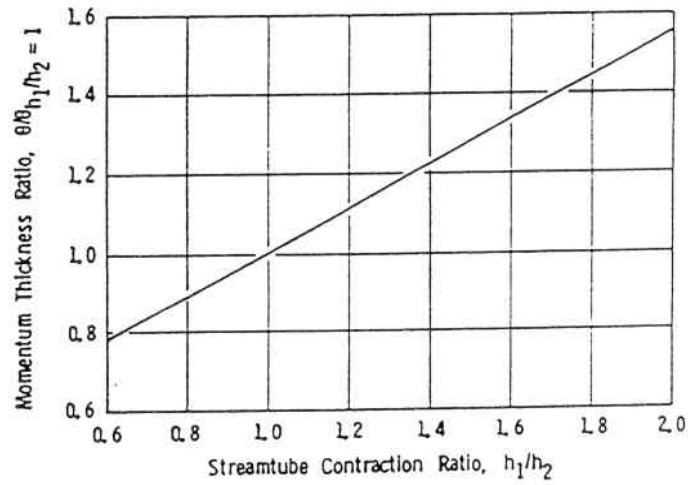


Figure D.9 Effect of Streamtube Contraction on Calculated Trailing-Edge Momentum Thickness.

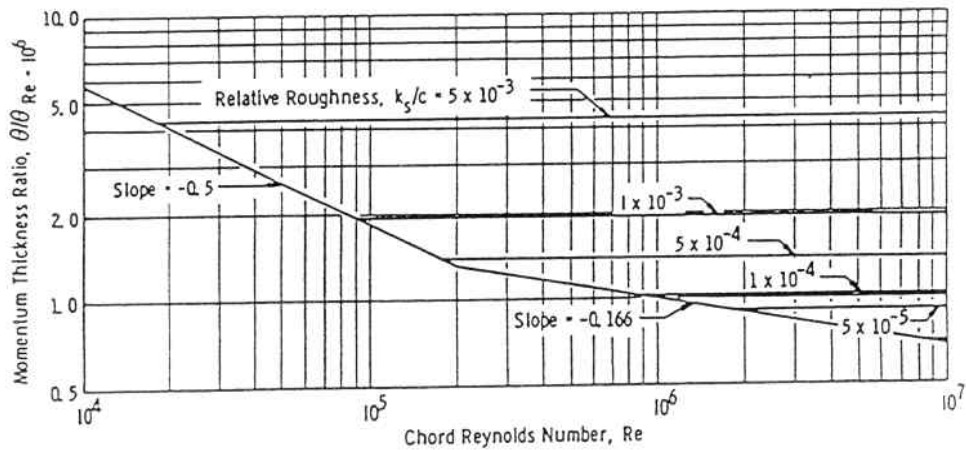


Figure D.10 Effect of Reynolds Number and Surface on Calculated Trailing-Edge Momentum Thickness.

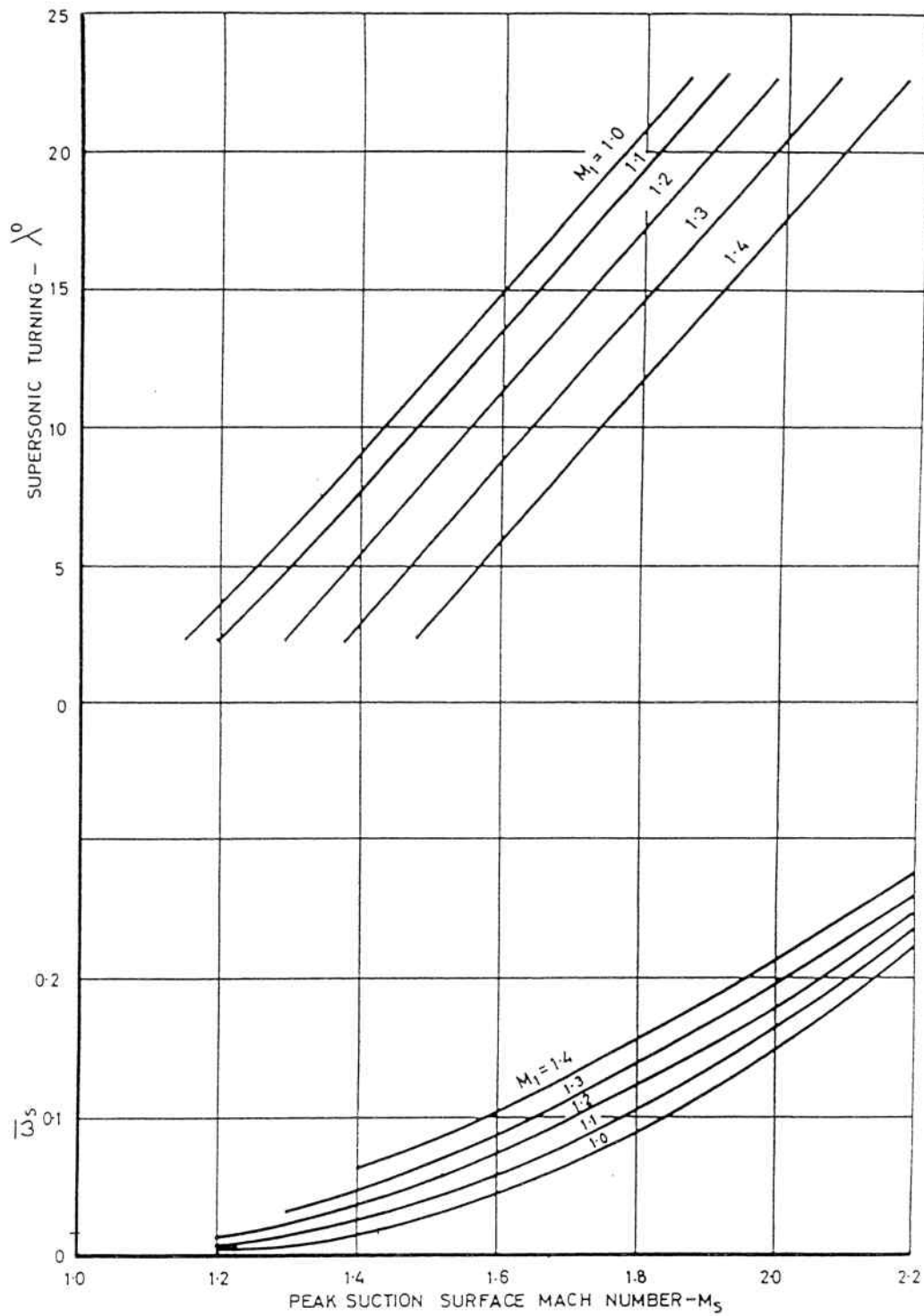


Figure D.11 Shock Loss Coefficient.

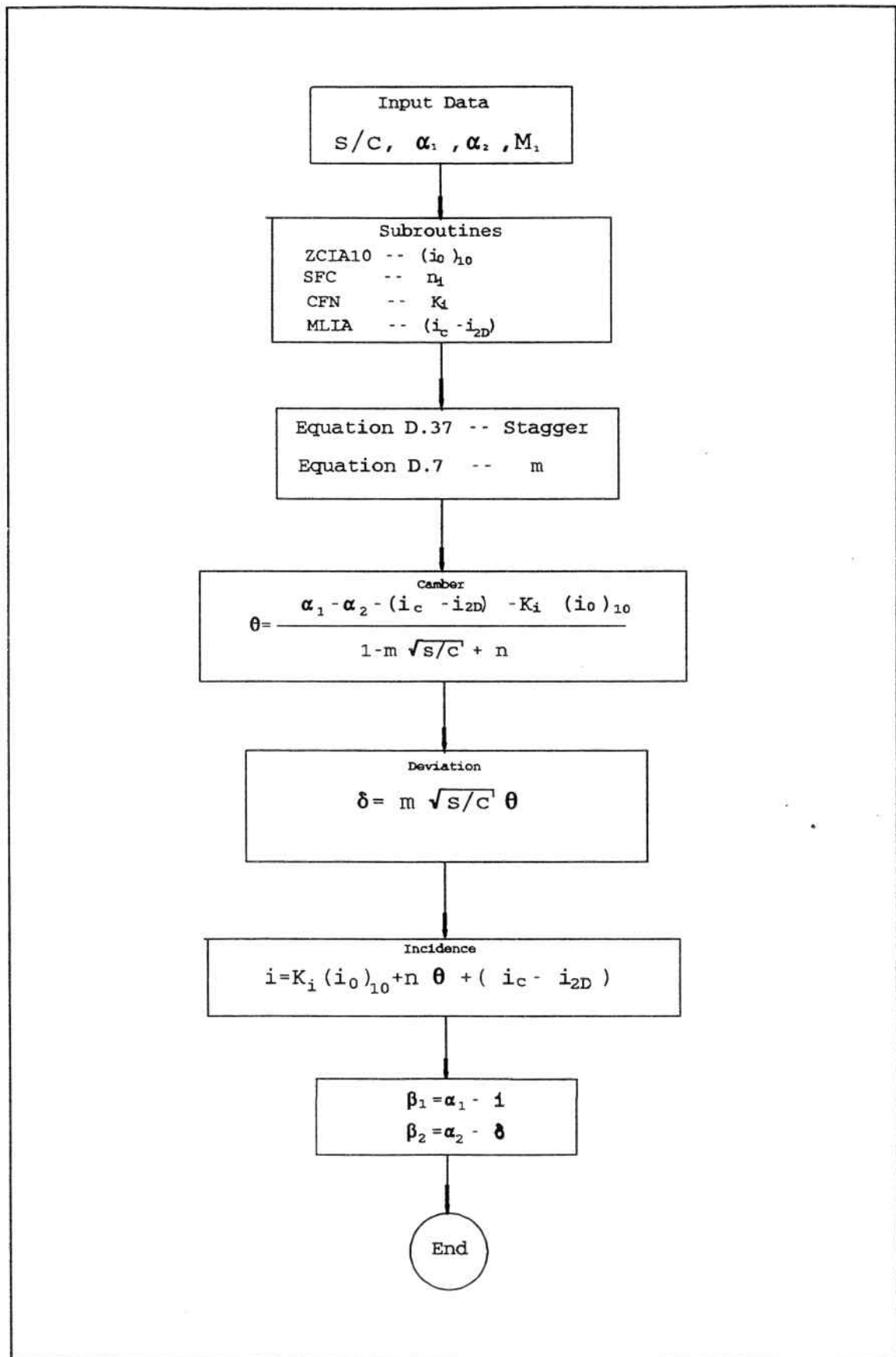


Figure D.14 Flow Diagram of Rotor Blade Angles Calculation.

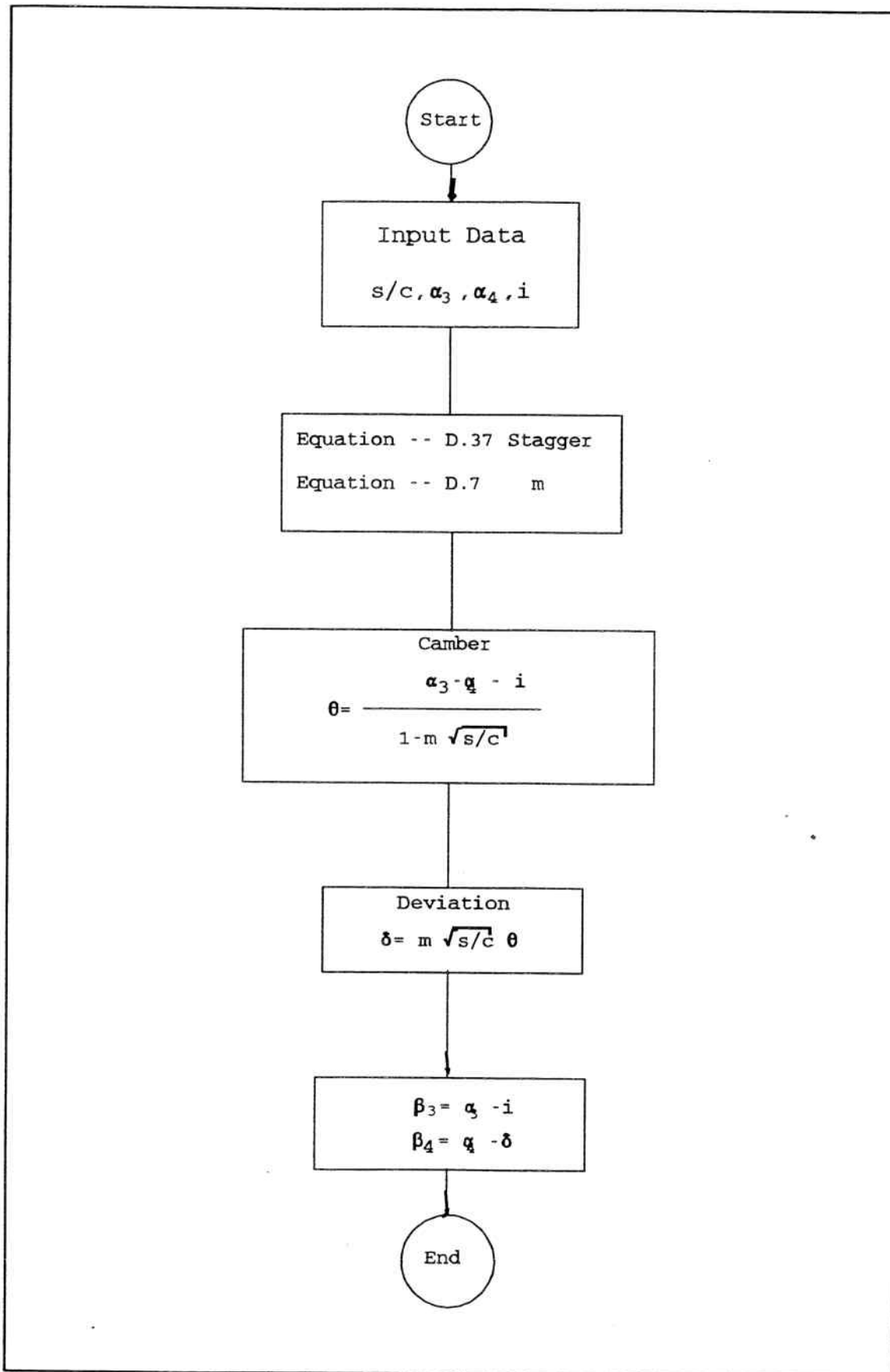


Figure D.15 Flow Diagram of Stator Blade Angles Calculation.

INPUT FILE REQUIREMENTS FOR "COMPDES" PROGRAMME

Note: The commands for running COMPDES are:

```
$ASSIGN SYS_$ _COMMAND: SYS_$ _INPUT
$RUN USER7:[ME001T]COMPDES
```

TITLE	40 characters
NUMSTG	Number of stages (maximum 9).
IRAD	Annulus geometry selector
	1 = Constant hub radius
	2 = Constant mean radius
	3 = Constant tip radius.
IGV	Radius type
	0 = no IGVs
	1 = with IGVs
HTR0	Hub tip ratio
EFFISEN	Design isentropic efficiency
VA(N) N=1, (NUMSTG+2)	Inlet axial velocities [m/s]
DT(N) N=1, NUNSTG	Stage temperature raise of each one [K]
TINL	Inlet total temperature [K]
PINL	Inlet total pressure [atm]
PR	Design pressure raise
MF	Inlet air mass flow ratio [kg/s]
IVORTX	Vortex design selector
	1 = Free Vortex
	2 = Exponential
	3 = Graded Work
	4 = Forced Vortex
	5 = Infinite aspect ratio

Figure D.16 COMPDES Input File Requirements.

ALO(N) N=1, (NUMSTG+2) .. Stage inlet swirl angle from IGV to Exit [deg]. Specify if IVORTX = 1 or 6, otherwise give any value.

RPM Rotational speed [rpm]

WD(I) I=1, NUMSTG Stage work done factors

SCR(I) I=1, NUMSTG*2 Rotor and stator (s/c) ratio respectively.

AR(I) I=1, NUMSTG Rotor (h/c) ratio

REACTM(I) I=1, NUMSTG ... BMH degree of reaction values [decimal]. Specify if IVORTX = 2,3,4 or 5., otherwise give any value.

TIPC(I) I=1, NUMSTG Tip clearance values for rotors [decimal].

XKS Equivalent sand roughness (microns) Note: Average roughness 0.0001 microns

Figure D.16 COMPDES Input File Requirements.

```

$
$Run COMPDES

*****
*   The programme prints on the screen your   *
*   input file as well as a message if the   *
*   reading is carried out successfully.     *
*   If it does not happen you should check up *
*   your input file.                         *
*****

*****
*                               *
*   Give input filename.         *
*                               *
*****
COMPDESLP.DAT

*****
*   This programme gives you an output data  *
*   called BLADE.INP to be used as input    *
*   data of BLADE programme.                *
*   Enter:                                  *
*                                           *
*       0= if BLADE.INP is not require.     *
*                                           *
*       1= if BLADE.INP is require.         *
*****
0

*****
*   This programme gives you an output data  *
*   called PERFORM.INP to be used as input   *
*   data of HOWELL programme.                *
*   Enter:                                  *
*                                           *
*       0= if PERFORM.INP is not require.   *
*                                           *
*       1= if PERFORM.INP is require.       *
*           [ Howell Multistage Compressor   *
*             Performance Prediction Programme] *
*****
0

FORTRAN STOP
$

```

Figure D.17 COMPDES Screen Information.

INPUT FILE REQUIREMENTS FOR "BLADE" PROGRAMME

Note: The commands for running BLADE are:

```
$ASSIGN SYS_$ COMMAND: SYS_$ INPUT
$RUN USER7:[ME001T]BLADE
```

TITLE	40 Characters
RPM	Rotational speed [rpm]
DOTM	Design mass flow ratio [kg/s]
BLN	Number of rotor blades
SBLM	Number of stator blades
RAD(1) RAD(NSC)	Root and tip rotor radii [mm]
SRAD(1) SRAD(NSC) ..	Root and tip stator radii [mm]
ALPHA1	Rotor inlet angles at each section from root to tip [deg].
ALPHA2	Rotor outlet angles at each section from root to tip [deg].
ALPHA3	Stator inlet angles at each section from root to tip [deg].
ALPHA4	Stator outlet angles at each section from root to tip [deg].
SCR	Rotor space/chord ratio at each section from root to tip.
SSCR	Stator space/chord ratio at each section from root to tip.
M1	Rotor inlet Mach numbers at each section from root to tip.

Figure D.18 BLADE Input File Requirements.

VA1	Relative inlet axial velocities [m/s] at each section from root to tip.
VA2	Relative outlet axial velocities [m/s] at each section from root to tip.
PS1	Inlet static pressure (atm). at each section from root to tip.
PS2	Outlet static pressure (atm). at each section from root to tip.
ISECT	Enter: 0= to end the file data 1= for subsequent stages data set.

Figure D.18 BLADE Input File Requirements.

```

$
$ Run BLADE

*** Enter Elastic Modulus [GN/meter square] ***
102

*** Enter Rigidity Modulus [GN/meter square] ***
27.6

*** Enter Material Density [Mg/meter cubic] ***
7.6

*** Enter Tangential & Lean angles [deg] ***
-0.014 0.15

*** Enter:

        0 - for no vibration data
        1 - for vibration data
0

** Rotational speeds for centrifugal stiffening **
***          Enter six values          ***
1000. 2000. 3000. 4000. 4550. 5000.

**** Enter number of section (maximum 11) ****
5

*** Enter number of section (maximum 11) ****
*** The values of (t/c) from hub to tip ****
*** are:

        ROTOR} - 0.1 0.08 0.07 0.05 0.03
        STATOR} - 0.03 0.045 0.06 0.075 0.09

*** Would you like to place your on values? (Y/N) ***
N

*** Give Input filename ***
BLADE.INP

*** Give Output filename ***
BLADE.OUT

$ FORTRAN STOP

```

Figure D.19 BLADE Screen Information.

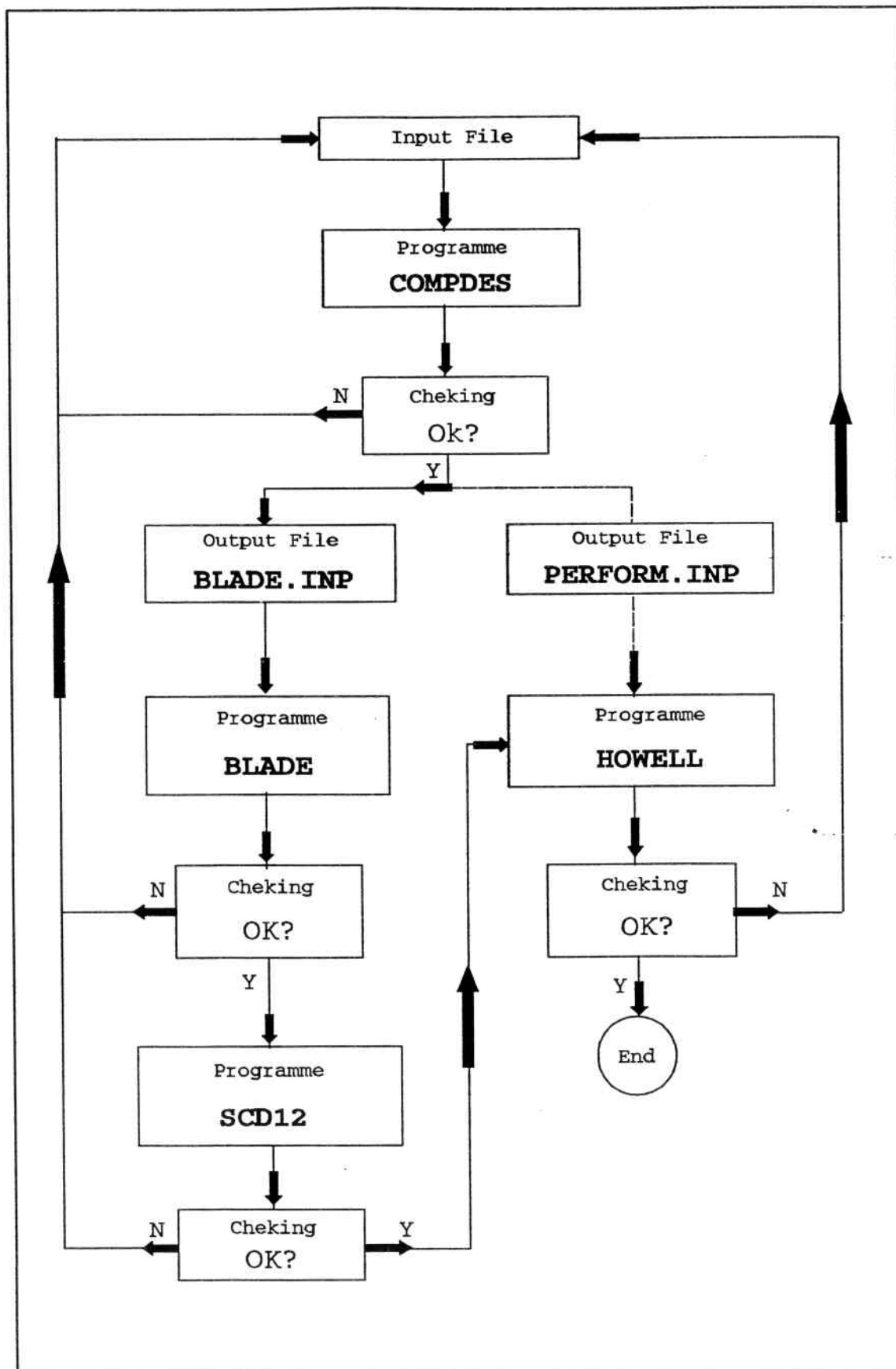


Figure D.20 New Flow Chart of Compressor Design Method.

APPENDIX E

Turbine Design and Off-design simulation

In this Appendix a preliminary aerodynamic design and off-design performance prediction for an axial-flow turbine are presented. The methodology employed and the computer design sequence are outlined.

E.1- Turbine Aerodynamic Design

In the following sections a description of individual programmes is given as well as their sequence of use for a turbine design.

E.1.1- TURBGEO Computer Programme

TURBGEO is a turbine pitch line design computer programme for multistage axial-flow turbines. Its calculations are based literally on the theory given in Chapter 7 (Reference 15). It is meant to calculate the annulus geometry and to estimate the flow conditions at the mean diameter of the annulus. In essence these calculations use mass flow continuity and the Euler stage work equation to solve the average conditions at various planes through the turbine between the blades and vanes. Thus, it is helpful for the very beginning of turbine preliminary design.

AT theory (Cohen)

TURBGEO requires a very simple input data file (the instructions are given in Figure E.1) , and a constant tip, mean, and hub diameter can be selected for

design and analysis. In addition, it supplies an output data file called *TURBDES.INP*, to be used as an input data file for *TURBDES* computer programme as the next step of the design procedure.

The design analysis, on *TURBGEO*, may be carried out on the blade loading coefficient, flow coefficient, degree of reaction and Mach numbers. Due to its short running time *TURBGEO* allows the users to run it more times. This means that, the users can better assess the design parameters.

Once the design parameters are selected with *TURBGEO* the next step is to refine them and allow for three-dimensional effects. It is done by the computer programme presented in the next section.

E.1.2- *TURBDES* Computer Programme

TURBDES is a new version of the original free vortex design programme written by Barthwick (47) and modified by Dener (24) who named it *CEMTURB*. It uses basically a free vortex solution to allow for a three-dimensional effect and it is meant to be a middle ground between a very simple turbine design and a sophisticated one. Dener has introduced some new loss correlations and off-design performance predictions. Further details about the previous version, can be found in Reference 24.

In general, the programme was found in a good condition and a close examination over the calculation procedure was carried out and no serious problem was found. However, the input data file instructions were found to be very difficult to prepare, especially for a multistage turbine, due to lack of directions. In addition to that, a hand calculation of annulus geometry was necessary to start with making it even more difficult to establish interrelations for optimisation studies. As a result of this, a pitch line turbine design computer programme, called *TURBGEO*, was written as stated in the foregoing section and another input data file instruction (Figure E.2) was rewritten with more details.

The programme also provides, when required, two output data files named *TURBOFFC.DAT* and *TURBOFFE.DAT* for plotting. The first one is to plot non-dimensional mass flow ($m\sqrt{T/P}$) against pressure ratio (PR) for different operating speed line. The second one is to plot the isentropic efficiency (η) against pressure ratio (PR) for different operating speed line.

During the running time the master segment calls subroutines, which represent the correlations for losses, incidence, gas angle, total pressure, velocity, Mach number etc (References 28, 29) and compressible flow properties (Reference 48). The iteration procedure stops when the maximum limit work output at the rotor blade exit is achieved. It means that, beyond this point the turbine isentropic efficiency reduces substantially and no work output can be extracted by the turbine as the pressure ratio increases.

To sum up, the three computer programmes form a complete package for the design and off-design performance prediction with reasonable accuracy for a preliminary design approach. The short running time of the programmes enables the user to concentrate more on the design optimisation instead of the mathematics behind it. Figure E.5 presents the flow diagram showing the running sequence to carry out the aerodynamic design.

E.2- Ainley and Mathieson's Method

Ainley and Mathieson's method is a mean line off-design performance prediction calculation of a conventional axial-flow turbine. It makes use of data obtained from the analysis of a large number of turbine tests and other associated test work reported elsewhere. The method enables the performance of a turbine to be calculated over a large part of its full operating range.

Basically, the method procedure is divided into two parts as follows:

- 1- Determination of the stagnation pressure loss coefficients and gas outlet

angles for rotor and stator over a wide range of operation conditions.

2- Determination of the performance of the stage at particular entry flow conditions and particular blade speed.

Repeating the second part for different values of these parameters the overall performance map can be built up.

The method considers the flow path through each stage at one diameter only. This diameter is named the "reference diameter" and it is the arithmetical mean of the rotor and stator row inner and outer diameters. This approach assumes that the total pressure, total temperature, and axial velocity are equal at all points. Despite such assumptions being different from the fact, the accuracy of the performance will not be reduced. Steps in the calculation of one operating point are given in the next sections.

E.2.1- Notation and Convention of Signs

A full notation of signs for the gas velocity triangles of each blade row are given in Figure E.6. Figure E.7 gives details of turbine nomenclature, convention and trailing edge illustrating curvature of the upper (convex) surface between blade throat and trailing edge. Figure E.8 shows the reference diameter and the reference stations between adjacent blades.

E.2.2- Outlet Gas Angle from a Blade Row

The gas outlet angle for exit Mach number less than 0.5 ($M_2 < 0.5$) is given by:

$$\alpha_2 = \alpha_2^* - 4\left(\frac{S}{\theta}\right) \quad (\text{E.1})$$

where α_2^* is given in Figure E.9 (Reference 28) and may be replaced by:

$$\alpha_2^* = 11.15385 - 1.153846 \cos^{-1} \left(\frac{o}{s} \right), \quad (E.2)$$

s is the blade pitch, o is the width of the throat between the blades, and e is the mean radius of curvature of the suction surface of the blade profile between throat and trailing edge.

At an exit Mach number of unity (M_2) the gas outlet angle is:

$$\alpha_2 = -\cos^{-1} \left(\frac{A_t}{A_{n2}} \right) \quad (E.3)$$

where A_t is the passage throat area and A_{n2} is the annulus area at exit.

For flared annulus the throat area may be calculated by the following empirical equation:

$$A_t = \frac{(o/s)[5A_{n2} + A_{no}]}{6} \quad (E.4)$$

where A_{no} is annulus area at the inlet and A_{n2} is the annulus area at the exit.

For exit Mach number $0.5 < M_2 < 1.0$ a linear interpolation is assumed with reasonable accuracy.

For blades having tip or shroud clearance, the outlet gas angle is corrected by the following equations:

1- At the exit Mach number less than 0.5 ($M_2 < 0.5$)

$$\alpha_2 = \tan^{-1} \left\{ \left[1 - X \left(\frac{k}{h} \right) \left(\frac{\cos \beta_1}{\cos \alpha_2^*} \right) \right] \tan \alpha_2^* + X \left(\frac{k}{h} \right) \left(\frac{\cos \beta_1}{\cos \alpha_2^*} \right) \tan \beta_1 \right\} \quad (E.5)$$

where $X=1.35$ for radial tip clearance, $X=0.70$ for simple shrouded blade, k is the

clearance, h is the blade height α_2^* is the outlet gas angle for clearance zero and β_1 is the blade inlet angle.

2- At exit Mach number of unity ($M_2=1$), equation (E.3) is allowed for the tip clearance by correcting the throat area as follows:

$$A_t = A_t^* [1 - (\frac{k}{h})] + A_k \quad (E.6)$$

where A_t^* is the throat area for zero tip clearance, A_k is the flow area in the clearance space ($A_k = \pi Dk$), D is the tip diameter, k is the tip clearance and for shroud clearance the equation is replaced by:

$$A_t = A_t^* + A_k \quad (E.7)$$

E.2.3- Profile Loss

The profile-loss coefficient (Reference 28) for zero incidence is a function of the exit gas angle (α_2) for two special cases of blade inlet angle ($\beta_1=0$ and $\beta_1=\beta_2$ Figure E.10). For any other combination of angles, these graphs are interpolated by the following equation:

$$Y_{p(i=0)} = \{Y_{p(\beta_1=0)} + (\frac{\beta_1}{\alpha_2})^2 [Y_{p(\beta_1=-\alpha_2)} - Y_{p(\beta_1=0)}]\} (\frac{t/c}{0.2})^{(-\beta_1/\alpha_2)} \quad (E.8)$$

where $Y_{p(i=0)}$ is the profile-loss coefficient for $\beta_1=0$ and $Y_{p(\beta_1=-\alpha_2)}$ is the profile-loss coefficient for $\beta_1=-\alpha_2$ both having the same α_2 and s/c as the actual blade.

Equation (E.8) is restricted to $0.15 < (t/c) < 0.25$. For (t/c) greater or less than the limits the loss should be taken as equal to the blade having (t/c) either 0.25 or 0.15 respectively.

E.2.4- Stalling Incidence

Stalling incidence (Reference 28) is function of exit gas angle α_2 , inlet blade angle β_1 and space chord ratio (s/c). Figure E.11 gives the stalling incidence for a wide range of blades having (s/c)=0.75 and the correction factor for blades with (s/c) different from 0.75.

For a given blade with α_2 , β_1 and (s/c), the following procedure is adopted to determine the stalling incidence:

1- With a given value of (s/c) and α_2 in Figure E.11 find the ratio $\alpha_{2(s/c=0.75)}/\alpha_2$ and hence $\alpha_{2(s/c=0.75)}$.

2- Calculate $\beta_1/\alpha_{2(s/c=0.75)}$ and in Figure E.11(b) find $i_{s(s/c=0.75)}$.

3- With actual value of (s/c) and α_2 find Δi_s in Figure E.11(a) and thus, determine i_s by:

$$i_s = i_{s(s/c=0.75)} + \Delta i_s \quad (E.9)$$

With i_s and profile-loss coefficient for zero incidence, the profile-loss coefficient (Y_p) for a wide range of incidence can be found in Figure E.12.

E.2.5- Secondary Loss

The secondary losses arise from the interaction between the flow trough the blade passage and the end wall boundary layer.

An update empirical formula for secondary flow has been suggested by Dunhan and Came (29) as a replacement of the one in Reference 28.

The expression is:

$$Y_s = 0.0334 \left(\frac{c}{h} \right) \left(\frac{\cos \alpha_2}{\cos \beta_1} \right) Z \quad (E.10)$$

where the numerical constant was chosen from comparisons with overall efficiency data, and it also compensates for the use of reference diameter blade loading coefficient values, where c is the blade true chord, h is the blade mid height, α_2 is the exit gas angle, β_1 is the inlet blade angle,

$$Z = \left(\frac{C_L}{s/c} \right)^2 \left(\frac{\cos^2 \alpha_2}{\cos^3 \alpha_m} \right), \quad (E.11)$$

$$\left(\frac{C_L}{s/c} \right) = 2(\tan \alpha_1 - \tan \alpha_2) \cos \alpha_m, \quad (E.13)$$

$$\alpha_m = \tan^{-1} \left[\left(\frac{\tan \alpha_1 + \tan \alpha_2}{2} \right) \right], \quad (E.12)$$

C_L is the lift coefficient and α_1 is the inlet gas angle.

E.2.6- Tip Clearance Loss

The tip clearance loss is defined as the increase in end loss due to tip clearance. Thus, it depends on the blade loading, the size and nature of the clearance.

Dunhan and Came suggest another formula for tip clearance as a replacement for the one in Reference 28. The expression is:

$$Y_k = B \left(\frac{c}{h} \right) \left(\frac{k}{c} \right)^{0.78} Z \quad (E.14)$$

where $B=0.47$ for plain tip clearance, $B=0.37$ for Shrouded blade, h is the blade mid height, c is the blade true chord, k is the tip clearance and Z is calculated by equations from (E.11) to (E.13).

For shrouded blades having multiple tip seal, the following expression is given to correct the tip clearance.

$$k = k_a (N)^{-0.42} \quad (\text{E.15})$$

where k_a is the shrouded clearance without seals and N is the number of seals.

Equations (E.10) and (E.14) are restricted within $-1.5 \leq (i/i_s) \leq 1.0$. At values of $(i/i_s) > 1.0$ and $(i/i_s) < -1.5$, secondary and clearance losses are assumed to remain constant and equal to the value when $(i/i_s) = 1$ and $(i/i_s) = -1.5$, respectively.

E.2.7- Total Loss Coefficient

The total loss coefficient (References 28, 29) is assumed to be the sum of profile loss, secondary and tip or shroud clearance losses and corrected by Mach number, Reynolds number effect and trailing edge loss effect as follows:

$$Y_t = [(Y_p f_M + Y_s) f_{Re} + Y_k] Y_{TET} \quad (\text{E.16})$$

where f_M is the Mach number correction factor suggested in Reference 29, which is given by:

$$f_M = 1 + 60(M_e - 1)^2 \quad (\text{E.17})$$

being applied only for $M_e > 1$, M_e is the relative Mach number at exit from blade row, f_{Re} is the Reynolds number correction factor (Reference 29) given by:

$$f_{Re} = \left(\frac{R_e}{2 \times 10^5} \right)^{-0.2} \quad (\text{E.18})$$

and Y_{TET} is the trailing edge loss multiplier used for trailing-edge thickness pitch ratio (t_e/s) rather than 0.02. Figure E.13 provides Y_{TET} for different values of (t_e/s) .

E.2.8- Stage Performance Prediction Procedure

An outlined description of the method is given here. Further details can be seen in Reference 28. Before starting the performance calculation procedure, the following should have been determined:

- 1- The reference diameter of the stage.*
- 2- Annulus areas at the reference stations mid-way in the axial clearance spaces between adjacent blade rows (see Figure E.8).*
- 3- Gas outlet angle as a function of outlet Mach number at the reference diameter for each row.*
- 4- Variation of total loss coefficient (Y_t) with incidence for each blade row.*

The calculation process is described for one stage and for further stages and rotational speed it is treat in a similar way. The specific heat at constant pressure (C_p) and specific heat ratio (γ) are assumed to remain constant throughout expansion of the gas through the stage. An inlet pressure and temperature are arbitrarily selected at the inlet of turbine.

The non-dimensional speed ($N/\sqrt{T}$) of a turbine is a function of the non-dimensional mass flow quantity ($m\sqrt{T}/PA$) and pressure ratio so that for a fixed turbine entry values of P, T , and A only the mass flow vary to obtain one operating speed line.

i)- Unchoked blade row

For a suitable turbine entry value of inlet temperature, inlet pressure and a first value of arbitrarily selected mass flow, the following steps are given below to determine the gas outlet conditions for the nozzle.

1- Select a first approximation to outlet gas angle (α_0).

2- From the inlet conditions find $(m\sqrt{T_i}/A_0P_i)$ where $A_0=A_{n0}\cos \alpha_0$ and m is mass flow.

3- With the value of Y_i in Figures E.14 and E.15 find the value of $(m\sqrt{T_i}/A_0P_i)_{\max}$ and $(\bar{\omega}/P_i)_{\text{crit}}$ Where $\bar{\omega}$ is pressure loss.

4- Calculate $(m\sqrt{T_i}/A_0P_i)/(m\sqrt{T_i}/A_0P_i)_{\max}$ and thus from Figure E.16 find the corresponding value of $(\bar{\omega}/P_i)/(\bar{\omega}/P_i)_{\text{crit}}$

5- From items 3 and 4 find $(\bar{\omega}/P_i)$.

6- Determine $(m\sqrt{T_0}/A_0P_0)$ where P_0 is given by $P_0=P_i-\bar{\omega}$. For an uncooled nozzle row $T_0=T_i$.

7- With the values of $(m\sqrt{T_0}/A_0P_0)$ and M_0 find the value of $V_0/\sqrt{T_0}$ in any compressible flow tables.

If the value of α_0 corresponding to the value of M_0 found in 7 differs from the initial assumption in 1 by more than about 0.2 degree then the process is repeated using a second approximation to α_0 .

The following equations determine the gas conditions relative to the rotor row at inlet. From the velocity triangle yields:

$$V_{a1}=V_0\cos\alpha_0 \quad (\text{E.19})$$

where V_{a1} is the axial velocity at inlet of rotor row, V_0 is the absolute velocity at exit of nozzle blade row, and α_0 is the nozzle exit gas angle. α_1 is determined as follows:

$$\tan \alpha_1 = \frac{U}{V_{a1}} - \tan \alpha_0 \quad (\text{E.20})$$

where U is the rotor blade speed at reference diameter. From the energy equation, the total temperature relative to the rotor at inlet is given by:

$$T_0 - T_1 = \frac{V_0^2 (1 - [\frac{\cos \alpha_0}{\cos \alpha_1}]^2)}{2C_p} \quad (\text{E.21})$$

where T_0 is the nozzle outlet total temperature, T_1 is the rotor inlet total temperature and V_0 is the nozzle outlet absolute velocity. The total pressure relative to the rotor at inlet is given by:

$$\frac{P_1}{P_0} = \left(\frac{T_1}{T_0} \right)^{\frac{\gamma}{\gamma-1}} \quad (\text{E.22})$$

where P_1 is the rotor inlet total pressure and P_0 is the nozzle outlet total pressure.

Having determined the gas conditions at the rotor entry, the next steps are to calculate the gas conditions at the rotor exit.

Knowing the gas conditions at the rotor entry and assuming no heat is transferred to or from the blade, so that the gas total temperature relative to the row does not change ($T_2 = T_1$), the rotor exit gas conditions are determined as follows:

- 1- Select a first approximation to outlet angle (α_2).
- 2- Determine the value of $(m\sqrt{T_1}/A_2P_1)$, where $A_2 = A_{n2} \cos \alpha_2$ and A_{n2} is the annulus area at the exit.
- 3- With the value of Y_1 find in Figures E.14 and E.11 the values for of $(m\sqrt{T_1}/A_2P_1)_{\max}$ and $(\bar{\omega}/P_1)_{\text{crit}}$

4- Calculate the ratio $(m\sqrt{T_1}/A_2P_1)/(m\sqrt{T_1}/A_2P_1)_{\max}$ and find in Figure E.16 the value of $(\bar{\omega}/P_1)/(\bar{\omega}/P_1)_{\max}$.

5- From items 3 and 4 calculate the value of $(\bar{\omega}/P_1)$.

6- Determine $(m\sqrt{T_2}/A_2P_2)$, where $P_2=P_1-\bar{\omega}$.

7- With the value of $(m\sqrt{T_2}/A_2P_2)$ and M_2 , the value of $(V_2/\sqrt{T_2})$ may be found in any compressible flow tables.

If the value of α_2 corresponding to the value of M_2 differs from the value guessed in item 1 by more than about 0.2 degree, then the process is repeated using a second approximation to α_2 until the convergence is achieved.

The next step is to obtain the absolute gas outlet conditions from the rotor. So that, from the velocity triangle at the exit of the rotor, the axial velocity V_{a2} , absolute gas angle α_3 , total exit temperature T_3 , and total exit pressure P_3 are given by the following equations:

$$V_{a2} = V_2 \cos \alpha_2 \quad (\text{E.23})$$

$$\tan \alpha_3 = \frac{U}{V_{a2}} - \tan \alpha_2 \quad (\text{E.24})$$

$$T_2 - T_3 = \frac{V_2^2 \left[1 - \left(\frac{\cos \alpha_2}{\cos \alpha_3} \right)^2 \right]}{2C_p} \quad (\text{E.25})$$

$$\frac{P_3}{P_2} = \left\{ 1 - \left[\frac{(T_2 - T_3)}{T_2} \right] \right\}^{\frac{\gamma}{\gamma-1}} \quad (\text{E.26})$$

ii)- Choked blade row

Choking is assumed to occur in a blade row when the mass flow in nozzle or rotor achieves its maximum value, for a given Y_t and α_2 . In another words, when at the mean gas Mach number in the throat of the blade passage of a row becomes unity. So that, the flow upstream of the choked row then remains unchanged as the overall pressure ratio is increased, the flow downstream becomes supersonic and the gas direction changes to preserve continuity. The flow may choke again downstream as the pressure ratio is increased further.

When the outlet Mach number from any row exceeds the critical value, the following assumptions are made:

- 1- The non-dimensional mass flow ($m\sqrt{T/AP}$) at the turbine entry remains constant.
- 2- The gas outlet angle differs from the estimated value at the Mach number of unity adjusting itself sufficiently to keep constant flow.

For a choked rotor row, for instance, to calculate the gas conditions downstream to the choked row the following procedure may be applied.

- 1- Select the value of $M_2 \geq 1$ arbitrarily.
- 2- With the value of M_2 find in any compressible flow tables the value of total to static-pressure ratio $\frac{P}{P_{s_2}}$ and $(m\sqrt{T_2/A_2P_2})$.
- 3- Knowing the total loss coefficient Y_r , rotor inlet total P_1 , and P_2/p_{s2} find P_2 by:

$$\frac{\bar{\omega}}{P_2} = Y \left[1 - \frac{P_{s_2}}{P_2} \right] \quad (E.27)$$

$$\frac{\omega}{P_1} = \frac{\left(\frac{\bar{\omega}}{P_2} \right)}{\left(1 + \frac{\bar{\omega}}{P_2} \right)} \quad (E.28)$$

$$P_2 = P_1 - \bar{\omega} \quad (E.29)$$

and $T_1 = T_2$

4- With the values of m , T_2 , P_2 , and $(m\sqrt{T_2/A_2P_2})$ find the value of A_2 . Since $A_2 = A_{n2} \cos \alpha_2$, the gas outlet angle can be determined.

E.2.9- Overall Stage Isentropic Efficiency

The overall stage isentropic efficiency is given by:

$$\eta = \frac{\frac{(T_i - T_3)}{T_i}}{1 - \left(\frac{P_3}{P_i} \right)^{\frac{\gamma-1}{\gamma}}} \quad (E.30)$$

where T_i and P_i are the stage inlet total temperature and pressure, and T_3 and P_3 are the stage outlet total temperature and pressure, respectively.

Figure E.1 TURBGEO input file requirements

INPUT FILE REQUIREMENTS FOR "TURBGEO" PROGRAMME

Note: The commands for running TURBGEO are:

```
$ASSIGN SYS,$COMMAND: SYS,$INPUT
$RUN USER7:[ME001T]TURBGEO
```

TITLE..... 60 characters.

NSTAG..... Number of stages (max. 5).

FLOW Mass flow(kg/s).

PIN Inlet total pressure (N/m²).

TIN Inlet total temperature (K).

PR Pressure ratio.

FAR Fuel/air ratio.

VA Inlet axial velocity (m/s).

HTR Hub-tip ratio.
NOTE: HUB-TIP RATIO is equal to HUB-SHROUD RATIO.

RPM Rotational speed (rpm).

ETA Polytropic efficiency (%).

TD Temperature drop in each stage (K).

ALPH1 Inlet air angle of each stage(degree).

ALPH3 Exit air angle (degree).

IRAD Design type:
1=constant hub radius
2=constant mean radius
3=constant tip radius

ARN,ARR Nozzle/rotor aspect ratio of each stage.

SCRN,SCRR Nozzle/rotor space-chord ratio of each stage.

LOSSN Nozzle loss coefficient.
Note:Normally the value is 0.05

Figure E.2 TURBDES input file requirements.

 INPUT FILE REQUIREMENTS FOR "TURBDES" PROGRAMME

Note: The commands for running TURBDES are:

```
$ASSIGN SYS_$COMMAND: SYS_$INPUT
$RUN USER7:[ME001T]TURBDES
```

Input Block 1

NSTAG Number of stages

TITLE 80 Characters

Input Block 2

POWER Turbine Power (MW) (STOP if POWER=0.0)

DOTM Inlet Mass Flow (kg/s)

REVS Rotational Speed (rev/min)

ETAT Total-Total Effy (decimal)

ETAN Total-Static Effy (decimal)

ALF3 Exit swirl Angle (deg)

WDF Work Done Factor (decimal)
Note: if left blank WDF=1.0 (default)

FAR Fuel-air Ratio

RHOMET Blade material density (Mg/m3)

TAPER Taper Factor
Note: See appendix A2.1 equation A2 of
Ref(26). TAPER=1 for a parallel blade.

SHROUD Additional Blade Length to Compensate for a Shroud. (mm)

Input Block 3

** Design Options **

IT Inlet Total Temperature Profile:
IT=0 constant values.
IT=1 Root, Mid, and Tip values.
IT=2 5 radial values.
Note: It is used in conjunction with TIN in block 4.

IVAR Rotor exit axial velocity:
IVAR=1 root & tip radii given, axial velocity calculate.
IVAR=2 axial velocity given, geometry calculate.

IVAN NGV exit axial velocity: The same as IVAR.

IRDEFR Datum root radius:
 IRDEFR=1 constant hub.
 IRDEFR=2 constant mean.
 IRDEFR=3 constant tip.

IRDEFN Datum NGV radius: The same as IRDEFR

NO No of radial positions:
 No=1 Three stations; Root, Mean, and Tip.
 No=2 Five stations; Root, 25%, Mean, 75%, and Tip.

IETA "Va specified" route efficiency:
 IETA=0 non-isentropic turbine.
 IETA=1 isentropic turbine.

MULTY Multi-stage?:
 MULTY=0 single stage turbine.
 MULTY=1 first stage of multi stage turbine.
 MULTY=2 second stage or subsequent stage of multi stage turbine.

IREPEAT..... Range of design:
 IREPEAT=0 single set of data.
 IREPEAT=1 changes in input data.

ICALIN NGV inlet plane:
 ICALIN=0 inlet root & tip radii given, inlet velocity calculate.
 ICALIN=1 inlet velocity given, annulus area calculate.
 Note: see variables VINFIN and ALINF below.

LOAD(1)..... NGV blade loading:
 LOAD(1)=0 axial chord & h/c input or found internally.
 LOAD(1)=1 blade numbers given, axial velocity calc from Zweifel coefficient.
 LOAD(1)=2 blade numbers specified, loading variable.

LOAD(2)..... Rotor blade loading: The same as LOAD(1)

IDELTA Three-dimensional losses:
 IDELTA=0 not required.
 IDELTA=1 required.

 Input Block 4 (Skipped if 2nd stage of multi-stage)

TIN Inlet Total Temperatures (K)
 Note: If the variable IT above is set:
 IT=1 only one value is need.
 IT=2 three values are need.
 IT=5 five values are need.

PIN Inlet Total Pressures (kPa)
 Note: Five values are required from Root to Tip.

 Input Block 5 (Skipped if multi-stage)

ALINF Inlet Swirl Angles (deg)
 Note: Five values are required from root to tip. If ICALIN=0 only the Mean value needs to be given
 and the rest equal zero. Eg. (0. 0. 30. 0. 0.)

VINF Inlet Axial Velocities (m/s)
 Note: Five values are required from root to tip. If ICALIN=0 only the Mean value needs to be given
 and the rest equal zero. Eg. (0. 0. 206. 0. 0.)

Input block 6 (Note: see Figure E.3)

X Axial station distances (mm) (four values)
 RAD Root radii (mm) (four values)
 RAD Tip radii (mm) (four values)

Input Block 7 (Note: specify the following if blade data is available. Give zero if not known.)

NBLADE NGV/Rotor Blade Numbers
 Note: If unknown give zero values.
 TCHORD NGV/Rotor True Chords (mm)
 EDGET NGV/Rotor Trailing Edge Thicknesses (mm)
 BLADET NGV/Rotor Maximum Blade Thicknesses (mm)
 CLEAR NGV/Rotor Blade Tip Clearances (mm)
 AR NGV/Rotor Axial Chord Aspect Ratios
 CAX NGV/Rotor Axial Chords (mm)
 NSEAL Number of Blade Tip Shroud Seals

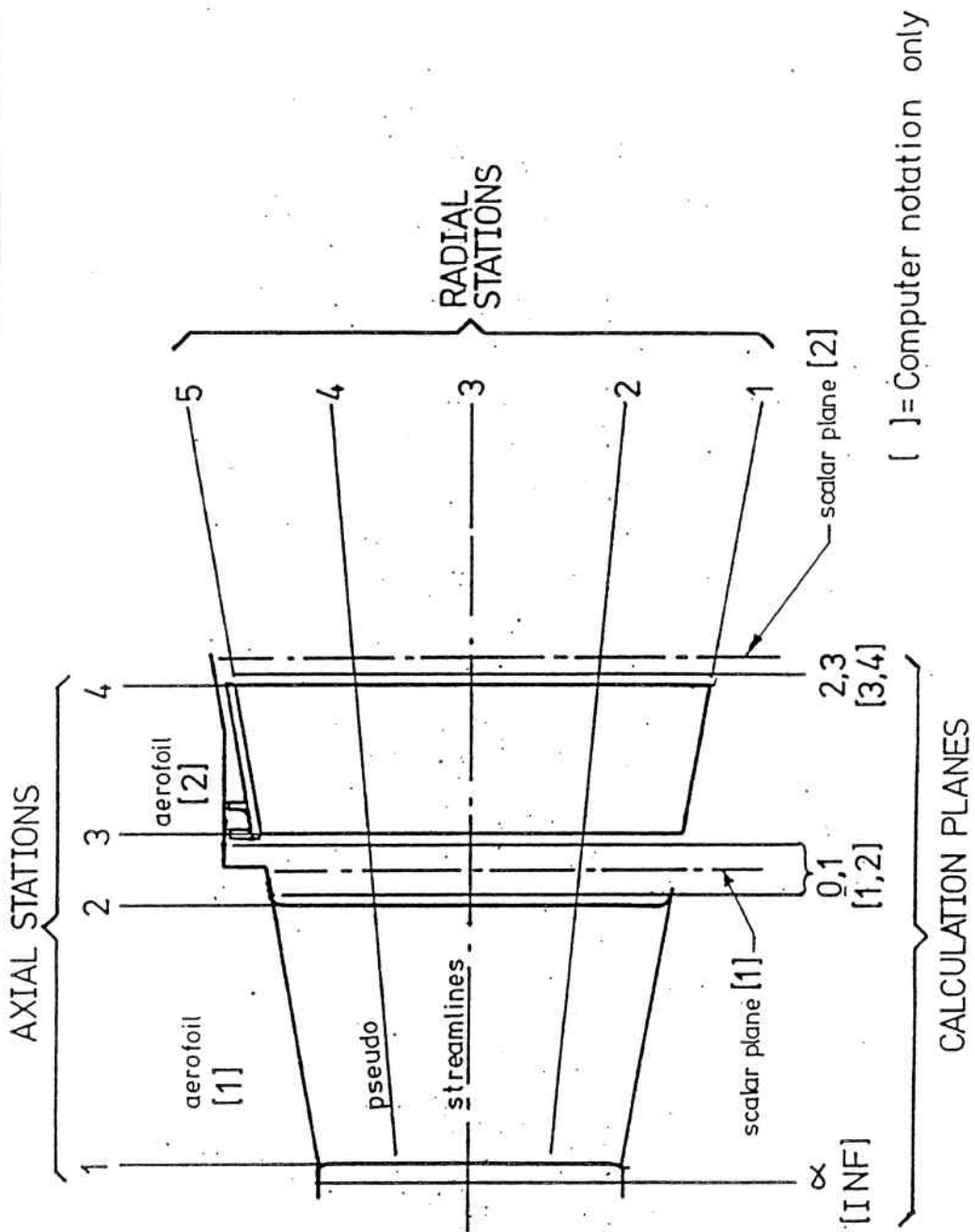


Figure E.3 Annulus notations.

Figure E.4 TURBOFF Input File Requirements.

 INPUT FILE REQUIREMENTS FOR "TURBOFF" PROGRAMME

Note: The commands for running TURBOFF are:

```
$ASSIGN SYS,$_COMMAND: SYS,$_INPUT
$RUN USER7:[ME001T]TURBOFF
```

 TURBINE INLET CONDITIONS

- 1) TITLE TITLE (60 CHARACTERS)
- 2) FAR FUEL AIR RATIO
- 3) TIN TURBINE INLET TOTAL TEMPERATURE (K)
- 4) PIN TURBINE INLET TOTAL PRESSURE (N/m²)
- 5) WF INLET MASS FLOW (kg/s)
- 6) RPMD DESIGN ROTATIONAL SPEED (rpm)
- 7) NRPM NUMBER OF ROTATIONAL SPEED
NOTE: NEGATIVE VALUE.
- 8) RPM(I) I=1,NRPM ROTATIONAL SPEEDS (rpm)
NOTE: ONE ABOVE THE OTHER AND
NEGATIVE VALUES.

 STAGE CONDITIONS

 ANNULUS GEOMETRY:

- 9) RAD TIP RADII (cm). FOUR VALUES.
- 10) RAD HUB RADII (cm). FOUR VALUES.

 NOZZLE:

- 11) BETAN BLADE INLET ANGLE (degree).
- 12) CN CHORD (m).
- 13) SN SPACE (m).
- 14) ON BLADE OPENING (m).

- 15) EN MEAN RADIUS OF CURVATURE OF THE
CONVEX SURFACE OF A BLADE BETWEEN
THE THROAT AND TRAILING EDGE (m).
- 16) TCRN THICKNESS/CHORD RATIO.
- 17) TEN BLADE TRAILING-EDGE THICKNESS, MEASURED
NORMAL TO CAMBER-LINE AT TRAILING EDGE
(m). SEE FIGURE E.7.
- 18) XHN BLADE HEIGHT (m).

ROTOR:

- 19) BETAR BLADE INLET ANGLE (degree).
- 20) CR CHORD (m).
- 21) SR SPACE (m).
- 22) OR BLADE OPENING (m).
- 23) ER MEAN RADIUS OF CURVATURE OF THE
CONVEX SURFACE OF A BLADE BETWEEN
THE THROAT AND TRAILING EDGE (m).
- 24) TCRR THICKNESS/CHORD RATIO.
- 25) TER BLADE TRAILING-EDGE THICKNESS, MEASURED
NORMAL TO CAMBER-LINE AT TRAILING EDGE
(m). SEE FIGURE E.7.
- 26) XK TIP CLEARANCE (m).
- 27) XHR BLADE HEIGHT (m).
- 28) B $B=0.47$ FOR PLAIN TIP CLEARANCE.
 $B=0.37$ FOR SHROUDED.
- 29) TS NUMBER OF TIP SEALS FOR SHROUDED BLADE.
- 28) INDEX 0= STOP
1= FOR SUBSEQUENT STAGE.
NOTE: START FROM LINE 9 FOR SUBSEQUENT STAGES.

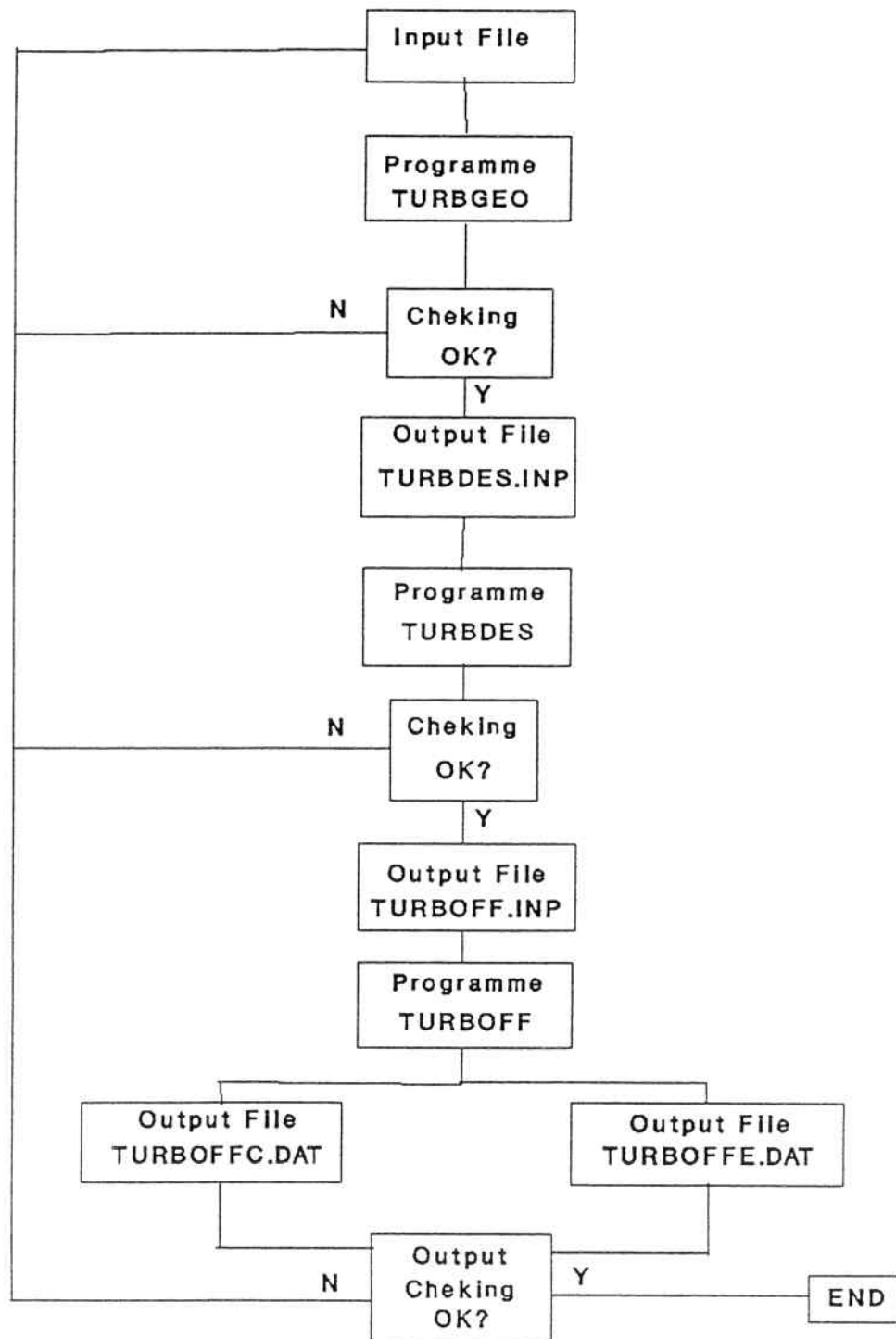


Figure E.5 Computer programme flow chart
for aerodynamic design

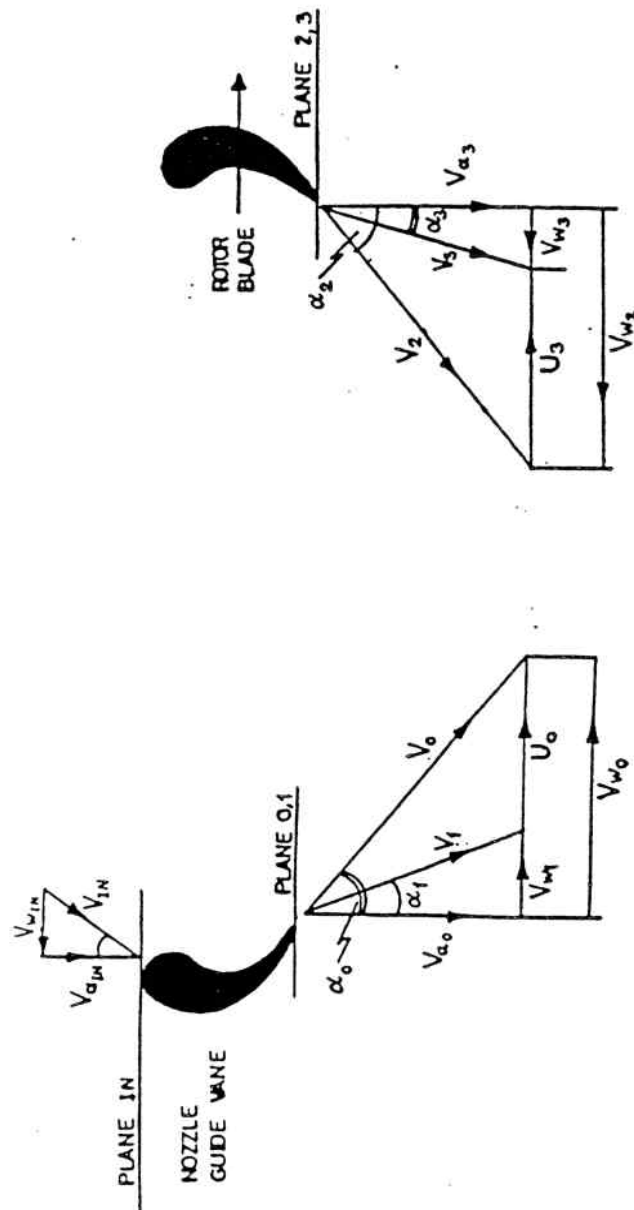
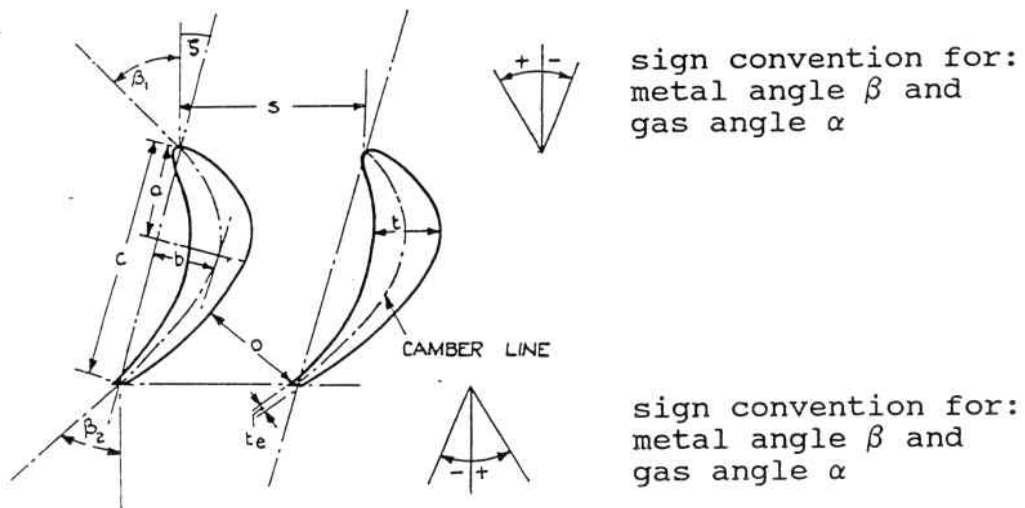


Figure E.6 Turbine velocity triangles.



- a = Distance of point of max camber from L.E.
- b = Maximum camber
- c = Chord
- e = Mean radius of curvature of upper blade surface between throat and T.E. ($e = j^2/8Z$)
- o = Blade opening
- s = Blade pitch
- t = Maximum thickness
- t_e = Trailing edge thickness
- β = Blade angle
- ξ = Stagger angle

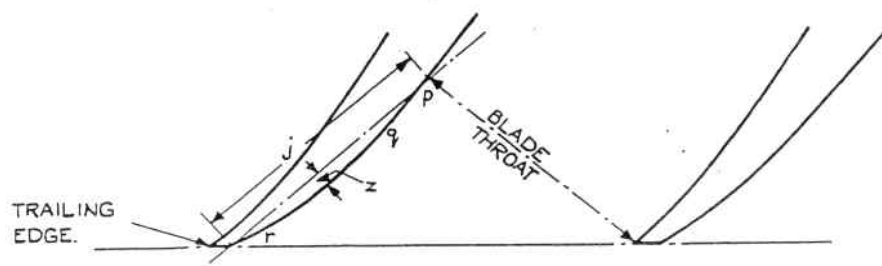


Figure E.7 Turbine Blade Nomenclature.

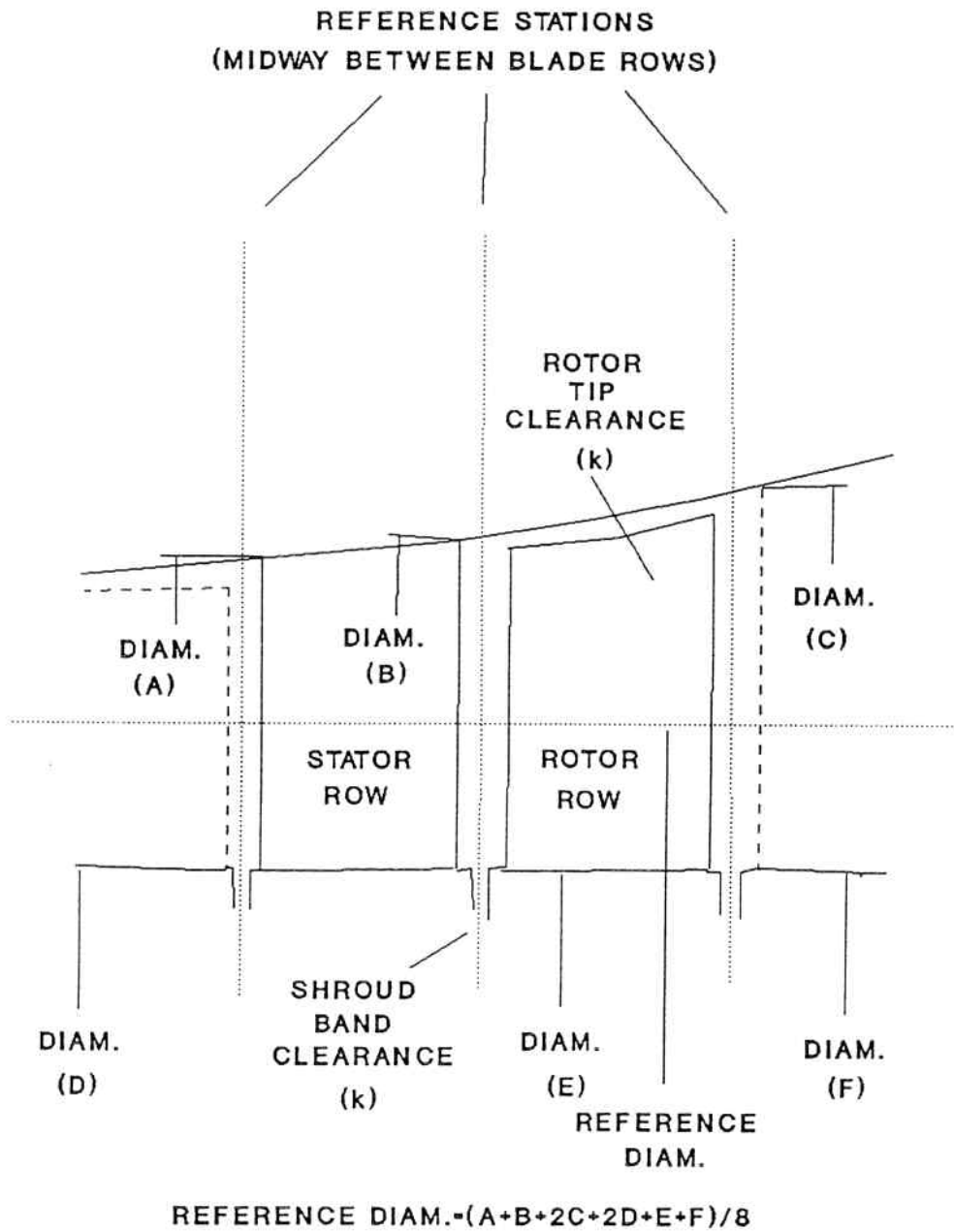


Figure E.8 Typical turbine stage with choice of reference stations and reference diameter for calculations.

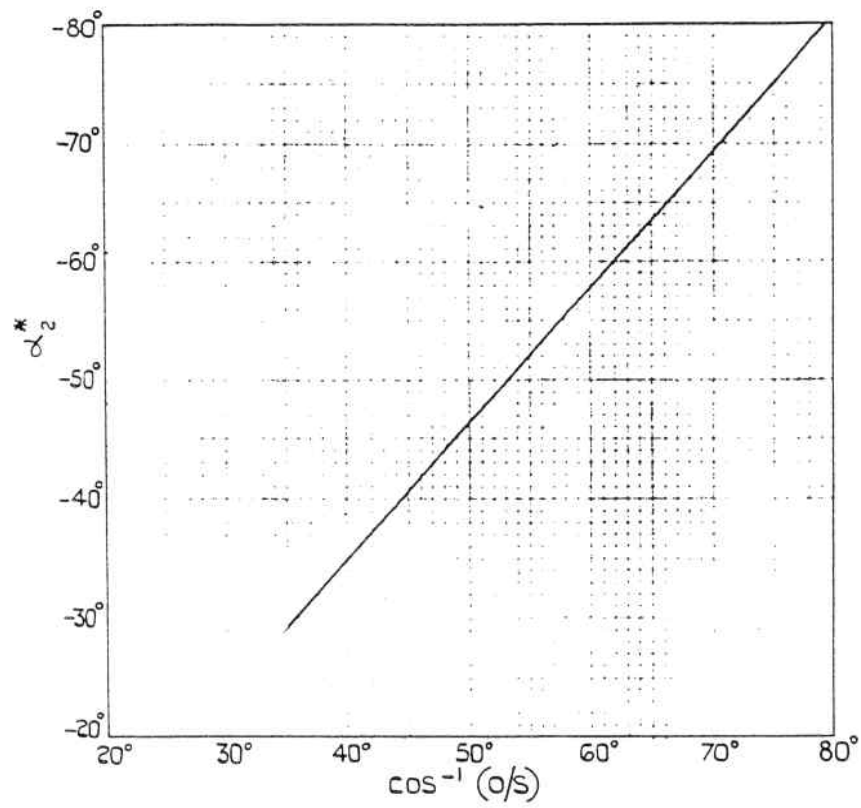


Figure E.9

Relationship between gas outlet angles and $\cos^{-1}(o/s)$ for "straight-backed" blades opening at low Mach numbers. ($R_e = 2 \times 10^5$; $M < 0.5$)

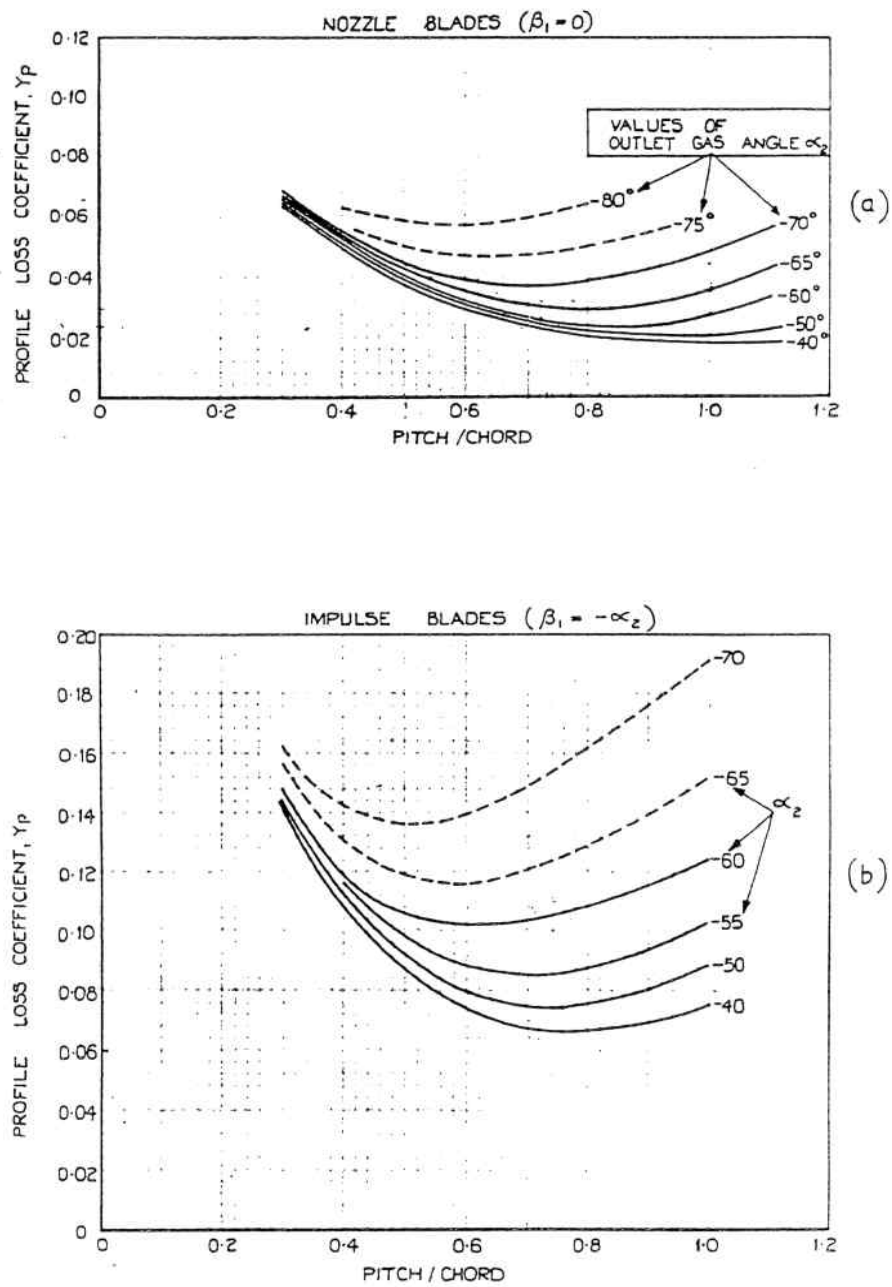


Figure E.10 Profile-loss coefficients for conventional section blades at zero incidence. ($t/c=0.02$; $R_e=2 \times 10^5$; $M < 0.6$)

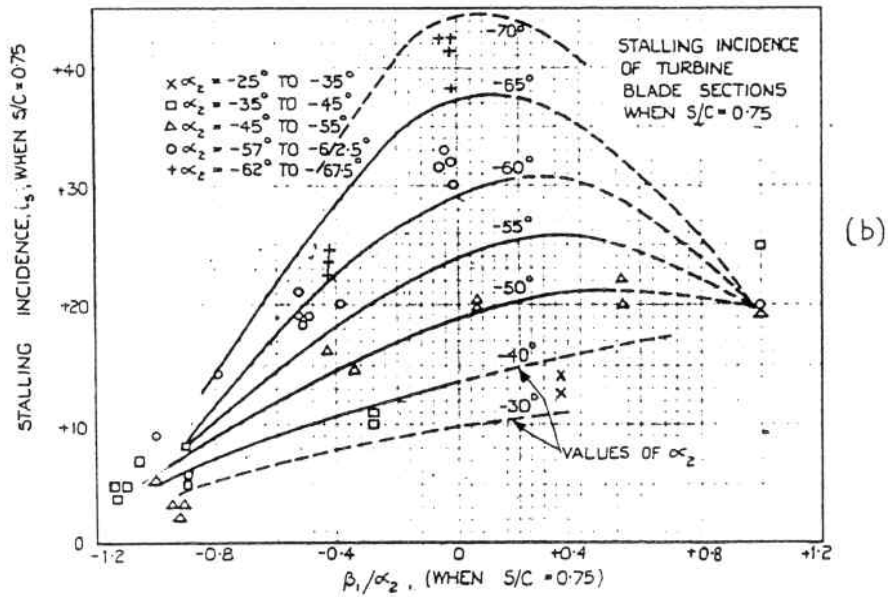
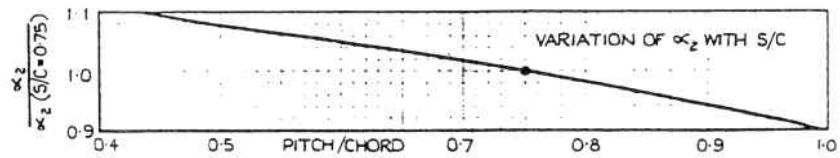
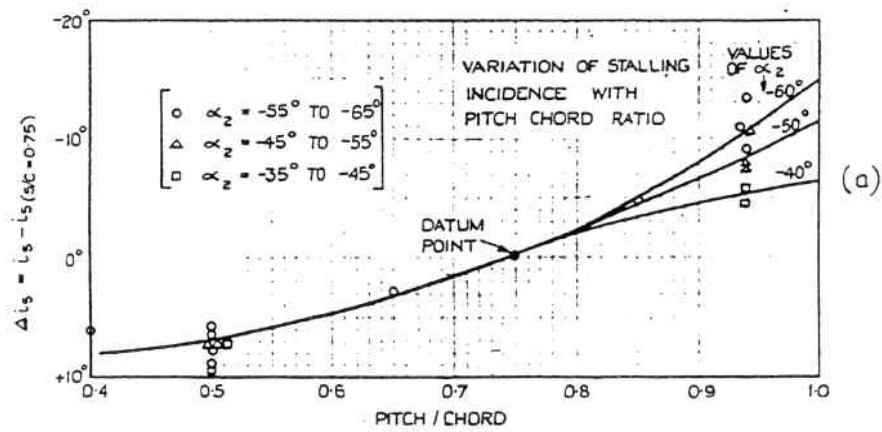


Figure E.11 Positive stalling incidence of cascade of turbine blades. ($R_c = 2 \times 10^5$; $M < 0.6$)

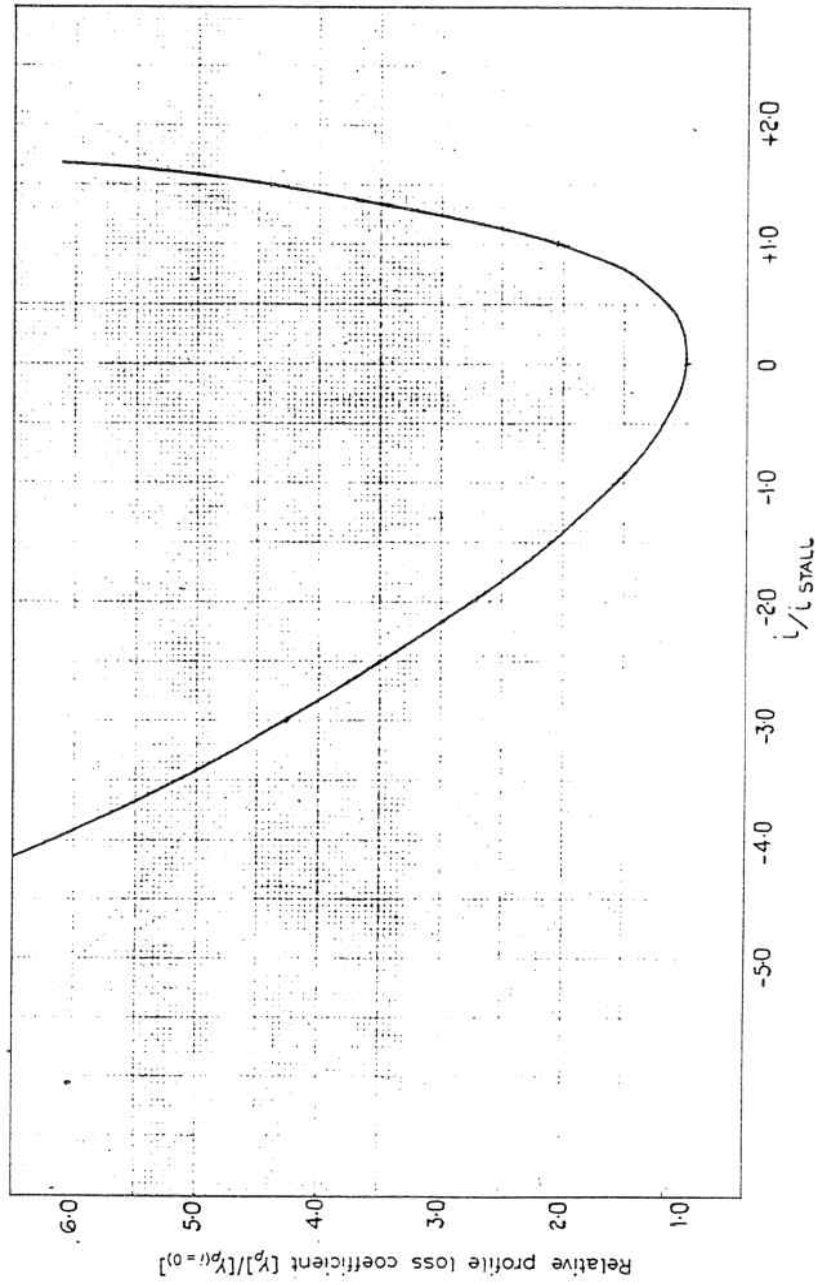


Figure E.12 Variation of profile loss with incidence for typical turbine blading.

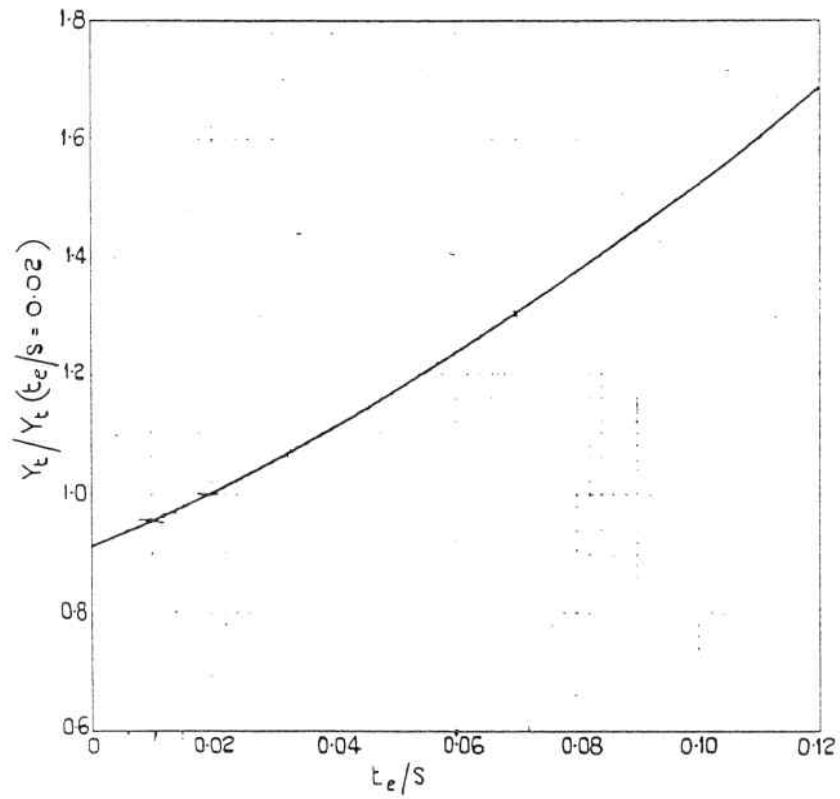


Figure E.13 Effect of trailing-edge thickness on
blade loss coefficients.

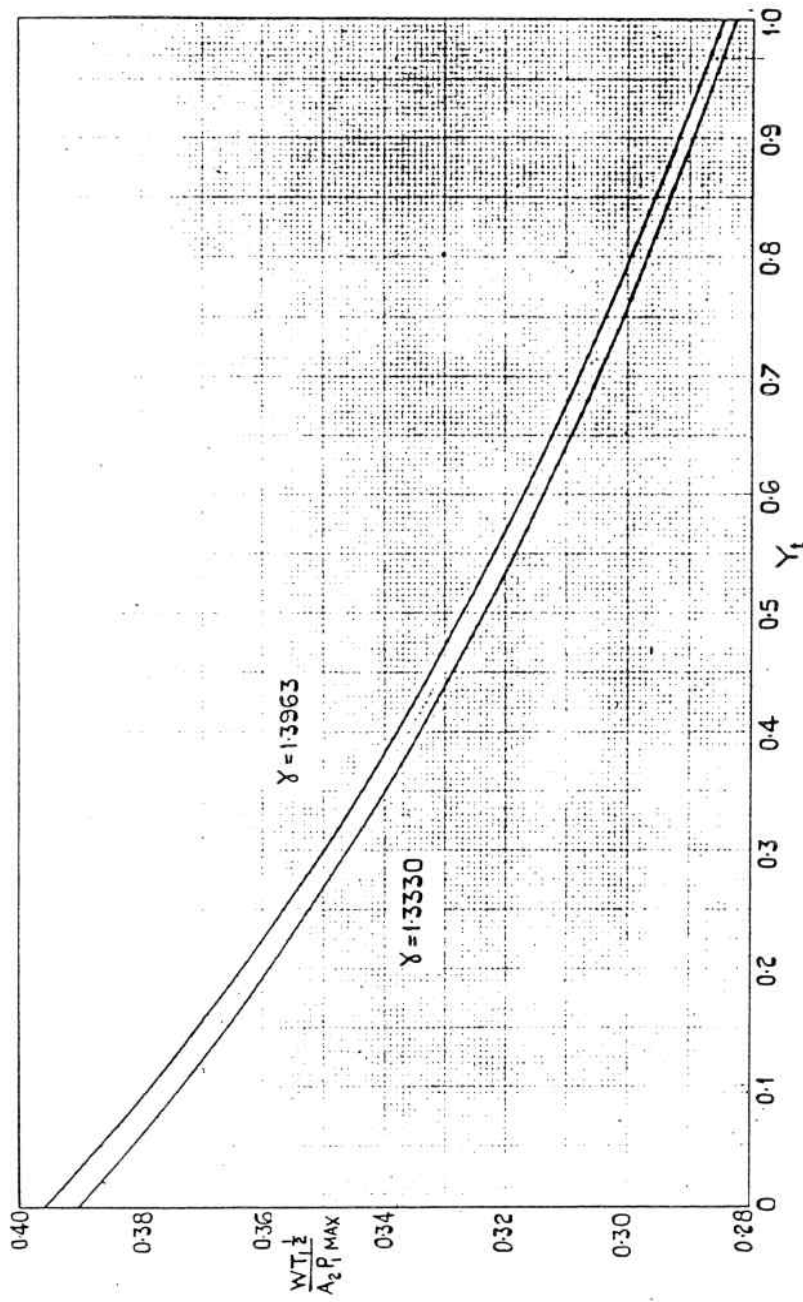


Figure E.14 Compressible adiabatic flow. (From

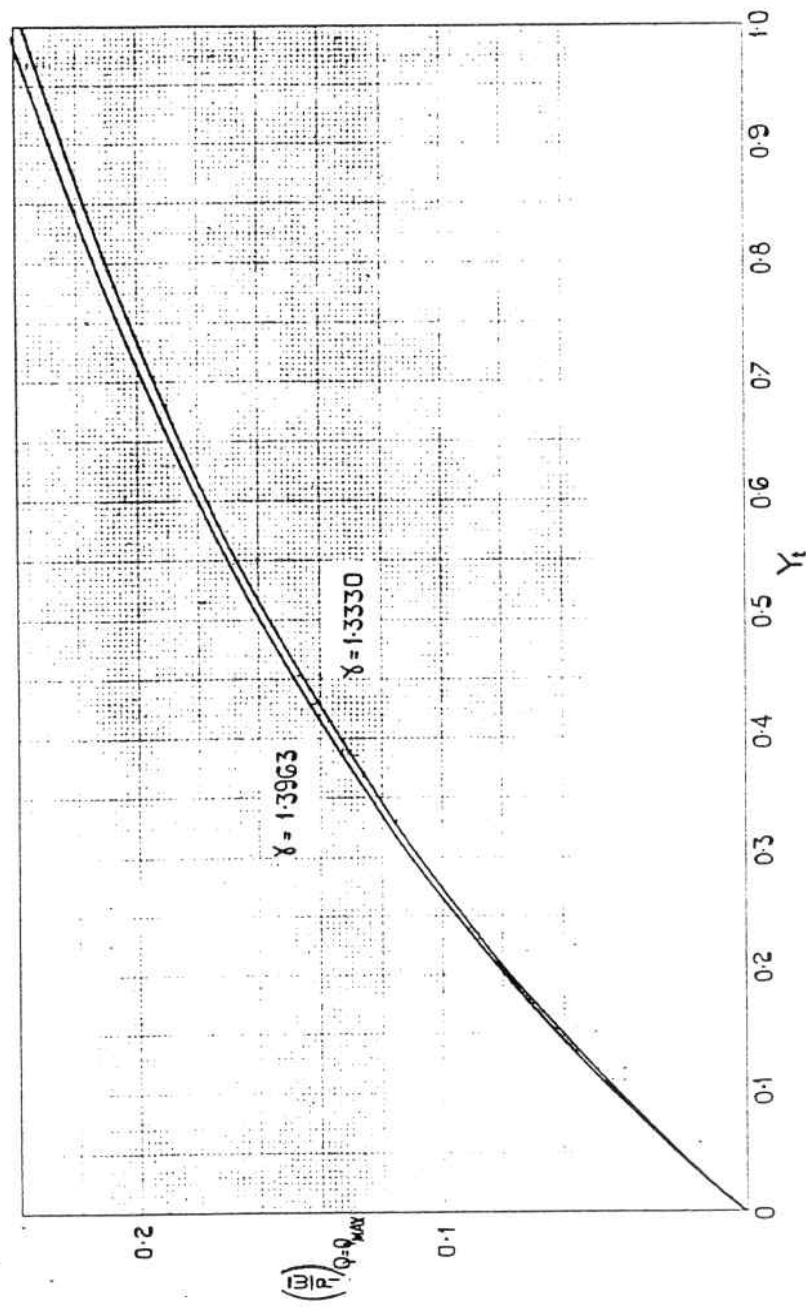


Figure E.15 Compressible adiabatic flow. (From

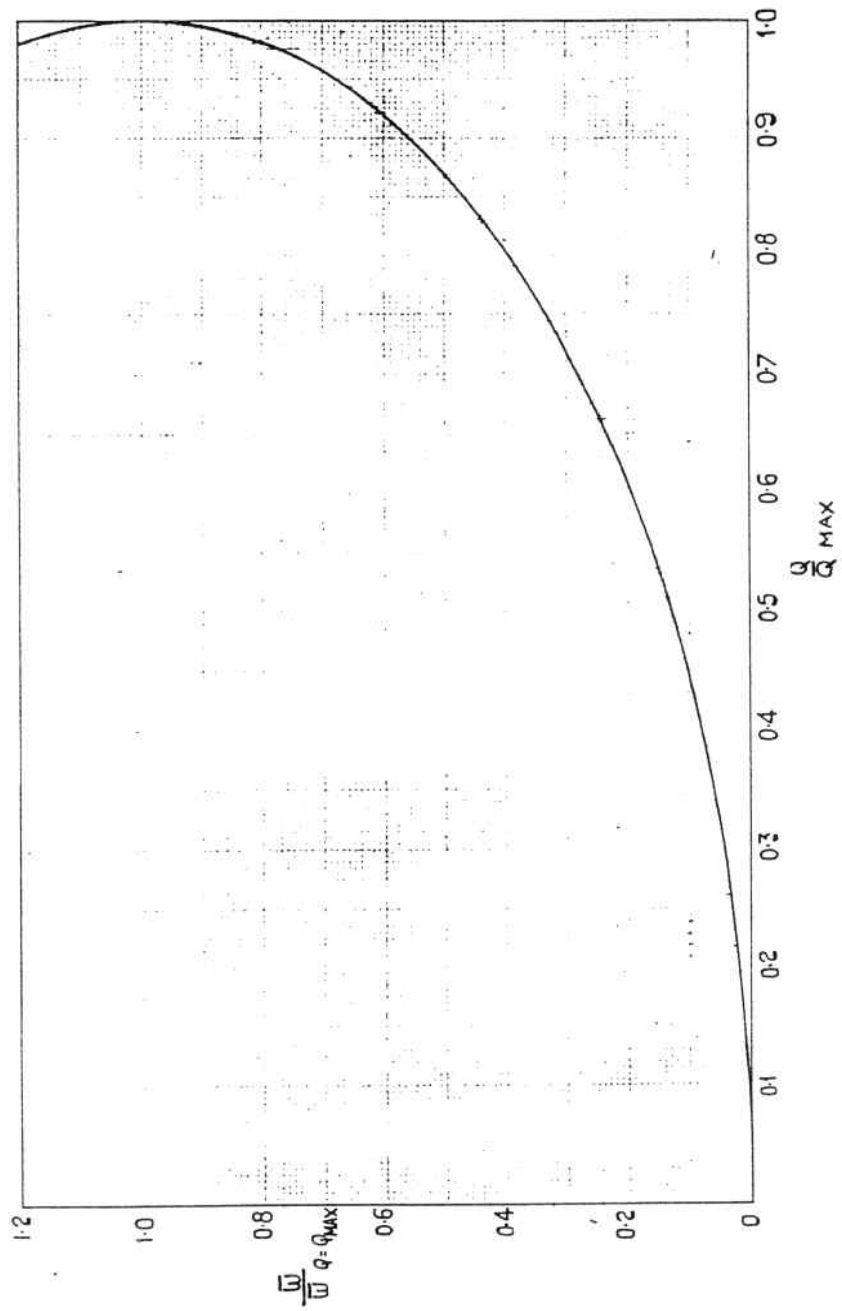


Figure E.16 Compressible adiabatic flow. (From

APPENDIX F

This appendix comprises input and output data for the computer programmes mentioned in this work.


```

5 STAGE LP COMPRESSOR      !TITLE
5                          !NUMBER OF STAGES (MAXIMUM 9)
2                          !RADIUS TYPE
0                          !INLET GUIDE VANE
0.43                      !REFERENCE RADIUS SELECTED
88.                      !ISENTROPIC EFFICIENCY
200. 200. 200. 200. 200. 200. 200. !INLET AXIAL VELOCITIES
23. 25.0 28.0 27. 26.    !TEMPERATURE RISE DISTRIBUTION
288.                    !INLET TOTAL TEMPERATURE
1.                      !INLET TOTAL PRESSURE
3.                      !PRESSURE RATIO
331.                    !INLET MASS FLOW RATIO
1                        !VORTEX DESIGN SELECTOR
0. 0. 0. 0. 0. 0. 0. 0. !SWIRL ANGLE DISTRIBUTION
4450.                  !ROTATIONAL SPEED
.98 .95 .92 .90 .88    !STAGES WORK DONE
.8 .7 .8 .7 .8 .7 .8 .7 .8 .7 ! (s/c)RATIO, ROTOR & STATOR
3. 3. 2.8 2.5 2.3     ! (h/c)RATIO, ROTOR & STATOR
0.80 0.70 0.6 0.6 0.5 ! BMH REACTION
.01 .01 .01 .01 .01   !TIP CLEARANCE
0.00001               !EQUIVALENT SAND ROUGHNESS

```

Figure F.1 LP Compressor Input Data File.

```

7 STAGE IP COMPRESSOR
7
1
0
.7
88.
190. 190. 190. 190. 190. 190. 190. 190. 190.
23. 25. 26. 27. 27. 26. 26.
406.8
2.35
3.13
243.4
1
0. 0. 5. 5. 5. 5. 5. 0.
4450.
.98 .95 .92 .90 .90 .88 .88
.8 .7 .8 .7 .8 .7 .8 .7 .8 .7 .8 .7 .8 .7
3.5 3.2 3. 2.9 2.9 2.8 2.8
0.5 0.5 0.5 0.5 0.5 .5 .5
.01 .01 .01 .01 .01 .01 .01
0.00001

```

Figure F.2 IP Compressor Input Data File.

```

7 STAGE HP COMPRESSOR
7
3
0
.74
88.
220. 220. 220. 220. 220. 220. 220. 220. 220.
38. 40. 43. 44. 44. 44. 43.
580.32
7.361
3.96
173.86
1
.0 .0 9. 10. 12. 13. 14. 13. 0.
10000.
.99 .95 .92 .90 .88 .88 .88
.8 .75 .8 .75 .8 .75 .8 .75 .8 .75 .8 .75
2. 2. 2. 2. 1.9 1.9 1.8
0.5 0.5 0.5 0.5 0.5 .5 .5
.01 .01 .01 .01 .01 .01 .01
0.00001

```

Figure F.3 HP Compressor COMPDES Input Data File.

Figure F.4 COMPDES Output Data for LP Compressor.

PROPERTY DISTRIBUTION THROUGH A 5 STAGE AXIAL COMPRESSOR WITHOUT IGVs

CONSTANT MEAN RADIUS TYPE

THE FOLLOWING CONSTANT DATA HAS BEEN USED FOR THIS RUN:

MASS FLOW (kg/s)-----	331.00									
OVERALL PRESSURE RATIO-----	3.00									
INLET AXIAL VELOCITY (m/s)-----	200.00									
ISENTROPIC EFFICIENCY (%)-----	88.00									
OVERALL TEMPERATURE RISE (K)-----	129.00									
INLET TOTAL TEMPERATURE (K)-----	288.00									
INLET TOTAL PRESSURE (atm)-----	1.00									
ROTATIONAL SPEED (rpm)-----	4450.00									
INLET IGV'S	STA 1	STA 2	STA 3	STA 4	STA 5	STA 6	STA 7	STA 8	STA 9	
TOTAL TEMPERATURE RISE (K)	23.00	25.00	28.00	27.00	26.00	.00	.00	.00	.00	
OUTLET TOTAL TEMPERATURE (K)	288.00	288.00	311.00	336.00	364.00	391.00	417.00	.00	.00	.00
AVERAGE TOTAL TEMPERATURE (K)	299.50	323.50	350.00	377.50	404.00	.00	.00	.00	.00	
OUTLET STATIC TEMPERATURE (K)	268.06	268.06	291.09	316.12	344.17	371.23	397.30	.00	.00	.00
OUTLET TOTAL PRESSURE (atm)	1.000	1.000	1.274	1.625	2.093	2.626	3.222	.000	.000	.000
OUTLET STATIC PRESSURE (atm)	.778	.778	1.010	1.313	1.719	2.187	2.715	.000	.000	.000
OUTLET DENSITY (kg/cubic m)	1.0249	1.0249	1.2255	1.4658	1.7635	2.0798	2.4124	.0000	.0000	.0000
OUTLET SPECIFIC HEAT (kJ/kg K)	1.0030	1.0030	1.0043	1.0061	1.0087	1.0117	1.0151	.0000	.0000	.0000
AVERAGE SPECIFIC HEAT (kJ/kg K)	1.0036	1.0051	1.0073	1.0101	1.0134	.0000	.0000	.0000	.0000	
OUTLET ANNULUS AREA (square m)	1.6147	1.6147	1.3505	1.1291	.9385	.7957	.6860	.0000	.0000	.0000
OUTLET INNER RADIUS (cm)	34.15	34.15	37.85	40.95	43.62	45.62	47.16	.00	.00	.00
OUTLET OUTER RADIUS (cm)	79.41	79.41	75.71	72.60	69.93	67.93	66.39	.00	.00	.00
BLADE HEIGHT (cm)	45.26	45.26	37.86	31.65	26.31	22.31	19.23	.00	.00	.00
OUTLET HUB/TIP RATIO	.430	.430	.500	.564	.624	.672	.710	.000	.000	.000
ABSOLUTE OUTLET VELOCITY (m/s)	200.00	200.00	200.00	200.00	200.00	200.00	200.00	.00	.00	.00
OUTLET SWIRL ANGLE (DEGREES)	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00
AXIAL VELOCITY (m/s)	200.00	200.00	200.00	200.00	200.00	200.00	200.00	.00	.00	.00
PRESSURE RATIO	1.2737	1.2761	1.2879	1.2544	1.2269	.0000	.0000	.0000	.0000	
ENTHALPY RISE (kJ/kg)	23.083	25.128	28.205	27.274	26.347	.000	.000	.000	.000	
COMPRESSOR POWER (kW)-----	43042.170									
OVERALL PRESSURE RATIO-----	3.2217									

5 STAGE LP COMPRESSOR

AXIAL COMPRESSOR/FAN "FREE VORTEX" DESIGN

CONSTANT MEAN RADIUS TYPE

	STAGE 1	STAGE 2	STAGE 3	STAGE 4	STAGE 5	STAGE 6	STAGE 7	STAGE 8	STAGE 9				
ALL ANNULUS DIMENSIONS IN CENTIMETRES													
OUTSIDE RADIUS	79.41	75.71	72.60	69.93	67.93	66.39	.00	.00	.00	.00			
HUB RADIUS	34.15	37.85	40.95	43.62	45.62	47.16	.00	.00	.00	.00			
INLET HEIGHT	45.263	37.856	31.649	26.306	22.306	19.231	.000	.000	.000	.000			
INLET HUB/TIP RATIO	.4300	.5000	.5641	.6238	.6716	.7103	.0000	.0000	.0000	.0000			

ASPECT RATIO	3.0000	3.0000	2.8000	2.5000	2.3000	.0000	.0000	.0000	.0000
MID CHORD HEIGHT-ROTOR	44.049	36.829	30.750	25.637	21.795	.000	.000	.000	.000
MID CHORD HEIGHT-STATOR	40.303	33.689	28.047	23.616	20.242	.000	.000	.000	.000
TRUE CHORD-ROTOR	14.683	12.276	10.982	10.255	9.476	.000	.000	.000	.000
TRUE CHORD-STATOR	13.434	11.230	10.017	9.446	8.801	.000	.000	.000	.000
ROTOR AXIAL CHORD	10.643	9.021	8.231	7.673	7.078	.000	.000	.000	.000
ROTOR AXIAL SPACE	4.405	3.683	3.295	3.076	2.843	.000	.000	.000	.000
STATOR AXIAL CHORD	13.399	11.174	9.920	9.360	8.724	.000	.000	.000	.000
STATOR AXIAL SPACE	4.030	3.369	3.005	2.834	2.640	.000	.000	.000	.000
TOTAL STAGE LENGTH	32.478	27.247	24.451	22.944	21.286	.000	.000	.000	.000
NUMBER OF ROTOR BLADES	30.4	36.3	40.6	43.5	47.1	.0	.0	.0	.0
NUMBER OF STATOR BLADES	37.9	45.4	50.9	53.9	57.9	.0	.0	.0	.0

TOTAL COMPRESSOR LENGTH 128.405

		STAGE 1							
		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP			
BLADE SPEED	(U)	161.9	213.3	264.6	315.9	367.2	M/S		
ABS. STATOR OUTLET VELOCITY	(V0)	200.0	200.0	200.0	200.0	200.0	M/S		
REL. ROTOR INLET VELOCITY	(V1)	257.3	292.4	331.7	373.9	418.2	M/S		
REL. ROTOR OUTLET VELOCITY	(V2)	200.7	224.9	266.1	313.4	363.1	M/S		
ABS. STATOR INLET VELOCITY	(V3)	247.3	228.5	218.9	213.4	210.0	M/S		
ABS. STATOR OUTLET VELOCITY	(V4)	200.0	200.0	200.0	200.0	200.0	M/S		
AXIAL VELOCITY AT ROTOR INLET	(VA1)	200.0	200.0	200.0	200.0	200.0	M/S		
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	200.0	200.0	200.0	200.0	200.0	M/S		
STATIC PRESSURE AT ROTOR INLET	(PS1)	.666	.597	.522	.446	.374	ATM		
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	.876	.925	.950	.964	.972	ATM		
REL. ROTOR INLET MACH NUMBER	(M1REL)	.78	.89	1.01	1.14	1.27			
ABS. STATOR INLET MACH NUMBER	(M3ABS)	.74	.68	.64	.63	.62			
ROTOR DECELERATION	(V2/V1)	.78	.77	.80	.84	.87	DE HALLER NUMBER		
PRESSURE RISE COEFFICIENT	(DP/D)	.39	.41	.36	.30	.25			
STATOR DECELERATION	(V0/V3)	.81	.88	.91	.94	.95	DE HALLER NUMBER		
STAGE LOADING (DELTA H/U SQUARED)		.90	.52	.34	.24	.17			
FLOW COEFFICIENT (VA/U)		1.23	.94	.76	.63	.54			
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	.0	.0	.0	.0	.0	DEGREES		
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	39.0	46.8	52.9	57.7	61.4	DEGREES		
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	4.7	27.2	41.3	50.4	56.6	DEGREES		
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	36.0	28.9	24.0	20.4	17.8	DEGREES		
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	.0	.0	.0	.0	.0	DEGREES		
ROTOR DEFLECTION		34.3	19.6	11.6	7.3	4.8	DEGREES		
STATOR DEFLECTION		36.0	28.9	24.0	20.4	17.8	DEGREES		
REACTION		55.1	74.1	83.2	88.2	91.3	PERCENT		
SPACE CHORD RATIO (ROTOR)		.648	.749	.800	.826	.840			
SPACE CHORD RATIO (STATOR)		.443	.573	.700	.826	.951			
DIFFUSION FACTOR (ROTOR)		.403	.372	.305	.244	.196			
DIFFUSION FACTOR (STATOR)		.321	.263	.229	.207	.193			
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.659	1.610	1.494	1.400	1.333			
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.544	1.448	1.400	1.371	1.353			
SHOCK LOSS COEFFICIENT (ROTOR)		.00000	.00304	.02332	.03509	.06640			
SHOCK LOSS COEFFICIENT (STATOR)		.00000	.00000	.00000	.00000	.00000			
REYNOLDS NUMBER (ROTOR)		.20E+06	.21E+06	.22E+06	.23E+06	.24E+06			
REYNOLDS NUMBER (STATOR)		.30E+06	.31E+06	.32E+06	.33E+06	.33E+06			
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000			
AXIAL VELOCITY RATIO (STATOR)		1.000	1.000	1.000	1.000	1.000			

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.02388	.02071	.02239	.02503	.02820
PROFILE LOSS COEFFICIENT (STATOR)	.03518	.02880	.02430	.02082	.01826
SECONDARY LOSS COEFFICIENT (ROTOR)			.00617		
SECONDARY LOSS COEFFICIENT (STATOR)			.00854		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00797		
ANNULUS LOSS COEFFICIENT (STATOR)			.00598		
TOTAL LOSS COEFFICIENT (ROTOR)			.06375		
TOTAL LOSS COEFFICIENT (STATOR)			.03999		

EFFICIENCY LOSS DUE TO TIP CLEARANCE .025

ROTOR ISENTROPIC EFFICIENCY 86.08 PERCENT

STAGE ISENTROPIC EFFICIENCY 84.31 PERCENT

STAGE PRESSURE RATIO 1.26

		STAGE 2							
		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP			
BLADE SPEED	(U)	178.8	221.7	264.6	307.5	350.4	M/S		
ABS. STATOR OUTLET VELOCITY	(V0)	200.0	200.0	200.0	200.0	200.0	M/S		
REL. ROTOR INLET VELOCITY	(V1)	268.3	298.6	331.7	366.8	403.5	M/S		
REL. ROTOR OUTLET VELOCITY	(V2)	202.4	224.7	259.0	298.4	340.0	M/S		
ABS. STATOR INLET VELOCITY	(V3)	248.8	232.9	223.6	217.7	213.8	M/S		

ABS. STATOR OUTLET VELOCITY (V4)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	.837	.764	.687	.608	.532	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	1.137	1.187	1.216	1.234	1.245	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	.78	.87	.97	1.07	1.18	
ABS. STATOR INLET MACH NUMBER (M3ABS)	.71	.66	.63	.61	.60	
ROTOR DECELERATION (V2/V1)	.75	.75	.78	.81	.84	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT (DP/D)	.43	.43	.39	.34	.29	
STATOR DECELERATION (V0/V3)	.80	.86	.89	.92	.94	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	.83	.54	.38	.28	.22	
FLOW COEFFICIENT (VA/U)	1.12	.90	.76	.65	.57	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	.0	.0	.0	.0	.0	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	41.8	47.9	52.9	57.0	60.3	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	8.8	27.1	39.5	47.9	54.0	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	36.5	30.8	26.6	23.3	20.7	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	.0	.0	.0	.0	.0	DEGREES
ROTOR DEFLECTION	33.0	20.8	13.5	9.0	6.3	DEGREES
STATOR DEFLECTION	36.5	30.8	26.6	23.3	20.7	DEGREES
REACTION	58.6	73.1	81.1	86.0	89.2	PERCENT
SPACE CHORD RATIO (ROTOR)	.629	.761	.800	.821	.833	
SPACE CHORD RATIO (STATOR)	.483	.593	.700	.806	.912	
DIFFUSION FACTOR (ROTOR)	.436	.400	.340	.283	.235	
DIFFUSION FACTOR (STATOR)	.340	.293	.262	.241	.225	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.714	1.654	1.548	1.455	1.385	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.566	1.489	1.443	1.413	1.393	
SHOCK LOSS COEFFICIENT (ROTOR)	.00000	.00244	.00569	.02412	.04107	
SHOCK LOSS COEFFICIENT (STATOR)	.00000	.00000	.00000	.00000	.00000	
REYNOLDS NUMBER (ROTOR)	.22E+06	.22E+06	.24E+06	.25E+06	.26E+06	
REYNOLDS NUMBER (STATOR)	.32E+06	.34E+06	.34E+06	.35E+06	.35E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	1.090	1.000	1.000	1.000	1.000	

 CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.02178	.01987	.02117	.02352	.02624
PROFILE LOSS COEFFICIENT (STATOR)	.03185	.02727	.02372	.02097	.01879
SECONDARY LOSS COEFFICIENT (ROTOR)			.00768		
SECONDARY LOSS COEFFICIENT (STATOR)			.01039		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00765		
ANNULUS LOSS COEFFICIENT (STATOR)			.00584		
TOTAL LOSS COEFFICIENT (ROTOR)			.05251		
TOTAL LOSS COEFFICIENT (STATOR)			.04075		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.026

ROTOR ISENTROPIC EFFICIENCY

90.80 PERCENT

STAGE ISENTROPIC EFFICIENCY

87.09 PERCENT

STAGE PRESSURE RATIO

1.27

STAGE 3						
	BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	192.9	228.8	264.6	300.4	336.2	M/S
ABS. STATOR OUTLET VELOCITY (V0)	200.0	200.0	200.0	200.0	200.0	M/S
REL. ROTOR INLET VELOCITY (V1)	277.9	303.9	331.7	360.9	391.2	M/S
REL. ROTOR OUTLET VELOCITY (V2)	202.9	221.3	249.2	281.7	316.3	M/S
ABS. STATOR INLET VELOCITY (V3)	255.4	240.7	231.1	224.5	219.8	M/S
ABS. STATOR OUTLET VELOCITY (V4)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	1.065	.990	.910	.828	.748	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	1.473	1.530	1.567	1.591	1.608	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	.78	.85	.93	1.01	1.10	
ABS. STATOR INLET MACH NUMBER (M3ABS)	.70	.66	.63	.61	.60	
ROTOR DECELERATION (V2/V1)	.73	.73	.75	.78	.81	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT (DP/D)	.47	.47	.44	.39	.35	
STATOR DECELERATION (V0/V3)	.78	.83	.87	.89	.91	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	.82	.59	.44	.34	.27	
FLOW COEFFICIENT (VA/U)	1.04	.87	.76	.67	.59	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	.0	.0	.0	.0	.0	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	44.0	48.8	52.9	56.3	59.3	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	9.7	25.3	36.6	44.8	50.8	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	38.5	33.8	30.1	27.0	24.5	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	.0	.0	.0	.0	.0	DEGREES
ROTOR DEFLECTION	34.3	23.5	16.3	11.6	8.5	DEGREES
STATOR DEFLECTION	38.5	33.8	30.1	27.0	24.5	DEGREES
REACTION	58.8	70.7	78.1	83.0	86.4	PERCENT
SPACE CHORD RATIO (ROTOR)	.718	.771	.800	.816	.826	
SPACE CHORD RATIO (STATOR)	.517	.609	.700	.790	.879	
DIFFUSION FACTOR (ROTOR)	.475	.442	.388	.335	.288	
DIFFUSION FACTOR (STATOR)	.378	.339	.310	.289	.272	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.761	1.725	1.627	1.535	1.459	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.627	1.555	1.508	1.476	1.453	
SHOCK LOSS COEFFICIENT (ROTOR)	.00000	.00177	.00439	.02121	.02577	

SHOCK LOSS COEFFICIENT (STATOR)	.00000	.00000	.00000	.00000	.00000
REYNOLDS NUMBER (ROTOR)	.25E+06	.26E+06	.27E+06	.28E+06	.29E+06
REYNOLDS NUMBER (STATOR)	.37E+06	.38E+06	.39E+06	.40E+06	.40E+06
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000
AXIAL VELOCITY RATIO (STATOR)	1.000	1.000	1.000	1.000	1.000

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.02030	.01897	.01961	.02149	.02377
PROFILE LOSS COEFFICIENT (STATOR)	.02879	.02551	.02287	.02072	.01893
SECONDARY LOSS COEFFICIENT (ROTOR)			.01011		
SECONDARY LOSS COEFFICIENT (STATOR)			.01319		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00772		
ANNULUS LOSS COEFFICIENT (STATOR)			.00603		
TOTAL LOSS COEFFICIENT (ROTOR)			.04928		
TOTAL LOSS COEFFICIENT (STATOR)			.04259		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.027

ROTOR ISENTROPIC EFFICIENCY

92.12 PERCENT

STAGE ISENTROPIC EFFICIENCY

88.43 PERCENT

STAGE PRESSURE RATIO

1.28

		STAGE 4				
		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP
BLADE SPEED (U)	204.8	234.7	264.6	294.5	324.3	M/S
ABS. STATOR OUTLET VELOCITY (V0)	200.0	200.0	200.0	200.0	200.0	M/S
REL. ROTOR INLET VELOCITY (V1)	286.5	308.4	331.7	356.0	381.0	M/S
REL. ROTOR OUTLET VELOCITY (V2)	207.9	226.2	250.0	276.9	305.5	M/S
ABS. STATOR INLET VELOCITY (V3)	248.8	238.1	230.5	224.9	220.8	M/S
ABS. STATOR OUTLET VELOCITY (V4)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	1.380	1.301	1.218	1.133	1.048	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	1.919	1.968	2.002	2.026	2.044	ATM
REL. ROTOR INLET MACH NUMBER (MREL)	.77	.83	.89	.96	1.03	
ABS. STATOR INLET MACH NUMBER (M3ABS)	.65	.62	.60	.59	.58	
ROTOR DECELERATION (V2/V1)	.73	.73	.75	.78	.80	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT (DP/D)	.47	.46	.43	.39	.36	
STATOR DECELERATION (V0/V3)	.80	.84	.87	.89	.91	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	.72	.55	.43	.35	.29	
FLOW COEFFICIENT (VA/U)	.98	.85	.76	.68	.62	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	.0	.0	.0	.0	.0	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	45.7	49.6	52.9	55.8	58.3	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	15.9	27.8	36.9	43.8	49.1	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	36.5	32.8	29.8	27.2	25.0	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	.0	.0	.0	.0	.0	DEGREES
ROTOR DEFLECTION	29.8	21.7	16.0	12.1	9.2	DEGREES
STATOR DEFLECTION	36.5	32.8	29.8	27.2	25.0	DEGREES
REACTION	63.9	72.5	78.4	82.5	85.6	PERCENT
SPACE CHORD RATIO (ROTOR)	.739	.777	.800	.814	.823	
SPACE CHORD RATIO (STATOR)	.546	.624	.700	.775	.850	
DIFFUSION FACTOR (ROTOR)	.465	.429	.384	.340	.299	
DIFFUSION FACTOR (STATOR)	.359	.329	.306	.288	.274	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.766	1.702	1.621	1.543	1.478	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.591	1.539	1.503	1.476	1.456	
SHOCK LOSS COEFFICIENT (ROTOR)	.00000	.00100	.00310	.00529	.01621	
SHOCK LOSS COEFFICIENT (STATOR)	.00000	.00000	.00000	.00000	.00000	
REYNOLDS NUMBER (ROTOR)	.31E+06	.32E+06	.33E+06	.34E+06	.35E+06	
REYNOLDS NUMBER (STATOR)	.47E+06	.48E+06	.48E+06	.49E+06	.50E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01917	.01870	.01941	.02094	.02280
PROFILE LOSS COEFFICIENT (STATOR)	.02718	.02457	.02243	.02061	.01907
SECONDARY LOSS COEFFICIENT (ROTOR)			.00989		
SECONDARY LOSS COEFFICIENT (STATOR)			.01295		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00869		
ANNULUS LOSS COEFFICIENT (STATOR)			.00678		
TOTAL LOSS COEFFICIENT (ROTOR)			.04391		
TOTAL LOSS COEFFICIENT (STATOR)			.04250		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.026

ROTOR ISENTROPIC EFFICIENCY

92.66 PERCENT

STAGE ISENTROPIC EFFICIENCY

88.84 PERCENT

STAGE PRESSURE RATIO

1.25

		STAGE 5					
		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	213.8	239.2	264.6	290.0	315.4	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	200.0	200.0	200.0	200.0	200.0	M/S
REL. ROTOR INLET VELOCITY	(V1)	292.8	311.8	331.7	352.3	373.4	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	213.2	230.2	250.9	273.6	297.6	M/S
ABS. STATOR INLET VELOCITY	(V3)	244.2	235.9	229.6	225.1	221.4	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET	(PS1)	1.747	1.665	1.580	1.492	1.403	ATM
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	2.419	2.462	2.494	2.518	2.536	ATM
REL. ROTOR INLET MACH NUMBER	(M1REL)	.76	.81	.86	.91	.97	
ABS. STATOR INLET MACH NUMBER	(M3ABS)	.62	.60	.58	.57	.56	
ROTOR DECELERATION	(V2/V1)	.73	.74	.76	.78	.80	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT	(DP/D)	.47	.45	.43	.40	.36	
STATOR DECELERATION	(V0/V3)	.82	.85	.87	.89	.90	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)		.65	.52	.43	.36	.30	
FLOW COEFFICIENT	(VA/U)	.94	.84	.76	.69	.63	
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	.0	.0	.0	.0	.0	DEGREES
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	46.9	50.1	52.9	55.4	57.6	DEGREES
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	20.2	29.7	37.1	43.0	47.8	DEGREES
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	35.0	32.0	29.5	27.3	25.4	DEGREES
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	.0	.0	.0	.0	.0	DEGREES
ROTOR DEFLECTION		26.7	20.4	15.8	12.4	9.8	DEGREES
STATOR DEFLECTION		35.0	32.0	29.5	27.3	25.4	DEGREES
REACTION		67.3	73.8	78.6	82.2	84.9	PERCENT
SPACE CHORD RATIO (ROTOR)		.752	.781	.800	.813	.821	
SPACE CHORD RATIO (STATOR)		.569	.635	.700	.765	.829	
DIFFUSION FACTOR (ROTOR)		.452	.418	.380	.342	.307	
DIFFUSION FACTOR (STATOR)		.344	.321	.302	.287	.274	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.744	1.683	1.614	1.548	1.491	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.566	1.527	1.497	1.474	1.457	
SHOCK LOSS COEFFICIENT (ROTOR)		.00000	.00000	.00201	.00380	.00564	
SHOCK LOSS COEFFICIENT (STATOR)		.00000	.00000	.00000	.00000	.00000	
REYNOLDS NUMBER (ROTOR)		.37E+06	.38E+06	.39E+06	.40E+06	.41E+06	
REYNOLDS NUMBER (STATOR)		.56E+06	.57E+06	.58E+06	.58E+06	.59E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		1.000	1.000	1.000	1.000	1.000	

 CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01862	.01843	.01928	.02056	.02211
PROFILE LOSS COEFFICIENT (STATOR)	.02603	.02390	.02210	.02054	.01919
SECONDARY LOSS COEFFICIENT (ROTOR)			.00967		
SECONDARY LOSS COEFFICIENT (STATOR)			.01270		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00950		
ANNULUS LOSS COEFFICIENT (STATOR)			.00739		
TOTAL LOSS COEFFICIENT (ROTOR)			.04126		
TOTAL LOSS COEFFICIENT (STATOR)			.04245		

EFFICIENCY LOSS DUE TO TIP CLEARANCE		.025	
ROTOR ISENTROPIC EFFICIENCY		92.80	PERCENT
STAGE ISENTROPIC EFFICIENCY		88.85	PERCENT
STAGE PRESSURE RATIO		1.22	

OVERALL PRESSURE RATIO	3.13		
COMPRESSOR ISENTROPIC EFFICIENCY	85.23	PERCENT	
COMPRESSOR POLYTROPIC EFFICIENCY	87.36	PERCENT	

Figure F.5 COMPDES Output Data for IP Compressor.

PROPERTY DISTRIBUTION THROUGH A 7 STAGE AXIAL COMPRESSOR WITHOUT IGVs

CONSTANT HUB RADIUS TYPE

THE FOLLOWING CONSTANT DATA HAS BEEN USED FOR THIS RUN:

MASS FLOW (kg/s)	243.40									
OVERALL PRESSURE RATIO	3.31									
INLET AXIAL VELOCITY (m/s)	190.00									
ISENTROPIC EFFICIENCY (%)	88.00									
OVERALL TEMPERATURE RISE (K)	180.00									
INLET TOTAL TEMPERATURE (K)	406.80									
INLET TOTAL PRESSURE (atm)	2.35									
ROTATIONAL SPEED (rpm)	4450.00									
INLET IGV'S	STA 1	STA 2	STA 3	STA 4	STA 5	STA 6	STA 7	STA 8	STA 9	
TOTAL TEMPERATURE RISE (K)	23.00	25.00	26.00	27.00	27.00	26.00	26.00	.00	.00	
OUTLET TOTAL TEMPERATURE (K)	406.80	406.80	429.80	454.80	480.80	507.80	534.80	560.80	586.80	.00 .00
AVERAGE TOTAL TEMPERATURE (K)	418.30	442.30	467.80	494.30	521.30	547.80	573.80	.00	.00	
OUTLET STATIC TEMPERATURE (K)	388.99	388.99	411.92	436.98	463.06	490.15	517.24	543.33	569.56	.00 .00
OUTLET TOTAL PRESSURE (atm)	2.350	2.350	2.800	3.354	4.008	4.780	5.653	6.597	7.651	.000 .000
OUTLET STATIC PRESSURE (atm)	2.006	2.006	2.408	2.909	3.505	4.210	5.011	5.881	6.862	.000 .000
OUTLET DENSITY (kg/cubic m)	1.8210	1.8210	2.0641	2.3506	2.6720	3.0322	3.4204	3.8216	4.2537	.0000 .0000
OUTLET SPECIFIC HEAT (kJ/kg K)	1.0137	1.0137	1.0170	1.0209	1.0253	1.0304	1.0358	1.0414	1.0472	.0000 .0000
AVERAGE SPECIFIC HEAT (kJ/kg K)	1.0153	1.0189	1.0230	1.0278	1.0331	1.0386	1.0443	.0000	.0000	
OUTLET ANNULUS AREA (square m)	.7035	.7035	.6206	.5450	.4794	.4225	.3745	.3352	.3012	.0000 .0000
OUTLET INNER RADIUS (cm)	46.38	46.38	46.38	46.38	46.38	46.38	46.38	46.38	46.38	.00 .00
OUTLET OUTER RADIUS (cm)	66.26	66.26	64.24	62.34	60.64	59.13	57.82	56.73	55.77	.00 .00
BLADE HEIGHT (cm)	19.88	19.88	17.86	15.96	14.26	12.75	11.44	10.35	9.38	.00 .00
OUTLET HUB/TIP RATIO	.700	.700	.722	.744	.765	.784	.802	.818	.832	.000 .000
ABSOLUTE OUTLET VELOCITY (m/s)	190.00	190.00	190.73	190.73	190.73	190.73	190.73	190.73	190.00	.00 .00
OUTLET SWIRL ANGLE (DEGREES)	.00	.00	5.00	5.00	5.00	5.00	5.00	5.00	.00	.00 .00
AXIAL VELOCITY (m/s)	190.00	190.00	190.00	190.00	190.00	190.00	190.00	190.00	190.00	.00 .00
PRESSURE RATIO	1.1913	1.1979	1.1952	1.1925	1.1826	1.1670	1.1596	.0000	.0000	
ENTHALPY RISE (kJ/kg)	23.352	25.471	26.599	27.751	27.893	27.003	27.151	.000	.000	
COMPRESSOR POWER (kW)	45082.410									
OVERALL PRESSURE RATIO	3.2557									

7 STAGE IP COMPRESSOR

AXIAL COMPRESSOR/FAN "FREE VORTEX" DESIGN

CONSTANT HUB RADIUS TYPE

ALL ANNULUS DIMENSIONS IN CENTIMETRES

	STAGE 1	STAGE 2	STAGE 3	STAGE 4	STAGE 5	STAGE 6	STAGE 7	STAGE 8	STAGE 9
OUTSIDE RADIUS	66.26	64.24	62.34	60.64	59.13	57.82	56.73	55.77	.00 .00

HUB RADII	46.38	46.38	46.38	46.38	46.38	46.38	46.38	46.38	.00	.00
INLET HEIGHT	19.879	17.858	15.956	14.259	12.745	11.440	10.348	9.384	.00	.00
INLET HUB/TIP RATIO	.7000	.7220	.7441	.7649	.7845	.8022	.8176	.8317	.0000	.0000
ASPECT RATIO	3.5000	3.2000	3.0000	2.9000	2.8000	2.8000	2.8000	.0000	.0000	.0000
MID CHORD HEIGHT-ROTOR	19.558	17.538	15.665	13.995	12.516	11.248	10.178	.000	.000	.000
MID CHORD HEIGHT-STATOR	18.940	16.960	14.811	13.225	11.860	10.698	9.693	.000	.000	.000
TRUE CHORD-ROTOR	5.588	5.481	5.222	4.824	4.326	4.017	3.635	.000	.000	.000
TRUE CHORD-STATOR	5.297	5.181	4.937	4.563	4.089	3.821	3.462	.000	.000	.000
ROTOR AXIAL CHORD	3.966	4.185	4.086	3.861	3.495	3.279	2.993	.000	.000	.000
ROTOR AXIAL SPACE	1.676	1.644	1.567	1.448	1.295	1.205	1.090	.000	.000	.000
STATOR AXIAL CHORD	5.251	5.070	4.803	4.413	3.949	3.689	3.373	.000	.000	.000
STATOR AXIAL SPACE	1.589	1.554	1.481	1.369	1.227	1.146	1.039	.000	.000	.000
TOTAL STAGE LENGTH	12.483	12.454	11.937	11.091	9.565	8.319	6.495	.000	.000	.000
NUMBER OF ROTOR BLADES	70.9	79.0	81.5	88.9	95.8	101.7	111.2	.0	.0	.0
NUMBER OF STATOR BLADES	94.5	94.7	97.8	104.3	116.8	121.5	132.8	.0	.0	.0

TOTAL COMPRESSOR LENGTH 75.744

		STAGE 1		STAGE 2		STAGE 3		STAGE 4	
		BLADE ROOT	75%	BLADE MID HEIGHT	75%	BLADE TIP	75%	BLADE TIP	75%
BLADE SPEED	(U)	216.9	239.7	262.5	285.3	308.0	M/S		
ABS. STATOR OUTLET VELOCITY	(V0)	190.0	190.0	190.0	190.0	190.0	M/S		
REL. ROTOR INLET VELOCITY	(V1)	288.3	305.9	324.0	342.7	361.9	M/S		
REL. ROTOR OUTLET VELOCITY	(V2)	218.1	234.2	256.1	277.3	298.9	M/S		
ABS. STATOR INLET VELOCITY	(V3)	219.5	214.4	210.4	207.6	205.1	M/S		
ABS. STATOR OUTLET VELOCITY	(V4)	190.0	190.0	190.0	190.0	190.0	M/S		
AXIAL VELOCITY AT ROTOR INLET	(VA1)	190.0	190.0	190.0	190.0	190.0	M/S		
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	180.0	180.0	180.0	180.0	180.0	M/S		
STATIC PRESSURE AT ROTOR INLET	(P51)	1.645	1.582	1.511	1.439	1.366	ATM		
STATIC PRESSURE AT ROTOR OUTLET	(P52)	2.289	2.311	2.327	2.340	2.350	ATM		
REL. ROTOR INLET MACH NUMBER (MREL)		.73	.78	.82	.87	.92			
ABS. STATOR INLET MACH NUMBER (MABS)		.54	.53	.52	.51	.51			
ROTOR DECELERATION	(V2/V1)	.76	.77	.79	.81	.83	DE HALLER NUMBER		
PRESSURE RISE COEFFICIENT	(P52/P51)	.43	.46	.48	.50	.52			
STATOR DECELERATION	(V0/V3)	.87	.89	.90	.92	.93	DE HALLER NUMBER		
STAGE LOADING (DELTA H/U SQUARED)		.51	.43	.35	.29	.25			
FLOW COEFFICIENT	(VA/U)	.88	.79	.72	.67	.62			
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)		.0	.0	.0	.0	.0	DEGREES		
REL. ROTOR AIR INLET ANGLE (ALPHA1)		48.8	51.4	54.1	56.3	58.3	DEGREES		
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)		29.4	36.4	42.1	46.7	50.5	DEGREES		
ABS. STATOR AIR INLET ANGLE (ALPHA3)		30.0	27.6	25.5	23.7	22.2	DEGREES		
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)		5.9	5.4	5.0	4.6	4.3	DEGREES		
ROTOR DEFLECTION		19.4	15.2	12.0	9.6	7.8	DEGREES		
STATOR DEFLECTION		24.1	22.2	20.5	19.1	17.8	DEGREES		
REACTION		74.7	79.3	82.7	85.4	87.4	PERCENT		
SPACE CHORD RATIO (ROTOR)		.762	.784	.800	.811	.819			
SPACE CHORD RATIO (STATOR)		.579	.640	.700	.760	.819			
DIFFUSION FACTOR (ROTOR)		.389	.355	.322	.290	.262			
DIFFUSION FACTOR (STATOR)		.278	.262	.249	.237	.228			
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.635	1.576	1.518	1.468	1.424			
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.431	1.406	1.386	1.371	1.359			
SHOCK LOSS COEFFICIENT (ROTOR)		.00000	.00000	.00071	.00230	.00392			
SHOCK LOSS COEFFICIENT (STATOR)		.00000	.00000	.00000	.00000	.00000			
REYNOLDS NUMBER (ROTOR)		.20E+06	.21E+06	.21E+06	.22E+06	.22E+06			
REYNOLDS NUMBER (STATOR)		.30E+06	.30E+06	.30E+06	.31E+06	.31E+06			
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000			
AXIAL VELOCITY RATIO (STATOR)		.995	.996	.996	.997	.997			

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.02103	.02182	.02329	.02459	.02626
PROFILE LOSS COEFFICIENT (STATOR)	.02794	.02559	.02370	.02192	.02047
SECONDARY LOSS COEFFICIENT (ROTOR)			.00687		
SECONDARY LOSS COEFFICIENT (STATOR)			.00650		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00688		
ANNULUS LOSS COEFFICIENT (STATOR)			.00522		
TOTAL LOSS COEFFICIENT (ROTOR)			.03852		
TOTAL LOSS COEFFICIENT (STATOR)			.03564		

EFFICIENCY LOSS DUE TO TIP CLEARANCE	.027	PERCENT
ROTOR ISENTROPIC EFFICIENCY	92.70	PERCENT
STAGE ISENTROPIC EFFICIENCY	89.51	PERCENT
STAGE PRESSURE RATIO	1.19	

		STAGE 2		STAGE 3		STAGE 4	
		BLADE ROOT	75%	BLADE MID HEIGHT	75%	BLADE TIP	75%
BLADE SPEED	(U)	216.9	237.3	257.8	278.2	298.6	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	191.0	190.9	190.7	190.6	190.5	M/S
REL. ROTOR INLET VELOCITY	(V1)	273.8	290.1	307.0	324.3	341.9	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	203.7	217.7	234.3	252.6	271.9	M/S

ABS. STATOR INLET VELOCITY (V3)	238.0	238.8	225.1	220.4	216.7	M/S
ABS. STATOR OUTLET VELOCITY (V4)	190.0	190.0	190.0	190.0	190.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	190.0	190.0	190.0	190.0	190.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	190.0	190.0	190.0	190.0	190.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	2.065	1.993	1.918	1.841	1.762	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	2.683	2.720	2.749	2.772	2.791	ATM
REL. ROTOR INLET MACH NUMBER (MREL)	.67	.71	.76	.80	.84	
ABS. STATOR INLET MACH NUMBER (MABS)	.58	.56	.54	.53	.52	
ROTOR DECELERATION (V2-V1)	.74	.75	.76	.78	.80	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT (P/D)	.45	.44	.42	.39	.37	
STATOR DECELERATION (V3-V2)	.80	.81	.84	.86	.88	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	.57	.48	.40	.35	.30	
FLOW COEFFICIENT (VA/U)	.88	.80	.74	.68	.64	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	5.9	5.4	5.0	4.6	4.3	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	46.1	49.1	51.8	54.1	56.2	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	21.2	20.2	35.4	41.2	45.7	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	37.0	34.6	32.4	30.5	28.7	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	5.8	5.4	5.0	4.7	4.4	DEGREES
ROTOR DEFLECTION	24.9	19.9	15.9	12.9	10.6	DEGREES
STATOR DEFLECTION	31.2	29.2	27.4	25.8	24.4	DEGREES
REACTION	62.4	68.6	73.4	77.1	80.2	PERCENT
SPACE CHORD RATIO (ROTOR)	.766	.788	.800	.808	.812	
SPACE CHORD RATIO (STATOR)	.586	.643	.700	.756	.812	
DIFFUSION FACTOR (ROTOR)	.439	.403	.372	.341	.311	
DIFFUSION FACTOR (STATOR)	.354	.334	.318	.303	.291	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.702	1.456	1.403	1.348	1.499	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.581	1.346	1.518	1.486	1.478	
SHOCK LOSS COEFFICIENT (ROTOR)	.00000	.00000	.00000	.00000	.00142	
SHOCK LOSS COEFFICIENT (STATOR)	.00000	.00000	.00000	.00000	.00000	
REYNOLDS NUMBER (ROTOR)	.25E+06	.25E+06	.26E+06	.27E+06	.27E+06	
REYNOLDS NUMBER (STATOR)	.33E+06	.34E+06	.34E+06	.35E+06	.35E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	.995	.996	.996	.997	.997	

CASCAD PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01945	.01952	.02035	.02158	.02305
PROFILE LOSS COEFFICIENT (STATOR)	.02543	.02366	.02214	.02082	.01964
SECONDARY LOSS COEFFICIENT (ROTOR)			.00833		
SECONDARY LOSS COEFFICIENT (STATOR)			.01145		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00672		
ANNULUS LOSS COEFFICIENT (STATOR)			.00535		
TOTAL LOSS COEFFICIENT (ROTOR)			.03713		
TOTAL LOSS COEFFICIENT (STATOR)			.03914		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.026

ROTOR ISENTROPIC EFFICIENCY

93.99 PERCENT

STAGE ISENTROPIC EFFICIENCY

90.34 PERCENT

STAGE PRESSURE RATIO

1.20

STAGE 3

	BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	216.8	235.1	253.3	271.6	289.8	M/S
ABS. STATOR OUTLET VELOCITY (V0)	191.0	190.8	190.7	190.6	190.6	M/S
REL. ROTOR INLET VELOCITY (V1)	274.0	288.5	303.5	318.9	334.5	M/S
REL. ROTOR OUTLET VELOCITY (V2)	200.9	212.1	226.1	241.8	258.7	M/S
ABS. STATOR INLET VELOCITY (V3)	243.8	236.5	230.6	225.8	221.7	M/S
ABS. STATOR OUTLET VELOCITY (V4)	190.0	190.0	190.0	190.0	190.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	190.0	190.0	190.0	190.0	190.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	190.0	190.0	190.0	190.0	190.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	2.518	2.444	2.366	2.286	2.205	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	3.317	3.260	3.295	3.323	3.346	ATM
REL. ROTOR INLET MACH NUMBER (MREL)	.66	.69	.73	.76	.80	
ABS. STATOR INLET MACH NUMBER (MABS)	.57	.56	.54	.53	.52	
ROTOR DECELERATION (V2-V1)	.73	.73	.74	.76	.77	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT (P/D)	.46	.46	.45	.42	.40	
STATOR DECELERATION (V3-V2)	.78	.80	.82	.84	.86	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	.61	.52	.45	.38	.34	
FLOW COEFFICIENT (VA/U)	.88	.81	.75	.70	.66	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	5.8	5.4	5.0	4.7	4.4	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	46.1	48.8	51.2	53.4	55.4	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	18.6	26.4	32.8	38.2	42.7	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	38.8	36.6	34.5	32.7	31.0	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	5.7	5.3	5.0	4.7	4.4	DEGREES
ROTOR DEFLECTION	27.5	22.5	18.4	15.2	12.7	DEGREES
STATOR DEFLECTION	33.1	31.2	29.5	28.0	26.6	DEGREES
REACTION	60.3	66.2	70.9	74.7	77.8	PERCENT
SPACE CHORD RATIO (ROTOR)	.770	.788	.800	.807	.812	
SPACE CHORD RATIO (STATOR)	.595	.646	.700	.752	.803	
DIFFUSION FACTOR (ROTOR)	.456	.433	.405	.374	.346	
DIFFUSION FACTOR (STATOR)	.383	.365	.349	.336	.324	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	3.747	1.708	1.658	1.606	1.557	
EQUIVALENT DIFFUSION FACTOR (STATOR)	3.628	1.593	1.545	1.542	1.523	

SHOCK LOSS COEFFICIENT (ROTOR)	.00000	.00000	.00000	.00000	.00000
SHOCK LOSS COEFFICIENT (STATOR)	.00000	.00000	.00000	.00000	.00000
REYNOLDS NUMBER (ROTOR)	.29E+06	.29E+06	.30E+06	.31E+06	.32E+06
REYNOLDS NUMBER (STATOR)	.38E+06	.38E+06	.39E+06	.39E+06	.38E+06
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000
AXIAL VELOCITY RATIO (STATOR)	.995	.996	.996	.997	.997

 CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01877	.01871	.01910	.02002	.02120
PROFILE LOSS COEFFICIENT (STATOR)	.02419	.02269	.02136	.02024	.01922
SECONDARY LOSS COEFFICIENT (ROTOR)			.01121		
SECONDARY LOSS COEFFICIENT (STATOR)			.01324		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00681		
ANNULUS LOSS COEFFICIENT (STATOR)			.00558		
TOTAL LOSS COEFFICIENT (ROTOR)			.03757		
TOTAL LOSS COEFFICIENT (STATOR)			.04036		

EFFICIENCY LOSS DUE TO TIP CLEARANCE	.026				
ROTOR ISENTROPIC EFFICIENCY	94.23	PERCENT			
STAGE ISENTROPIC EFFICIENCY	90.44	PERCENT			
STAGE PRESSURE RATIO	1.20				

		STAGE 4					
		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	216.8	233.1	249.4	265.7	282.0	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	191.0	190.8	190.7	190.6	190.6	M/S
REL. ROTOR INLET VELOCITY	(V1)	274.2	287.1	300.5	314.1	327.9	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	197.9	207.3	218.1	232.5	247.1	M/S
ABS. STATOR INLET VELOCITY	(V3)	249.3	242.1	236.2	231.2	226.9	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	190.0	190.0	190.0	190.0	190.0	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	190.0	190.0	190.0	190.0	190.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	190.0	190.0	190.0	190.0	190.0	M/S
STATIC PRESSURE AT ROTOR INLET	(P01)	3.060	2.983	2.904	2.822	2.738	ATM
STATIC PRESSURE AT ROTOR OUTLET	(P02)	3.851	3.900	3.941	3.974	4.002	ATM
REL. ROTOR INLET MACH NUMBER	(MREL)	.64	.61	.70	.73	.76	
ABS. STATOR INLET MACH NUMBER	(MABS)	.57	.55	.54	.53	.52	
ROTOR DECELERATION	(V2/V1)	.72	.72	.73	.74	.75	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT	(DE/D)	.48	.48	.47	.45	.43	
STATOR DECELERATION	(V0/V3)	.76	.78	.80	.82	.84	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)		.66	.57	.50	.44	.39	
FLOW COEFFICIENT	(VA/U)	.88	.82	.76	.72	.67	
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	5.7	5.3	5.0	4.7	4.4	DEGREES
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	46.1	48.6	50.8	52.8	54.6	DEGREES
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	16.3	23.6	29.9	35.2	39.7	DEGREES
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	40.3	38.3	36.4	34.7	33.1	DEGREES
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	5.7	5.3	5.0	4.7	4.5	DEGREES
ROTOR DEFLECTION		29.9	25.0	20.9	17.6	14.9	DEGREES
STATOR DEFLECTION		34.7	33.0	31.4	30.0	28.7	DEGREES
REACTION		58.4	64.0	68.5	72.3	75.4	PERCENT
SPACE CHORD RATIO (ROTOR)		.772	.789	.800	.807	.812	
SPACE CHORD RATIO (STATOR)		.603	.652	.700	.748	.795	
DIFFUSION FACTOR (ROTOR)		.478	.460	.435	.408	.382	
DIFFUSION FACTOR (STATOR)		.610	.593	.579	.566	.554	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.786	1.754	1.720	1.662	1.614	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.672	1.638	1.610	1.586	1.566	
SHOCK LOSS COEFFICIENT (ROTOR)		.00000	.00000	.00000	.00000	.00000	
SHOCK LOSS COEFFICIENT (STATOR)		.00000	.00000	.00000	.00000	.00000	
REYNOLDS NUMBER (ROTOR)		.32E+06	.32E+06	.34E+06	.34E+06	.35E+06	
REYNOLDS NUMBER (STATOR)		.41E+06	.42E+06	.42E+06	.43E+06	.43E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		.995	.996	.996	.997	.997	

 CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01831	.01810	.01834	.01884	.01976
PROFILE LOSS COEFFICIENT (STATOR)	.02372	.02193	.02079	.01978	.01888
SECONDARY LOSS COEFFICIENT (ROTOR)			.01312		
SECONDARY LOSS COEFFICIENT (STATOR)			.01495		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00671		
ANNULUS LOSS COEFFICIENT (STATOR)			.00565		
TOTAL LOSS COEFFICIENT (ROTOR)			.03850		
TOTAL LOSS COEFFICIENT (STATOR)			.04152		

EFFICIENCY LOSS DUE TO TIP CLEARANCE	.027				
ROTOR ISENTROPIC EFFICIENCY	94.36	PERCENT			
STAGE ISENTROPIC EFFICIENCY	90.46	PERCENT			

STAGE PRESSURE RATIO

1.18

		STAGE 5				
		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP
BLADE SPEED	(U)	216.7	231.3	245.8	260.4	275.0
ABS. STATOR OUTLET VELOCITY	(V0)	190.9	190.8	190.7	190.6	190.4
REL. ROTOR INLET VELOCITY	(V1)	274.2	285.9	297.7	309.8	322.1
REL. ROTOR OUTLET VELOCITY	(V2)	197.7	206.0	216.2	227.8	240.5
ABS. STATOR INLET VELOCITY	(V3)	248.6	243.1	237.6	232.9	228.8
ABS. STATOR OUTLET VELOCITY	(V4)	190.0	190.0	190.0	190.0	190.0
AXIAL VELOCITY AT ROTOR INLET	(VA1)	190.0	190.0	190.0	190.0	190.0
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	190.0	190.0	190.0	190.0	190.0
STATIC PRESSURE AT ROTOR INLET	(PS1)	3.706	3.628	3.547	3.463	3.377
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	4.620	4.661	4.705	4.740	4.770
REL. ROTOR INLET MACH NUMBER (M1REL)		.62	.65	.67	.70	.73
ABS. STATOR INLET MACH NUMBER (M3ABS)		.54	.54	.53	.52	.51
ROTOR DECELERATION	(V2/V1)	.72	.72	.73	.74	.75
PRESSURE RISE COEFFICIENT	(DP/D)	.48	.48	.47	.46	.44
STATOR DECELERATION	(V0/V3)	.76	.76	.80	.82	.83
STAGE LOADING (DELTA H/U SQUARED)		.66	.58	.51	.46	.41
FLOW COEFFICIENT	(VA/U)	.84	.82	.77	.73	.69
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)		5.7	5.3	5.0	4.7	4.5
REL. ROTOR AIR INLET ANGLE (ALPHA1)		48.2	48.3	50.3	52.2	53.9
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)		16.1	22.7	28.5	33.5	37.8
ABS. STATOR AIR INLET ANGLE (ALPHA3)		40.4	38.6	36.9	35.3	33.9
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)		5.6	5.3	5.0	4.7	4.5
ROTOR DEFLECTION		30.1	35.6	21.8	18.7	16.0
STATOR DEFLECTION		34.8	33.3	31.9	30.6	29.4
REACTION		58.3	63.4	67.6	71.1	74.1
SPACE CHORD RATIO (ROTOR)		.774	.789	.800	.807	.812
SPACE CHORD RATIO (STATOR)		.612	.656	.700	.744	.787
DIFFUSION FACTOR (ROTOR)		.461	.464	.443	.420	.395
DIFFUSION FACTOR (STATOR)		.414	.399	.386	.374	.363
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.795	1.742	1.724	1.681	1.637
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.676	1.647	1.621	1.598	1.580
SHOCK LOSS COEFFICIENT (ROTOR)		.00000	.00000	.00000	.00000	.00000
SHOCK LOSS COEFFICIENT (STATOR)		.00000	.00000	.00000	.00000	.00000
REYNOLDS NUMBER (ROTOR)		.22E+06	.36E+06	.36E+06	.37E+06	.38E+06
REYNOLDS NUMBER (STATOR)		.44E+06	.45E+06	.45E+06	.45E+06	.46E+06
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000
AXIAL VELOCITY RATIO (STATOR)		.995	.996	.998	.997	.997

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01808	.01786	.01803	.01835	.01909
PROFILE LOSS COEFFICIENT (STATOR)	.02269	.02156	.02034	.01962	.01881
SECONDARY LOSS COEFFICIENT (ROTOR)			.01372		
SECONDARY LOSS COEFFICIENT (STATOR)			.01536		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00659		
ANNULUS LOSS COEFFICIENT (STATOR)			.00562		
TOTAL LOSS COEFFICIENT (ROTOR)			.03858		
TOTAL LOSS COEFFICIENT (STATOR)			.04165		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.027

ROTOR ISENTROPIC EFFICIENCY

84.45 PERCENT

STATOR ISENTROPIC EFFICIENCY

90.68 PERCENT

STAGE PRESSURE RATIO

1.18

		STAGE 6				
		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP
BLADE SPEED	(U)	216.6	229.7	242.8	255.9	269.0
ABS. STATOR OUTLET VELOCITY	(V0)	190.9	190.8	190.7	190.7	190.6
REL. ROTOR INLET VELOCITY	(V1)	274.4	284.8	295.4	306.2	317.2
REL. ROTOR OUTLET VELOCITY	(V2)	188.2	205.6	214.4	224.8	236.0
ABS. STATOR INLET VELOCITY	(V3)	248.6	242.8	237.8	232.5	227.7
ABS. STATOR OUTLET VELOCITY	(V4)	190.0	190.0	190.0	190.0	190.0
AXIAL VELOCITY AT ROTOR INLET	(VA1)	190.0	190.0	190.0	190.0	190.0
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	190.0	190.0	190.0	190.0	190.0
STATIC PRESSURE AT ROTOR INLET	(PS1)	4.446	4.368	4.283	4.197	4.109
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	5.448	5.498	5.542	5.579	5.611
REL. ROTOR INLET MACH NUMBER (M1REL)		.61	.63	.65	.68	.70
ABS. STATOR INLET MACH NUMBER (M3ABS)		.54	.53	.52	.51	.50
ROTOR DECELERATION	(V2/V1)	.72	.72	.73	.73	.74
PRESSURE RISE COEFFICIENT	(DP/D)	.48	.48	.47	.46	.45
STATOR DECELERATION	(V0/V3)	.76	.78	.80	.81	.83
STAGE LOADING (DELTA H/U SQUARED)		.60	.56	.52	.47	.42
FLOW COEFFICIENT	(VA/U)	.86	.83	.78	.74	.71
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)		5.6	5.3	5.0	4.7	4.5

REL. ROTOR AIR INLET ANGLE (ALPHA1)	46.2	46.2	50.0	51.4	53.2	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	16.5	22.5	27.7	32.3	36.4	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	40.2	38.2	37.0	35.5	34.2	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	5.5	5.2	5.0	4.8	4.6	DEGREES
ROTOR DEFLECTION	29.7	25.7	22.3	19.3	16.8	DEGREES
STATOR DEFLECTION	34.6	33.2	32.0	30.8	29.6	DEGREES
REACTION	58.7	63.2	67.1	70.4	73.2	PERCENT
SPACE CHORD RATIO (ROTOR)	.776	.790	.800	.807	.813	
SPACE CHORD RATIO (STATOR)	.620	.640	.700	.740	.779	
DIFFUSION FACTOR (ROTOR)	.478	.453	.445	.424	.402	
DIFFUSION FACTOR (STATOR)	.412	.399	.387	.376	.366	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.785	1.760	1.726	1.688	1.649	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.673	1.646	1.622	1.602	1.585	
SHOCK LOSS COEFFICIENT (ROTOR)	.00000	.00000	.00000	.00000	.00000	
SHOCK LOSS COEFFICIENT (STATOR)	.00000	.00000	.00000	.00000	.00000	
REYNOLDS NUMBER (ROTOR)	.39E+06	.41E+06	.40E+06	.41E+06	.42E+06	
REYNOLDS NUMBER (STATOR)	.48E+06	.49E+06	.50E+06	.50E+06	.51E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	.995	.996	.996	.997	.997	

 CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01778	.01760	.01772	.01795	.01855
PROFILE LOSS COEFFICIENT (STATOR)	.02215	.02115	.02024	.01942	.01867
SECONDARY LOSS COEFFICIENT (ROTOR)			.01389		
SECONDARY LOSS COEFFICIENT (STATOR)			.01544		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00676		
ANNULUS LOSS COEFFICIENT (STATOR)			.00582		
TOTAL LOSS COEFFICIENT (ROTOR)			.03857		
TOTAL LOSS COEFFICIENT (STATOR)			.04158		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.027

ROTOR ISENTROPIC EFFICIENCY

94.33 PERCENT

STAGE ISENTROPIC EFFICIENCY

90.21 PERCENT

STAGE PRESSURE RATIO

1.17

		STAGE 7				
		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP
BLADE SPEED	(U)	216.5	228.4	240.3	252.1	264.0
ABS. STATOR OUTLET VELOCITY	(V0)	190.8	190.8	190.7	190.7	190.6
REL. ROTOR INLET VELOCITY	(V1)	274.5	283.5	293.5	303.2	313.1
REL. ROTOR OUTLET VELOCITY	(V2)	296.0	294.6	212.5	221.5	231.3
ABS. STATOR INLET VELOCITY	(V3)	249.0	243.7	239.0	235.0	231.4
ABS. STATOR OUTLET VELOCITY	(V4)	190.0	190.0	190.0	190.0	190.0
AXIAL VELOCITY AT ROTOR INLET	(VA1)	190.0	190.0	190.0	190.0	190.0
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	190.0	190.0	190.0	190.0	190.0
STATIC PRESSURE AT ROTOR INLET (PS1)		5.252	5.370	5.085	4.998	4.909
STATIC PRESSURE AT ROTOR OUTLET (PS2)		6.373	6.423	6.470	6.509	6.543
REL. ROTOR INLET MACH NUMBER (MREL)		.59	.61	.63	.65	.67
ABS. STATOR INLET MACH NUMBER (MSAS)		.53	.52	.51	.50	.49
ROTOR DECELERATION (V2/V1)		.72	.72	.72	.73	.74
PRESSURE RISE COEFFICIENT (DP/D)		.48	.46	.48	.47	.45
STATOR DECELERATION (V0/V3)		.76	.78	.79	.81	.82
STAGE LOADING (DELTA H/U SQUARED)		.66	.59	.53	.49	.44
FLOW COEFFICIENT (VA/U)		.88	.85	.79	.75	.72
ABS. STATOR AIR OUTLET ANGLE (ALPHA5)		5.5	5.2	5.0	4.8	4.6
REL. ROTOR AIR INLET ANGLE (ALPHA1)		46.2	48.0	49.6	51.2	52.6
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)		16.3	21.6	26.6	30.9	34.8
ABS. STATOR AIR INLET ANGLE (ALPHA3)		40.2	38.8	37.4	36.0	34.8
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)		.0	.0	.0	.0	.0
ROTOR DEFLECTION		29.7	26.2	22.0	20.3	17.9
STATOR DEFLECTION		40.2	38.8	37.4	36.0	34.8
REACTION		58.6	62.8	66.4	69.4	72.1
SPACE CHORD RATIO (ROTOR)		.778	.790	.800	.807	.812
SPACE CHORD RATIO (STATOR)		.628	.644	.700	.736	.771
DIFFUSION FACTOR (ROTOR)		.481	.467	.451	.432	.413
DIFFUSION FACTOR (STATOR)		.417	.404	.393	.383	.374
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.788	1.767	1.737	1.703	1.668
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.715	1.692	1.668	1.649	1.632
SHOCK LOSS COEFFICIENT (ROTOR)		.00000	.00000	.00000	.00000	.00000
SHOCK LOSS COEFFICIENT (STATOR)		.00000	.00000	.00000	.00000	.00000
REYNOLDS NUMBER (ROTOR)		.42E+06	.42E+06	.42E+06	.44E+06	.46E+06
REYNOLDS NUMBER (STATOR)		.52E+06	.52E+06	.53E+06	.53E+06	.53E+06
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000
AXIAL VELOCITY RATIO (STATOR)		1.000	1.000	1.000	1.000	1.000

 CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01759	.01741	.01748	.01775	.01809
PROFILE LOSS COEFFICIENT (STATOR)	.02208	.02109	.02030	.01958	.01891
SECONDARY LOSS COEFFICIENT (ROTOR)			.01440		
SECONDARY LOSS COEFFICIENT (STATOR)			.01864		
ANNULUS LOSS COEFFICIENT (ROTOR)			.00666		

ANNULUS LOSS COEFFICIENT (STATOR)	.00553	
TOTAL LOSS COEFFICIENT (ROTOR)	.03872	
TOTAL LOSS COEFFICIENT (STATOR)	.04579	

EFFICIENCY LOSS DUE TO TIP CLEARANCE	.027	
ROTOR ISENTROPIC EFFICIENCY	94.37	PERCENT
STAGE ISENTROPIC EFFICIENCY	89.81	PERCENT
STAGE PRESSURE RATIO	1.16	

.....

OVERALL PRESSURE RATIO	3.30	
COMPRESSOR ISENTROPIC EFFICIENCY	87.13	PERCENT
COMPRESSOR POLYTROPIC EFFICIENCY	89.01	PERCENT

Figure F.6 COMPDES Output Data for HP Compressor.

PROPERTY DISTRIBUTION THROUGH A 7 STAGE AXIAL COMPRESSOR WITHOUT IGVs										
CONSTANT OUTER RADIUS TYPE										
THE FOLLOWING CONSTANT DATA HAS BEEN USED FOR THIS RUN:										
MASS FLOW (kg/s)	173.86									
OVERALL PRESSURE RATIO	3.96									
INLET AXIAL VELOCITY (m/s)	220.00									
ISENTROPIC EFFICIENCY (%)	88.00									
OVERALL TEMPERATURE RISE (K)	296.00									
INLET TOTAL TEMPERATURE (K)	580.32									
INLET TOTAL PRESSURE (atm)	1.36									
ROTATIONAL SPEED (rpm)	10000.00									
INLET IGV'S	STA 1	STA 2	STA 3	STA 4	STA 5	STA 6	STA 7	STA 8	STA 9	
TOTAL TEMPERATURE RISE (K)	38.00	40.00	43.00	44.00	44.00	44.00	43.00	.00	.00	
OUTLET TOTAL TEMPERATURE (K)	580.32	580.32	618.32	658.32	701.32	745.32	789.32	833.32	876.32	.00
AVERAGE TOTAL TEMPERATURE (K)	599.32	638.32	679.82	723.32	767.32	811.32	854.82	.00	.00	
OUTLET STATIC TEMPERATURE (K)	557.18	557.18	594.80	634.87	677.79	721.84	765.87	810.26	854.59	.00
OUTLET TOTAL PRESSURE (atm)	1.361	1.361	1.976	11.184	13.836	17.011	20.707	24.879	29.744	.000
OUTLET STATIC PRESSURE (atm)	6.347	6.347	7.871	9.777	12.176	15.071	18.454	22.420	27.003	.000
OUTLET DENSITY (kg/cubic m)	4.0214	4.0214	4.6718	5.4369	6.3422	7.3713	8.5070	9.7690	11.1354	.0000
OUTLET SPECIFIC HEAT (kJ/kg K)	1.0457	1.0457	1.0546	1.0643	1.0749	1.0856	1.0960	1.1055	1.1137	.0000
AVERAGE SPECIFIC HEAT (kJ/kg K)	1.0501	1.0594	1.0695	1.0803	1.0908	1.1009	1.1098	.0000	.0000	
OUTLET ANNULUS AREA (square m)	.1965	.1965	.1482	.1454	.1248	.1072	.0928	.0809	.0708	.0000
OUTLET INNER RADIUS (cm)	27.52	27.52	29.06	30.33	31.40	32.27	32.97	33.54	34.02	.00
OUTLET OUTER RADIUS (cm)	37.18	37.18	37.18	37.18	37.18	37.18	37.18	37.18	37.18	.00
BLADE HEIGHT (cm)	9.67	9.67	8.13	6.83	5.78	4.92	4.22	3.64	3.17	.00
OUTLET HUB/TIP RATIO	.740	.740	.781	.816	.844	.868	.887	.902	.915	.000
ABSOLUTE OUTLET VELOCITY (m/s)	220.00	220.00	222.74	223.39	224.51	225.79	226.74	225.79	220.00	.00
OUTLET SWIRL ANGLE (DEGREES)	.00	.00	9.00	10.00	12.00	13.00	13.00	.00	.00	.00
AXIAL VELOCITY (m/s)	220.00	220.00	220.00	220.00	220.00	220.00	220.00	220.00	220.00	.00
PRESSURE RATIO	1.2336	1.2332	1.2371	1.2295	1.2179	1.2043	1.1914	.0000	.0000	
ENTHALPY RISE (kJ/kg)	39.904	42.375	49.990	47.532	47.998	48.438	47.720	.000	.000	
COMPRESSOR POWER (kW)	55627.860									
OVERALL PRESSURE RATIO	4.0438									

7 STAGE HP COMPRESSOR

AXIAL COMPRESSOR/FAN "FREE VORTEX" DESIGN

CONSTANT OUTER RADIUS TYPE

ALL ANNULUS DIMENSIONS IN CENTIMETRES

	STAGE 1	STAGE 2	STAGE 3	STAGE 4	STAGE 5	STAGE 6	STAGE 7	STAGE 8	STAGE 9
OUTSIDE RADIUS	37.18	37.18	37.18	37.18	37.18	37.18	37.18	37.18	.00
HUB RADIUS	27.52	29.06	30.33	31.40	32.27	32.97	33.54	34.02	.00
INLET HEIGHT	9.668	8.129	6.853	5.783	4.913	4.215	3.641	3.167	.00

INLET HUB/TIP RATIO	.7400	.7814	.8157	.8445	.8679	.8866	.9021	.9148	.0000
ASPECT RATIO	2.0000	2.0000	2.0000	2.0000	1.9000	1.9000	1.8000	.0000	.0000
MID CHORD HEIGHT-ROTOR	9.428	7.918	6.674	5.636	4.795	4.118	3.563	.000	.000
MID CHORD HEIGHT-STATOR	8.630	7.273	6.133	5.197	4.442	3.828	3.324	.000	.000
TRUE CHORD-ROTOR	4.714	3.959	3.337	2.816	2.524	2.167	1.980	.000	.000
TRUE CHORD-STATOR	4.325	3.637	3.067	2.590	2.338	2.023	1.847	.000	.000
ROTOR AXIAL CHORD	3.141	2.834	2.408	2.058	1.847	1.587	1.421	.000	.000
ROTOR AXIAL SPACE	1.424	1.188	1.001	.845	.757	.650	.594	.000	.000
STATOR AXIAL CHORD	4.241	3.474	2.880	2.407	2.147	1.850	1.770	.000	.000
STATOR AXIAL SPACE	1.297	1.091	.920	.779	.701	.604	.554	.000	.000
TOTAL STAGE LENGTH	10.084	8.586	7.208	6.090	5.453	4.692	4.332	.000	.000
NUMBER OF ROTOR BLADES	54.1	65.8	79.7	95.8	108.3	127.3	140.5	.0	.0
NUMBER OF STATOR BLADES	43.6	77.3	83.7	111.5	129.3	146.7	161.1	.0	.0

TOTAL COMPRESSOR LENGTH 44.443

		STAGE 1					
		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	289.4	314.1	338.8	363.5	388.2	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	220.0	220.0	220.0	220.0	220.0	M/S
REL. ROTOR INLET VELOCITY	(V1)	263.5	283.5	303.9	324.9	346.2	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	266.3	287.8	311.0	334.9	359.9	M/S
ABS. STATOR INLET VELOCITY	(V3)	260.4	254.7	250.1	246.4	243.3	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	220.0	220.0	220.0	220.0	220.0	M/S
STATIC PRESSURE AT ROTOR INLET	(P01)	4.977	4.774	4.566	4.356	4.144	ATM
STATIC PRESSURE AT ROTOR OUTLET	(P02)	7.461	7.527	7.578	7.620	7.654	ATM
REL. ROTOR INLET MACH NUMBER	(MREL)	.77	.82	.86	.90	.95	
ABS. STATOR INLET MACH NUMBER	(M000)	.54	.53	.52	.51	.50	
ROTOR DECELERATION	(V2/V1)	.73	.75	.77	.79	.81	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT	(DP/D)	.46	.44	.41	.38	.35	
ROTOR DECELERATION	(V0/V3)	.84	.86	.88	.89	.90	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)		.48	.41	.35	.31	.27	
FLOW COEFFICIENT (VA/D)		.75	.70	.65	.61	.57	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)		.0	.0	.0	.0	.0	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)		52.8	55.0	57.0	58.8	60.5	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)		34.3	40.2	45.0	48.9	52.3	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)		32.3	30.3	28.4	26.6	25.3	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)		10.2	9.6	9.0	8.5	8.1	DEGREES
ROTOR DEFLECTION		18.4	14.8	12.0	9.9	8.2	DEGREES
STATOR DEFLECTION		22.1	20.7	19.4	18.2	17.2	DEGREES
REACTION		75.9	79.6	82.4	84.7	86.6	PERCENT
SPACE CHORD RATIO (ROTOR)		.782	.793	.800	.805	.808	
SPACE CHORD RATIO (STATOR)		.642	.696	.750	.803	.857	
DIFFUSION FACTOR (ROTOR)		.427	.382	.348	.317	.288	
DIFFUSION FACTOR (STATOR)		.327	.312	.299	.288	.279	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.480	1.615	1.554	1.505	1.460	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.452	1.429	1.410	1.394	1.381	
SHOCK LOSS COEFFICIENT (ROTOR)		.00000	.00000	.00202	.00352	.00504	
SHOCK LOSS COEFFICIENT (STATOR)		.00000	.00000	.00000	.00000	.00000	
REYNOLDS NUMBER (ROTOR)		.45E+06	.44E+06	.47E+06	.48E+06	.49E+06	
REYNOLDS NUMBER (STATOR)		.67E+06	.68E+06	.69E+06	.69E+06	.69E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		.984	.984	.988	.988	.990	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01803	.01906	.02063	.02181	.02339
PROFILE LOSS COEFFICIENT (STATOR)	.02180	.02032	.01915	.01794	.01695
SECONDARY LOSS COEFFICIENT (ROTOR)			.00808		
SECONDARY LOSS COEFFICIENT (STATOR)			.00647		
ANNULUS LOSS COEFFICIENT (ROTOR)			.01252		
ANNULUS LOSS COEFFICIENT (STATOR)			.00920		
TOTAL LOSS COEFFICIENT (ROTOR)			.04330		
TOTAL LOSS COEFFICIENT (STATOR)			.03490		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.001

ROTOR ISENTROPIC EFFICIENCY

92.45 PERCENT

STAGE ISENTROPIC EFFICIENCY

90.09 PERCENT

STAGE PRESSURE RATIO

1.23

		STAGE 2					
		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	305.4	326.1	346.8	367.6	388.3	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	223.8	223.1	222.7	222.4	222.2	M/S
REL. ROTOR INLET VELOCITY	(V1)	345.0	363.2	381.8	400.5	419.5	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	250.5	267.6	286.4	306.4	327.3	M/S
ABS. STATOR INLET VELOCITY	(V3)	287.9	280.4	274.1	268.7	264.0	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	220.0	220.0	220.0	220.0	220.0	M/S

STATIC PRESSURE AT ROTOR INLET (PSI)	8.504	8.244	8.042	5.835	5.605	ATM
STATIC PRESSURE AT ROTOR OUTLET (PSI)	8.039	8.044	8.136	8.211	9.274	ATM
REL. ROTOR INLET MACH NUMBER (MREL)	.71	.75	.78	.83	.87	
ABS. STATOR INLET MACH NUMBER (MABS)	.56	.57	.51	.54	.53	
ROTOR DECELERATION (V2/V1)	.73	.74	.75	.77	.78	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT (DP/D)	.47	.48	.44	.41	.35	
STATOR DECELERATION (V0/V2)	.76	.78	.60	.82	.83	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	.46	.42	.37	.33	.30	
FLOW COEFFICIENT (VA/U)	.72	.67	.65	.60	.57	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	10.2	9.6	9.0	8.5	8.1	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	50.4	52.1	54.8	56.7	58.4	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	28.6	34.7	39.8	44.1	47.8	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	40.2	38.3	36.8	35.0	33.6	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	11.1	10.5	10.0	9.5	9.1	DEGREES
ROTOR DEFLECTION	21.8	18.0	15.0	12.6	10.6	DEGREES
STATOR DEFLECTION	29.1	27.8	26.8	25.5	24.5	DEGREES
REACTION	63.1	67.7	71.4	74.3	77.2	PERCENT
SPACE CHORD RATIO (ROTOR)	.794	.800	.811	.795	.794	
SPACE CHORD RATIO (STATOR)	.656	.703	.751	.797	.843	
DIFFUSION FACTOR (ROTOR)	.442	.414	.382	.354	.325	
DIFFUSION FACTOR (STATOR)	.402	.387	.373	.361	.350	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.723	1.472	1.412	1.570	1.525	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.633	1.604	1.578	1.556	1.540	
SHOCK LOSS COEFFICIENT (ROTOR)	.00000	.00000	.00100	.00090	.00221	
SHOCK LOSS COEFFICIENT (STATOR)	.00000	.00000	.00100	.00000	.00000	
REYNOLDS NUMBER (ROTOR)	.43E+04	.50E+04	.51E+04	.53E+04	.54E+04	
REYNOLDS NUMBER (STATOR)	.64E+04	.67E+04	.67E+04	.68E+04	.68E+04	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.001	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	.961	.963	.962	.966	.967	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01703	.01754	.01861	.01987	.02128
PROFILE LOSS COEFFICIENT (STATOR)	.01978	.01877	.01788	.01708	.01638
SECONDARY LOSS COEFFICIENT (ROTOR)			.01804		
SECONDARY LOSS COEFFICIENT (STATOR)			.02230		
ANNULUS LOSS COEFFICIENT (ROTOR)			.01134		
ANNULUS LOSS COEFFICIENT (STATOR)			.01859		
TOTAL LOSS COEFFICIENT (ROTOR)			.04068		
TOTAL LOSS COEFFICIENT (STATOR)			.03887		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.001

ROTOR ISENTROPIC EFFICIENCY

92.95 PERCENT

STAGE ISENTROPIC EFFICIENCY

90.75 PERCENT

STAGE PRESSURE RATIO

1.23

	STAGE 3					
	BLADE ROOT	25%	BLADE HGT HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	318.4	336.0	353.5	371.0	388.5	M/S
ABS. STATOR OUTLET VELOCITY (V0)	224.2	222.8	223.4	223.1	222.8	M/S
REL. ROTOR INLET VELOCITY (V1)	352.4	368.2	384.0	400.0	416.1	M/S
REL. ROTOR OUTLET VELOCITY (V2)	249.9	264.3	280.1	296.8	314.3	M/S
ABS. STATOR INLET VELOCITY (V3)	297.3	290.4	284.4	279.1	274.4	M/S
ABS. STATOR OUTLET VELOCITY (V4)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	220.0	220.0	220.0	220.0	220.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	8.085	7.860	7.633	7.401	7.168	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	11.084	11.203	11.305	11.394	11.472	ATM
REL. ROTOR INLET MACH NUMBER (MREL)	.71	.74	.77	.80	.83	
ABS. STATOR INLET MACH NUMBER (MABS)	.56	.57	.54	.55	.54	
ROTOR DECELERATION (V2/V1)	.71	.72	.73	.74	.76	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT (DP/D)	.50	.48	.47	.45	.43	
STATOR DECELERATION (V0/V2)	.74	.76	.77	.79	.80	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	.49	.44	.41	.36	.33	
FLOW COEFFICIENT (VA/U)	.49	.45	.42	.39	.37	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	11.1	10.5	10.0	9.5	9.1	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	51.4	53.3	55.0	56.6	58.1	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	28.3	33.7	38.2	42.2	45.6	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	42.3	40.8	39.3	38.0	36.7	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	13.0	12.5	12.0	11.5	11.1	DEGREES
ROTOR DEFLECTION	23.1	19.7	16.6	14.5	12.5	DEGREES
STATOR DEFLECTION	29.2	28.3	27.3	26.4	25.6	DEGREES
REACTION	61.9	65.7	69.0	71.9	74.3	PERCENT
SPACE CHORD RATIO (ROTOR)	.801	.801	.800	.797	.794	
SPACE CHORD RATIO (STATOR)	.469	.710	.751	.790	.829	
DIFFUSION FACTOR (ROTOR)	.469	.444	.418	.392	.367	
DIFFUSION FACTOR (STATOR)	.437	.424	.412	.402	.393	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.771	1.725	1.677	1.631	1.588	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.674	1.646	1.624	1.607	1.589	
SHOCK LOSS COEFFICIENT (ROTOR)	.00000	.00000	.00000	.00000	.00110	
SHOCK LOSS COEFFICIENT (STATOR)	.00000	.00000	.00000	.00000	.00000	
REYNOLDS NUMBER (ROTOR)	.51E+04	.52E+04	.52E+04	.54E+04	.55E+04	
REYNOLDS NUMBER (STATOR)	.68E+04	.68E+04	.69E+04	.69E+04	.70E+04	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	

AXIAL VELOCITY RATIO (STATOR)

.974 .975 .976 .980 .981

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01469	.01723	.01786	.01889	.02004
PROFILE LOSS COEFFICIENT (STATOR)	.01906	.01821	.01747	.01681	.01621
SECONDARY LOSS COEFFICIENT (ROTOR)			.01147		
SECONDARY LOSS COEFFICIENT (STATOR)			.01338		
ANNULUS LOSS COEFFICIENT (ROTOR)			.01093		
ANNULUS LOSS COEFFICIENT (STATOR)			.00853		
TOTAL LOSS COEFFICIENT (ROTOR)			.04096		
TOTAL LOSS COEFFICIENT (STATOR)			.03945		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.001

ROTOR ISENTROPIC EFFICIENCY

84.26 PERCENT

STAGE ISENTROPIC EFFICIENCY

91.04 PERCENT

STAGE PRESSURE RATIO

1.24

STAGE 4

	BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	329.6	344.4	359.1	373.9	388.6	M/S
ABS. STATOR OUTLET VELOCITY (V0)	225.8	225.3	224.9	224.5	224.2	M/S
REL. ROTOR INLET VELOCITY (V1)	355.0	368.5	362.1	395.7	408.5	M/S
REL. ROTOR OUTLET VELOCITY (V2)	249.8	262.0	275.2	289.2	303.8	M/S
ABS. STATOR INLET VELOCITY (V3)	305.0	298.8	293.2	288.2	283.7	M/S
ABS. STATOR OUTLET VELOCITY (V4)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	220.0	220.0	220.0	220.0	220.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	10.185	9.855	9.720	9.482	9.242	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	13.694	13.827	13.934	14.035	14.124	ATM
REL. ROTOR INLET MACH NUMBER (MREL)	.69	.72	.74	.77	.79	
ABS. STATOR INLET MACH NUMBER (M3ABS)	.58	.57	.56	.55	.54	
ROTOR DECELERATION (V2/V1)	.70	.71	.72	.73	.74	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT (DP/D)	.50	.49	.48	.47	.45	
STATOR DECELERATION (V0/V3)	.72	.74	.75	.76	.78	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	.49	.45	.43	.38	.35	
FLOW COEFFICIENT (VA/U)	.67	.64	.61	.59	.57	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	13.0	12.5	12.0	11.5	11.1	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	31.7	33.3	34.8	36.2	37.5	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	29.3	32.9	36.9	40.5	43.6	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	43.8	42.6	41.4	40.2	39.2	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	13.9	13.4	13.0	12.6	12.2	DEGREES
ROTOR DEFLECTION	23.4	20.3	17.8	15.8	13.9	DEGREES
STATOR DEFLECTION	29.9	29.1	28.4	27.7	27.0	DEGREES
REACTION	60.2	63.6	66.5	69.1	71.4	PERCENT
SPACE CHORD RATIO (ROTOR)	.803	.802	.800	.797	.793	
SPACE CHORD RATIO (STATOR)	.681	.716	.750	.784	.818	
DIFFUSION FACTOR (ROTOR)	.478	.456	.434	.411	.390	
DIFFUSION FACTOR (STATOR)	.458	.447	.438	.429	.420	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.786	1.744	1.705	1.665	1.624	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.714	1.692	1.672	1.654	1.638	
SHOCK LOSS COEFFICIENT (ROTOR)	.00000	.00000	.00000	.00000	.00000	
SHOCK LOSS COEFFICIENT (STATOR)	.00000	.00000	.00000	.00000	.00000	
REYNOLDS NUMBER (ROTOR)	.55E+06	.55E+06	.54E+06	.52E+06	.50E+06	
REYNOLDS NUMBER (STATOR)	.70E+06	.70E+06	.71E+06	.71E+06	.72E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	.971	.973	.974	.976	.977	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01451	.01698	.01761	.01827	.01923
PROFILE LOSS COEFFICIENT (STATOR)	.01854	.01771	.01709	.01654	.01603
SECONDARY LOSS COEFFICIENT (ROTOR)			.01260		
SECONDARY LOSS COEFFICIENT (STATOR)			.01470		
ANNULUS LOSS COEFFICIENT (ROTOR)			.01066		
ANNULUS LOSS COEFFICIENT (STATOR)			.00842		
TOTAL LOSS COEFFICIENT (ROTOR)			.04098		
TOTAL LOSS COEFFICIENT (STATOR)			.04032		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.001

ROTOR ISENTROPIC EFFICIENCY

94.44 PERCENT

STAGE ISENTROPIC EFFICIENCY

91.05 PERCENT

STAGE PRESSURE RATIO

1.23

STAGE 5

	BLADE ROOT	254	BLADE MID HEIGHT	754	BLADE TIP	
BLADE SPEED (U)	338.4	351.1	363.7	376.2	388.8	M/S
ABS. STATOR OUTLET VELOCITY (V0)	226.7	236.2	245.8	255.4	265.1	M/S
REL. ROTOR INLET VELOCITY (V1)	359.2	370.8	382.5	394.2	406.0	M/S
REL. ROTOR OUTLET VELOCITY (V2)	252.0	262.5	273.7	285.6	298.0	M/S
ABS. STATOR INLET VELOCITY (V3)	308.1	302.7	297.8	293.4	289.3	M/S
ABS. STATOR OUTLET VELOCITY (V4)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	220.0	220.0	220.0	220.0	220.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	12.698	12.462	12.223	11.980	11.735	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	16.844	16.973	17.088	17.185	17.290	ATM
REL. ROTOR INLET MACH NUMBER (MREL)	.68	.70	.72	.74	.77	
ABS. STATOR INLET MACH NUMBER (MABS)	.57	.56	.55	.54	.54	
ROTOR DECELERATION (V2/V1)	.70	.71	.72	.73	.73	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT (DP/D)	.51	.50	.49	.48	.46	
STATOR DECELERATION (V0/V3)	.71	.73	.74	.75	.76	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	.48	.44	.41	.38	.36	
FLOW COEFFICIENT (VA/U)	.65	.63	.60	.58	.57	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	13.8	13.4	13.0	12.6	12.2	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	52.2	53.6	54.9	56.1	57.2	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	29.2	30.1	30.5	30.6	30.4	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	44.4	43.4	42.4	41.4	40.3	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	14.8	14.4	14.0	13.6	13.3	DEGREES
ROTOR DEFLECTION	23.0	20.6	18.4	16.5	14.8	DEGREES
STATOR DEFLECTION	28.4	29.0	28.4	27.8	27.2	DEGREES
REACTION	60.1	62.9	65.4	67.7	69.7	PERCENT
SPACE CHORD RATIO (ROTOR)	.805	.803	.800	.797	.793	
SPACE CHORD RATIO (STATOR)	.691	.721	.750	.779	.808	
DIFFUSION FACTOR (ROTOR)	.479	.460	.442	.422	.403	
DIFFUSION FACTOR (STATOR)	.467	.450	.430	.412	.403	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.789	1.754	1.716	1.684	1.650	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.724	1.706	1.689	1.673	1.659	
SHOCK LOSS COEFFICIENT (ROTOR)	.00000	.00000	.00000	.00000	.00000	
SHOCK LOSS COEFFICIENT (STATOR)	.00000	.00000	.00000	.00000	.00000	
REYNOLDS NUMBER (ROTOR)	.61E+06	.61E+06	.62E+06	.63E+06	.64E+06	
REYNOLDS NUMBER (STATOR)	.77E+06	.78E+06	.78E+06	.76E+06	.79E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	.967	.969	.970	.972	.973	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01634	.01677	.01730	.01781	.01860
PROFILE LOSS COEFFICIENT (STATOR)	.01800	.01740	.01676	.01628	.01585
SECONDARY LOSS COEFFICIENT (ROTOR)			.01304		
SECONDARY LOSS COEFFICIENT (STATOR)			.01499		
ANNULUS LOSS COEFFICIENT (ROTOR)			.01113		
ANNULUS LOSS COEFFICIENT (STATOR)			.00888		
TOTAL LOSS COEFFICIENT (ROTOR)			.04294		
TOTAL LOSS COEFFICIENT (STATOR)			.04073		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.001

ROTOR ISENTROPIC EFFICIENCY

94.37

PERCENT

STAGE ISENTROPIC EFFICIENCY

90.86

PERCENT

STAGE PRESSURE RATIO

1.22

	BLADE ROOT	254	BLADE MID HEIGHT	754	BLADE TIP	
BLADE SPEED (U)	345.8	358.4	367.3	378.1	388.9	M/S
ABS. STATOR OUTLET VELOCITY (V0)	227.4	227.1	226.7	226.4	226.0	M/S
REL. ROTOR INLET VELOCITY (V1)	362.0	372.0	382.2	392.3	402.5	M/S
REL. ROTOR OUTLET VELOCITY (V2)	254.7	263.8	273.4	283.8	294.3	M/S
ABS. STATOR INLET VELOCITY (V3)	309.3	304.8	300.5	296.6	292.9	M/S
ABS. STATOR OUTLET VELOCITY (V4)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	220.0	220.0	220.0	220.0	220.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	15.679	15.438	15.194	14.947	14.697	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	20.554	20.682	20.808	20.901	21.007	ATM
REL. ROTOR INLET MACH NUMBER (MREL)	.66	.68	.70	.72	.74	
ABS. STATOR INLET MACH NUMBER (MABS)	.56	.55	.54	.53	.53	
ROTOR DECELERATION (V2/V1)	.70	.71	.72	.72	.73	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT (DP/D)	.51	.50	.49	.48	.47	
STATOR DECELERATION (V0/V3)	.71	.72	.73	.74	.75	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	.46	.43	.41	.38	.36	
FLOW COEFFICIENT (VA/U)	.64	.62	.60	.58	.57	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	14.8	14.4	14.0	13.6	13.3	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	52.6	53.7	54.9	56.9	58.9	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	30.2	30.5	30.5	30.2	30.6	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	44.7	43.8	42.9	42.1	41.3	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	13.7	13.3	13.0	12.7	12.4	DEGREES
ROTOR DEFLECTION	22.3	20.2	18.4	16.7	15.2	DEGREES
STATOR DEFLECTION	31.0	30.3	29.9	29.4	28.9	DEGREES
REACTION	60.1	62.5	64.7	66.7	68.5	PERCENT

SPACE CHORD RATIO (ROTOR)	.805	.803	.800	.797	.793
SPACE CHORD RATIO (STATOR)	.700	.725	.750	.775	.800
DIFFUSION FACTOR (ROTOR)	.473	.457	.441	.425	.408
DIFFUSION FACTOR (STATOR)	.469	.462	.455	.448	.442
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.779	1.749	1.718	1.688	1.659
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.750	1.734	1.719	1.705	1.692
SHOCK LOSS COEFFICIENT (ROTOR)	.00000	.00000	.00000	.00000	.00000
SHOCK LOSS COEFFICIENT (STATOR)	.00000	.00000	.00000	.00000	.00000
REYNOLDS NUMBER (ROTOR)	.64E+06	.65E+06	.65E+06	.66E+06	.67E+06
REYNOLDS NUMBER (STATOR)	.81E+06	.81E+06	.81E+06	.82E+06	.82E+06
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000
AXIAL VELOCITY RATIO (STATOR)	.972	.973	.974	.976	.977

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.01637	.01674	.01724	.01767	.01834
PROFILE LOSS COEFFICIENT (STATOR)	.01758	.01710	.01664	.02045	.01991
SECONDARY LOSS COEFFICIENT (ROTOR)			.01303		
SECONDARY LOSS COEFFICIENT (STATOR)			.01638		
ANNULUS LOSS COEFFICIENT (ROTOR)			.01112		
ANNULUS LOSS COEFFICIENT (STATOR)			.00872		
TOTAL LOSS COEFFICIENT (ROTOR)			.04143		
TOTAL LOSS COEFFICIENT (STATOR)			.04348		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.001

ROTOR ISENTROPIC EFFICIENCY

94.40 PERCENT

STAGE ISENTROPIC EFFICIENCY

90.62 PERCENT

STAGE PRESSURE RATIO

1.21

		STAGE 1					
		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	351.7	361.0	370.3	379.7	389.0	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	226.4	226.1	225.8	225.5	225.3	M/S
REL. ROTOR INLET VELOCITY	(V1)	370.4	379.3	388.0	396.7	405.5	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	262.9	271.3	279.9	288.9	296.1	M/S
ABS. STATOR INLET VELOCITY	(V3)	302.5	298.9	295.5	292.2	289.2	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	220.0	220.0	220.0	220.0	220.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	220.0	220.0	220.0	220.0	220.0	M/S
STATIC PRESSURE AT ROTOR INLET	(PS1)	18.991	18.747	18.500	18.250	17.999	ATM
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	24.921	25.035	25.142	25.241	25.333	ATM
REL. ROTOR INLET MACH NUMBER	(MREL)	.66	.68	.69	.71	.72	
ABS. STATOR INLET MACH NUMBER	(MABS)	.53	.53	.52	.51	.51	
ROTOR DECELERATION	(V2/V1)	.71	.72	.72	.73	.74	DE HALLER NUMBER
PRESSURE RISE COEFFICIENT	(DP/D)	.50	.49	.48	.47	.46	
STATOR DECELERATION	(V0/V3)	.73	.74	.74	.75	.76	DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)		.44	.42	.40	.38	.36	
FLOW COEFFICIENT	(VA/U)	.63	.61	.59	.58	.57	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)		13.7	13.3	13.0	12.7	12.4	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)		53.6	54.5	55.5	56.3	57.1	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)		33.2	35.8	38.2	40.4	42.4	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)		43.4	42.6	41.9	41.2	40.5	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)		.0	.0	.0	.0	.0	DEGREES
ROTOR DEFLECTION		20.4	18.7	17.3	15.9	14.7	DEGREES
STATOR DEFLECTION		43.4	42.6	41.9	41.2	40.5	DEGREES
REACTION		42.9	44.8	46.5	48.1	49.6	PERCENT
SPACE CHORD RATIO (ROTOR)		.804	.802	.800	.797	.795	
SPACE CHORD RATIO (STATOR)		.710	.730	.750	.770	.790	
DIFFUSION FACTOR (ROTOR)		.456	.444	.429	.415	.401	
DIFFUSION FACTOR (STATOR)		.454	.447	.441	.435	.430	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.749	1.723	1.697	1.671	1.646	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.838	1.823	1.810	1.797	1.785	
SHOCK LOSS COEFFICIENT (ROTOR)		.00000	.00000	.00000	.00000	.00000	
SHOCK LOSS COEFFICIENT (STATOR)		.00000	.00000	.00000	.00000	.00000	
REYNOLDS NUMBER (ROTOR)		.70E+06	.71E+06	.72E+06	.72E+06	.73E+06	
REYNOLDS NUMBER (STATOR)		.91E+06	.91E+06	.92E+06	.92E+06	.93E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	.02094	.02144	.02186	.02161	.02141
PROFILE LOSS COEFFICIENT (STATOR)	.02314	.02266	.02222	.02139	.02104
SECONDARY LOSS COEFFICIENT (ROTOR)			.01332		
SECONDARY LOSS COEFFICIENT (STATOR)			.02637		
ANNULUS LOSS COEFFICIENT (ROTOR)			.01209		
ANNULUS LOSS COEFFICIENT (STATOR)			.00811		
TOTAL LOSS COEFFICIENT (ROTOR)			.04450		
TOTAL LOSS COEFFICIENT (STATOR)			.05656		

EFFICIENCY LOSS DUE TO TIP CLEARANCE

.001 PERCENT

ROTOR ISENTROPIC EFFICIENCY

93.72 PERCENT

STAGE ISENTROPIC EFFICIENCY	81.87	PERCENT
STAGE PRESSURE RATIO	1.18	

OVERALL PRESSURE RATIO	4.08	
COMPRESSOR ISENTROPIC EFFICIENCY	81.45	PERCENT
COMPRESSOR POLYTROPIC EFFICIENCY	88.11	PERCENT

Figure F.7 HOWELL Input Data File for LP Compressor.

```

5 STAGE LP-COMPRESSOR
0      5      1      101.0000      288.0000      4450.000
0
0.00  0.00  0.00  0.00
43.44749  40.47592  19.99451  -4.624632
794.0858  778.7074  772.3427  752.9825
341.4569  356.8353  363.1999  382.5602
0.80  0.07  0.50  0.00  30
0.70  0.06  0.50  0.00  38
0.000  0.000  0.000  0.000  0.000  0.000
0.000  1.026  1.010  0.000  0.000
0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
0.000  0.000  0.000  0.000  0.001  1.000
44.22551  37.72222  22.55855  -5.254201
757.0522  744.0179  738.6966  722.5512
378.4906  391.5249  396.8462  412.9915
0.80  0.07  0.50  0.00  36
0.70  0.06  0.50  0.00  45
0.000  0.000  0.000  0.000  0.000  0.000
0.000  1.026  1.010  0.000  0.000
0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
0.000  0.000  0.000  0.000  0.001  1.000
45.30348  33.55001  26.08606  -6.136202
726.0168  714.1237  709.3634  695.0295
409.5259  421.4189  426.1793  440.5132
0.80  0.07  0.50  0.00  40
0.70  0.06  0.50  0.00  50
0.000  0.000  0.000  0.000  0.000  0.000
0.000  1.026  1.010  0.000  0.000
0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
0.000  0.000  0.000  0.000  0.001  1.000
45.73852  33.71489  25.79863  -6.063631
699.3037  688.2164  683.7712  670.2473
436.2391  447.3263  451.7715  465.2954
0.80  0.07  0.50  0.00  43
0.70  0.06  0.50  0.00  53
0.000  0.000  0.000  0.000  0.000  0.000
0.000  1.026  1.010  0.000  0.000
0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
0.000  0.000  0.000  0.000  0.001  1.000
46.21382  33.87150  25.50100  -5.988618
679.3011  669.0737  664.5662  652.3607
456.2416  466.4691  470.5766  483.1821
0.80  0.07  0.50  0.00  47
0.70  0.06  0.50  0.00  57
0.000  0.000  0.000  0.000  0.000  0.000
0.000  1.026  1.010  0.000  0.000
0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
0.000  0.000  0.000  0.000  0.001  1.000
9
4450.000  -5  100%
4005.000  -5  100%
3560.000  -5  100%
3115.000  -5  100%
2670.000  -5  100%
2225.000  -5  100%
0

```

Figure F.8 HOWELL Input Data File For IP Compressor.

```

7 STAGE IP-COMPRESSOR
0 7 1 101.0000 288.0000 4450.000
1
0.70 0.00 1.00 0.00
47.26053 40.11669 21.53921 1.179484
662.6313 658.1331 656.2317 650.2759
463.8419 463.8419 463.8419 463.8419
0.80 0.07 0.50 0.00 79
0.70 0.06 0.50 0.00 94
0.000 0.000 0.000 0.000 0.000 0.000
0.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
47.24094 31.70748 28.41376 -0.5916066
642.4235 637.6764 635.8116 630.0620
463.8419 463.8419 463.8419 463.8419
0.80 0.07 0.50 0.00 79
0.70 0.06 0.50 0.00 94
0.000 0.000 0.000 0.000 0.000 0.000
5.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
47.91200 27.53031 30.53448 -1.172682
623.3994 616.4472 613.7822 605.6112
463.8419 463.8419 463.8419 463.8419
0.80 0.07 0.50 0.00 54
0.70 0.06 0.50 0.00 65
0.000 0.000 0.000 0.000 0.000 0.000
5.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
48.52930 23.45931 32.43694 -1.711268
606.4291 602.0495 600.4075 595.4022
463.8419 463.8419 463.8419 463.8419
0.80 0.07 0.50 0.00 86
0.70 0.06 0.50 0.00 104
0.000 0.000 0.000 0.000 0.000 0.000
5.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
48.79615 21.62072 32.90532 -1.846623
591.2939 587.3303 585.8618 581.3831
463.8419 463.8419 463.8419 463.8419
0.80 0.07 0.50 0.00 96
0.70 0.06 0.50 0.00 115
0.000 0.000 0.000 0.000 0.000 0.000
5.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
48.93535 20.57988 32.96568 -1.864211
576.2457 574.5270 573.1601 568.9765
463.8419 463.8419 463.8419 463.8419
0.80 0.07 0.50 0.00 101
0.70 0.06 0.50 0.00 121
0.000 0.000 0.000 0.000 0.000 0.000
5.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
49.12591 19.12743 33.35656 -8.040668
567.3186 563.9239 562.6871 558.8621
463.8419 463.8419 463.8419 463.8419
0.80 0.07 0.50 0.00 111
0.70 0.06 0.50 0.00 132
0.000 0.000 0.000 0.000 0.000 0.000
5.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
3744.258 -5 1100%
3369.832 -5 190%
2995.407 -5 180%
2620.981 -5 170%
2246.555 -5 160%
1872.129 -5 150%
0

```


Figure F.9 HOWELL Input Data File for HP Compressor.

```

7 STAGE HP-COMPRESSOR 19/07/90
0 1 101.0000 288.0000 10000.00
1
0.70 0.00 1.00 0.00
49.38477 43.18531 24.40508 5.179709
371.8446 371.8446 371.8446 371.8446
275.1650 278.5939 280.1380 284.7686
0.80 0.07 0.50 0.00 54
0.75 0.06 0.50 0.00 63
0.000 0.000 0.000 0.000 0.000 0.000
0.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
49.63995 36.06042 32.61103 4.091167
371.8446 371.8446 371.8446 371.8446
290.5576 293.6512 294.9479 298.7405
0.80 0.07 0.50 0.00 66
0.75 0.06 0.50 0.00 77
0.000 0.000 0.000 0.000 0.000 0.000
0.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
50.92754 33.41769 35.32058 5.728447
371.8446 371.8446 371.8446 371.8446
303.3160 305.9453 307.0383 310.1826
0.80 0.07 0.50 0.00 79
0.75 0.06 0.50 0.00 93
0.000 0.000 0.000 0.000 0.000 0.000
10.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
51.72231 31.36478 37.38110 6.304191
371.8446 371.8446 371.8446 371.8446
314.0143 316.2611 317.1841 319.8125
0.80 0.07 0.50 0.00 95
0.75 0.06 0.50 0.00 111
0.000 0.000 0.000 0.000 0.000 0.000
12.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
52.46388 30.54779 38.38360 7.214350
371.8446 371.8446 371.8446 371.8446
322.7115 324.7279 325.5545 327.8987
0.80 0.07 0.50 0.00 100
0.75 0.06 0.50 0.00 125
0.000 0.000 0.000 0.000 0.000 0.000
13.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
52.97825 30.29635 38.93698 5.763213
371.8446 371.8446 371.8446 371.8446
329.6943 331.4272 332.1371 334.1573
0.80 0.07 0.50 0.00 127
0.75 0.06 0.50 0.00 146
0.000 0.000 0.000 0.000 0.000 0.000
14.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
53.58485 32.33856 37.87492 -9.841449
371.8446 371.8446 371.8446 371.8446
335.4377 336.9892 337.6375 339.5697
0.80 0.07 0.50 0.00 140
0.75 0.06 0.50 0.00 161
0.000 0.000 0.000 0.000 0.000 0.000
13.000 1.026 1.010 0.000 0.000
0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000
0.000 0.000 0.000 0.000 0.000 1.000
10
7044.699 -5 1100%
6340.229 -5 190%
5635.759 -5 180%
4931.290 -5 170%
4226.820 -5 160%
3522.350 -5 150%
0

```



```

HP TURBINE DESIGN POINT
1          !NUMBER OF STAGE
152.73     !MASS FLW
28.27E5    !INLET TOTAL PRESSURE
1800.      !INLET TOTAL TEMPERATURE
0.02767    !FUEL AIR RATIO
240.       !INLET AXIAL VELOCITY
0.875      !HUB-TIP RATIO
10000.     !RPM
0.90       !POLYTROPIC EFFICIENCY
277.       !TEMPERATURE DROP
0.0        !INLET AIR ANGLE
37.0       !EXIT AIR ANGLE
1          !DESIG TYPE
1.2 2.0    !ASPEC RATIO
1.0 0.95   !SPACE-CHORD RATIO
.05        !NOZZLE LOSS COEFFICIENT

```

Figure F.10 TURBGEO Input Data for HP Turbine.

Figure F.11 TURBGEO Output Data for HP Turbine.

```

*****
TURBINE FITCH-LINE DESIGN CALCULATION

      HP TURBINE DESIGN POINT

            1 STAGE

DESIGN INLET CONDITIONS

MASS FLOW (Kg/s) ..... 152.73
INLET TOTAL PRESSURE (N/m²) ..... 2827000.00
OUTLET TOTAL PRESSURE (N/m²) ..... 1326292.00
INLET TOTAL TEMPERATURE (K) ..... 1800.00
TEMPERATURE DROP (Tin-Tout) ..... 277.00
FUEL AIR RATIO ..... .02767
AXIAL VELOCITY (m/s) ..... 240.00
INLET HUB-TIP RATIO ..... .8750
ROTATIONAL SPEED ..... 16000.00
ISENTROPIC EFFICIENCY (%) ..... 90.00
INLET AIR ANGLE (degree) ..... .00
OUTLET AIR ANGLE (degree) ..... 37.00
NOZZLE ASPECT RATIO ..... 1.20
ROTOR ASPECT RATIO ..... 1.15
NOZZLE/ROTOR SPACE-CHORD RATIO ..... 1.00/ .95
NOZZLE LOSS COEFFICIENT ..... .0500
ROTOR LOSS COEFFICIENT ..... .1220
GAMMA ..... 1.329
SPECIFIC HEAT [Cpg] (J/kgK) ..... 1133.8830


ANNULUS GEOMETRY

TIP RADIUS (cm) ..... 40.59 42.03 42.39 44.53
HUB RADIUS (cm) ..... 35.52 35.52 35.52 35.52
NOZZLE AXIAL CHORD (cm) ..... 4.0387
NOZZLE CHORD (cm) ..... 4.8299
ROTOR AXIAL CHORD (cm) ..... 5.9798
ROTOR CHORD (cm) ..... 6.9002
INTER GAP (cm) ..... 1.0097
OUTLET GAP (cm) ..... 1.4949
OUTLET HUB-TIP RATIO ..... .7975
ANNULUS INLET AREA (m²) ..... .1213
NOZZLE EXIT AREA (m²) ..... .0610
ROTOR EXIT AREA (m²) ..... .1811
ANNULUS EXIT AREA (m²) ..... .2268
ANNULUS FLARE ANGLE (degree) ..... 19.68
NUMBER OF BLADES (NOZZLE) ..... 52
NUMBER OF BLADES (ROTOR) ..... 38

DWEIFEL S CORRELATION
NOZZLE/ROTOR SPACE-CHORD RATIO ..... 1.57/ 1.24


PITCH LINE GAS PARAMETERS

NOZZLE INLET MACH NUMBER ..... .2916
NOZZLE OUTLET MACH NUMBER ..... .7812
ROTOR INLET MACH NUMBER ..... .3444
ROTOR INLET ROOT MACH NUMBER ..... .4417
ROTOR EXIT MACH NUMBER ..... .3981
MEAN BLADE SPEED (m/s) ..... 419.14
FLOW COEFFICIENT ..... .57
BLADE LOADING COEFFICIENT ..... 1.79
REACTION % ..... 53.7547

```

REACTION AT THE ROOT %	36.6181
NOZZLE ABS. INLET VELOCITY (m/s)	240.00
NOZZLE ABS. OUTLET VELOCITY (m/s)	617.10
ROTOR RELAT. INLET VELOCITY (m/s)	282.69
ROTOR RELAT. OUTLET VELOCITY (m/s)	646.21
ROTOR ABS. OUTLET VELOCITY (m/s)	300.51
ANNULUS EXIT AXIAL VELOCITY (m/s)	240.00
NOZZLE ABS. INLET AIR ANGLE (degree)	.00
NOZZLE ABS. EXIT AIR ANGLE (degree)	67.11
ROTOR RELAT. INLET AIR ANGLE (degree)	31.90
ROTOR RELAT. EXIT AIR ANGLE (degree)	68.20
ROTOR ABS. EXIT AIR ANGLE (degree)	37.00
NOZZLE INLET BLADE ANGLE (degree)	.00
NOZZLE EXIT BLADE ANGLE (degree)	52.68
ROTOR INLET BLADE ANGLE (degree)	31.90
ROTOR EXIT BLADE ANGLE (degree)	60.59

STAGE OVERALL CONDITION

PRESSURE RATIO	2.1315
PREDICTED ISENT. EFFICIENCY (%)	90.2020

This is the warning & error sections which
give information about flow conditions for
each appropriate section.

WARNING: ANNULUS INLET SUBSONIC FLOW
WARNING: ANNULUS OUTLET SUBSONIC FLOW
WARNING: NOZZLE OUTLET SUBSONIC FLOW

TURBINE OVERALL CONDITION

OVERALL PRESSURE RATIO (Pin/Pout)	2.1275
OVERALL ISENT. EFFICIENCY (%)	90.20
POLYTROPIC EFFICIENCY (%)	89.33
POWER OUTPUT (KJ/S)	47970.30

Figure F.12 TURBDES Output Data for HP turbine.

Free Vortex Turbine Design Program

HP TURBINE DESIGN POINT				STAGE	1				
Input Data									
Inlet Mass Flow Rate	152.73 kg/s								
Power Output	47.9703 MW								
Shaft Rotational Speed	10000.0 rev/min								
Inlet Swirl Angle	.00 deg								
Turbine T/T Isentropic Efficiency	.90								
Outlet Swirl Angle	37.00 deg								
Inlet Fuel/Air Ratio	.0277								
Work Done Factor	1.00								
	P (inf)	T (inf)							
	(kPa)	(K)							
Tip	2827.0	1800.0							
Mean	2827.0	1800.0							
Root	2827.0	1800.0							
Blade Stress Input Information									
Metal Density	8.00	Mg/cu.m	Taper Factor	1.000	Shroud Equivalent Length .00 mm				
Plane Number	1	2	3	4					
Axial Distance	.0	40.4	50.6	120.4	(mm)				
Tip Radius	405.89	420.34	423.95	445.33	(mm)				
Mean Radius	380.52	387.75	389.55	400.24	(mm)				
Root Radius	355.16	355.16	355.16	355.16	(mm)				
N.G.V. Aerofoil Data			Rotor Aerofoil Data						
True Chord	48.40	(mm)	True Chord	62.65	(mm)				
Axial Chord	40.39	(mm)	Axial Chord	59.80	(mm)				
Max Thickness	9.69	(mm)	Max Thickness	12.51	(mm)				
T.E. Thickness	.94	(mm)	T.E. Thickness	1.28	(mm)				
			Number of Seals	0					
			Tip Seal Clearance	.25	(mm)				
Ainley, Mathieson, Dunham & Cane Loss Coefficients									
N.G.V.			Rotor						
Profile Loss (YF)	.038		Profile Loss (YF)	.074					
Secondary Loss (YS)	.057		Secondary Loss (YS)	.088					
Tip (YK)	.000		Tip (YK)	.037					
Total + R.E.	.095		Total + R.E.	.199					
Reynolds Number	2.06E+06		Reynolds Number	2.01E+06					
Number of N.G.V.'s	52	Space/Axial Chord Ratio 1.16	Axial Chord Aspect Ratio 1.61						
		Space/Chord Ratio .97	Aspect Ratio 1.35						
Number of Rotor 's	38	Space/Axial Chord Ratio 1.09	Axial Chord Aspect Ratio 1.33						
		Space/Chord Ratio 1.04	Aspect Ratio 1.27						
Nozzle Guide Vane Outlet Triangles,									
	V(0)	V(1)	VW(0)	ALFA(0)	ALFA(1)	MA(0)	MA(1)	VA(0,1)	
	(m/s)	(m/s)	(m/s)	(deg)	(deg)	()	(-)	(-)	(m/s)
Tip	581.5	254.8	527.7	65.16	16.54	.741	.325	244.3	
Mean	622.0	291.2	572.0	66.88	32.98	.797	.373	244.3	
Root	670.6	351.4	624.5	68.64	45.96	.865	.453	244.3	
Rotor Outlet Triangles									
	V(3)	V(2)	VW(3)	ALFA(3)	ALFA(2)	MA(3)	MA(2)	VA(2,3)	
	(m/s)	(m/s)	(m/s)	(deg)	(deg)	()	(-)	(-)	(m/s)
Tip	296.5	668.1	166.2	34.11	68.45	.395	.891	245.5	
Mean	307.3	646.9	185.0	37.00	67.70	.410	.863	245.5	
Root	322.0	630.1	208.4	40.34	67.07	.430	.841	245.5	
Stagnation Temperatures & Pressures									

	T(0) (K)	T(1) (K)	T(2) (K)	T(3) (K)	P(0) (kPa)	P(1) (kPa)	P(2) (kPa)	P(3) (kPa)
Tip	1800.0	1694.4	1694.4	1555.9	2751.7	2096.0	1947.6	1326.4
Mean	1800.0	1683.3	1683.3	1558.1	2742.7	2027.8	1890.5	1334.7
Root	1800.0	1673.9	1673.9	1560.6	2731.7	1969.8	1841.2	1342.5

	Velocity Ratios (0)/(inf) (2)/(1) (-)		Deflections (inf,0)1) (1,2) (deg) (deg)		Blade Stress Tensile Bending (MPa) (MPa)		Outlet Static Pressure (kPa)
Tip	2.42	2.62	65.2	85.0	.0	.0	1200.9
Mean	2.38	2.22	66.9	100.7	.0	29.1	1199.6
Root	2.79	1.79	68.6	113.0	.0	92.0	1194.1

	Stage Parameters u dVw (m/s) (m/s)		dH/u.sq (-)	Reaction (%)	Zweifel Stator (-)	Lift Coef Rotor (-)
Tip	455.2	693.9	1.52	60.4	.79	.75
Mean	413.5	757.0	1.83	53.3	.71	.81
Root	371.9	833.0	2.24	44.2	.63	.85

ETA(TT) = 87.5 % CP dT/T = 362.8 N/ RT = .0
ETA(TS) = 77.5 % Va/u = .59

OVERALL PRESSURE RATIO..... 2.118
TURBINE ISENTROPIC EFFICIENCY..... 87.570
TURBINE POLYTROPIC EFFICIENCY..... 86.625

```

LP TURBINE DESIGN POINT
4
160.20
10.54E5
1210.76
0.0207
205.0
0.85      0.87
4450.
0.90
85. 82. 81. 79.35
0.0 21. 21. 21.  0.0 14.0 13. 12.0
19.0 12.0
2
2.0  2.29 2.38 2.99 3.07 3.57 3.2 3.6
1.1 0.95 1.11 1.1 1.11 1.0 1.12 1.0
0.05

```

Figure F.13 TURBGEO Input Data File for LP Turbine.

Figure F.14 TURBGEO Output Data for LP Turbine.

TURBINE PITCH-LINE DESIGN CALCULATION

LP TURBINE DESIGN POINT

1 STAGE

DESIGN INLET CONDITIONS

MASS FLOW (Kg/s)	160.20
INLET TOTAL PRESSURE (N/m ²)	1054000.00
OUTLET TOTAL PRESSURE (N/m ²)	773656.20
INLET TOTAL TEMPERATURE (K)	1210.76
TEMPERATURE DROP (T _{in} -T _{out})	85.00
FUEL AIR RATIO	.02070
AXIAL VELOCITY (m/s)	205.00
INLET HUB-TIP RATIO	.8500
ROTATIONAL SPEED	4450.00
ISENTROPIC EFFICIENCY (%)	90.00
INLET AIR ANGLE (degree)	.00
OUTLET AIR ANGLE (degree)	21.00
NOZZLE ASPECT RATIO	2.00
ROTOR ASPECT RATIO	2.15
NOZZLE/ROTOR SPACE-CHORD RATIO	1.10/ .95
NOZZLE LOSS COEFFICIENT	.0500
ROTOR LOSS COEFFICIENT	.0948
GAMMA	1.356
SPECIFIC HEAT [Cp] (J/kgK)	1083.6070

ANNULUS GEOMETRY

TIP RADIUS (cm)	55.60	56.10	56.22	56.77
HUB RADIUS (cm)	47.26	46.76	46.63	46.09
NOZZLE AXIAL CHORD (cm)	3.9929			
NOZZLE CHORD (cm)	4.4191			
ROTOR AXIAL CHORD (cm)	4.3611			
ROTOR CHORD (cm)	4.7194			
INTER GAP (cm)	.9982			
OUTLET GAP (cm)	1.0903			
OUTLET HUB-TIP RATIO	.8119			
ANNULUS INLET AREA (m ²)	.2695			
NOZZLE EXIT AREA (m ²)	.1681			
ROTOR EXIT AREA (m ²)	.3220			
ANNULUS EXIT AREA (m ²)	.3449			
ANNULUS FLARE ANGLE (degree)	14.23			
NUMBER OF BLADES (NOZZLE)	66			
NUMBER OF BLADES (ROTOR)	72			

SWEIPER'S CORRELATION

NOZZLE/ROTOR SPACE-CHORD RATIO	1.39/ 1.11
--------------------------------	------------

PITCH LINE GAS PARAMETERS

NOZZLE INLET MACH NUMBER	.3010
NOZZLE OUTLET MACH NUMBER	.5504
ROTOR INLET MACH NUMBER	.3165
ROTOR INLET ROOT MACH NUMBER	.3496
ROTOR EXIT MACH NUMBER	.3345
MEAN BLADE SPEED (m/s)	239.66
FLOW COEFFICIENT	.86

BLADE LOADING COEFFICIENT	1.60
REACTION %	52.6520
REACTION AT THE ROOT %	38.4658
NOZZLE ABS. INLET VELOCITY (m/s)	205.00
NOZZLE ABS. OUTLET VELOCITY (m/s)	368.02
ROTOR RELAT. INLET VELOCITY (m/s)	215.36
ROTOR RELAT. OUTLET VELOCITY (m/s)	378.64
ROTOR ABS. OUTLET VELOCITY (m/s)	219.58
ANNULUS EXIT AXIAL VELOCITY (m/s)	205.00
NOZZLE ABS. INLET AIR ANGLE (degree)	.00
NOZZLE ABS. EXIT AIR ANGLE (degree)	56.15
ROTOR RELAT. INLET AIR ANGLE (degree)	17.84
ROTOR RELAT. EXIT AIR ANGLE (degree)	57.22
ROTOR ABS. EXIT AIR ANGLE (degree)	21.00
NOZZLE INLET BLADE ANGLE (degree)	.00
NOZZLE EXIT BLADE ANGLE (degree)	43.48
ROTOR INLET BLADE ANGLE (degree)	17.84
ROTOR EXIT BLADE ANGLE (degree)	48.96

STAGE OVERALL CONDITION

PRESSURE RATIO	1.3624
PREDICTED ISENT. EFFICIENCY (%)	90.1620

This is the warning & error sections which
give information about flow conditions for
each appropriate section.

WARNING: ANNULUS INLET SUBSONIC FLOW
WARNING: ANNULUS OUTLET SUBSONIC FLOW
WARNING: NOZZLE OUTLET SUBSONIC FLOW

TURBINE PITCH-LINE DESIGN CALCULATION

LP TURBINE DESIGN POINT

2 STAGE

DESIGN INLET CONDITIONS

MASS FLOW (Kg/s)	160.20
INLET TOTAL PRESSURE (N/m ²)	773656.20
OUTLET TOTAL PRESSURE (N/m ²)	562549.10
INLET TOTAL TEMPERATURE (K)	1125.76
TEMPERATURE DROP (Tin-Tout)	82.00
FUEL AIR RATIO	.02070
AXIAL VELOCITY (m/s)	205.00
INLET HUB-TIP RATIO	.8119
ROTATIONAL SPEED	4450.00
ISENTROPIC EFFICIENCY (%)	90.00
INLET AIR ANGLE (degree)	21.00
OUTLET AIR ANGLE (degree)	21.00
NOZZLE ASPECT RATIO	2.29
ROTOR ASPECT RATIO	1.86
NOZZLE/ROTOR SPACE-CHORD RATIO	1.11/1.10
NOZZLE LOSS COEFFICIENT	.0500

ROTOR LOSS COEFFICIENT0920
 GAMMA 1.360
 SPECIFIC HEAT [Cpg] (J/kgK) 1074.6690

ANNULUS GEOMETRY

TIP RADIUS (cm)	56.77	57.34	57.48	58.26
HUB RADIUS (cm)	46.09	45.52	45.38	44.59
NOZZLE AXIAL CHORD (cm)	4.6427			
NOZZLE CHORD (cm)	4.9108			
ROTOR AXIAL CHORD (cm)	6.3757			
ROTOR CHORD (cm)	6.9217			
INTER GAP (cm)	1.1607			
OUTLET GAP (cm)	1.5939			
OUTLET HUB-TIP RATIO	.7654			
ANNULUS INLET AREA (m ²)	.3449			
NOZZLE EXIT AREA (m ²)	.2209			
ROTOR EXIT AREA (m ²)	.4124			
ANNULUS EXIT AREA (m ²)	.4417			
ANNULUS FLARE ANGLE (degree)	14.02			
NUMBER OF BLADES (NOZZLE)	59			
NUMBER OF BLADES (ROTOR)	42			

WEISSLER'S CORRELATION
 NOZZLE/ROTOR SPACE-CHORD RATIO 1.06/ 1.17

PITCH LINE GAS PARAMETERS

NOZZLE INLET MACH NUMBER	.3345
NOZZLE OUTLET MACH NUMBER	.5488
ROTOR INLET MACH NUMBER	.3210
ROTOR INLET ROOT MACH NUMBER	.3595
ROTOR EXIT MACH NUMBER	.3472
MEAN BLADE SPEED (m/s)	239.66
FLOW COEFFICIENT	.86
BLADE LOADING COEFFICIENT	1.53
REACTION %	56.1200
REACTION AT THE ROOT %	38.5028
NOZZLE ABS. INLET VELOCITY (m/s)	219.58
NOZZLE ABS. OUTLET VELOCITY (m/s)	354.34
ROTOR RELAT. INLET VELOCITY (m/s)	210.86
ROTOR RELAT. OUTLET VELOCITY (m/s)	378.64
ROTOR ABS. OUTLET VELOCITY (m/s)	219.58
ANNULUS EXIT AXIAL VELOCITY (m/s)	205.00
NOZZLE ABS. INLET AIR ANGLE (degree)	21.00
NOZZLE ABS. EXIT AIR ANGLE (degree)	54.65
ROTOR RELAT. INLET AIR ANGLE (degree)	13.54
ROTOR RELAT. EXIT AIR ANGLE (degree)	57.22
ROTOR ABS. EXIT AIR ANGLE (degree)	21.00
NOZZLE INLET BLADE ANGLE (degree)	21.00
NOZZLE EXIT BLADE ANGLE (degree)	47.02
ROTOR INLET BLADE ANGLE (degree)	13.54
ROTOR EXIT BLADE ANGLE (degree)	47.36

 STAGE OVERALL CONDITION
 PRESSURE RATIO 1.3753
 PREDICTED ISENT. EFFICIENCY (%) 90.1772

 This is the warning & error sections which
 give information about flow conditions for

each appropriate section.

WARNING: ANNULUS INLET SUBSONIC FLOW
 WARNING: ANNULUS OUTLET SUBSONIC FLOW
 WARNING: NOZZLE OUTLET SUBSONIC FLOW

TURBINE PITCH-LINE DESIGN CALCULATION

LP TURBINE DESIGN POINT

3 STAGE

DESIGN INLET CONDITIONS

MASS FLOW (Kg/s)	160.20
INLET TOTAL PRESSURE (N/m ²)	562549.10
OUTLET TOTAL PRESSURE (N/m ²)	401347.30
INLET TOTAL TEMPERATURE (K)	1043.76
TEMPERATURE DROP (Tin-Tout)	81.00
FUEL AIR RATIO	.02070
AXIAL VELOCITY (m/s)	205.00
INLET HUB-TIP RATIO	.7654
ROTATIONAL SPEED	4450.00
ISENTROPIC EFFICIENCY (%)	90.00
INLET AIR ANGLE (degree)	21.00
OUTLET AIR ANGLE (degree)	21.00
NOZZLE ASPECT RATIO	2.38
ROTOR ASPECT RATIO	1.78
NOZZLE/ROTOR SPACE-CHORD RATIO	1.11/1.00
NOZZLE LOSS COEFFICIENT	.0500
ROTOR LOSS COEFFICIENT	.0906
GAMMA	1.364
SPECIFIC HEAT [Cpg] (J/kgK)	1065.8130

ANNULUS GEOMETRY

TIP RADIUS (cm)	58.26	59.01	59.20	60.31
HUB RADIUS (cm)	44.59	43.85	43.66	42.55
NOZZLE AXIAL CHORD (cm)	5.7446			
NOZZLE CHORD (cm)	6.0573			
ROTOR AXIAL CHORD (cm)	8.5466			
ROTOR CHORD (cm)	9.3423			
INTER GAP (cm)	1.4362			
OUTLET GAP (cm)	2.1366			
OUTLET HUB-TIP RATIO	.7055			
ANNULUS INLET AREA (m ²)	.4417			
NOZZLE EXIT AREA (m ²)	.2884			
ROTOR EXIT AREA (m ²)	.5357			
ANNULUS EXIT AREA (m ²)	.5739			
ANNULUS FLARE ANGLE (degree)	14.82			
NUMBER OF BLADES (NOZZLE)	48			
NUMBER OF BLADES (ROTOR)	34			

SWEIFEL S CORRELATION	
NOZZLE/ROTOR SPACE-CHORD RATIO	1.06/ 1.20

PITCH LINE GAS PARAMETERS

NOZZLE INLET MACH NUMBER	.3472
NOZZLE OUTLET MACH NUMBER	.5601
ROTOR INLET MACH NUMBER	.3305
ROTOR INLET ROOT MACH NUMBER	.3832
ROTOR EXIT MACH NUMBER	.3614
MEAN BLADE SPEED (m/s)	239.66
FLOW COEFFICIENT	.86
BLADE LOADING COEFFICIENT	1.50
REACTION %	57.6800
REACTION AT THE ROOT %	35.0446
NOZZLE ABS. INLET VELOCITY (m/s)	219.58
NOZZLE ABS. OUTLET VELOCITY (m/s)	348.26
ROTOR RELAT. INLET VELOCITY (m/s)	209.23
ROTOR RELAT. OUTLET VELOCITY (m/s)	378.64
ROTOR ABS. OUTLET VELOCITY (m/s)	219.58
ANNULUS EXIT AXIAL VELOCITY (m/s)	205.00
NOZZLE ABS. INLET AIR ANGLE (degree)	21.00
NOZZLE ABS. EXIT AIR ANGLE (degree)	53.94
ROTOR RELAT. INLET AIR ANGLE (degree)	11.55
ROTOR RELAT. EXIT AIR ANGLE (degree)	57.22
ROTOR ABS. EXIT AIR ANGLE (degree)	21.00
NOZZLE INLET BLADE ANGLE (degree)	21.00
NOZZLE EXIT BLADE ANGLE (degree)	46.47
ROTOR INLET BLADE ANGLE (degree)	11.55
ROTOR EXIT BLADE ANGLE (degree)	47.39

 STAGE OVERALL CONDITION
 PRESSURE RATIO 1.4017
 PREDICTED ISENT. EFFICIENCY (%) 90.1943

 This is the warning & error sections which
 give information about flow conditions for
 each appropriate section.

WARNING: ANNULUS INLET SUBSONIC FLOW
 WARNING: ANNULUS OUTLET SUBSONIC FLOW
 WARNING: NOZZLE OUTLET SUBSONIC FLOW

 TURBINE FITCH-LINE DESIGN CALCULATION

LP TURBINE DESIGN POINT

4 STAGE

DESIGN INLET CONDITIONS

MASS FLOW (Kg/s)	160.20
INLET TOTAL PRESSURE (N/m ²)	401347.30
OUTLET TOTAL PRESSURE (N/m ²)	280942.70
INLET TOTAL TEMPERATURE (K)	962.76
TEMPERATURE DROP (Tin-Tout)	79.35
FUEL AIR RATIO	.02070
AXIAL VELOCITY (m/s)	205.00
INLET HUB-TIP RATIO	.7055
ROTATIONAL SPEED	4450.00
ISENTROPIC EFFICIENCY (%)	90.00
INLET AIR ANGLE (degree)	21.00
OUTLET AIR ANGLE (degree)	19.00

NOZZLE ASPECT RATIO	2.99
ROTOR ASPECT RATIO	2.32
NOZZLE/ROTOR SPACE-CHORD RATIO	1.12/1.00
NOZZLE LOSS COEFFICIENT	.0500
ROTOR LOSS COEFFICIENT	.0904
GAMMA	1.369
SPECIFIC HEAT [Cpg] (J/kgK)	1057.2190

ANNULUS GEOMETRY

TIP RADIUS (cm)	60.31	61.35	61.61	63.11
HUB RADIUS (cm)	42.55	41.51	41.25	39.74
NOZZLE AXIAL CHORD (cm)	5.9693			
NOZZLE CHORD (cm)	6.2876			
ROTOR AXIAL CHORD (cm)	8.6311			
ROTOR CHORD (cm)	9.4099			
INTER GAP (cm)	1.4923			
OUTLET GAP (cm)	2.1578			
OUTLET HUB-TIP RATIO	.6297			
ANNULUS INLET AREA (m ²)	.5739			
NOZZLE EXIT AREA (m ²)	.3792			
ROTOR EXIT AREA (m ²)	.7140			
ANNULUS EXIT AREA (m ²)	.7552			
ANNULUS FLARE ANGLE (degree)	19.78			
NUMBER OF BLADES (NOZZLE)	45			
NUMBER OF BLADES (ROTOR)	34			

DWEIFEL S CORRELATION	
NOZZLE/ROTOR SPACE-CHORD RATIO	1.06/ 1.21

PITCH LINE GAS PARAMETERS

NOZZLE INLET MACH NUMBER	.3614
NOZZLE OUTLET MACH NUMBER	.5809
ROTOR INLET MACH NUMBER	.3433
ROTOR INLET ROOT MACH NUMBER	.4261
ROTOR EXIT MACH NUMBER	.3722
MEAN BLADE SPEED (m/s)	239.66
FLOW COEFFICIENT	.86
BLADE LOADING COEFFICIENT	1.46
REACTION %	56.4227
REACTION AT THE ROOT %	25.8716
NOZZLE ABS. INLET VELOCITY (m/s)	219.58
NOZZLE ABS. OUTLET VELOCITY (m/s)	346.59
ROTOR RELAT. INLET VELOCITY (m/s)	208.83
ROTOR RELAT. OUTLET VELOCITY (m/s)	371.85
ROTOR ABS. OUTLET VELOCITY (m/s)	216.81
ANNULUS EXIT AXIAL VELOCITY (m/s)	205.00
NOZZLE ABS. INLET AIR ANGLE (degree)	21.00
NOZZLE ABS. EXIT AIR ANGLE (degree)	53.74
ROTOR RELAT. INLET AIR ANGLE (degree)	10.99
ROTOR RELAT. EXIT AIR ANGLE (degree)	56.54
ROTOR ABS. EXIT AIR ANGLE (degree)	19.00
NOZZLE INLET BLADE ANGLE (degree)	21.00
NOZZLE EXIT BLADE ANGLE (degree)	46.28
ROTOR INLET BLADE ANGLE (degree)	10.99
ROTOR EXIT BLADE ANGLE (degree)	46.74

STAGE OVERALL CONDITION

PRESSURE RATIO	1.4286
PREDICTED ISENT. EFFICIENCY (%)	90.2080

This is the warning & error sections which
give information about flow conditions for
each appropriate section.

WARNING: ANNULUS INLET SUBSONIC FLOW

WARNING: ANNULUS OUTLET SUBSONIC FLOW

WARNING: NOZZLE OUTLET SUBSONIC FLOW

TURBINE OVERALL CONDITION

OVERALL PRESSURE RATIO (Pin/Pout)	3.7410
OVERALL ISENT. EFFICIENCY (%)	91.36
POLYTROPIC EFFICIENCY (%)	89.83
POWER OUTPUT (KJ/S)	55442.11

Figure F.15 TURBIDES Output Data for LP Turbine.

Free Vortex Turbine Design Program

LP TURBINE DESIGN POINT

STAGE

1

Input Data

Inlet Mass Flow Rate	160.20 kg/s
Power Output	14.7555 MW
Shaft Rotational Speed	4450.0 rev/min
Inlet Swirl Angle	.00 deg
Turbine T/T Isentropic Efficiency	.90
Outlet Swirl Angle	21.00 deg
Inlet Fuel/Air Ratio	.0207
Work Done Factor	1.00

	P (inf)	T (inf)
	(kPa)	(K)
Tip	1054.0	1210.8
Mean	1054.0	1210.8
Root	1054.0	1210.8

Blade Stress Input Information

Metal Density	8.00	Mg/cu.m	Taper Factor	1.000	Shroud Equivalent Length	.00 mm
Plane Number	1	2	3	4		
Axial Distance	.0	39.9	59.9	103.5	(mm)	
Tip Radius	555.98	560.96	562.21	567.65	(mm)	
Mean Radius	514.28	514.28	514.28	514.28	(mm)	
Root Radius	472.58	467.60	466.35	460.90	(mm)	
N.G.V. Aerofoil Data						
True Chord	45.24 (mm)					
Axial Chord	39.93 (mm)					
Max Thickness	9.04 (mm)					
T.E. Thickness	.99 (mm)					
Rotor Aerofoil Data						
True Chord	46.32 (mm)					
Axial Chord	43.61 (mm)					
Max Thickness	9.27 (mm)					
T.E. Thickness	.90 (mm)					
Number of Seals	0					
Tip Seal Clearance	.25 (mm)					

Ainley, Mathieson, Dunham & Cane Loss Coefficients

N.G.V.			Rotor		
Profile Loss (YP)	.028		Profile Loss (YP)	.032	
Secondary Loss (YS)	.033		Secondary Loss (YS)	.042	
Tip (YK)	.000		Tip (YK)	.018	
Total + R.E.	.060		Total + R.E.	.092	
Reynolds Number	9.37E+05		Reynolds Number	8.78E+05	

Number of N.G.V.'s	56	Space/Axial Chord Ratio	1.23	Axial Chord Aspect Ratio	2.34
		Space/Chord Ratio	1.08	Aspect Ratio	2.06
Number of Rotor 's	72	Space/Axial Chord Ratio	1.03	Axial Chord Aspect Ratio	2.32
		Space/Chord Ratio	.97	Aspect Ratio	2.19

Nozzle Guide Vane Outlet Triangles

	V(0)	V(1)	VW(0)	ALFA(0)	ALFA(1)	MA(0)	MA(1)	VA(0,1)	
	(m/s)	(m/s)	(m/s)	(deg)	(deg)	(-)	(-)	(-)	(m/s)
Tip	347.1	205.9	279.9	53.75	4.64	.525	.312	205.2	
Mean	367.9	215.5	305.3	56.09	17.74	.558	.327	205.2	
Root	393.5	237.6	335.8	58.57	30.26	.599	.362	205.2	

Rotor Outlet Triangles

	V(3)	V(2)	VW(3)	ALFA(3)	ALFA(2)	MA(3)	MA(2)	VA(2,3)	
	(m/s)	(m/s)	(m/s)	(deg)	(deg)	(-)	(-)	(-)	(m/s)
Tip	217.9	393.1	71.6	19.18	58.42	.337	.607	205.9	
Mean	220.5	379.4	79.0	21.00	57.14	.340	.586	205.9	
Root	223.9	367.3	88.2	23.19	55.82	.346	.567	205.9	

Stagnation Temperatures & Pressures

	T(0) (K)	T(1) (K)	T(2) (K)	T(3) (K)	P(0) (kPa)	P(1) (kPa)	P(2) (kPa)	P(3) (kPa)
Tip	1210.6	1178.5	1178.6	1134.5	1043.8	931.6	914.0	778.0
Mean	1210.8	1174.1	1174.1	1134.9	1042.6	915.7	899.5	779.0
Root	1210.8	1170.2	1170.2	1135.3	1041.2	901.6	886.4	779.9

	Velocity Ratios		Deflections		Blade Stress		Outlet Static Pressure
	(0)/(inf) (-)	(2)/(1) (-)	(inf,0)1) (deg)	(1,2) (deg)	Tensile (MPa)	Bending (MPa)	
Tip	1.69	1.91	53.8	63.1	.0	.0	722.9
Mean	1.79	1.76	56.1	74.9	.0	31.0	722.6
Root	1.91	1.35	58.6	86.2	.0	111.9	721.7

	Stage Parameters		Reaction	Zweifel	Lift Coef	
	u	dVw			Stator	Rotor
	(m/s)	(m/s)	(-)	(%)	(-)	(-)
Tip	263.3	351.3	1.34	60.6	1.16	.97
Mean	239.7	364.3	1.60	52.9	1.05	1.04
Root	216.1	424.0	1.96	42.8	.93	1.11

ETA(TT) = 90.5 % CF dT/T = 147.1 W/ RT = .0
 ETA(TS) = 73.1 % Va/u = .86

Free Vortex Turbine Design Program

LP TURBINE DESIGN POINT

STAGE 2

Input Data

Inlet Mass Flow Rate 160.20 kg/s
 Power Output 14.1173 MW
 Shaft Rotational Speed 4450.0 rev/min
 Inlet Swirl Angle 21.00 deg
 Turbine T/T Isentropic Efficiency .90
 Outlet Swirl Angle 21.00 deg
 Inlet Fuel/Air Ratio .0207
 Work Done Factor 1.00

	P(inf) (kPa)	T(inf) (K)
Tip	779.0	1135.3
Mean	779.0	1134.9
Root	779.0	1134.5

Blade Stress Input Information

Metal Density 8.00	Mg/cu.m	Taper Factor 1.000	Shroud Equivalent Length .60 mm	
Plane Number	1	2	3	4
Axial Distance	125.8	172.2	195.4	259.2 (mm)
Tip Radius	567.65	573.36	574.79	582.63 (mm)
Mean Radius	514.28	514.28	514.28	514.28 (mm)
Root Radius	460.90	455.20	453.77	445.93 (mm)

N.G.V. Aerofoil Data

True Chord 52.23 (mm)
 Axial Chord 46.43 (mm)
 Max Thickness 10.44 (mm)
 T.E. Thickness 1.10 (mm)

Rotor Aerofoil Data

True Chord 68.70 (mm)
 Axial Chord 63.76 (mm)
 Max Thickness 13.74 (mm)
 T.E. Thickness 1.54 (mm)
 Number of Seals 0
 Tip Seal Clearance .25 (mm)

Ainley, Mathieson, Dunham & Cane Loss Coefficients

N.G.V.

Profile Loss (YF) .036
 Secondary Loss (YS) .047
 Tip (YK) .000
 Total + R.E. .083
 Reynolds Number 8.33E+05

Rotor

Profile Loss (YF) .035
 Secondary Loss (YS) .045
 Tip (YK) .014
 Total + R.E. .094
 Reynolds Number 1.05E+06

Number of N.G.V.'s 59 Space/Axial Chord Ratio 1.18 Axial Chord Aspect Ratio 2.55
 Space/Chord Ratio 1.05 Aspect Ratio 2.26

Number of Rotor's 42 Space/Axial Chord Ratio 1.21 Axial Chord Aspect Ratio 2.02
 Space/Chord Ratio 1.12 Aspect Ratio 1.88

Nozzle Guide Vane Outlet Triangles,

	V(0) (m/s)	V(1) (m/s)	VW(0) (m/s)	ALFA(0) (deg)	ALFA(1) (deg)	MA(0) (-)	MA(1) (-)	VA(0,1) (-)	(m/s)
Tip	330.4	205.9	258.7	51.52	-3.07	.515	.321	205.6	
Mean	354.2	211.3	288.4	54.51	13.33	.554	.330	205.6	
Root	385.3	236.2	325.8	57.75	29.47	.605	.371	205.6	

Rotor Outlet Triangles

	V(3) (m/s)	V(2) (m/s)	VW(3) (m/s)	ALFA(3) (deg)	ALFA(2) (deg)	MA(3) (-)	MA(2) (-)	VA(2,3) (-)	(m/s)
Tip	219.2	397.6	70.0	18.72	58.69	.348	.634	206.6	
Mean	221.3	380.1	79.3	21.00	57.06	.353	.606	206.6	
Root	226.0	365.2	91.5	23.88	55.54	.360	.582	206.6	

Stagnation Temperatures & Pressures

	T(0) (K)	T(1) (K)	T(2) (K)	T(3) (K)	P(0) (kPa)	P(1) (kPa)	P(2) (kPa)	P(3) (kPa)
Tip	1134.5	1106.7	1106.7	1060.7	769.1	693.3	676.9	568.5
Mean	1134.9	1101.2	1101.2	1061.4	767.7	677.0	663.9	569.3
Root	1135.3	1096.7	1096.7	1062.4	765.8	662.8	650.5	570.0

	Velocity Ratios (0)/(inf) (2)/(1) (-)	Deflections (inf,0)1) (1,2) (deg) (deg)	Blade Stress Tensile Bending (MPa) (MPa)	Outlet Static Pressure (kPa)
Tip	1.50	1.93	70.7 55.6	525.3
Mean	1.61	1.80	75.5 70.4	525.0
Root	1.75	1.55	80.9 85.0	523.8

	Stage Parameters u (m/s)	dVw (m/s)	dH/u.sq (-)	Reaction Zweifel (%)	Lift Coef Stator Rotor (-) (-)
Tip	269.7	328.7	1.22	65.3	1.49 1.05
Mean	239.7	367.7	1.53	56.6	1.32 1.16
Root	209.6	417.2	1.99	44.4	1.13 1.26

ETA(TT) = 89.2 % CP dT/T = 148.1 N/ RT = .0
 ETA(TS) = 71.5 % Vs/u = .86

Warning Rotor Space/True Chord Ratio outside Ainl ey's Correlation

Free Vortex Turbine Design Program

LP TURBINE DESIGN POINT

STAGE

3

Input Data

Inlet Mass Flow Rate 160.20 kg/s
 Power Output 13.8302 MW
 Shaft Rotational Speed 4450.0 rev/min
 Inlet Swirl Angle 21.00 deg
 Turbine T/T Isentropic Efficiency .90
 Outlet Swirl Angle 21.00 deg
 Inlet Fuel/Air Ratio .0207
 Work Done Factor 1.00

	P(inf) (kPa)	T(inf) (K)
Tip	569.3	1062.4
Mean	569.3	1061.4
Root	569.3	1060.7

Blade Stress Input Information

Metal Density 8.00 Mg/cu.m Taper Factor 1.000 Shroud Equivalent Length .00 mm
 Plane Number 1 2 3 4
 Axial Distance 286.8 344.2 372.9 458.4 (mm)

Tip Radius	582.63	590.10	591.96	603.08 (mm)
Mean Radius	514.28	514.28	514.28	(mm)
Root Radius	445.93	438.46	436.60	425.48 (mm)

N.G.V. Aerofoil Data

True Chord	64.40 (mm)
Axial Chord	57.45 (mm)
Max Thickness	12.88 (mm)
T.E. Thickness	1.35 (mm)

Rotor Aerofoil Data

True Chord	92.76 (mm)
Axial Chord	85.47 (mm)
Max Thickness	18.55 (mm)
T.E. Thickness	1.90 (mm)
Number of Seals	0
Tip Seal Clearance	.25 (mm)

Ainley, Mathieson, Dunham & Cane Loss Coefficients

N.G.V.	
Profile Loss (YP)	.035
Secondary Loss (YS)	.045
Tip (YK)	.000
Total + R.E.	.080
Reynolds Number	8.38E+05

Rotor	
Profile Loss (YP)	.029
Secondary Loss (YS)	.045
Tip (YK)	.011
Total + R.E.	.085
Reynolds Number	1.14E+06

Number of N.G.V.'s	48	Space/Axial Chord Ratio	1.17	Axial Chord Aspect Ratio	2.64
		Space/Chord Ratio	1.05	Aspect Ratio	2.35
Number of Rotor's	34	Space/Axial Chord Ratio	1.11	Axial Chord Aspect Ratio	1.95
		Space/Chord Ratio	1.02	Aspect Ratio	1.79

Nozzle Guide Vane Outlet Triangles,

	V(0) (m/s)	V(1) (m/s)	VW(0) (m/s)	ALFA(0) (deg)	ALFA(1) (deg)	MA(0) (-)	MA(1) (-)	VA(0,1) (-)	(m/s)
Tip	320.1	208.8	244.9	49.93	-9.23	.515	.336	206.1	
Mean	348.5	210.2	281.1	53.75	11.36	.563	.340	206.1	
Root	388.8	243.0	329.7	57.99	32.01	.632	.395	206.1	

Rotor Outlet Triangles

	V(3) (m/s)	V(2) (m/s)	VW(3) (m/s)	ALFA(3) (deg)	ALFA(2) (deg)	MA(3) (-)	MA(2) (-)	VA(2,3) (-)	(m/s)
Tip	217.0	402.8	67.5	18.13	59.20	.358	.665	206.2	
Mean	220.9	379.7	79.2	21.60	57.10	.365	.627	206.2	
Root	227.4	361.2	95.7	24.89	55.18	.375	.596	206.2	

Stagnation Temperatures & Pressures

	T(0) (K)	T(1) (K)	T(2) (K)	T(3) (K)	P(0) (kPa)	P(1) (kPa)	P(2) (kPa)	P(3) (kPa)
Tip	1060.7	1035.8	1035.8	987.2	562.2	509.8	499.4	409.6
Mean	1061.4	1028.8	1028.8	988.6	561.0	493.3	484.1	410.6
Root	1062.4	1023.5	1023.5	990.3	559.2	479.5	471.3	411.3

	Velocity Ratios		Deflections		Blade Stress		Outlet Static
	(0)/(inf)	(2)/(1)	(inf,0)	(1,2)	Tensile	Bending	Pressure
	(-)	(-)	(deg)	(deg)	(MPa)	(MPa)	(kPa)
Tip	1.45	1.93	68.6	50.0	.0	.0	376.6
Mean	1.58	1.81	74.8	68.5	.0	13.4	376.4
Root	1.76	1.49	81.9	87.2	.0	45.7	375.2

Stage Parameters

	u (m/s)	dVw (m/s)	dH/u.sq (-)	Reaction (%)	Zweifel (-)	Lift Coef (-)
Tip	278.4	312.5	1.12	68.2	1.55	.92
Mean	239.7	360.2	1.50	57.9	1.32	1.04
Root	200.9	425.4	2.12	41.8	1.09	1.15

ETA(TT) = 89.8 %
ETA(TS) = 71.7 %

CF dT/T = 153.0 N/ RT = .0
Va/u = .86

Free Vortex Turbine Design Program

LP TURBINE DESIGN POINT

STAGE

4

Input Data

Inlet Mass Flow Rate	160.20 kg/s
Power Output	13.4392 MW
Shaft Rotational Speed	4450.0 rev/min
Inlet Swirl Angle	21.00 deg
Turbine T/T Isentropic Efficiency	.90
Outlet Swirl Angle	19.00 deg
Inlet Fuel/Air Ratio	.0207
Work Done Factor	1.00

	P (inf) (kPa)	T (inf) (K)
Tip	410.6	990.3
Mean	410.6	988.6
Root	410.6	987.2

Blade Stress Input Information

Metal Density	8.00	Mg/cu.m	Taper Factor	1.000	Shroud Equivalent Length	.00 mm
---------------	------	---------	--------------	-------	--------------------------	--------

Plane Number	1	2	3	4	
Axial Distance	487.0	546.7	576.6	662.9	(mm)
Tip Radius	603.08	613.48	616.08	631.13	(mm)
Mean Radius	514.28	514.28	514.28	514.28	(mm)
Root Radius	425.48	415.08	412.47	397.43	(mm)

N.G.V. Aerofoil Data

True Chord	66.91 (mm)
Axial Chord	59.69 (mm)
Max Thickness	13.38 (mm)
T.E. Thickness	1.44 (mm)

Rotor Aerofoil Data

True Chord	93.62 (mm)
Axial Chord	86.31 (mm)
Max Thickness	18.73 (mm)
T.E. Thickness	1.90 (mm)
Number of Seals	0
Tip Seal Clearance	.25 (mm)

Ainley, Mathieson, Dunham & Cane Loss Coefficients

N.G.V.			Rotor		
Profile Loss (YP)	.036		Profile Loss (YP)	.029	
Secondary Loss (YS)	.036		Secondary Loss (YS)	.034	
Tip (YK)	.000		Tip (YK)	.008	
Total + R.E.	.072		Total + R.E.	.071	
Reynolds Number	6.94E+05		Reynolds Number	9.06E+05	

Number of N.G.V.'s	45	Space/Axial Chord Ratio	1.20	Axial Chord Aspect Ratio	3.32
		Space/Chord Ratio	1.07	Aspect Ratio	2.97
Number of Rotor's	34	Space/Axial Chord Ratio	1.10	Axial Chord Aspect Ratio	2.53
		Space/Chord Ratio	1.02	Aspect Ratio	2.34

Nozzle Guide Vane Outlet Triangles

	V(0) (m/s)	V(1) (m/s)	VW(0) (m/s)	ALFA(0) (deg)	ALFA(1) (deg)	MA(0) (-)	MA(1) (-)	VA(0,1) (m/s)
Tip	311.8	212.8	234.6	48.80	-15.26	.519	.355	205.3
Mean	347.1	269.2	279.8	53.73	11.07	.581	.350	205.3
Root	403.0	259.1	346.7	59.37	37.58	.680	.438	205.3

Rotor Outlet Triangles

	V(3) (m/s)	V(2) (m/s)	VW(3) (m/s)	ALFA(3) (deg)	ALFA(2) (deg)	MA(3) (-)	MA(2) (-)	VA(2,3) (m/s)
Tip	211.7	403.1	57.2	15.67	59.62	.363	.690	203.9
Mean	215.6	370.9	70.2	19.00	56.66	.369	.635	203.9
Root	223.2	346.0	90.8	24.02	53.90	.382	.592	203.9

Stagnation Temperatures & Pressures

	T(0) (K)	T(1) (K)	T(2) (K)	T(3) (K)	P(0) (kPa)	P(1) (kPa)	P(2) (kPa)	P(3) (kPa)
Tip	987.2	965.0	965.0	914.6	405.9	370.0	363.2	292.1
Mean	988.6	955.7	955.7	916.7	404.9	352.9	347.3	293.2
Root	990.3	949.5	949.5	919.6	403.1	339.8	335.0	294.1

	Velocity Ratios (0)/(inf) (2)/(1)	Deflections (inf,0/1) (1,2) (deg) (deg)	Blade Stress Tensile Bending (MPa) (MPa)	Outlet Static Pressure (kPa)
Tip	1.41 1.89	65.9 44.4	.0 .0	267.9
Mean	1.57 1.77	74.7 67.7	.0 16.8	268.1
Root	1.83 1.34	84.3 91.5	.0 54.4	267.3

	Stage Parameters					
	u	dVw	dH/u.sq	Reaction	Zweifel	Lift Coef
	(m/s)	(m/s)	(-)	(%)	Stator	Rotor
					(-)	(-)
Tip	290.6	291.8	1.00	69.1	1.64	.87
Mean	239.7	350.0	1.46	55.9	1.35	1.03
Root	188.7	437.6	2.32	31.8	1.04	1.19

ETA(IT) = 91.0 % CP dT/T = 157.2 N/ RT = .0
 ETA(TS) = 72.8 % Va/u = .85

OVERALL PRESSURE RATIO..... 3.595
 TURBINE ISENTROPIC EFFICIENCY..... 89.955
 TURBINE POLYTROPIC EFFICIENCY..... 88.409

Figure F.16 TURBOFF Input File Data for HP Turbine.

HP TURBINE DESIGN POINT	STAGE 1		
2.767000E-02			
1800.000000			
2827000.000000			
152.730000			
-10000.000000			
4			
-10000.000000			
-9000.000000			
-8000.000000			
-7000.000000			
40.589160	42.033610	42.394720	44.533410
35.515520	35.515520	35.515520	35.515520
0.000000E+00			
4.839778E-02			
5.614448E-02			
2.230997E-02			
1.871483E-01			
1.725639E-01			
9.370292E-04			
5.795863E-02			
32.979160			
6.265164E-02			
6.841183E-02			
2.639994E-02			
2.280394E-01			
1.828982E-01			
1.282251E-03			
2.500000E-04			
7.948540E-02			
4.700000E-01			
0			
0			

Figure F.17 TURBOFF Input Data for LP turbine.

LP TURBINE DESIGN POINT	STAGE			1
2.070000E-02				
1210.760000				
1054000.000000				
160.200000				
-4450.000000				
4				
-4450.000000				
-4005.000000				
-3560.000000				
-3115.000000				
55.597780	55.096320	55.220960	56.765480	
47.258110	46.759560	46.634930	46.090400	
0.000000E+00				
4.524139E-02				
5.547364E-02				
3.001772E-02				
1.849121E-01				
1.630241E-01				
9.791850E-04				
8.838214E-02				
17.740670				
4.632229E-02				
4.766926E-02				
2.515699E-02				
1.588975E-01				
1.943941E-01				
8.975862E-04				
2.500000E-04				
1.013055E-01				
4.700000E-01				
0				
1				
56.765480	57.936200	57.478890	58.262630	
46.090390	45.519670	45.376990	44.593240	
21.000000				
5.222671E-02				
6.160917E-02				
3.456471E-02				
2.053639E-01				
1.695339E-01				
1.095359E-03				
1.124581E-01				
13.334900				
6.869888E-02				
8.289967E-02				
4.382832E-02				
2.763322E-01				
1.657519E-01				
1.538719E-03				
2.500000E-04				
1.288564E-01				
4.700000E-01				
0				
1				
58.262630	59.009620	59.196340	60.307660	
44.593250	43.846280	43.659540	42.548230	
21.000000				
6.440225E-02				
7.547072E-02				
4.305970E-02				

2.515691E-01			
1.706505E-01			
1.346380E-03			
1.441636E-01			
11.360630			
9.275765E-02			
1.031478E-01			
5.448456E-02			
3.438259E-01			
1.798655E-01			
1.900771E-03			
2.500000E-04			
1.664812E-01			
4.700000E-01			
0			
1			
60.307660	61.348280	61.608430	63.113040
42.548230	41.507620	41.247460	39.742840
21.000010			
6.691496E-02			
8.049464E-02			
4.594707E-02			
2.683155E-01			
1.662007E-01			
1.436138E-03			
1.880004E-01			
11.074970			
9.362028E-02			
1.030876E-01			
5.504009E-02			
3.436252E-01			
1.816521E-01			
1.900771E-03			
2.500000E-04			
2.186558E-01			
4.700000E-01			
0			
0			

GAS OUTLET ANGLE (degree)..... -67.4591
 OUTLET MACH NUMBER3417
 INLET TOTAL PRESSURE (N/m²).....2552065.00
 INLET TOTAL TEMPERATURE (K)..... 1759.989
 OUTLET TOTAL PRESSURE (N/m²).....2302539.00
 OUTLET TOTAL TEMPERATURE (K)..... 1741.499
 TOTAL LOSS COEFFICIENT73533
 INCIDENCE ANGLE (degree)..... -66.64675

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .26080
 BLADE LOADING COEFFICIENT (DH/U)..... .40669
 GAMMA (mean) (SI)..... 1.33001
 Cp (mean) (SI)..... 1156.682
 PRESSURE RATIO8145
 ISENTROPIC EFFICIENCY (%) 65.47046

 SPEED= -10000.00 (rpm) NDMF=.001986493 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -66.9467
 OUTLET MACH NUMBER5480
 OUTLET TOTAL PRESSURE (N/m²).....2775507.00
 OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10899
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -67.4591
 OUTLET MACH NUMBER4617
 INLET TOTAL PRESSURE (N/m²).....2354875.00
 INLET TOTAL TEMPERATURE (K)..... 1728.176
 OUTLET TOTAL PRESSURE (N/m²).....2003241.00
 OUTLET TOTAL TEMPERATURE (K)..... 1679.816
 TOTAL LOSS COEFFICIENT35653
 INCIDENCE ANGLE (degree)..... -32.97579

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .34654
 BLADE LOADING COEFFICIENT (DH/U)..... .83454
 GAMMA (mean) (SI)..... 1.33052
 Cp (mean) (SI)..... 1155.346
 PRESSURE RATIO7086
 ISENTROPIC EFFICIENCY (%) 81.41821

 SPEED= -10000.00 (rpm) NDMF=(CHOKING POINT) .002292107 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -67.2820
 OUTLET MACH NUMBER7627
 OUTLET TOTAL PRESSURE (N/m²).....2738056.00
 OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10986
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -68.6025
 OUTLET MACH NUMBER 1.0000
 INLET TOTAL PRESSURE (N/m²).....2051246.00
 INLET TOTAL TEMPERATURE (K)..... 1675.699
 OUTLET TOTAL PRESSURE (N/m²).....1231649.00
 OUTLET TOTAL TEMPERATURE (K)..... 1502.878
 TOTAL LOSS COEFFICIENT16892
 INCIDENCE ANGLE (degree)..... -39486

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .66310
 BLADE LOADING COEFFICIENT (DH/U)..... 2.05915
 GAMMA (mean) (SI)..... 1.33139
 Cp (mean) (SI)..... 1153.089
 PRESSURE RATIO4357
 ISENTROPIC EFFICIENCY (%) 88.35586

SPEED= -10000.00 (rpm) NDMF=.002292107 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -67.2820
 OUTLET MACH NUMBER7627
 OUTLET TOTAL PRESSURE (N/m²).....2738056.00
 OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10386
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -67.4645
 OUTLET MACH NUMBER 1.2000
 INLET TOTAL PRESSURE (N/m²).....2051246.00
 INLET TOTAL TEMPERATURE (K)..... 1675.699
 OUTLET TOTAL PRESSURE (N/m²).....1091914.00
 OUTLET TOTAL TEMPERATURE (K)..... 1464.936
 TOTAL LOSS COEFFICIENT16892
 INCIDENCE ANGLE (degree)..... -39486

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .81080
 BLADE LOADING COEFFICIENT (DH/U)..... 2.32211
 GAMMA (mean) (SI)..... 1.33139
 Cp (mean) (SI)..... 1153.089
 PRESSURE RATIO3862
 ISENTROPIC EFFICIENCY (%) 88.29103

SPEED= -10000.00 (rpm) NDMF=.002292107 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -67.2820
 OUTLET MACH NUMBER7627
 OUTLET TOTAL PRESSURE (N/m²).....2738056.00

OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10986
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GRS OUTLET ANGLE (degree)..... -46.3968
 OUTLET MACH NUMBER 1.3000
 INLET TOTAL PRESSURE (N/m²)..... 2051246.00
 INLET TOTAL TEMPERATURE (K)..... 1675.699
 OUTLET TOTAL PRESSURE (N/m²)..... 1036088.00
 OUTLET TOTAL TEMPERATURE (K)..... 1448.748
 TOTAL LOSS COEFFICIENT16892
 INCIDENCE ANGLE (degree)..... -.39486

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .90262
 BLADE LOADING COEFFICIENT (DH/U)..... 2.43430
 GAMMA (mean) (SI)..... 1.33139
 Cp (mean) (SI)..... 1153.089
 PRESSURE RATIO3665
 ISENTROPIC EFFICIENCY (%) 88.26888

OUTLET DATA AT REFERENCE RADIUS

CALCULATION OF (N.D.M.F.) RANGE AT -9000.00 (rpm)

NDMF=.001375264 (SI) PR=.915 EFFIC= 52.413 %
 NDMF=.001451668 (SI) PR=.888 EFFIC= 59.196 %
 NDMF=.001528071 (SI) PR=.860 EFFIC= 63.542 %
 NDMF=.001604475 (SI) PR=.832 EFFIC= 66.613 %
 NDMF=.001680878 (SI) PR=.813 EFFIC= 71.666 %
 NDMF=.001757282 (SI) PR=.793 EFFIC= 76.114 %
 NDMF=.001833686 (SI) PR=.773 EFFIC= 79.850 %
 NDMF=.001910089 (SI) PR=.750 EFFIC= 82.872 %
 NDMF=.001986493 (SI) PR=.724 EFFIC= 85.230 %
 NDMF=.002062896 (SI) PR=.694 EFFIC= 86.716 %
 NDMF=.002139300 (SI) PR=.657 EFFIC= 87.632 %
 NDMF=.002215703 (SI) PR=.612 EFFIC= 88.149 %
 NDMF=.002292107 (SI) PR=.528 EFFIC= 86.768 %
 NDMF=.002368511 (SI) (CHOKING POINT) PR=.375 EFFIC= 80.757 %
 NDMF=.002368511 (SI) PR=.353 EFFIC= 80.340 %
 NDMF=.002368511 (SI) PR=.334 EFFIC= 79.967 %
 NDMF=.002368511 (SI) PR=.319 EFFIC= 79.609 %
 NDMF=.002368511 (SI) PR=.307 EFFIC= 79.230 %

SPEED= -9000.00 (rpm) NDMF=.001375264 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -66.8765
 OUTLET MACH NUMBER3242
 OUTLET TOTAL PRESSURE (N/m²).....2805954.00
 OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10881
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -67.4591
 OUTLET MACH NUMBER2463
 INLET TOTAL PRESSURE (N/m²).....2685444.00
 INLET TOTAL TEMPERATURE (K)..... 1780.528
 OUTLET TOTAL PRESSURE (N/m²).....2587862.00
 OUTLET TOTAL TEMPERATURE (K)..... 1779.550
 TOTAL LOSS COEFFICIENT73533
 INCIDENCE ANGLE (degree)..... -82.39256

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .21101
 BLADE LOADING COEFFICIENT (DH/U)..... .17865
 GAMMA (mean) (SI)..... 1.32968
 Cp (mean) (SI)..... 1157.531
 PRESSURE RATIO9154
 ISENTROPIC EFFICIENCY (%) 52.41513

SPEED= -9000.00 (rpm) NDMF=.001680878 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -66.8765
 OUTLET MACH NUMBER4228
 OUTLET TOTAL PRESSURE (N/m²).....2794431.00
 OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10881
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -67.4591
 OUTLET MACH NUMBER3386
 INLET TOTAL PRESSURE (N/m²).....2537683.00
 INLET TOTAL TEMPERATURE (K)..... 1757.526
 OUTLET TOTAL PRESSURE (N/m²).....2297085.00
 OUTLET TOTAL TEMPERATURE (K)..... 1735.104
 TOTAL LOSS COEFFICIENT59819
 INCIDENCE ANGLE (degree)..... -53.04691

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .28698
 BLADE LOADING COEFFICIENT (DH/U)..... .55624
 GAMMA (mean) (SI)..... 1.33005
 Cp (mean) (SI)..... 1156.579
 PRESSURE RATIO8126
 ISENTROPIC EFFICIENCY (%) 71.72333

SPEED= -9000.00 (rpm) NDMF=.001986493 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -66.9467
 OUTLET MACH NUMBER5480
 OUTLET TOTAL PRESSURE (N/m²).....2775507.00
 OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10899
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -67.4591
 OUTLET MACH NUMBER4495
 INLET TOTAL PRESSURE (N/m²).....2358830.00
 INLET TOTAL TEMPERATURE (K)..... 1728.895
 OUTLET TOTAL PRESSURE (N/m²).....2047353.00
 OUTLET TOTAL TEMPERATURE (K)..... 1681.632
 TOTAL LOSS COEFFICIENT24422
 INCIDENCE ANGLE (degree)..... -19.75298

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .37527
 BLADE LOADING COEFFICIENT (DH/U)..... 1.01476
 GAMMA (mean) (SI)..... 1.33051
 Cp (mean) (SI)..... 1155.377
 PRESSURE RATIO7242
 ISENTROPIC EFFICIENCY (%) 85.37534

SPEED= -9000.00 (rpm) NDMF=(CHOCKING POINT) .002368511 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -67.4816
 OUTLET MACH NUMBER 1.0000
 OUTLET TOTAL PRESSURE (N/m²).....2692380.00
 OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10881
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -62.9606
 OUTLET MACH NUMBER 1.8000
 INLET TOTAL PRESSURE (N/m²).....1819107.00
 INLET TOTAL TEMPERATURE (K)..... 1633.362
 OUTLET TOTAL PRESSURE (N/m²).....1060542.00
 OUTLET TOTAL TEMPERATURE (K)..... 1483.021
 TOTAL LOSS COEFFICIENT36393
 INCIDENCE ANGLE (degree)..... 16.34759

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .90661
 BLADE LOADING COEFFICIENT (DH/U)..... 2.70766
 GAMMA (mean) (SI)..... 1.33212
 Cp (mean) (SI)..... 1151.217
 PRESSURE RATIO3751
 ISENTROPIC EFFICIENCY (%) 81.20642

SPEED= -9000.00 (rpm) NDMF=.002368511 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -67.4816
 OUTLET MACH NUMBER 1.0000
 OUTLET TOTAL PRESSURE (N/m²)..... 2692380.00
 OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10881
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -60.9186
 OUTLET MACH NUMBER 1.2000
 INLET TOTAL PRESSURE (N/m²)..... 1819107.00
 INLET TOTAL TEMPERATURE (K)..... 1633.362
 OUTLET TOTAL PRESSURE (N/m²)..... 943983.90
 OUTLET TOTAL TEMPERATURE (K)..... 1453.322
 TOTAL LOSS COEFFICIENT36393
 INCIDENCE ANGLE (degree)..... 16.34759

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... 1.12814
 BLADE LOADING COEFFICIENT (DH/U)..... 2.96135
 GAMMA (mean) (SI)..... 1.33212
 Cp (mean) (SI)..... 1151.217
 PRESSURE RATIO3339
 ISENTROPIC EFFICIENCY (%) 80.49780

 SPEED= -9000.00 (rpm) NDMF=.002368511 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -67.4816
 OUTLET MACH NUMBER 1.0000
 OUTLET TOTAL PRESSURE (N/m²)..... 2692380.00
 OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10881
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -59.2247
 OUTLET MACH NUMBER 1.3000
 INLET TOTAL PRESSURE (N/m²)..... 1819107.00
 INLET TOTAL TEMPERATURE (K)..... 1633.362
 OUTLET TOTAL PRESSURE (N/m²)..... 900710.60
 OUTLET TOTAL TEMPERATURE (K)..... 1441.965
 TOTAL LOSS COEFFICIENT36393
 INCIDENCE ANGLE (degree)..... 16.34759

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... 1.26541
 BLADE LOADING COEFFICIENT (DH/U)..... 3.05836
 GAMMA (mean) (SI)..... 1.33212
 Cp (mean) (SI)..... 1151.217
 PRESSURE RATIO3196
 ISENTROPIC EFFICIENCY (%) 80.17012

OUTLET DATA AT REFERENCE RADIUS

CALCULATION OF (N.D.M.F.) RANGE AT -8000.00 (rpm)

```

NDMF=.001375264 (SI) PR=.895 EFFIC= 61.645 %
NDMF=.001451668 (SI) PR=.871 EFFIC= 64.727 %
NDMF=.001528071 (SI) PR=.854 EFFIC= 70.036 %
NDMF=.001604475 (SI) PR=.838 EFFIC= 74.999 %
NDMF=.001680878 (SI) PR=.822 EFFIC= 79.097 %
NDMF=.001757282 (SI) PR=.804 EFFIC= 82.285 %
NDMF=.001833686 (SI) PR=.784 EFFIC= 84.628 %
NDMF=.001910088 (SI) PR=.762 EFFIC= 86.145 %
NDMF=.001986493 (SI) PR=.736 EFFIC= 87.067 %
NDMF=.002062896 (SI) PR=.706 EFFIC= 87.570 %
NDMF=.002139300 (SI) PR=.670 EFFIC= 87.414 %
NDMF=.002215703 (SI) PR=.623 EFFIC= 86.539 %
NDMF=.002292107 (SI) (CHOCKING POINT) PR=.460 EFFIC= 82.182%

NDMF=.002292107 (SI) PR=.636 EFFIC= 81.841 %
NDMF=.002292107 (SI) PR=.614 EFFIC= 81.561 %
NDMF=.002292107 (SI) PR=.596 EFFIC= 81.325 %
NDMF=.002292107 (SI) PR=.580 EFFIC= 81.110 %

```

 SPEED= -8000.00 (rpm) NDMF=.001375264 (SI)

SINGLE STAGE

NOZZLE:

```

GAS OUTLET ANGLE (degree)..... -66.8765
OUTLET MACH NUMBER ..... .3243
OUTLET TOTAL PRESSURE (N/m²).....2805954.00
OUTLET TOTAL TEMPERATURE (K)..... 1800.000
TOTAL LOSS COEFFICIENT ..... .10881
INCIDENCE ANGLE (degree)..... .00000

```

ROTOR:

```

GAS OUTLET ANGLE (degree)..... -67.4591
OUTLET MACH NUMBER ..... .2486
INLET TOTAL PRESSURE (N/m²).....2683706.00
INLET TOTAL TEMPERATURE (K)..... 1776.946
OUTLET TOTAL PRESSURE (N/m²).....2530340.00
OUTLET TOTAL TEMPERATURE (K)..... 1769.904
TOTAL LOSS COEFFICIENT ..... .73533
INCIDENCE ANGLE (degree)..... -70.83978

```

STAGE OVERALL CONDITIONS:

```

FLOW COEFFICIENT (Va/U)..... .23927
BLADE LOADING COEFFICIENT (DH/U)..... .32712
GAMMA (mean) (SI)..... 1.32974
Cp (mean) (SI)..... 1157.384
PRESSURE RATIO ..... .8951
ISENTRIC EFFICIENCY (%) ..... 61.66081

```

SPEED= -8000.00 (rpm) NDMF=.001600878 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -66.8765
OUTLET MACH NUMBER4228
OUTLET TOTAL PRESSURE (N/m²)..... 2794431.00
OUTLET TOTAL TEMPERATURE (K)..... 1800.000
TOTAL LOSS COEFFICIENT10881
INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -67.4591
OUTLET MACH NUMBER3319
INLET TOTAL PRESSURE (N/m²)..... 2531707.00
INLET TOTAL TEMPERATURE (K)..... 1756.500
OUTLET TOTAL PRESSURE (N/m²)..... 2323469.00
OUTLET TOTAL TEMPERATURE (K)..... 1732.297
TOTAL LOSS COEFFICIENT39458
INCIDENCE ANGLE (degree)..... -36.67073

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .31646
BLADE LOADING COEFFICIENT (DH/U)..... .73532
GAMMA (mean) (SI)..... 1.33006
Cp (mean) (SI)..... 1156.536
PRESSURE RATIO8219
ISENTROPIC EFFICIENCY (%) 79.16553

SPEED= -8000.00 (rpm) NDMF=.001986493 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -66.9467
OUTLET MACH NUMBER5480
OUTLET TOTAL PRESSURE (N/m²)..... 2775507.00
OUTLET TOTAL TEMPERATURE (K)..... 1800.000
TOTAL LOSS COEFFICIENT10899
INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -67.4591
OUTLET MACH NUMBER4415
INLET TOTAL PRESSURE (N/m²)..... 2370719.00
INLET TOTAL TEMPERATURE (K)..... 1731.050
OUTLET TOTAL PRESSURE (N/m²)..... 2080107.00
OUTLET TOTAL TEMPERATURE (K)..... 1664.817
TOTAL LOSS COEFFICIENT18667
INCIDENCE ANGLE (degree)..... -7.80558

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .41523
BLADE LOADING COEFFICIENT (DH/U)..... 1.24985
GAMMA (mean) (SI)..... 1.33047
Cp (mean) (SI)..... 1155.468

PRESSURE RATIO7358
 ISENTROPIC EFFICIENCY (%) 87.21259

SPEED= -8000.00 (rpm) NDMF=(CHOCKING POINT) .002292107 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree) -67.2820
 OUTLET MACH NUMBER7627
 OUTLET TOTAL PRESSURE (N/m²) 2738056.00
 OUTLET TOTAL TEMPERATURE (K) 1800.000
 TOTAL LOSS COEFFICIENT10986
 INCIDENCE ANGLE (degree)00000

ROTOR:

GAS OUTLET ANGLE (degree) -68.3411
 OUTLET MACH NUMBER 1.0000
 INLET TOTAL PRESSURE (N/m²) 2118100.00
 INLET TOTAL TEMPERATURE (K) 1689.068
 OUTLET TOTAL PRESSURE (N/m²) 1301446.00
 OUTLET TOTAL TEMPERATURE (K) 1538.955
 TOTAL LOSS COEFFICIENT26380
 INCIDENCE ANGLE (degree) 11.70414

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)84180
 BLADE LOADING COEFFICIENT (DN/U) 2.82820
 GAMMA (mean) (SI) 1.33114
 Cp (mean) (SI) 1153.671
 PRESSURE RATIO4604
 ISENTROPIC EFFICIENCY (%) 82.63245

SPEED= -8000.00 (rpm) NDMF=.002292107 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree) -67.2820
 OUTLET MACH NUMBER7627
 OUTLET TOTAL PRESSURE (N/m²) 2738056.00
 OUTLET TOTAL TEMPERATURE (K) 1800.000
 TOTAL LOSS COEFFICIENT10986
 INCIDENCE ANGLE (degree)00000

ROTOR:

GAS OUTLET ANGLE (degree) -66.9703
 OUTLET MACH NUMBER 1.2000
 INLET TOTAL PRESSURE (N/m²) 2118100.00
 INLET TOTAL TEMPERATURE (K) 1689.068
 OUTLET TOTAL PRESSURE (N/m²) 1170683.00
 OUTLET TOTAL TEMPERATURE (K) 1508.973
 TOTAL LOSS COEFFICIENT26380
 INCIDENCE ANGLE (degree) 11.70414

STAGE OVERALL CONDITIONS:

```

FLOW COEFFICIENT (Va/U)..... 1.03861
BLADE LOADING COEFFICIENT (DH/U)..... 3.15303
GAMMA (mean) (SI)..... 1.33116
Cp (mean) (SI)..... 1153.671
PRESSURE RATIO ..... .4141
ISENTROPIC EFFICIENCY (%) ..... 82.09614

```

SPEED= -8000.00 (rpm) NDMF=.002292107 (SI)

SINGLE STAGE

NOZZLE:

```

GAS OUTLET ANGLE (degree)..... -67.2820
OUTLET MACH NUMBER ..... .7627
OUTLET TOTAL PRESSURE (N/m²)..... 2738056.00
OUTLET TOTAL TEMPERATURE (K)..... 1800.000
TOTAL LOSS COEFFICIENT ..... .10986
INCIDENCE ANGLE (degree)..... .00000

```

ROTOR:

```

GAS OUTLET ANGLE (degree)..... -65.7762
OUTLET MACH NUMBER ..... 1.3000
INLET TOTAL PRESSURE (N/m²)..... 2118100.00
INLET TOTAL TEMPERATURE (K)..... 1689.068
OUTLET TOTAL PRESSURE (N/m²)..... 1118417.00
OUTLET TOTAL TEMPERATURE (K)..... 1496.310
TOTAL LOSS COEFFICIENT ..... .26380
INCIDENCE ANGLE (degree)..... 11.70414

```

STAGE OVERALL CONDITIONS:

```

FLOW COEFFICIENT (Va/U)..... 1.16076
BLADE LOADING COEFFICIENT (DH/U)..... 3.29022
GAMMA (mean) (SI)..... 1.33116
Cp (mean) (SI)..... 1153.671
PRESSURE RATIO ..... .3956
ISENTROPIC EFFICIENCY (%) ..... 81.89532

```

OUTLET DATA AT REFERENCE RADIUS

CALCULATION OF (N.D.M.F.) RANGE AT -7000.00 (rpm)

```

NDMF=.001375264 (SI) PR= .888 EFFIC= 69.348 %
NDMF=.001451668 (SI) PR= .875 EFFIC= 74.502 %
NDMF=.001528071 (SI) PR= .861 EFFIC= 78.785 %
NDMF=.001604475 (SI) PR= .847 EFFIC= 82.055 %
NDMF=.001680878 (SI) PR= .831 EFFIC= 84.412 %
NDMF=.001757282 (SI) PR= .813 EFFIC= 85.908 %
NDMF=.001833686 (SI) PR= .794 EFFIC= 86.782 %
NDMF=.001910089 (SI) PR= .772 EFFIC= 87.234 %
NDMF=.001986493 (SI) PR= .747 EFFIC= 87.071 %
NDMF=.002062896 (SI) PR= .716 EFFIC= 86.276 %
NDMF=.002139300 (SI) PR= .678 EFFIC= 84.643 %
NDMF=.002215703 (SI) PR= .629 EFFIC= 82.733 %
NDMF=.002292107 (SI) (CHOCKING POINT) PR= .469 EFFIC= 76.7354

```


NDMF=.002292107 (SI) PR= .445 EFFIC= 76.198 %
 NDMF=.002292107 (SI) PR= .423 EFFIC= 75.759 %
 NDMF=.002292107 (SI) PR= .405 EFFIC= 75.386 %
 NDMF=.002292107 (SI) PR= .390 EFFIC= 75.046 %

SPEED= -7000.00 (rpm) NDMF=.001375264 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -66.8765
 OUTLET MACH NUMBER3242
 OUTLET TOTAL PRESSURE (N/m²).....2805954.00
 OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10881
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -67.4591
 OUTLET MACH NUMBER2450
 INLET TOTAL PRESSURE (N/m²).....2650748.00
 INLET TOTAL TEMPERATURE (K)..... 1774.851
 OUTLET TOTAL PRESSURE (N/m²).....2510153.00
 OUTLET TOTAL TEMPERATURE (K)..... 1763.725
 TOTAL LOSS COEFFICIENT61410
 INCIDENCE ANGLE (degree)..... -54.15941

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .27272
 BLADE LOADING COEFFICIENT (DE/U)..... .51493
 GAMMA (mean) (SI)..... 1.32977
 Cp (mean) (SI)..... 1137.295
 PRESSURE RATIO8879
 ISENTROPIC EFFICIENCY (%) 85.37508

SPEED= -7000.00 (rpm) NDMF=.001680878 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -66.8765
 OUTLET MACH NUMBER4228
 OUTLET TOTAL PRESSURE (N/m²).....2794431.00
 OUTLET TOTAL TEMPERATURE (K)..... 1800.000
 TOTAL LOSS COEFFICIENT10881
 INCIDENCE ANGLE (degree)..... .00000

ROTOR:

GAS OUTLET ANGLE (degree)..... -67.4591
 OUTLET MACH NUMBER3245
 INLET TOTAL PRESSURE (N/m²).....2534094.00
 INLET TOTAL TEMPERATURE (K)..... 1756.910
 OUTLET TOTAL PRESSURE (N/m²).....2348920.00
 OUTLET TOTAL TEMPERATURE (K)..... 1731.868
 TOTAL LOSS COEFFICIENT24375

INCIDENCE ANGLE (degree)..... -19.68511

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .33592
 BLADE LOADING COEFFICIENT (DH/U)..... .86936
 GAMMA (mean) (SI)..... 1.33006
 Cp (mean) (SI)..... 1122.553
 PRESSURE RATIO8309
 ISENTROPIC EFFICIENCY (%) 84.48820

SPEED= -7000.00 (rpm) NDMF=.001986493 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -66.9467
 OUTLET MACH NUMBER5461
 OUTLET TOTAL PRESSURE (N/m²)..... 2775507.01
 OUTLET TOTAL TEMPERATURE (K)..... 1800.001
 TOTAL LOSS COEFFICIENT10891
 INCIDENCE ANGLE (degree)..... .00001

ROTOR:

GAS OUTLET ANGLE (degree)..... -67.4591
 OUTLET MACH NUMBER4353
 INLET TOTAL PRESSURE (N/m²)..... 2390632.01
 INLET TOTAL TEMPERATURE (K)..... 1734.641
 OUTLET TOTAL PRESSURE (N/m²)..... 2110437.01
 OUTLET TOTAL TEMPERATURE (K)..... 1690.071
 TOTAL LOSS COEFFICIENT17481
 INCIDENCE ANGLE (degree)..... 2.20361

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)..... .45911
 BLADE LOADING COEFFICIENT (DH/U)..... 1.53817
 GAMMA (mean) (SI)..... 1.33041
 Cp (mean) (SI)..... 1152.620
 PRESSURE RATIO7465
 ISENTROPIC EFFICIENCY (%) 87.21118

SPEED= -7000.00 (rpm) NDMF=(CHOKING POINT) .002292107 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree)..... -67.2821
 OUTLET MACH NUMBER7627
 OUTLET TOTAL PRESSURE (N/m²)..... 2738056.01
 OUTLET TOTAL TEMPERATURE (K)..... 1800.001
 TOTAL LOSS COEFFICIENT10961
 INCIDENCE ANGLE (degree)..... .00001

ROTOR:

GAS OUTLET ANGLE (degree)..... -67.8378
 OUTLET MACH NUMBER 1.0001
 INLET TOTAL PRESSURE (N/m²)..... 2163192.01

INLET TOTAL TEMPERATURE (K) 1697.907
 OUTLET TOTAL PRESSURE (N/m²) 1325757.00
 OUTLET TOTAL TEMPERATURE (K) 1561.747
 TOTAL LOSS COEFFICIENT36413
 INCIDENCE ANGLE (degree) 16.35325

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U)98582
 BLADE LOADING COEFFICIENT (DH/U) 3.37256
 GAMMA (mean) (SI) 1.33101
 Cp (mean) (SI) 1154.053
 PRESSURE RATIO4690
 ISENTROPIC EFFICIENCY (%) 77.11234

SPEED= -7000.00 (rpm) NDMF=.002292107 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree) -67.2820
 OUTLET MACH NUMBER7627
 OUTLET TOTAL PRESSURE (N/m²) 2738056.00
 OUTLET TOTAL TEMPERATURE (K) 1800.000
 TOTAL LOSS COEFFICIENT10986
 INCIDENCE ANGLE (degree)00000

ROTOR:

GAS OUTLET ANGLE (degree) -66.2133
 OUTLET MACH NUMBER 1.2000
 INLET TOTAL PRESSURE (N/m²) 2163192.00
 INLET TOTAL TEMPERATURE (K) 1697.907
 OUTLET TOTAL PRESSURE (N/m²) 1196668.00
 OUTLET TOTAL TEMPERATURE (K) 1535.983
 TOTAL LOSS COEFFICIENT36413
 INCIDENCE ANGLE (degree) 16.35325

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U) 1.22694
 BLADE LOADING COEFFICIENT (DH/U) 3.73725
 GAMMA (mean) (SI) 1.33101
 Cp (mean) (SI) 1154.053
 PRESSURE RATIO4233
 ISENTROPIC EFFICIENCY (%) 75.20065

SPEED= -7000.00 (rpm) NDMF=.002292107 (SI)

SINGLE STAGE

NOZZLE:

GAS OUTLET ANGLE (degree) -67.2820
 OUTLET MACH NUMBER7627
 OUTLET TOTAL PRESSURE (N/m²) 2738056.00
 OUTLET TOTAL TEMPERATURE (K) 1800.000
 TOTAL LOSS COEFFICIENT10986
 INCIDENCE ANGLE (degree)00000

ROTOR:

GAS OUTLET ANGLE (degree).....	-64.8749
OUTLET MACH NUMBER	1.3000
INLET TOTAL PRESSURE (N/m ²).....	2163192.00
INLET TOTAL TEMPERATURE (K).....	1697.907
OUTLET TOTAL PRESSURE (N/m ²).....	1145462.00
OUTLET TOTAL TEMPERATURE (K).....	1525.257
TOTAL LOSS COEFFICIENT	.36413
INCIDENCE ANGLE (degree).....	16.35325

STAGE OVERALL CONDITIONS:

FLOW COEFFICIENT (Va/U).....	1.37637
BLADE LOADING COEFFICIENT (DN/U).....	3.88909
GAMMA (mean) (SI).....	1.33101
Cp (mean) (SI).....	1154.053
PRESSURE RATIO	.4052
ISENTROPIC EFFICIENCY (%)	75.85429

FIGURES

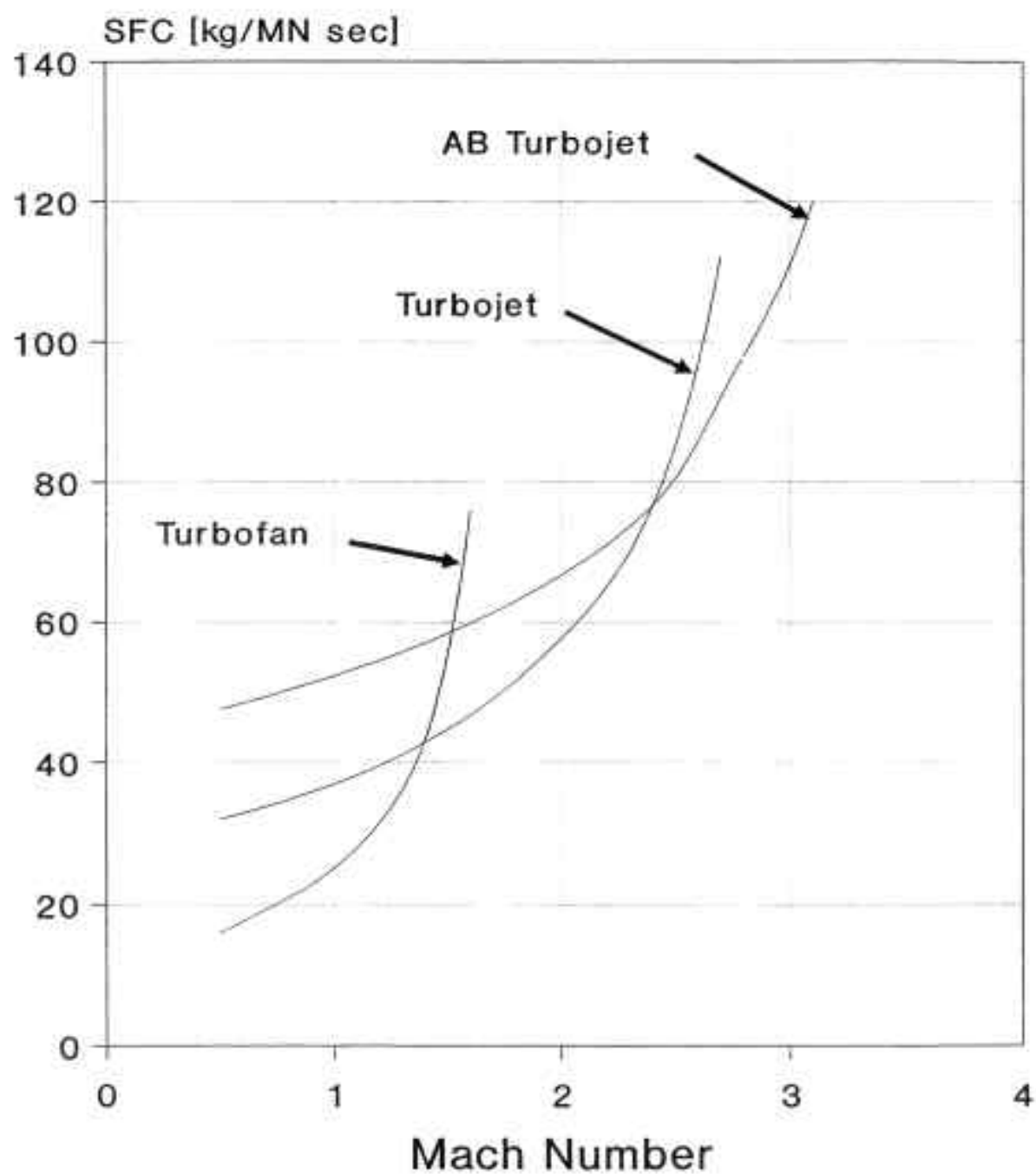
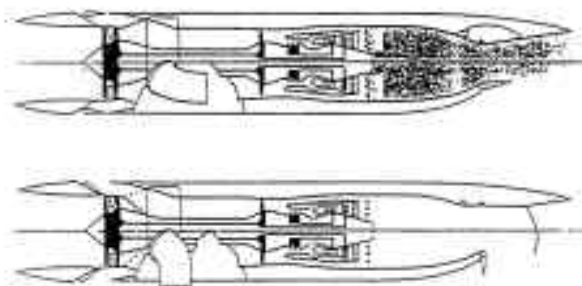
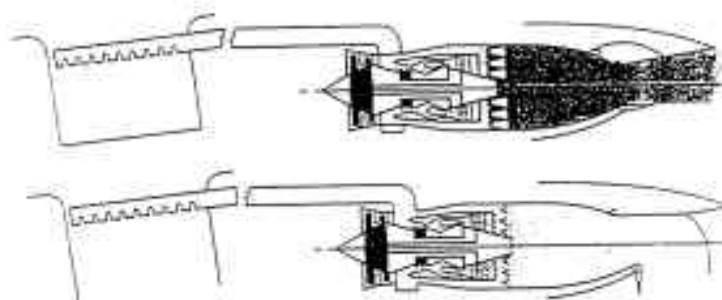


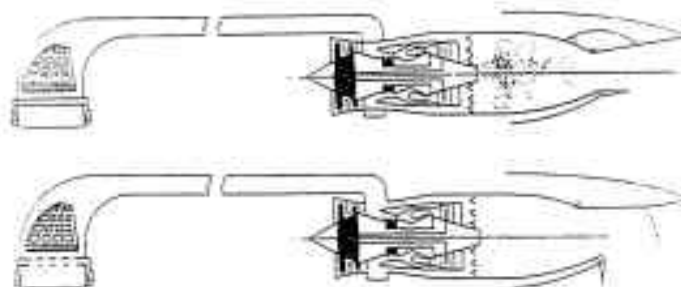
Figure 1.1 SFC of Jet Engines.



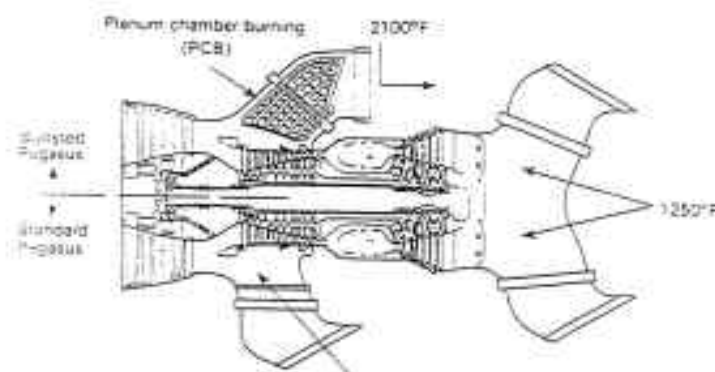
Hybrid Fan



Ejector Lift



Remote Augmented Lift System (RALS)



Vectored Thrust with PCB 224°F

Figure 1.2 Internal Engine Hover Cruise Scheme.

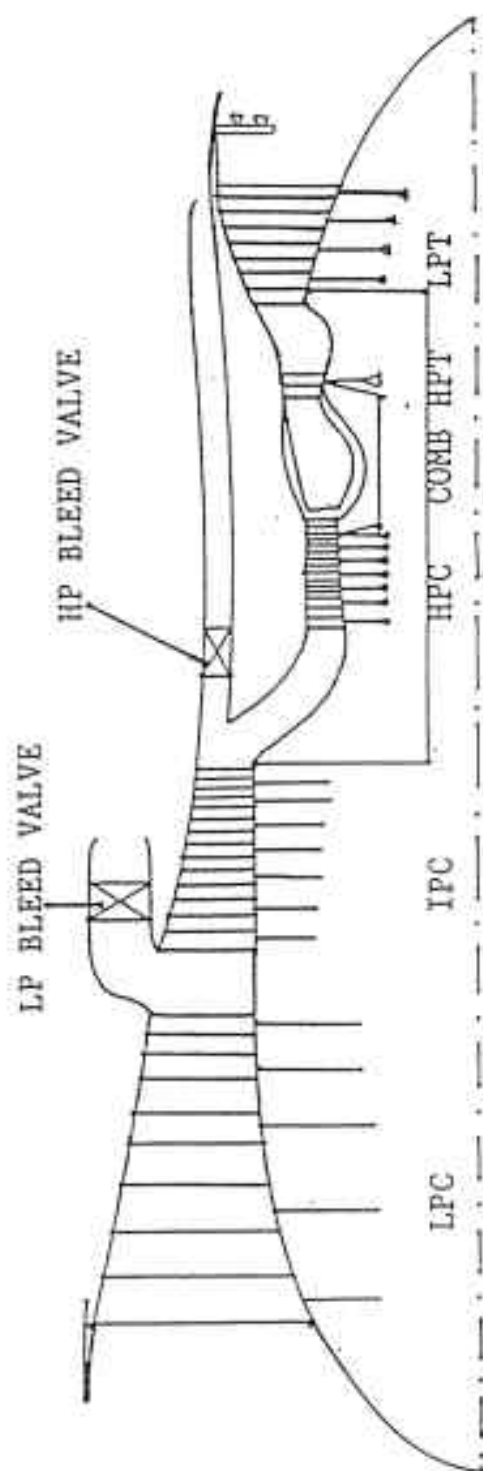
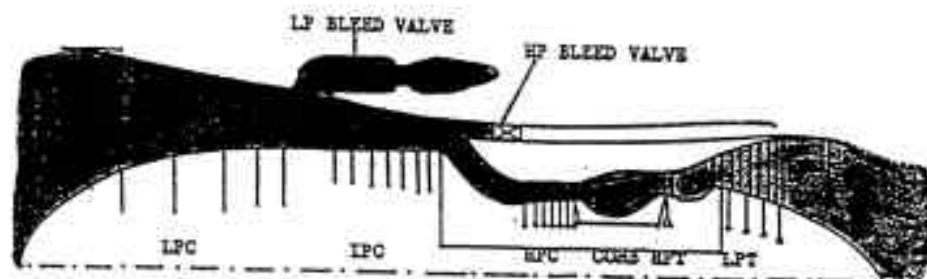
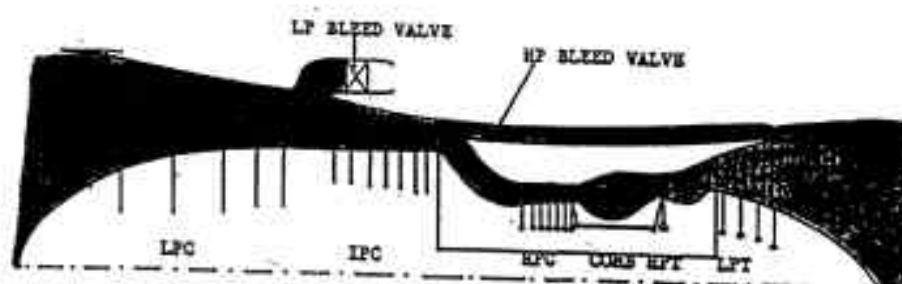


Figure 1.3 The Selective Bleed Engine.



a) LP Mode



b) HP Mode

Figure 2.1 The Selective Bleed Engine (LP and HP modes).

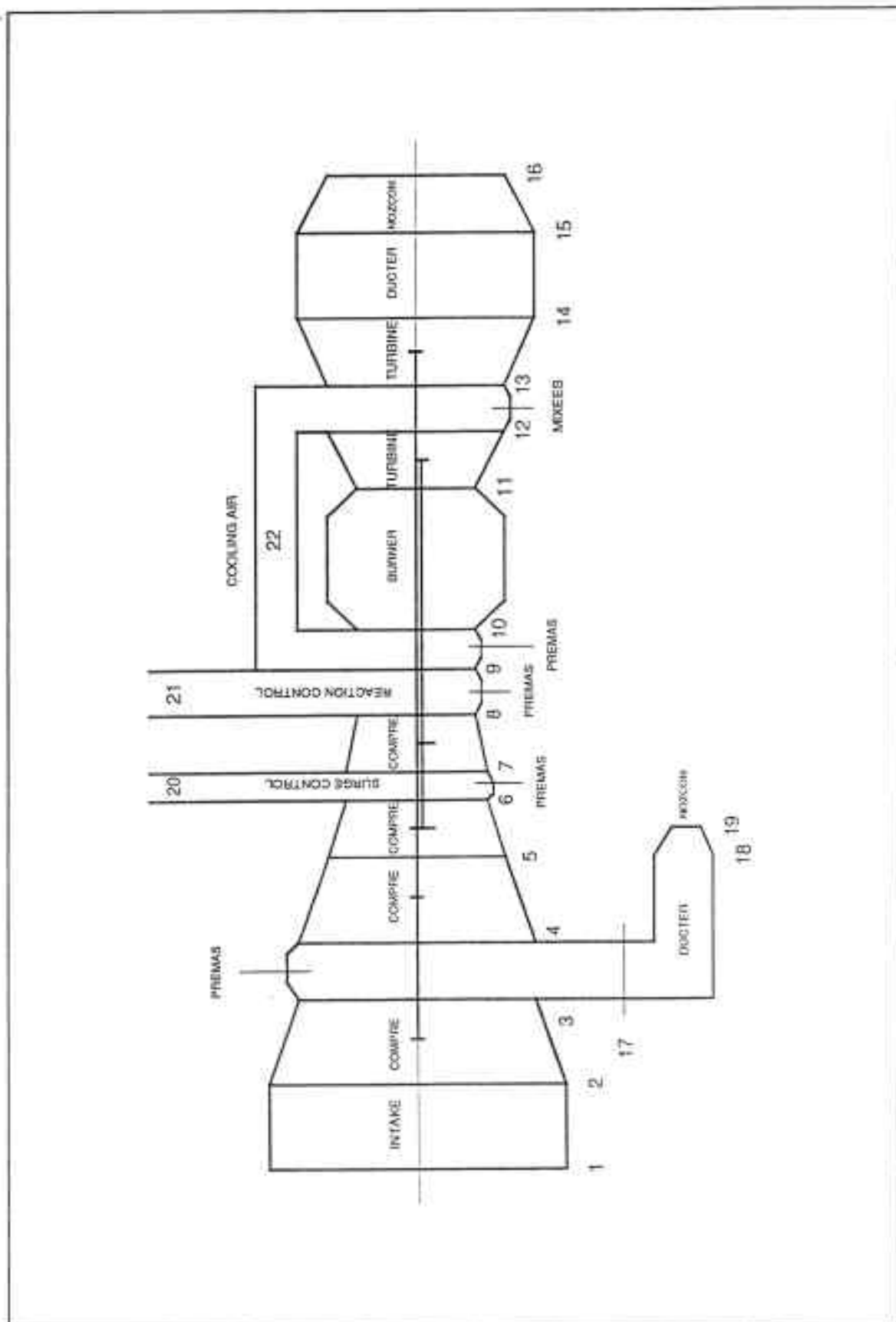


Figure 2.2 TURBOMATCH LP mode Configuration.

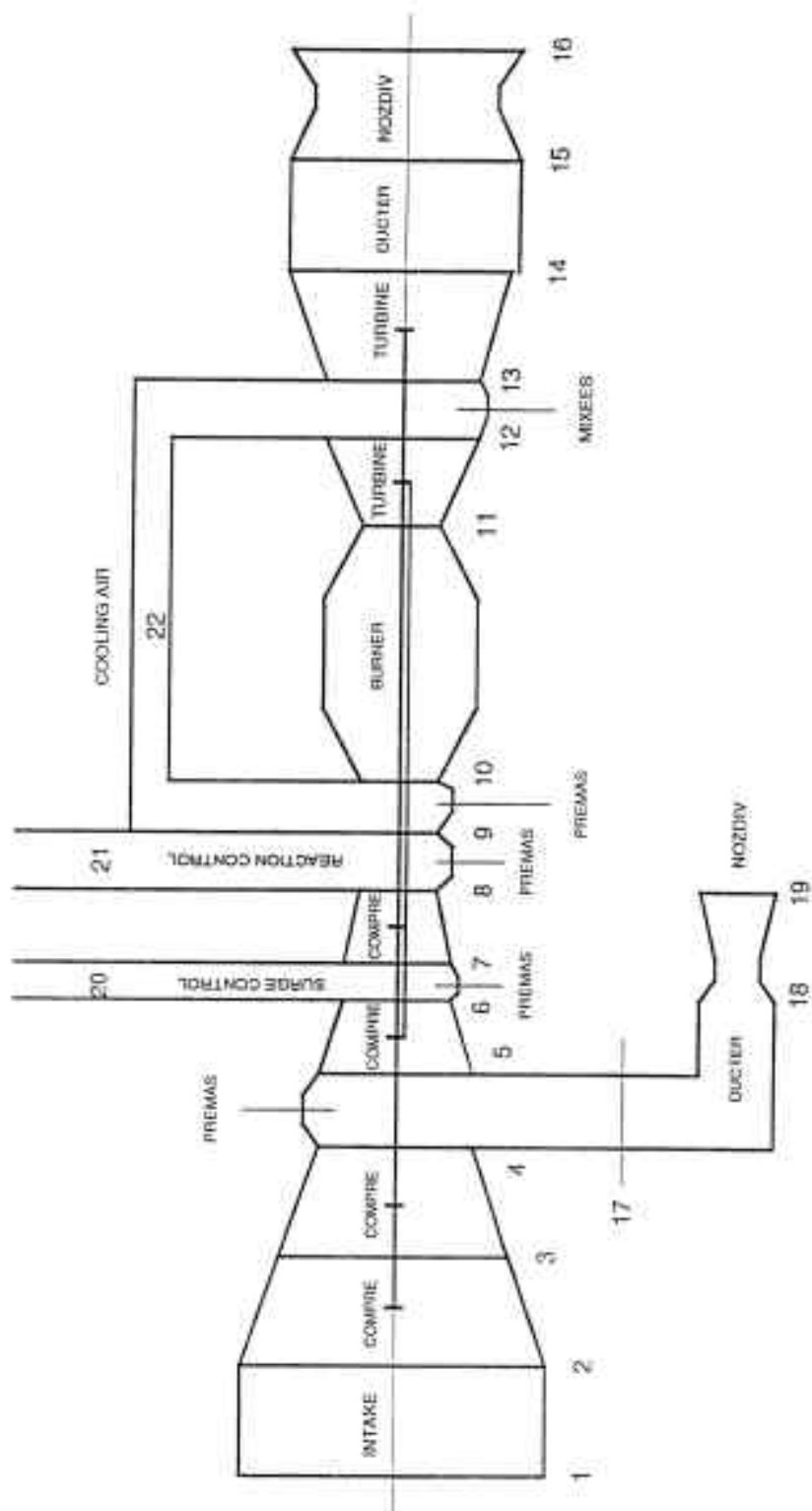


Figure 2.3 TURBOMATCH HP mode Configuration.

LP MODE OPTIMIZATION

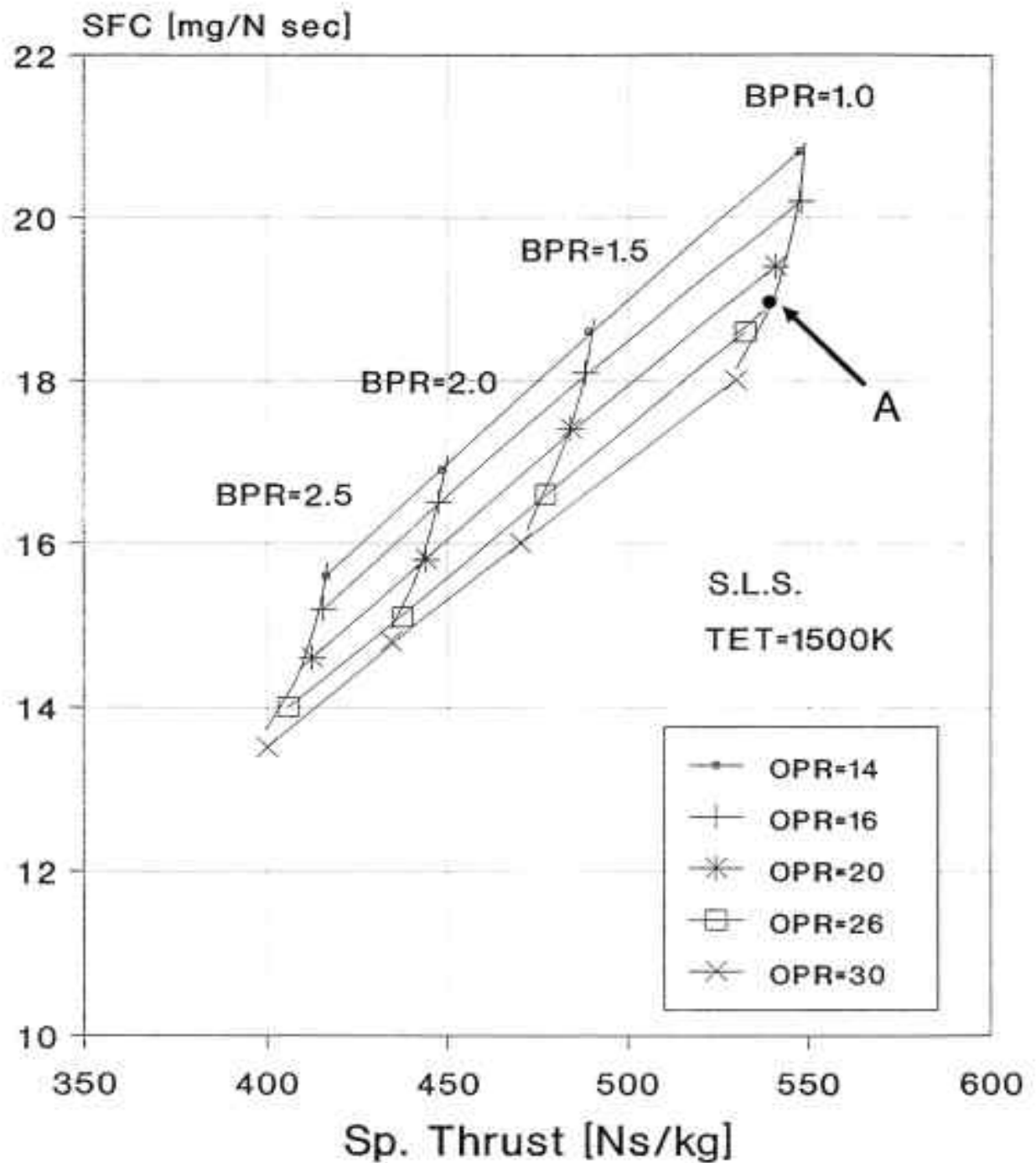


Figure 2.4 SFC versus Sp. Thrust.

LP MODE OPTIMIZATION

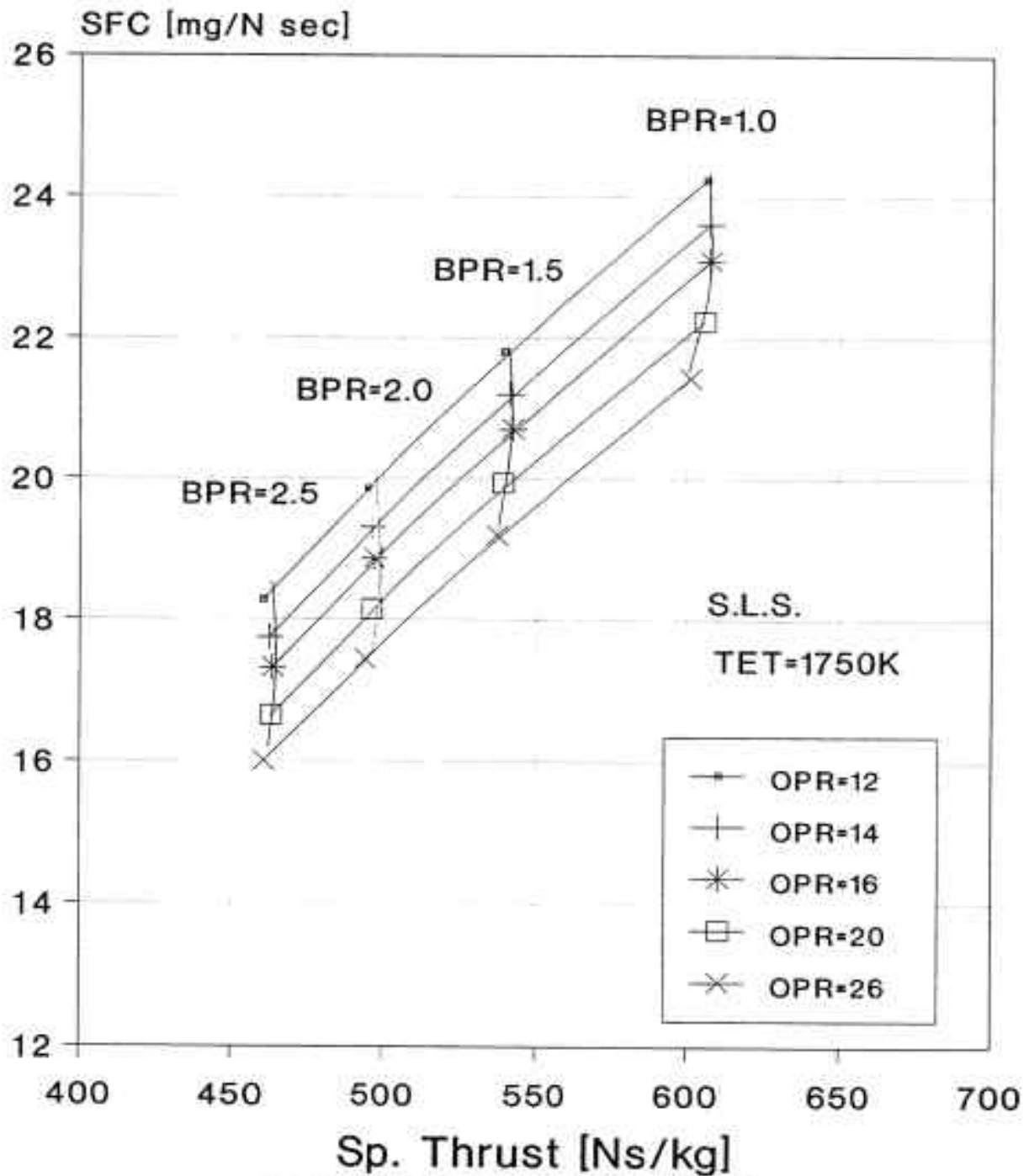


Figure 2.5 SFC versus Sp. Thrust.

LOW PRESSURE COMPRESSOR

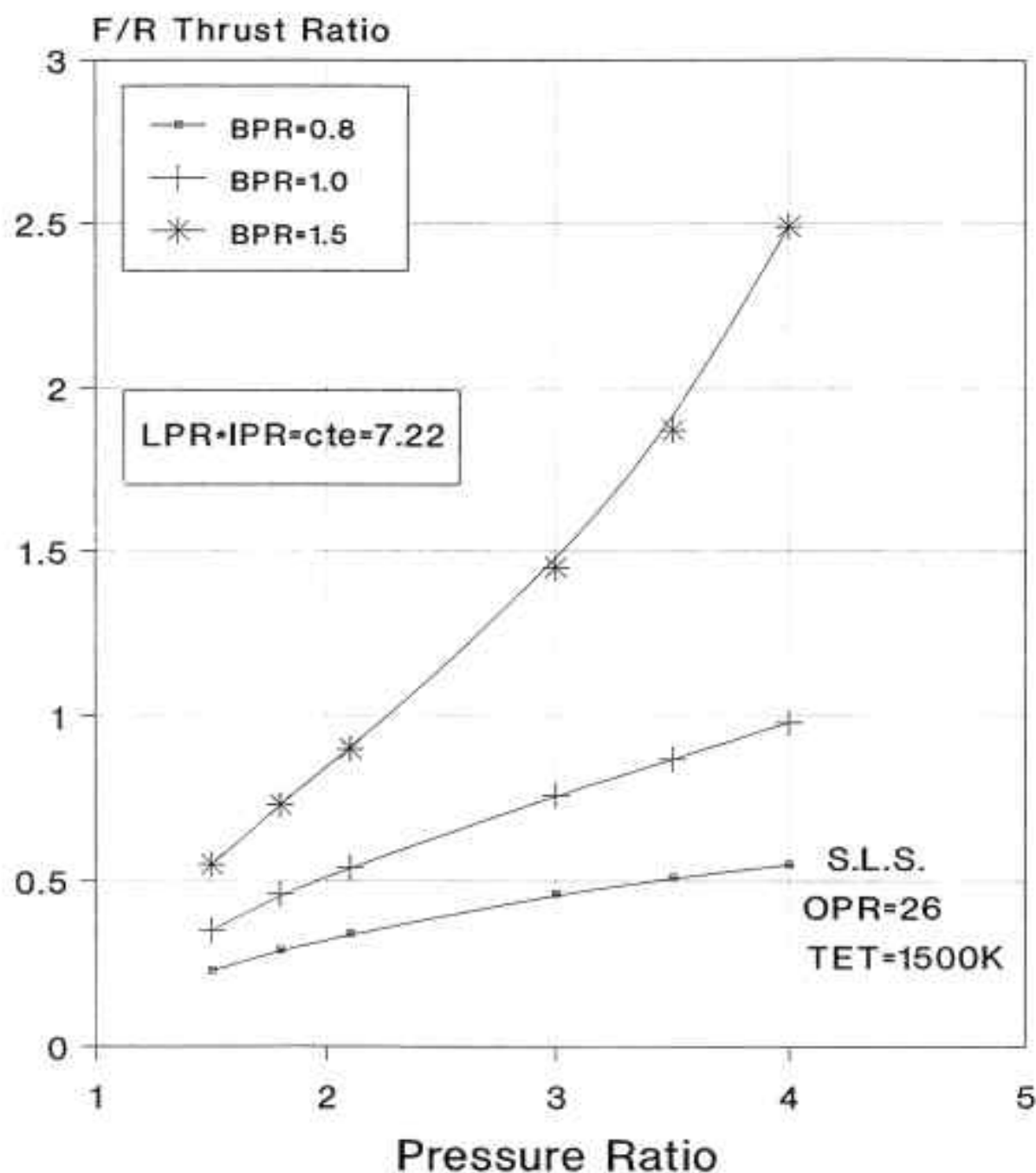


Figure 2.6 F/R Thrust Ratio versus Pressure Ratio (LP mode)

HP MODE OPTIMIZATION

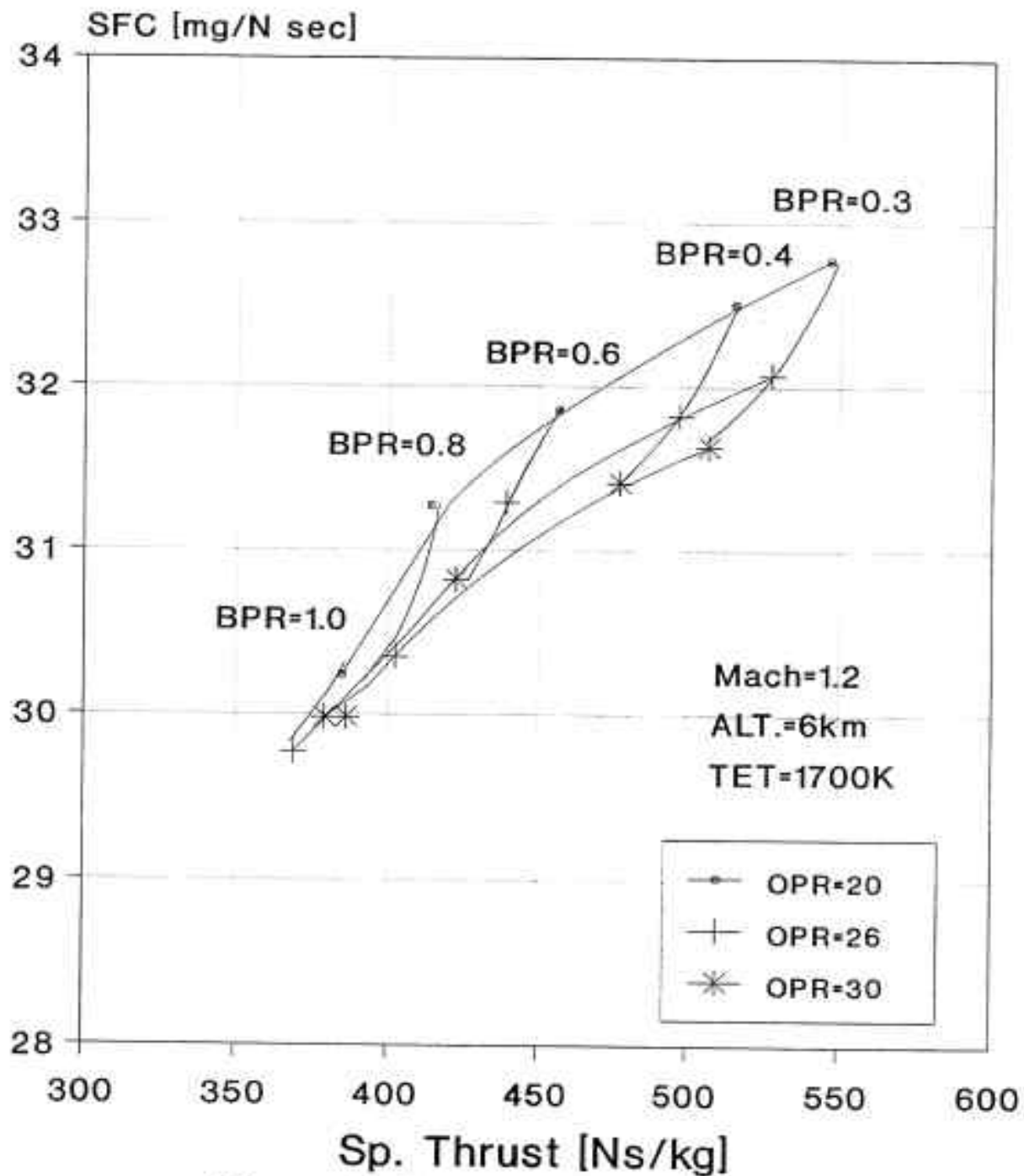


Figure 2.7 SFC versus Sp. Thrust.

HP MODE OPIMIZATION

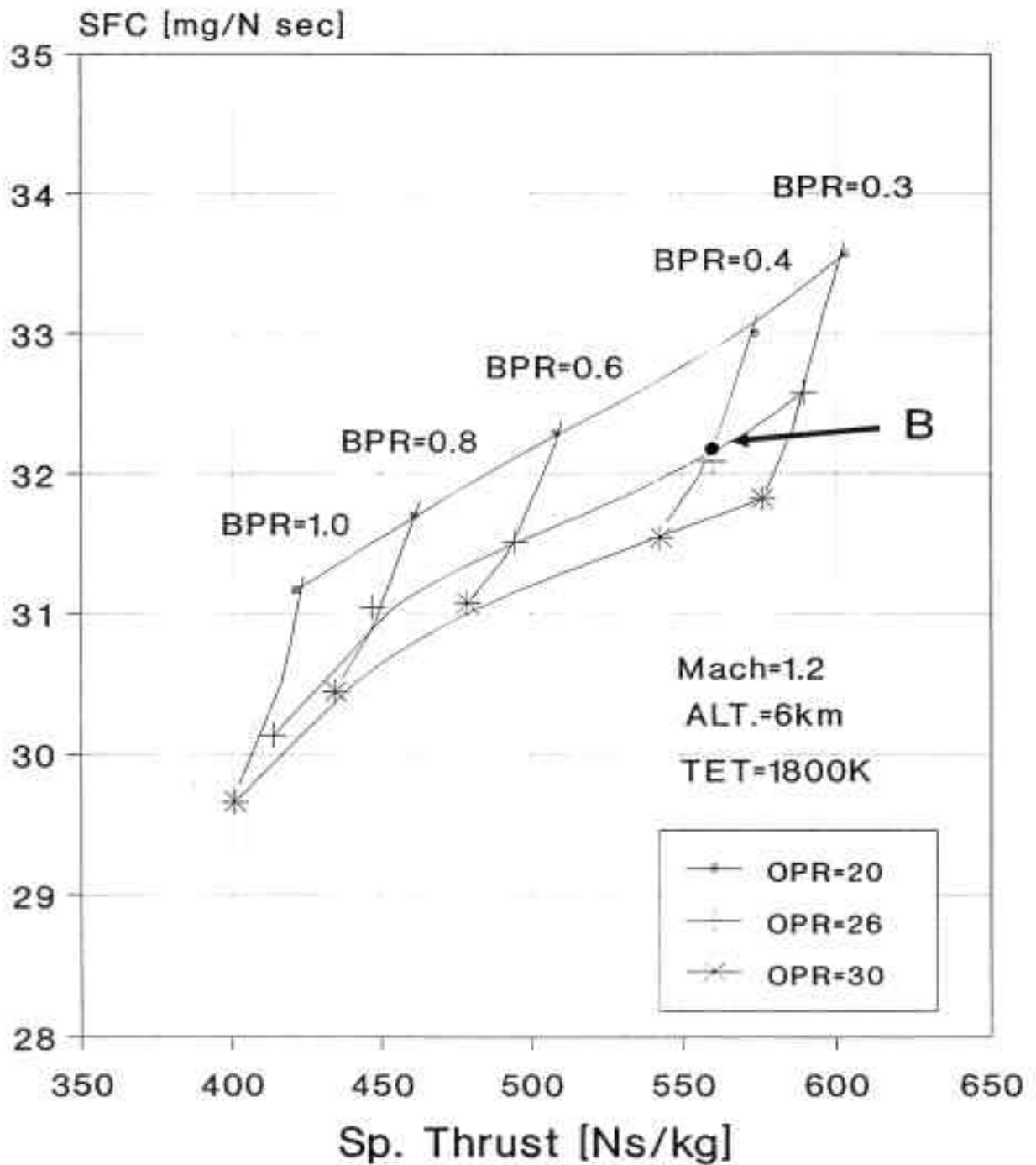


Figure 2.8 SFC versus Sp. Thrust.

LOW PRESSURE COMPRESSOR

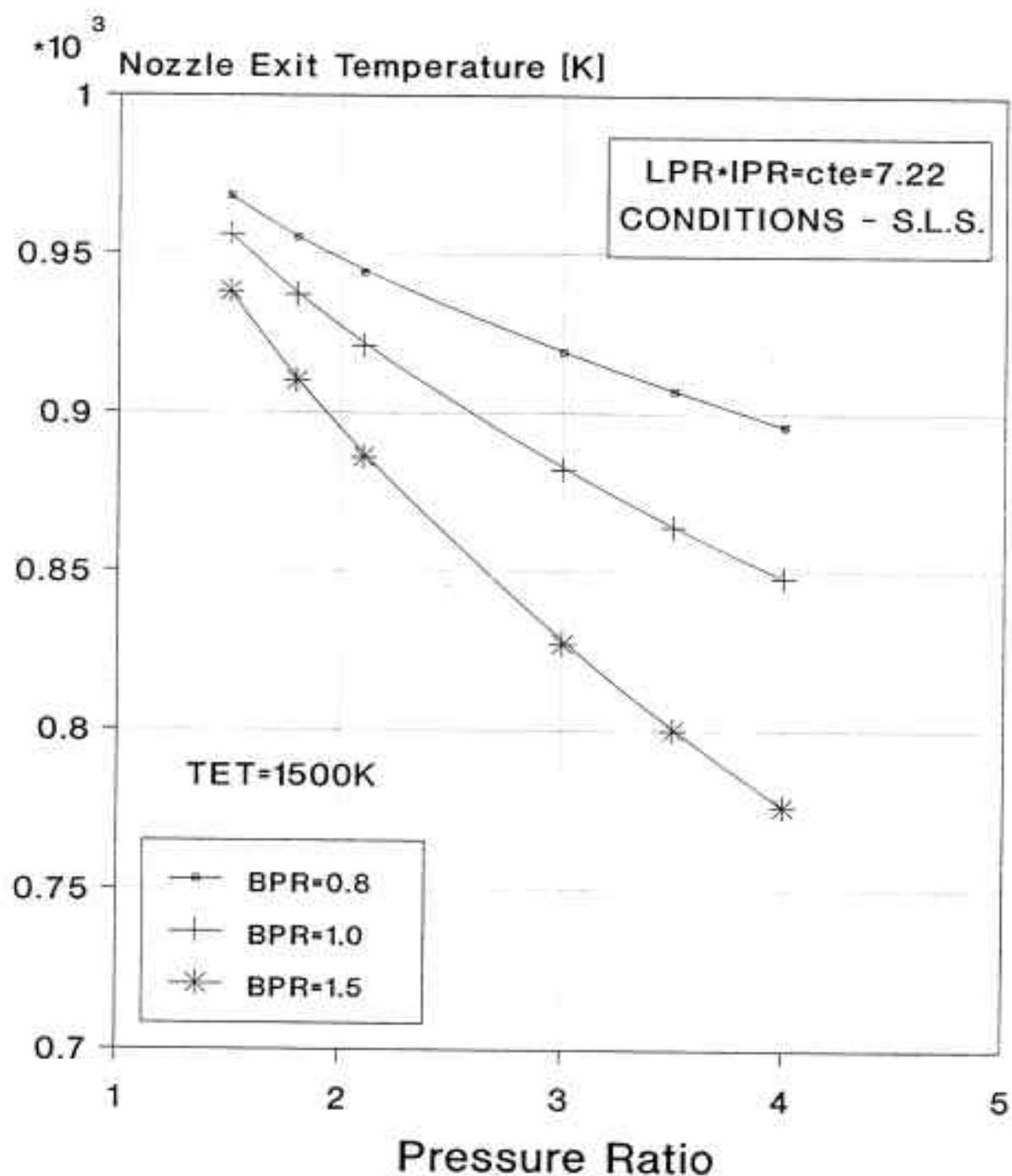


Figure 2.9 Nozzle Exit Temperature versus Pressure Ratio (LP mode)

LOW PRESSURE COMPRESSOR

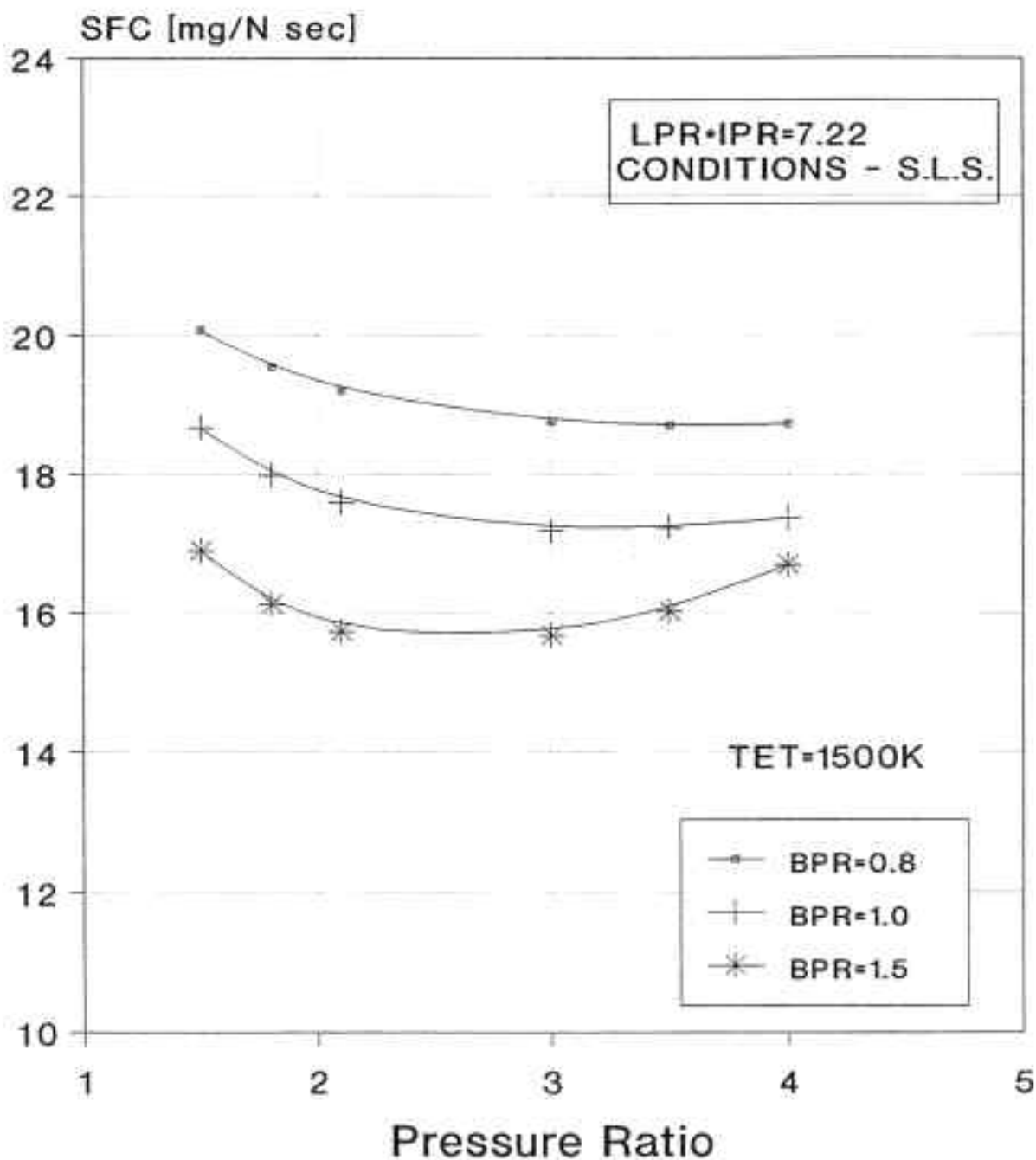


Figure 2.10 SFC versus Pressure Ratio
(LP mode)

COMPRESSORS

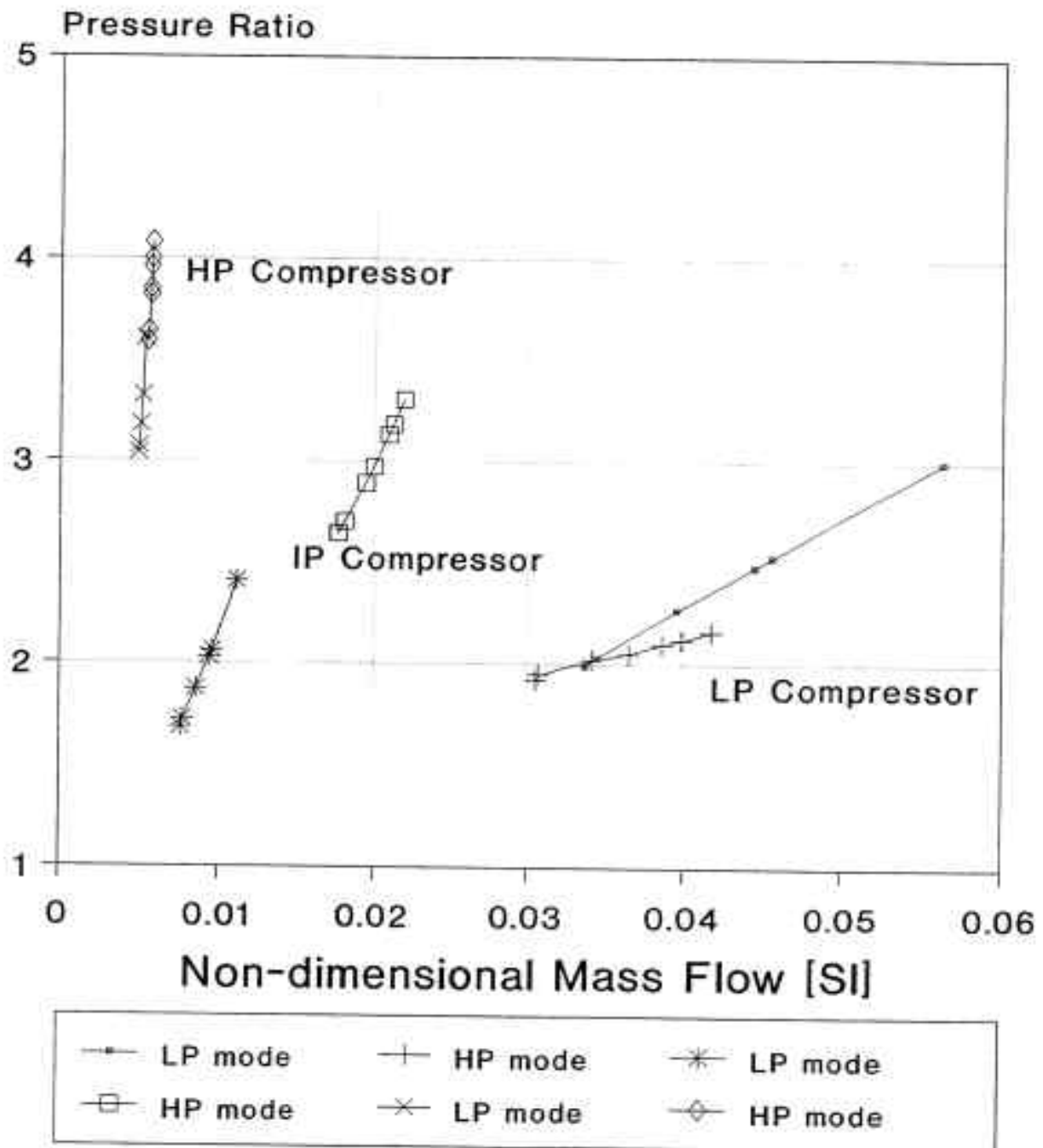


Figure 2.11 Compressor Operating Lines.

OFF-DESIGN PERFORMANCE

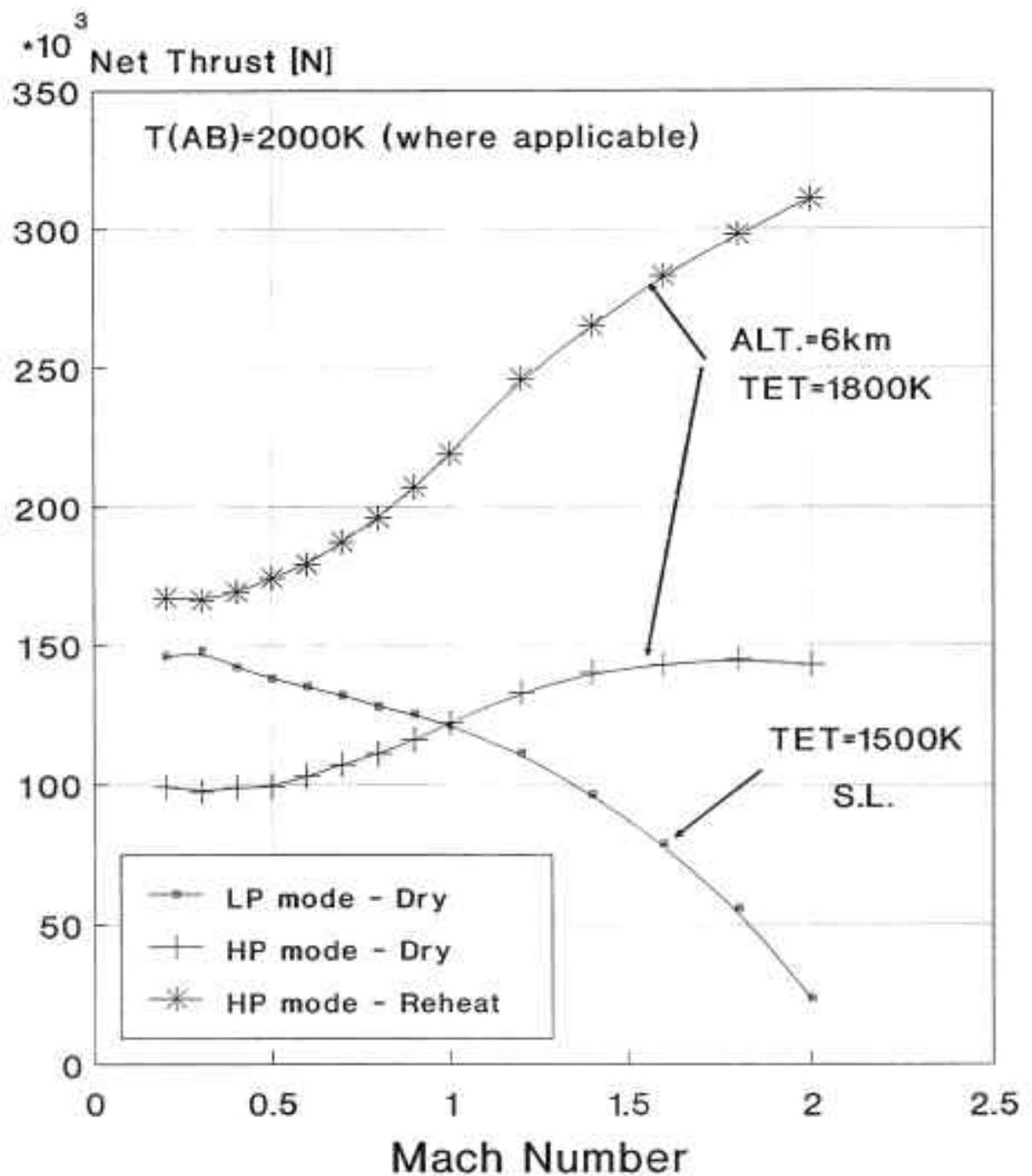


Figure 2.12 Variation of Thrust with Mach Number.

OFF-DESIGN PERFORMANCE

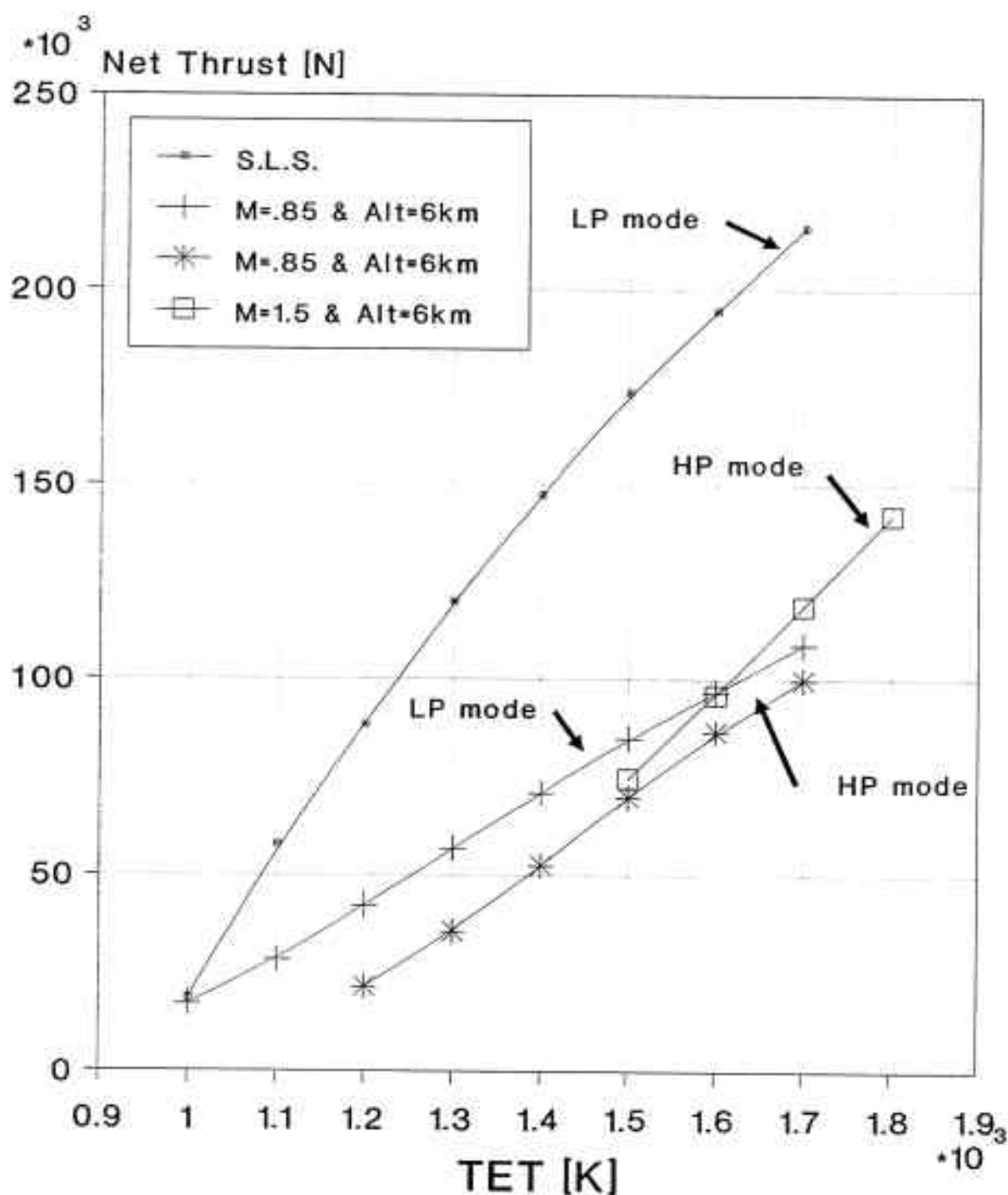


Figure 2.13 Change of Thrust with TET

OFF-DESIGN PERFORMANCE

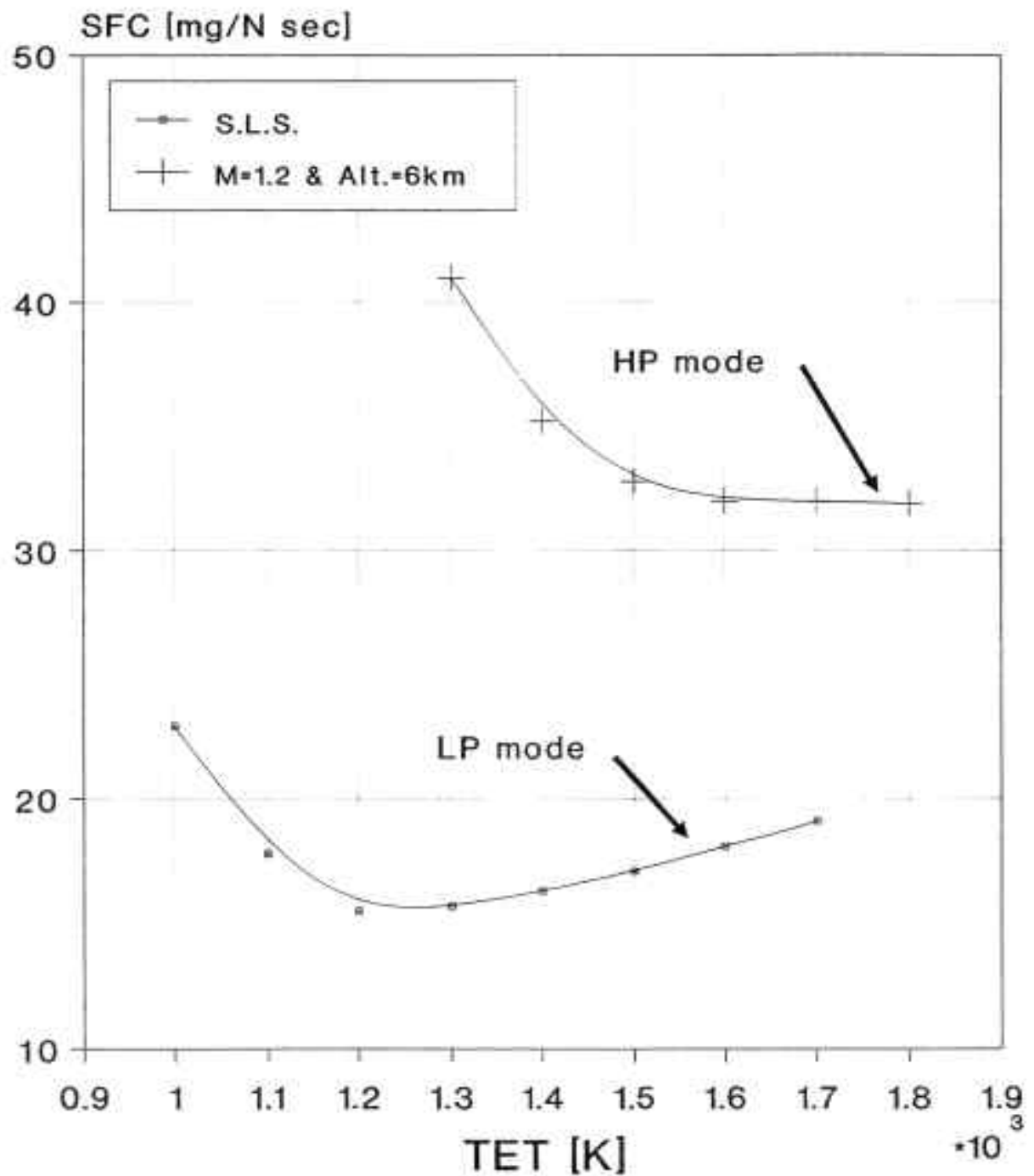


Figure 2.14 Change of SFC with TET

OFF-DESIGN PERFORMANCE

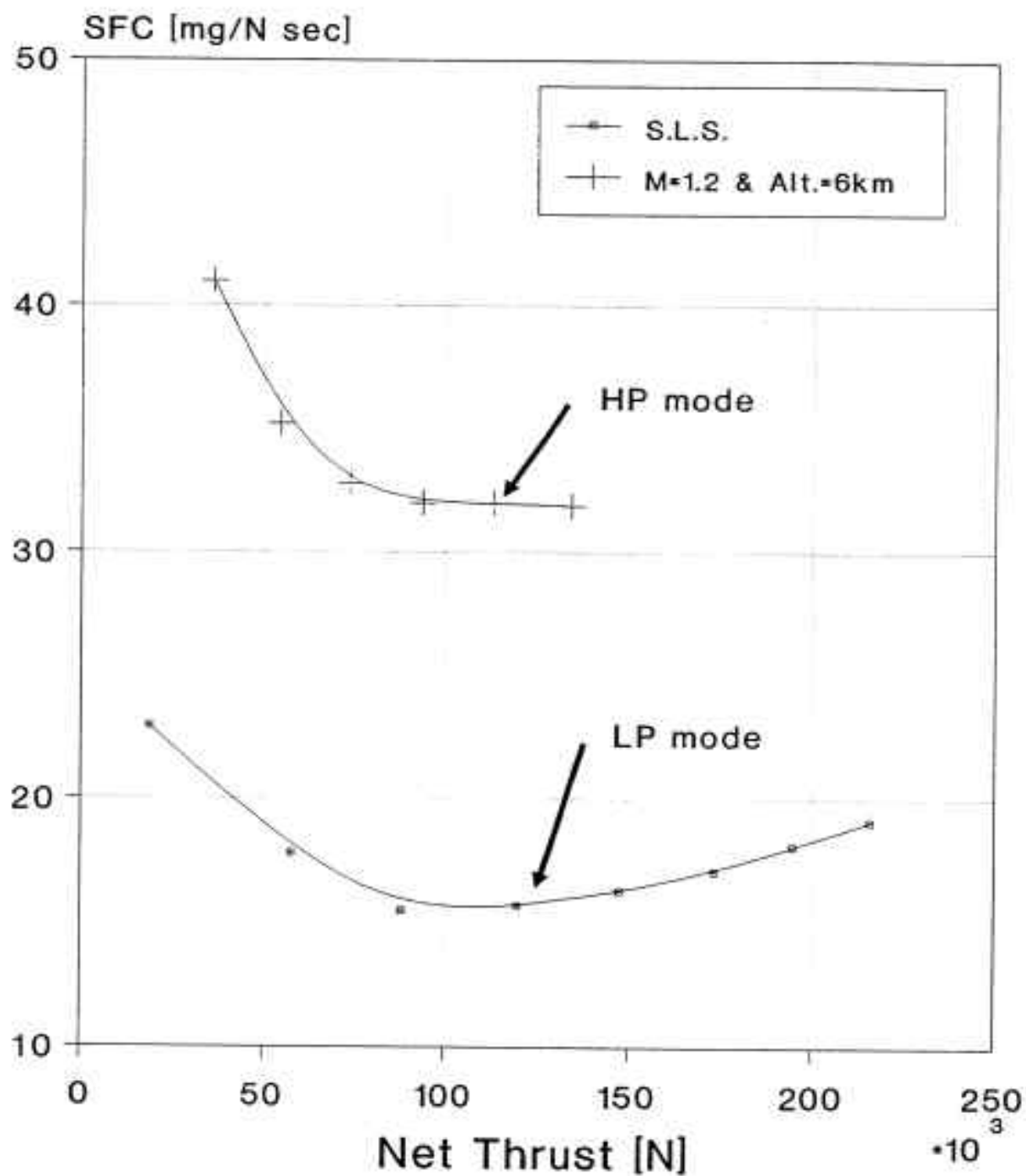


Figure 2.15 Change of SFC with Thrust

OFF-DESIGN PERFORMANCE

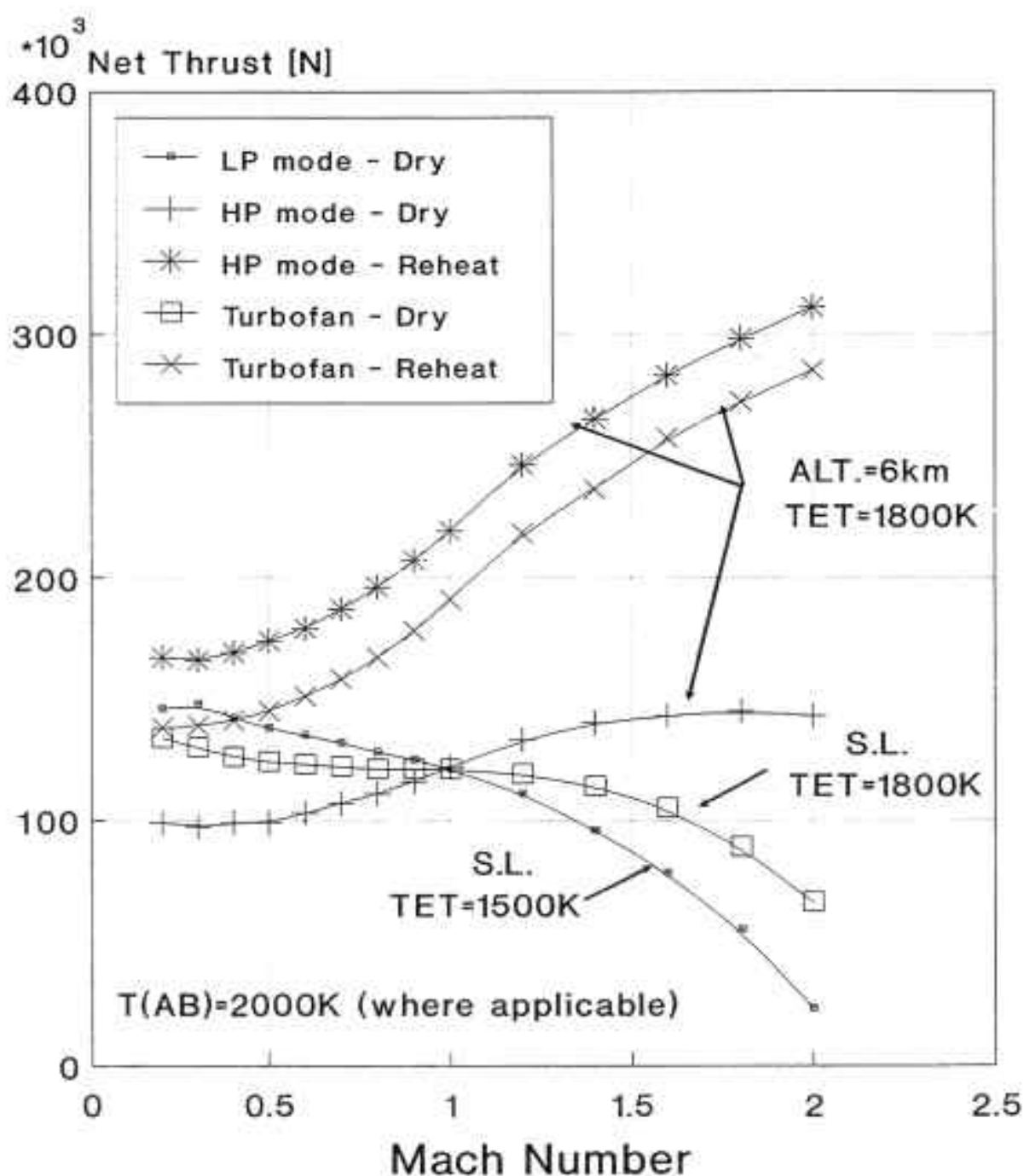


Figure 2.16 Comparison of the Selective Bleed Engine to a Turbofan.

OFF-DESIGN PERFORMANCE

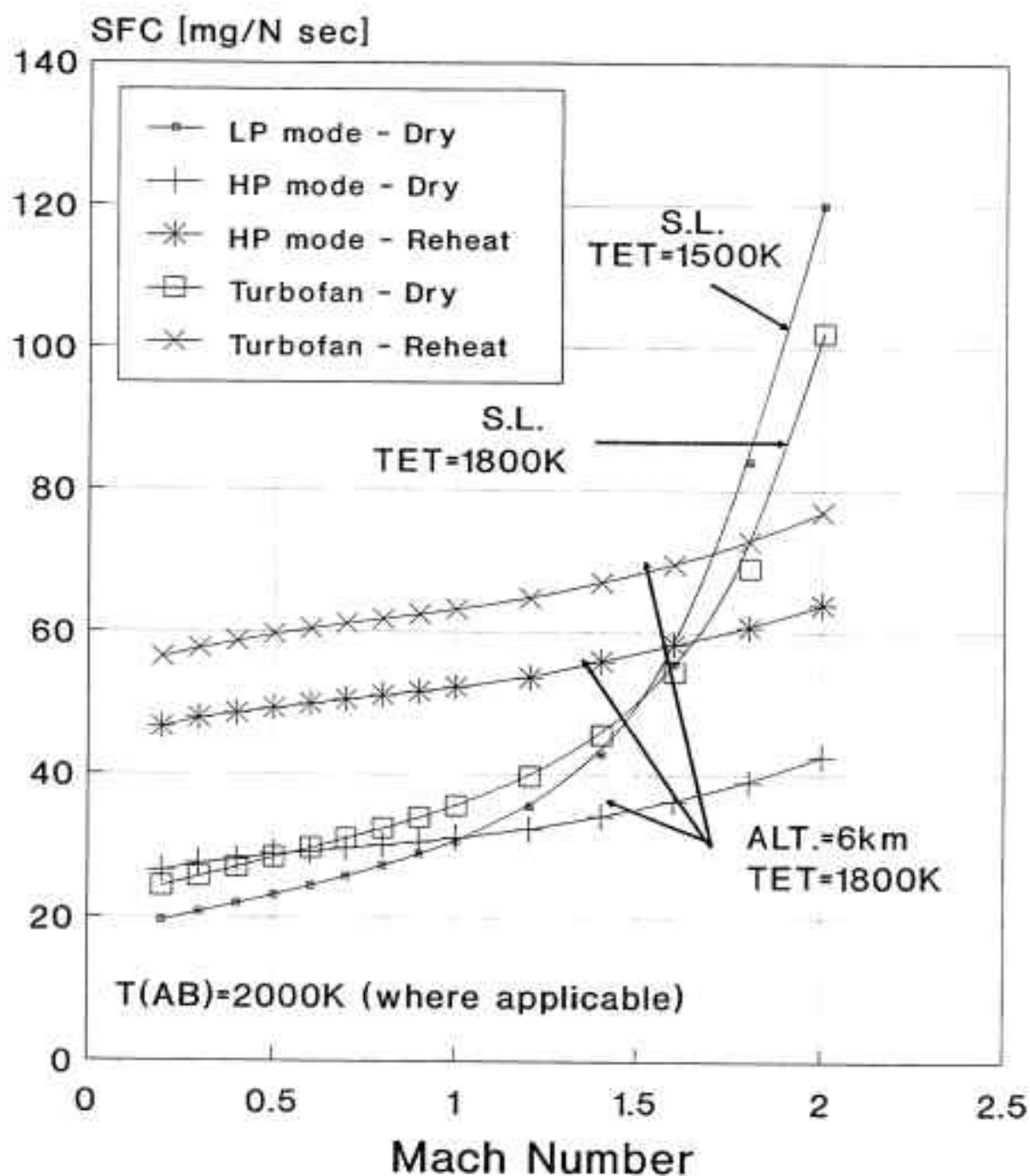


Figure 2.17 Comparison of the Selective Bleed Engine to a Turbofan.

Matching

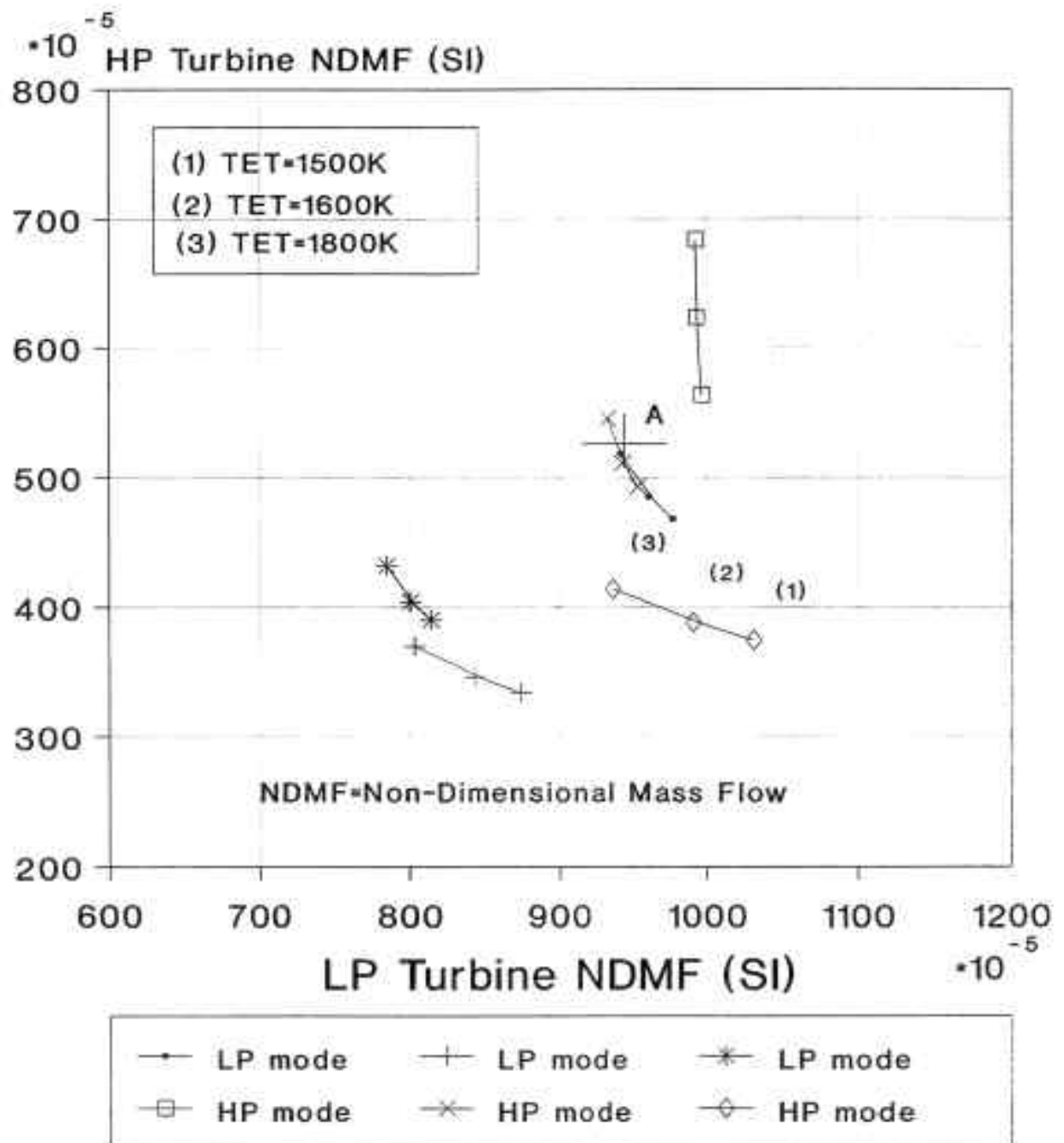


Figure 3.1 Crossplot of HP and LP Turbine Non-dimensional Mass Flows

Matching

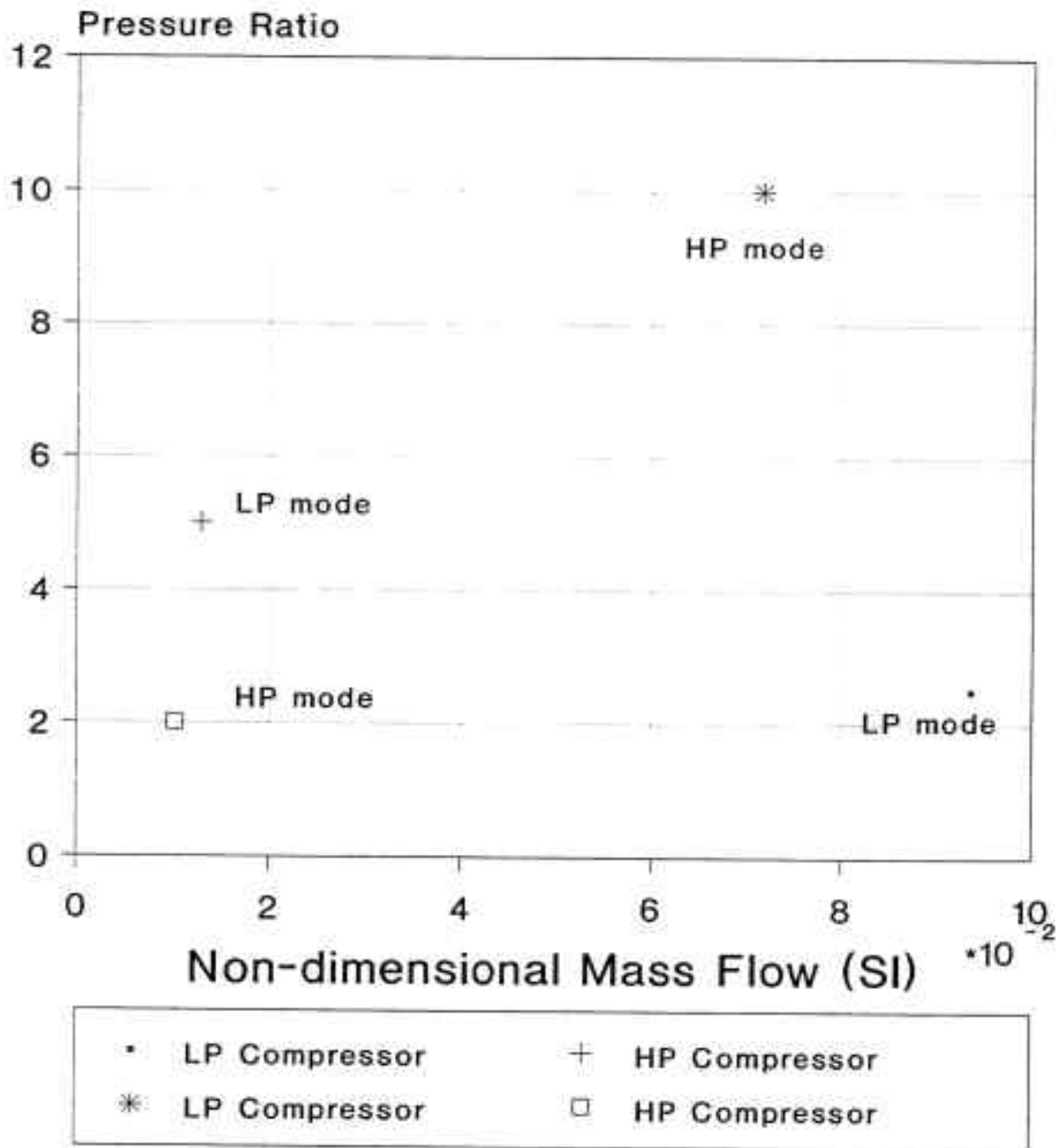


Figure 3.2 Compressor Design Point Characteristics.

Matching (HP Turbines)

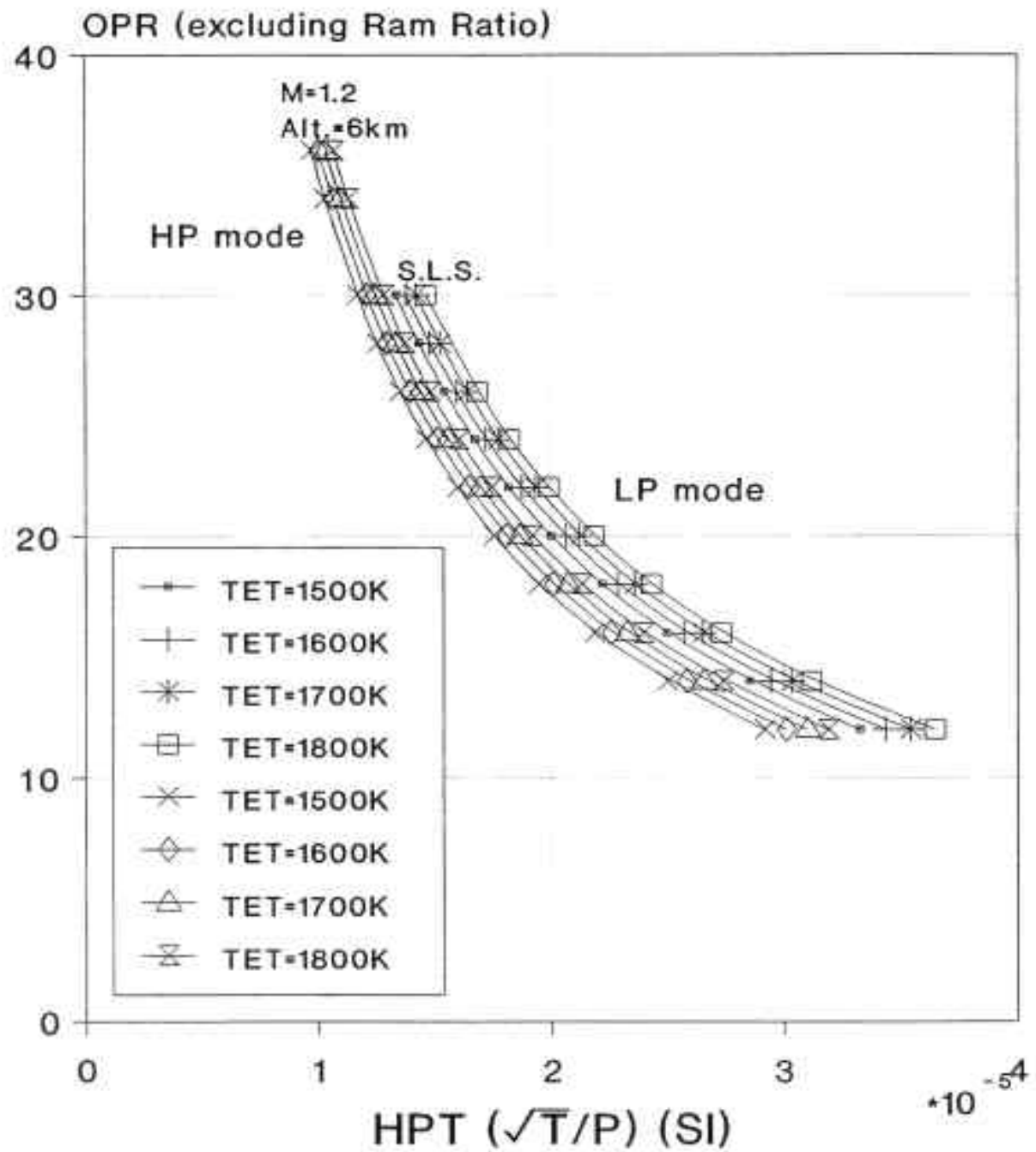


Figure 3.3 Effect of TET for Both Modes.

Matching (LP Turbines)

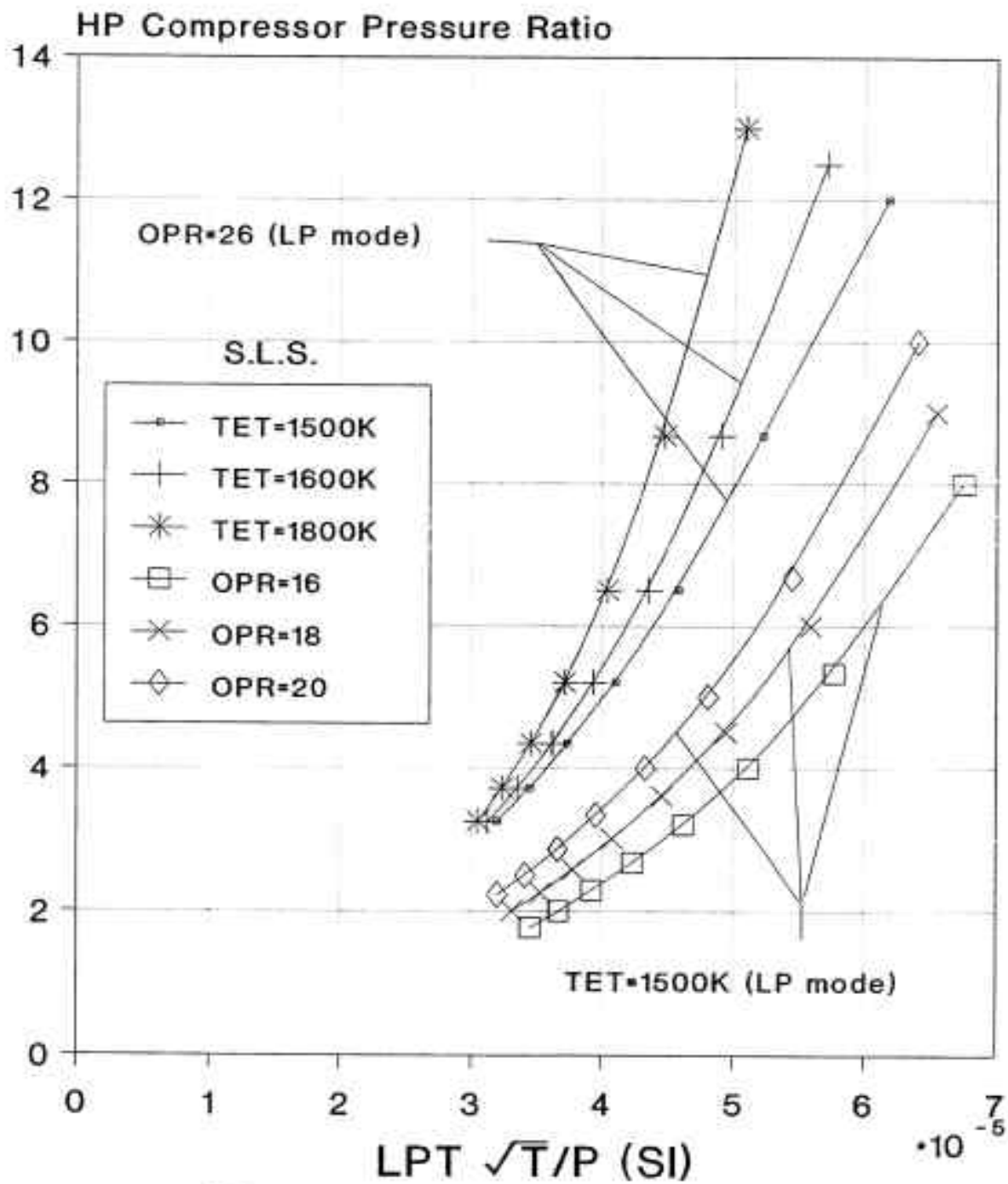


Figure 3.4 Effect of TET and OPR

LP Compressor Running Lines.

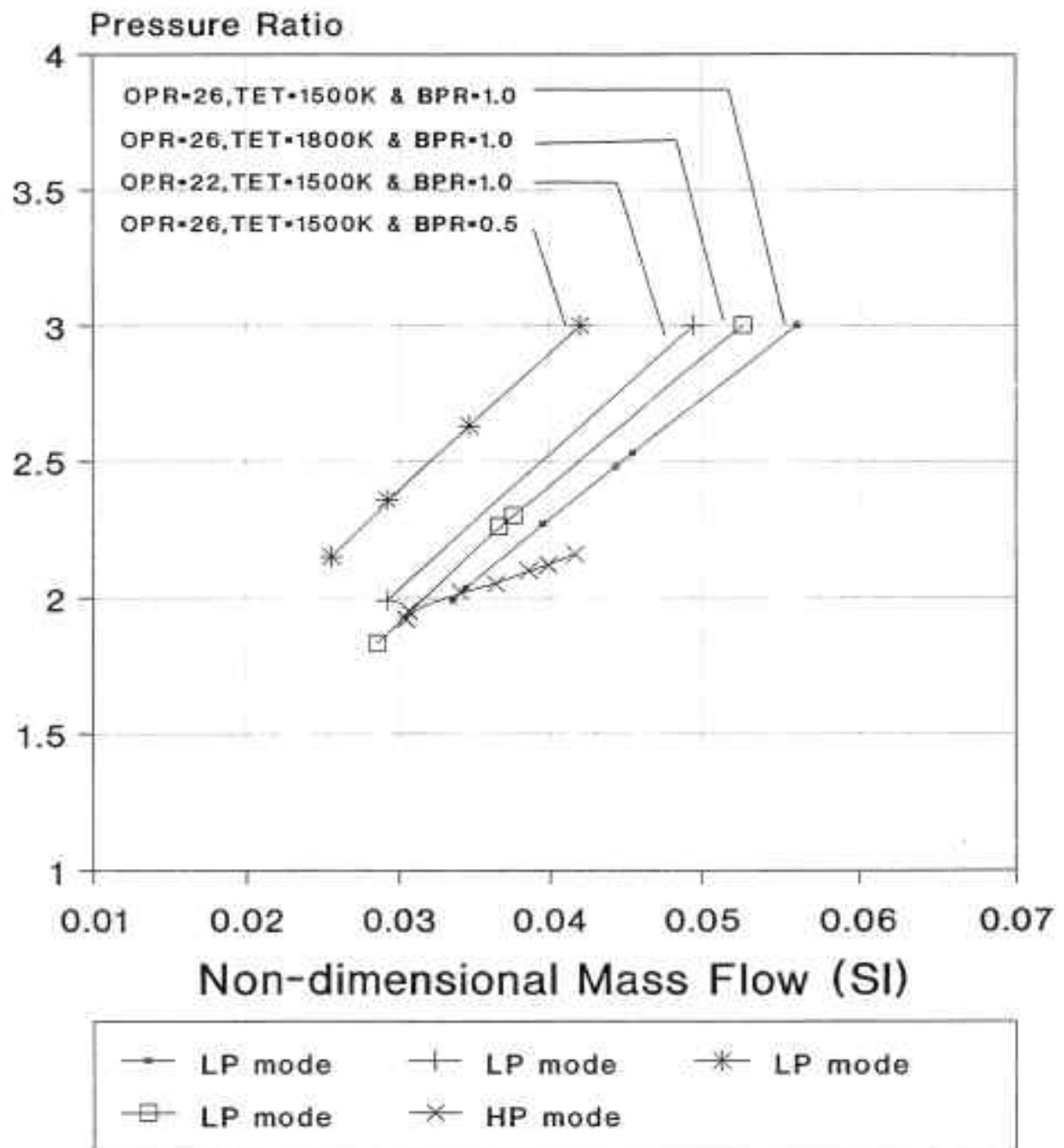


Figure 3.5 Effect of TET & BPR.

Matching

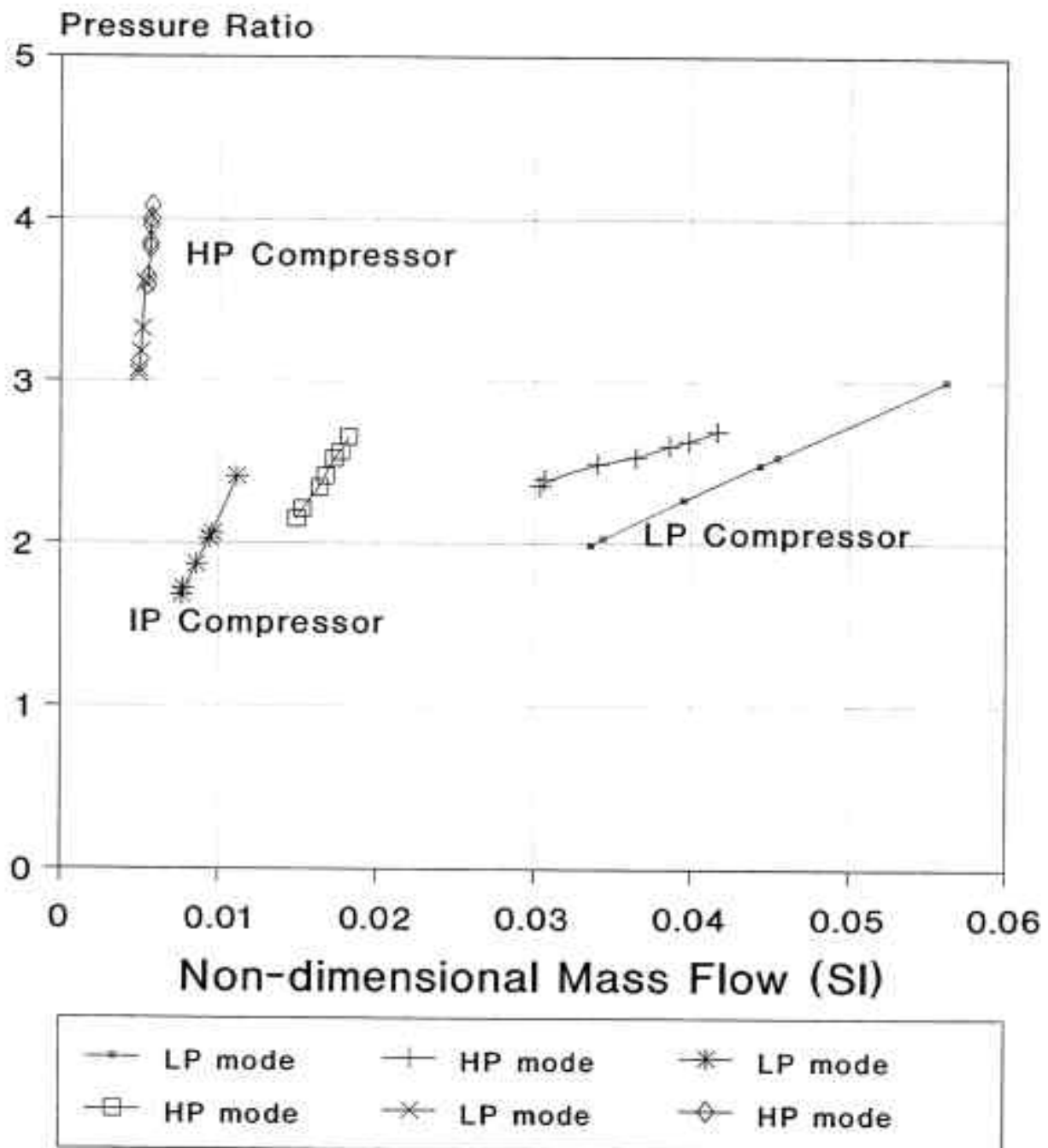


Figure 3.6 Compressor Operating Lines

Matching

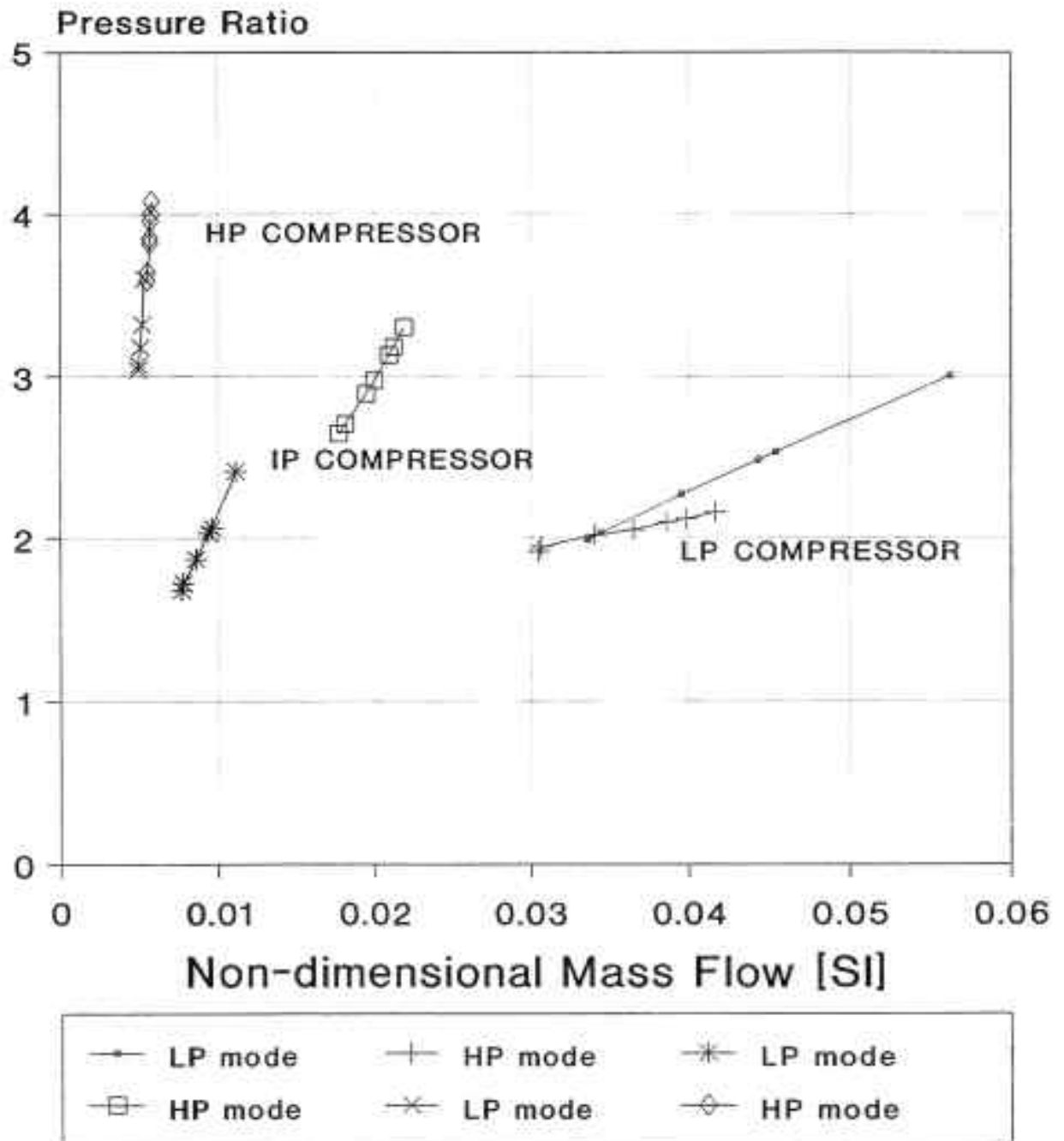


Figure 3.7 Compressor Operating Lines.

Matching

Example - IP Compressor

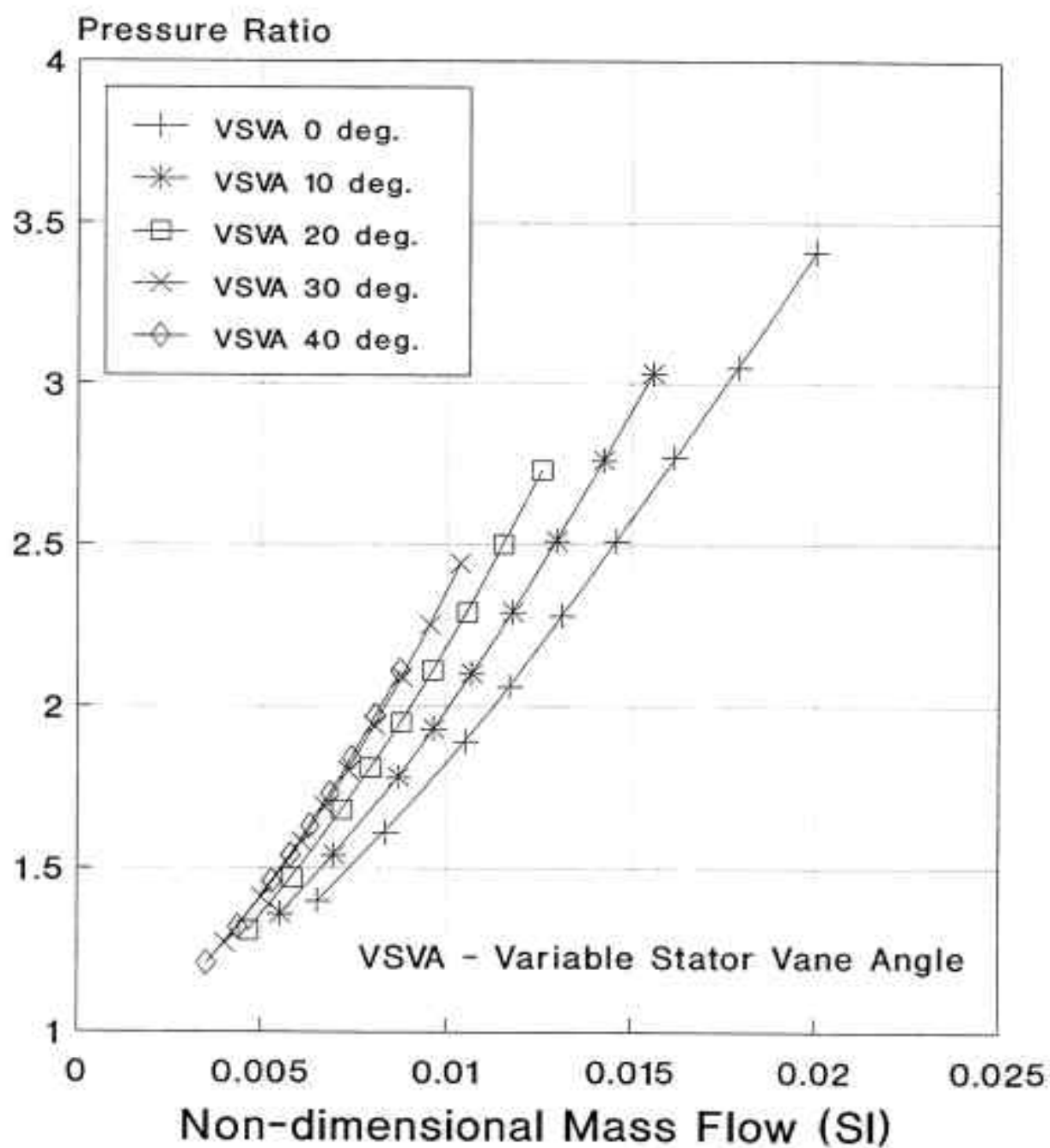


Figure 3.8 Surge Line Displacement.

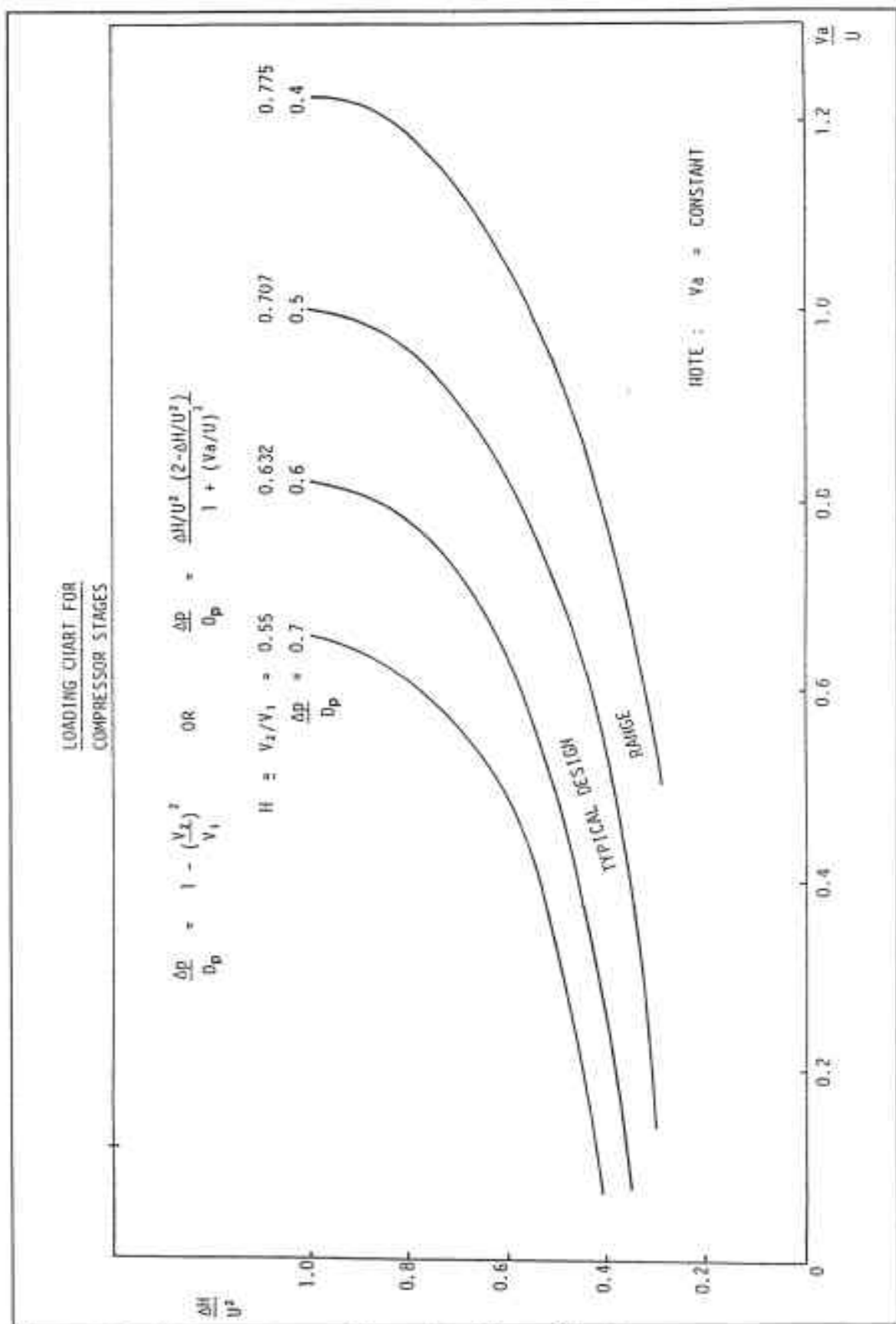


Figure 4.1 Compressor Stage Loading Chart.

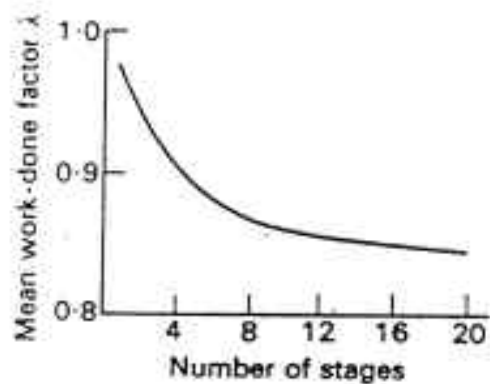


Figure 4.2 Variation of Mean Work-done Factor with Number of Stage.

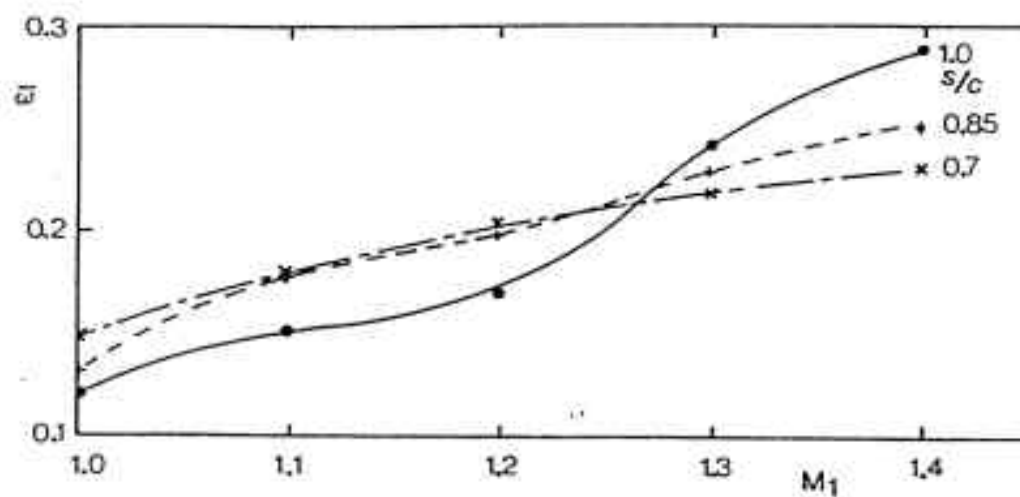


Figure 4.3 Effect of Inlet Mach number on the Loss Coefficient of Double-circular-arc Blading at Different Space/chord Ratios.

Compressor Characteristics

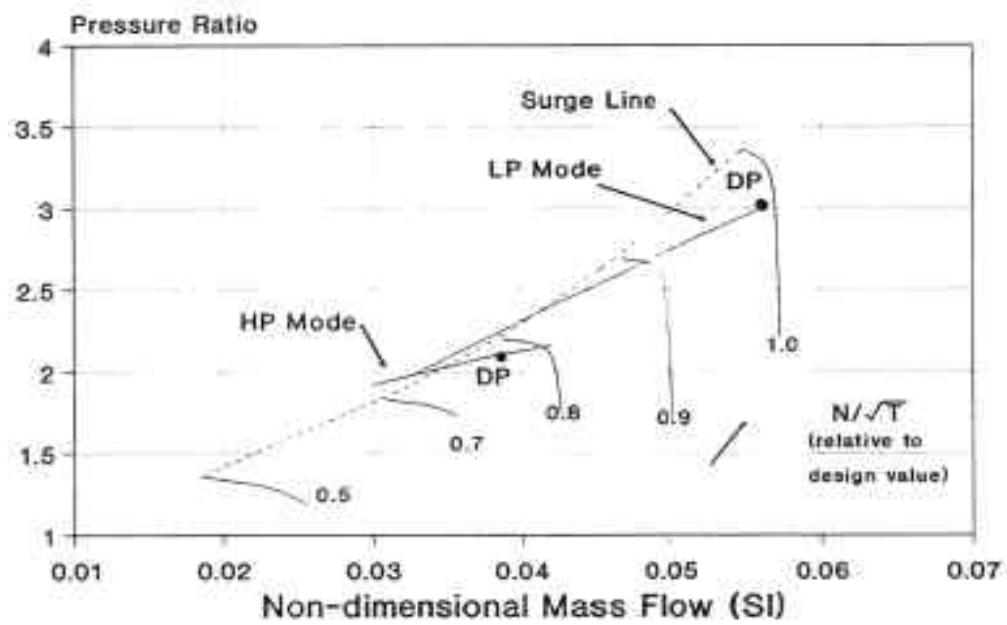
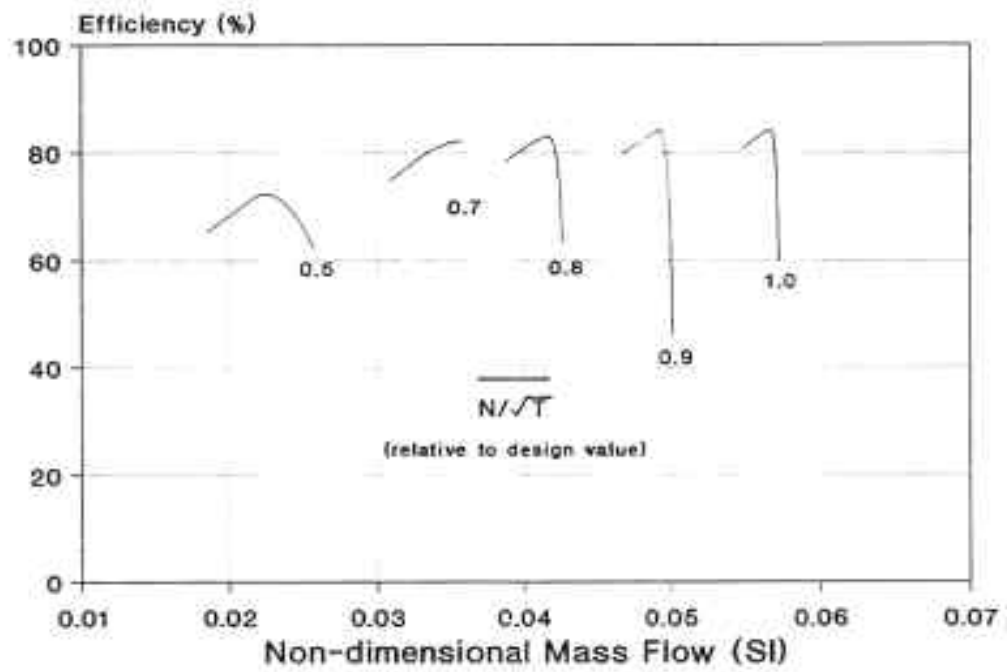


Figure 4.4 LP Compressor.

Compressor Characteristic

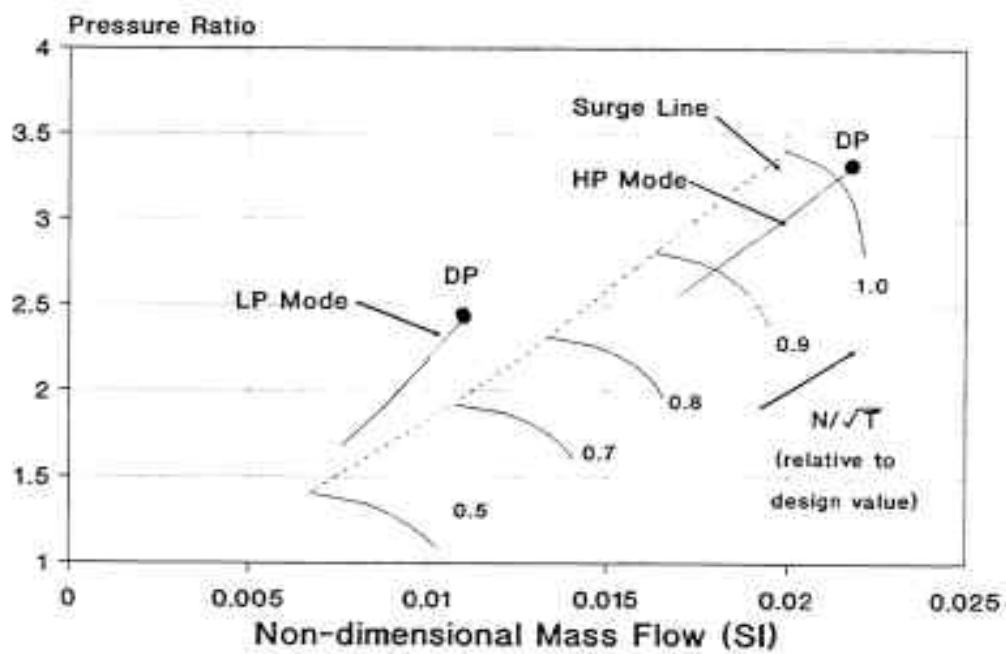
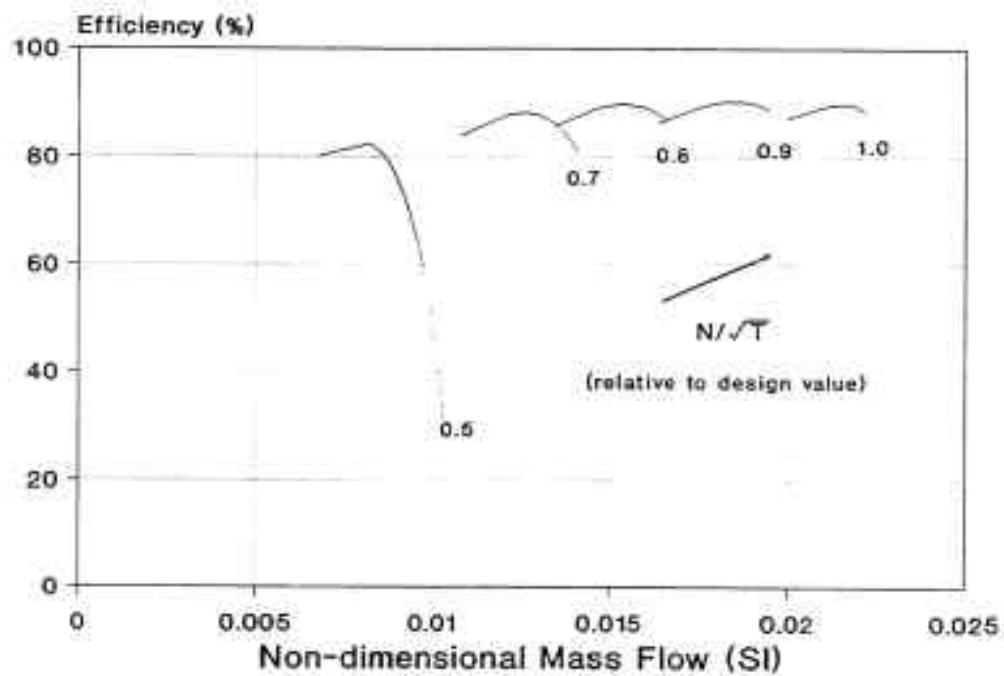


Figure 4.5 IP Compressor.

Compressor Characteristics

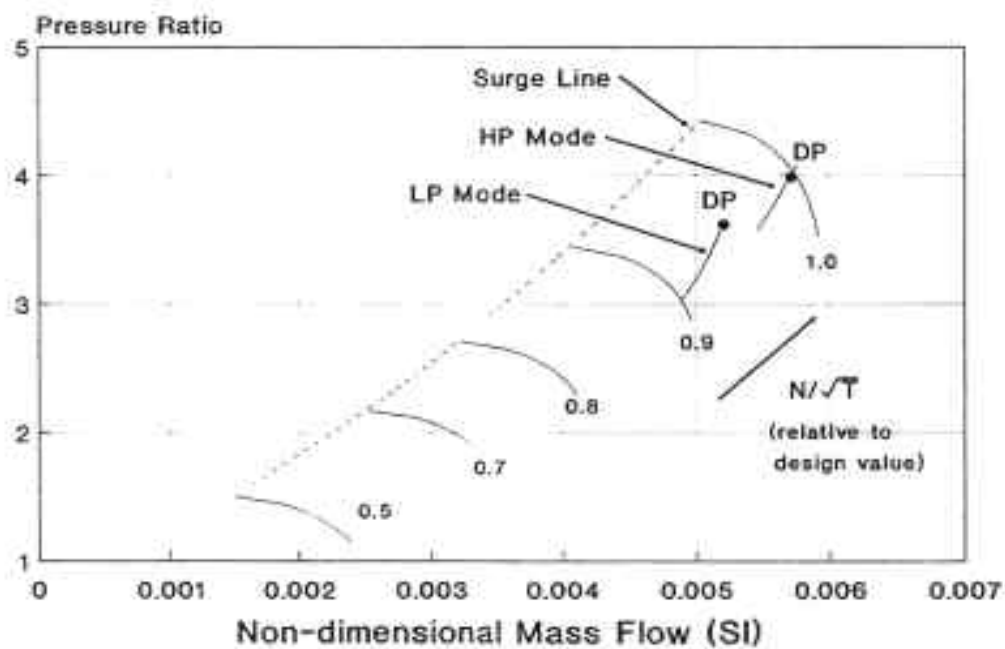
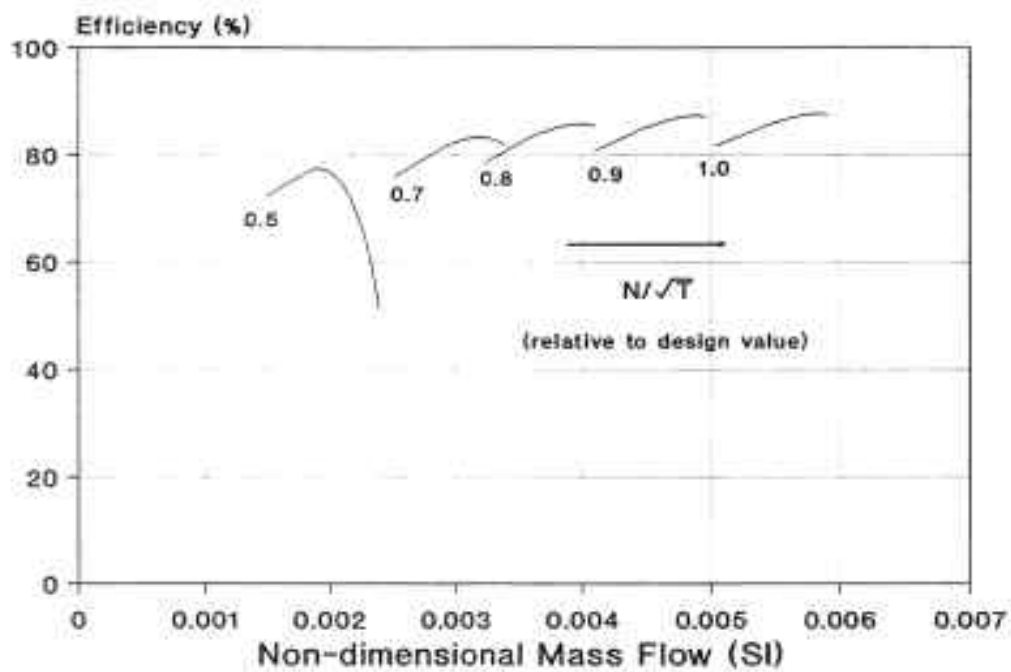


Figure 4.6 HP Compressor.

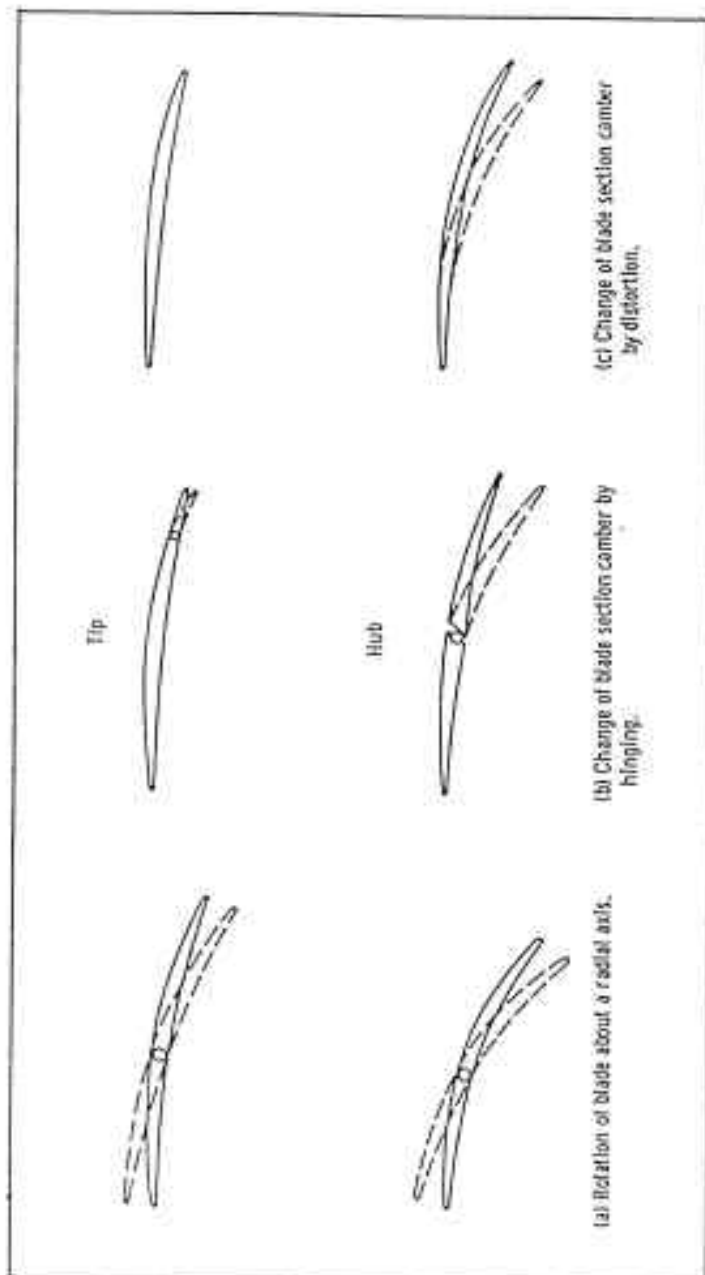


Figure 5.1 Guide Vane and Stator Adjustment Concepts.

Compressor Characteristics

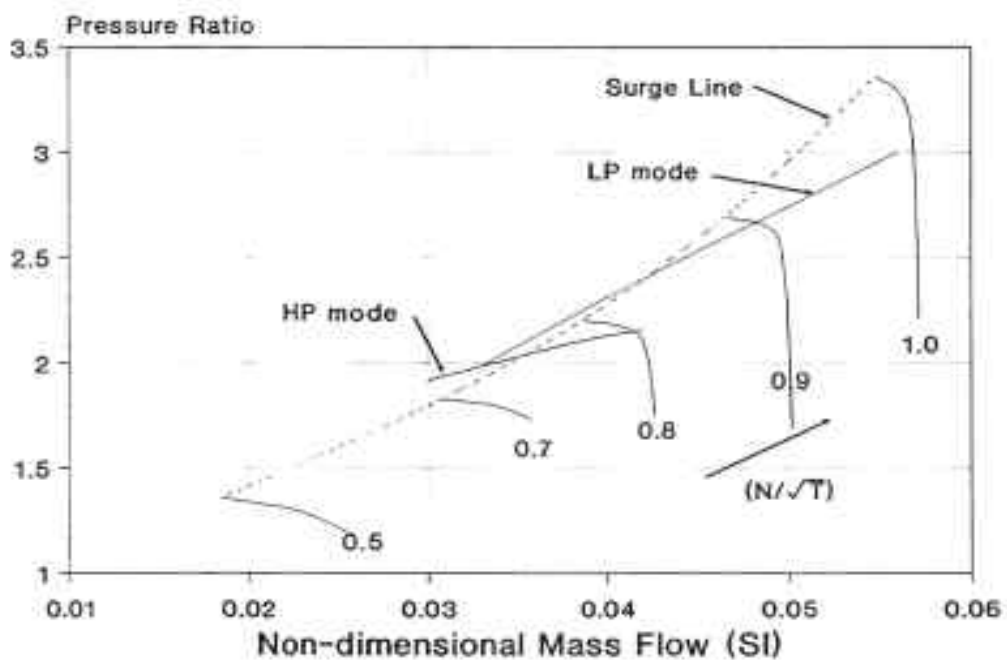
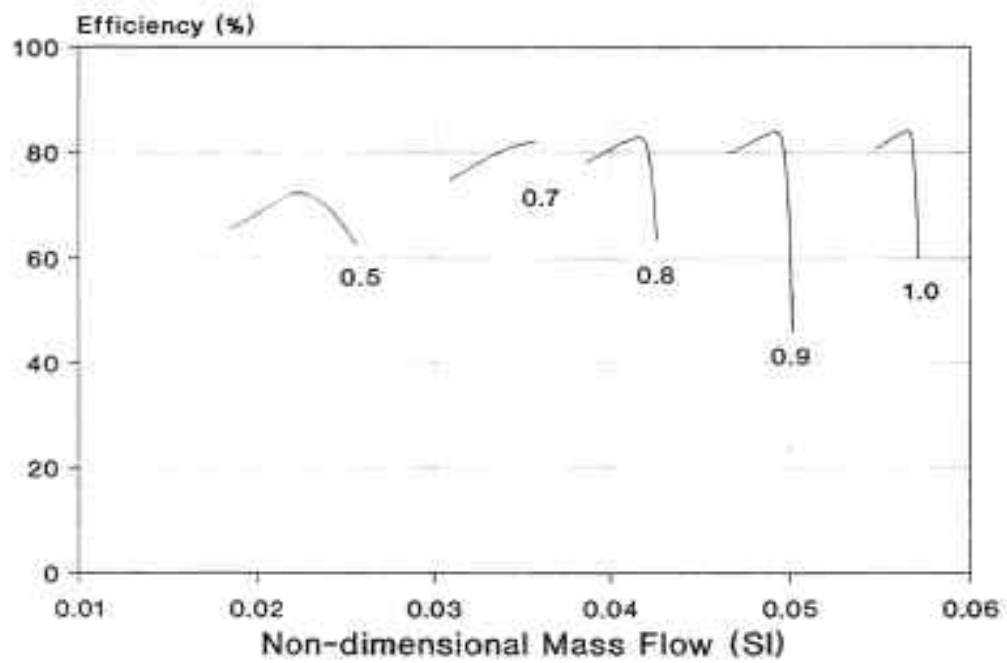


Figure 5.2 LP Compressor Fixed Geometry.

Compressor Characteristics

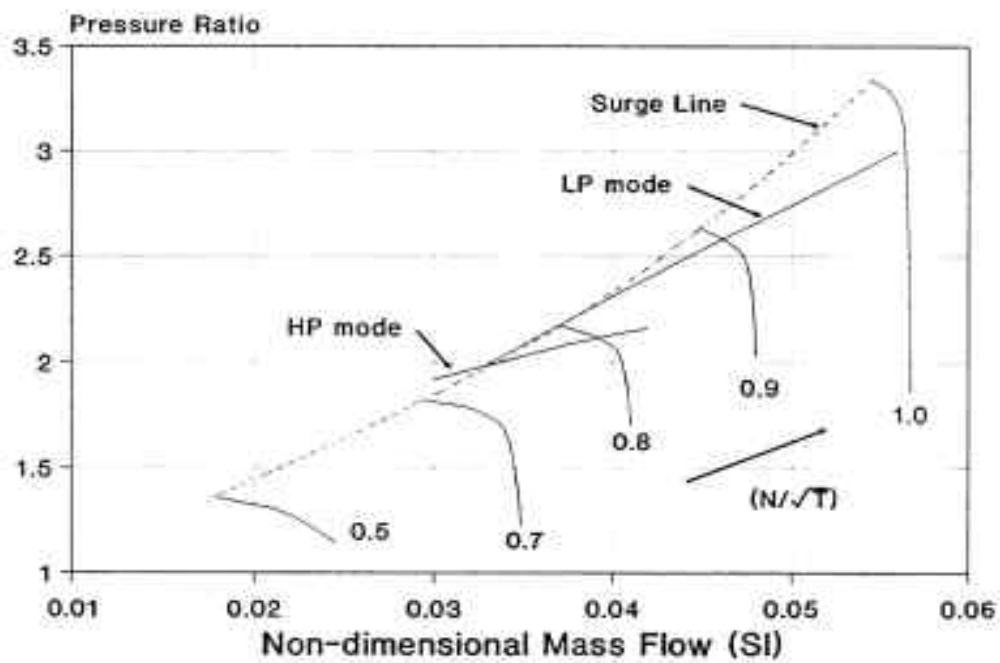
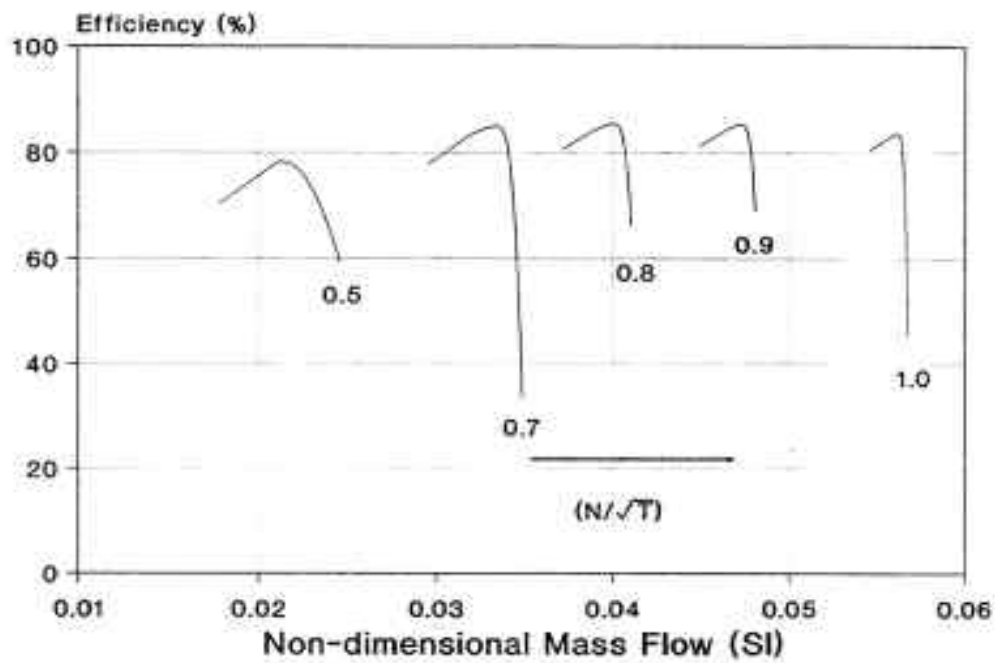


Figure 5.3 LP Compressor Variable IGV.

Compressor Characteristics

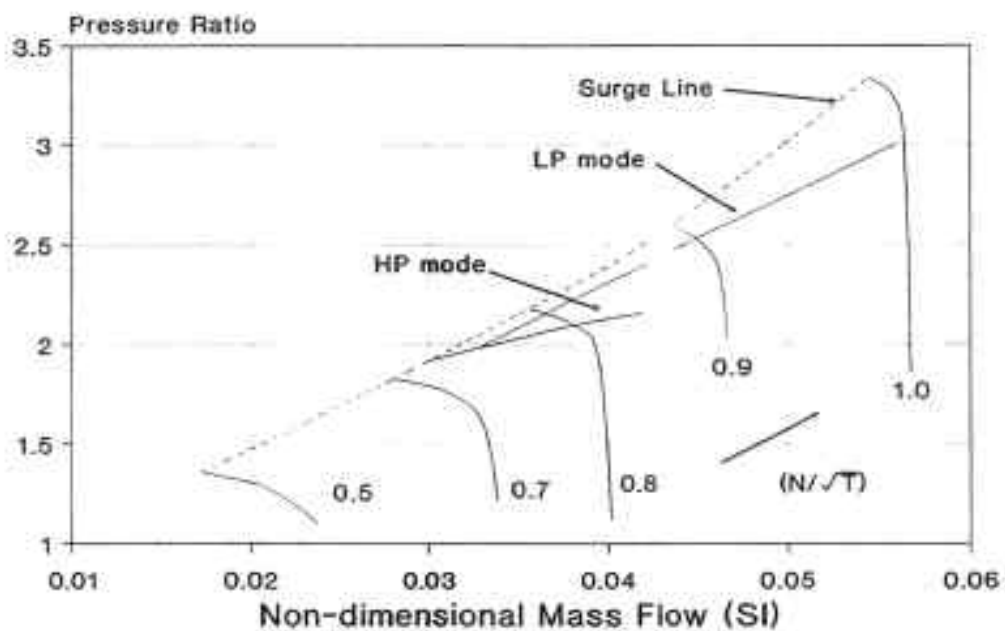
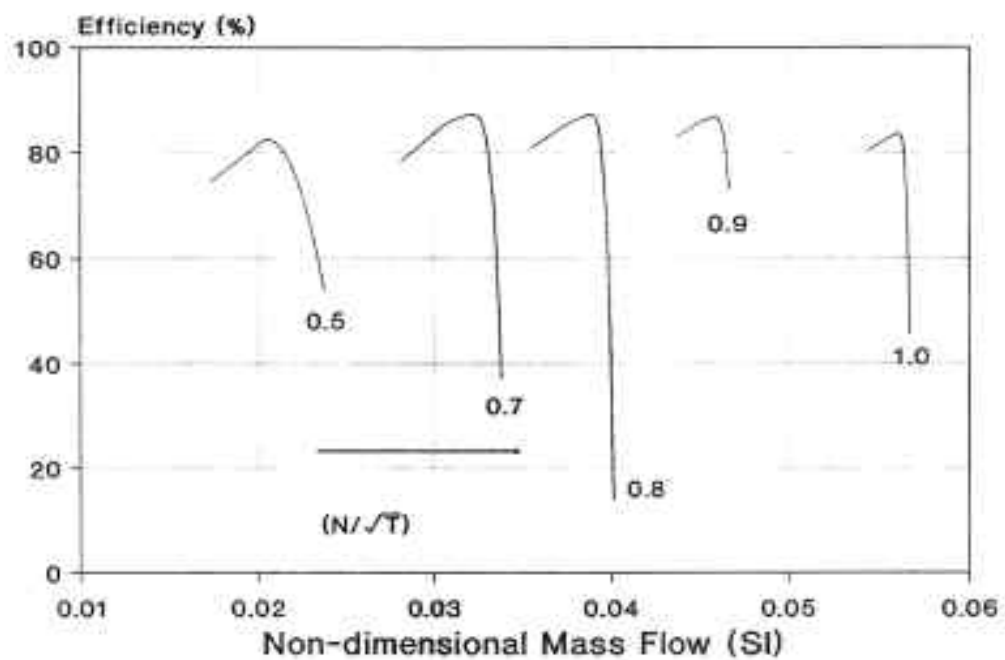


Figure 5.4 LP Compressor Variable IGV+ 1stator.

Compressor Characteristics

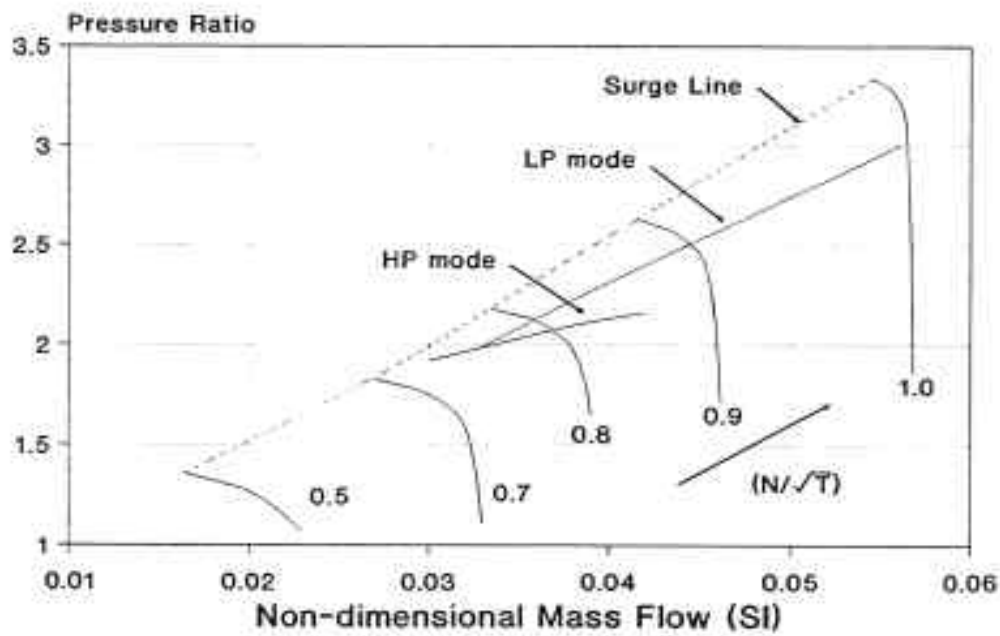
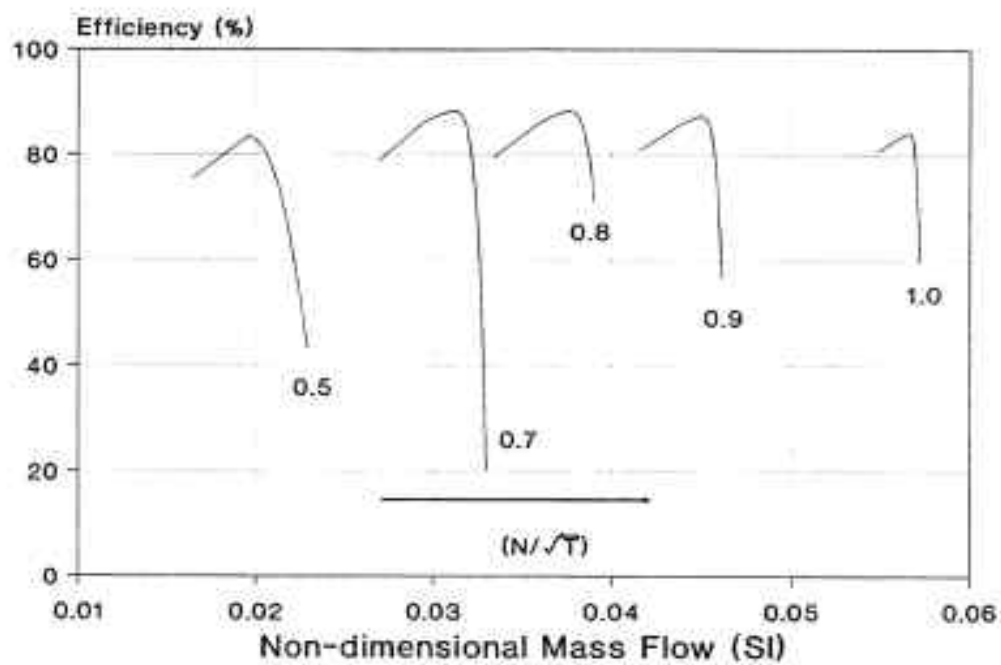


Figure 5.5 LP Compressor Variable IGV+
1stator+2stator.

Restaggering

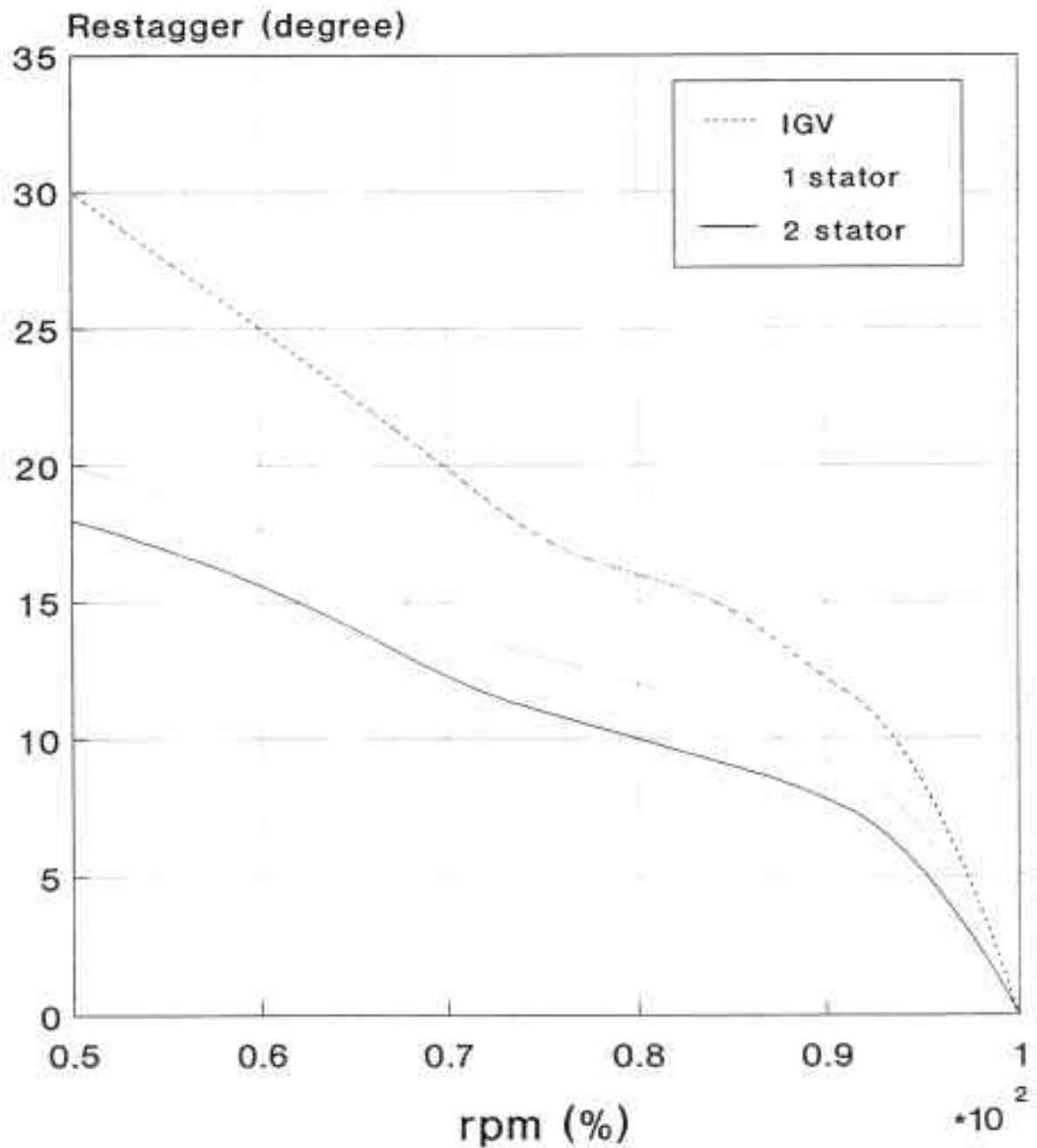


Figure 5.6 LP Compressor Schedule of Variable Stators.

Compressor Characteristics

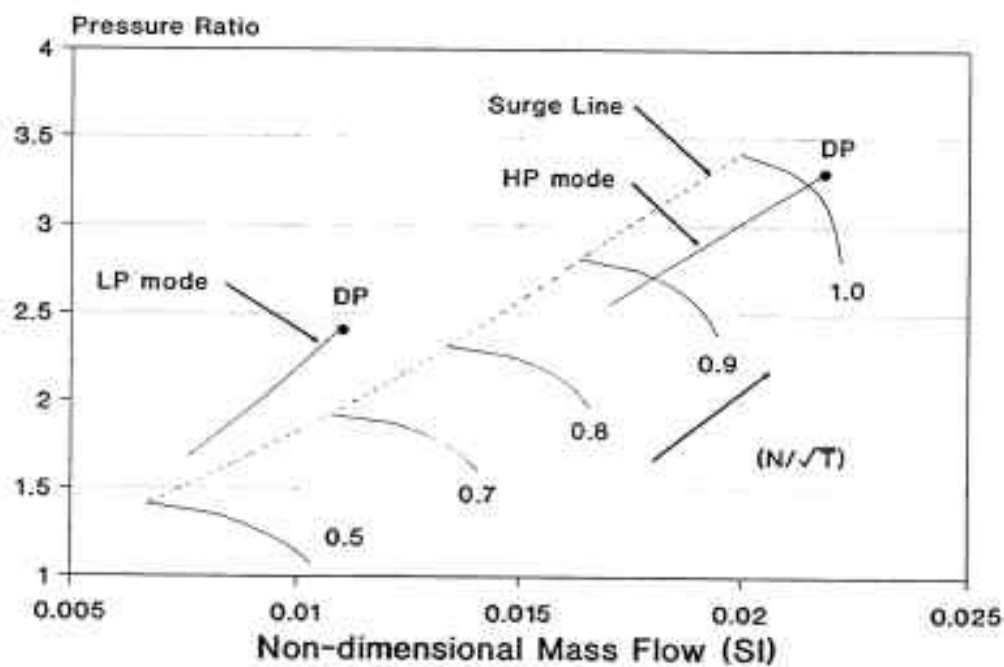
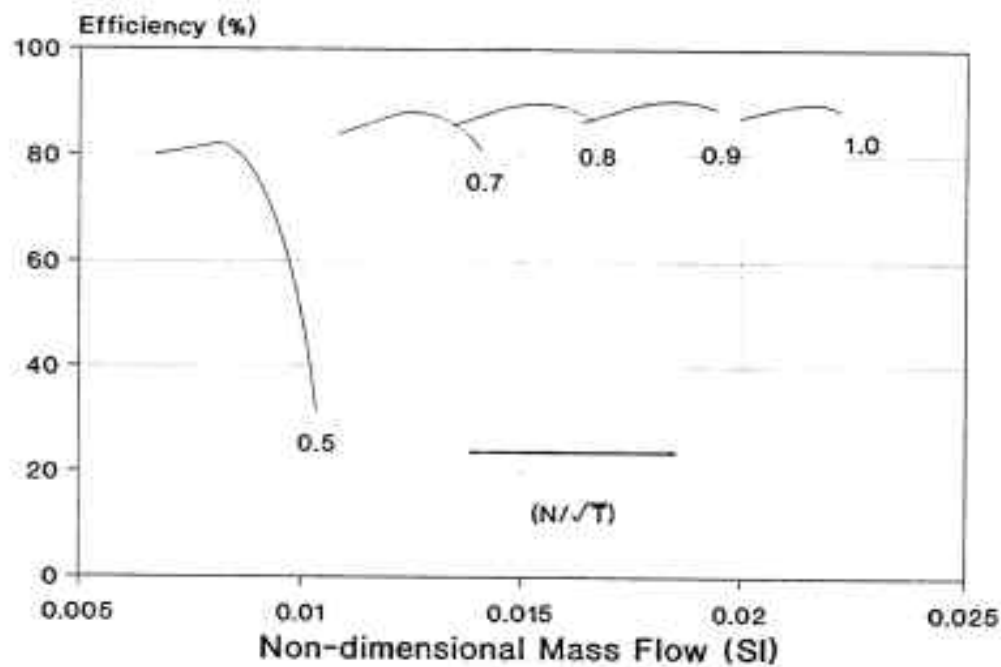


Figure 5.7 IP Compressor Fixed Geometry.

Compressor Characteristics

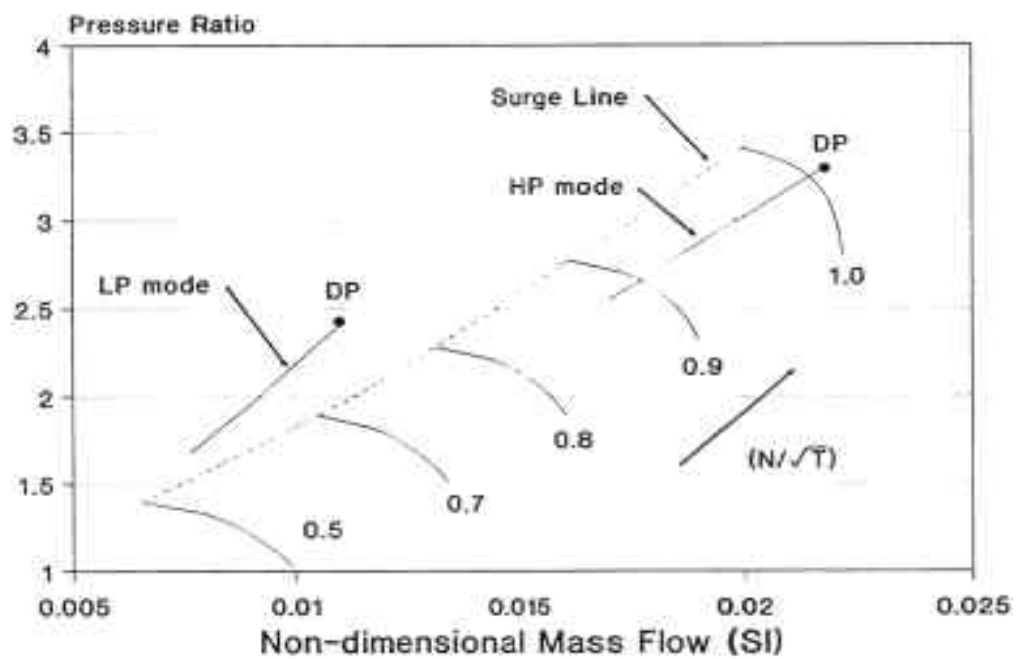
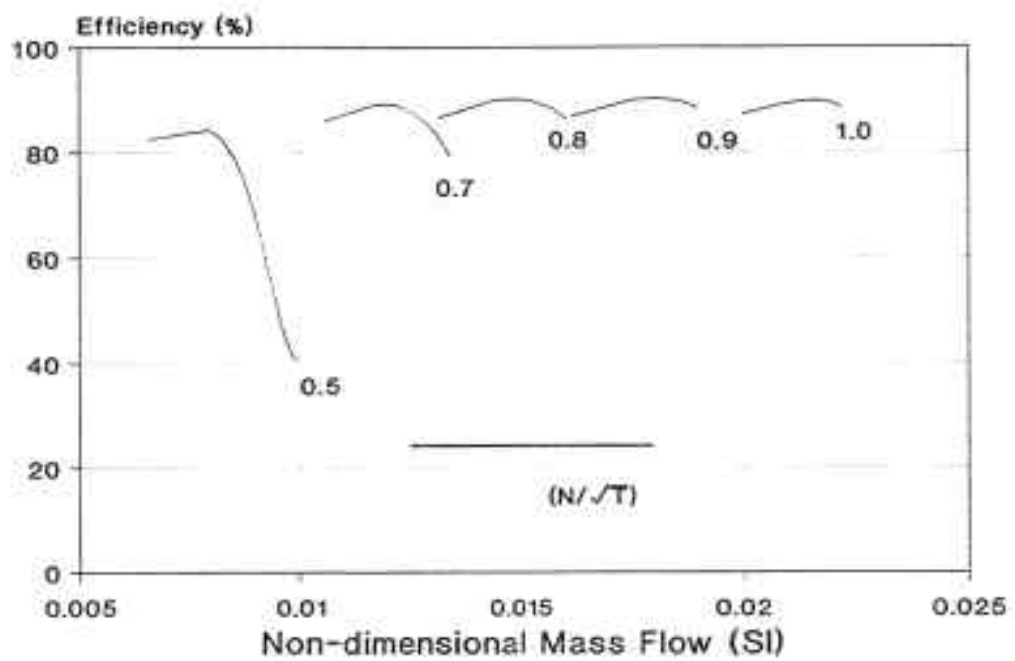


Figure 5.8 IP Compressor Variable IGV.

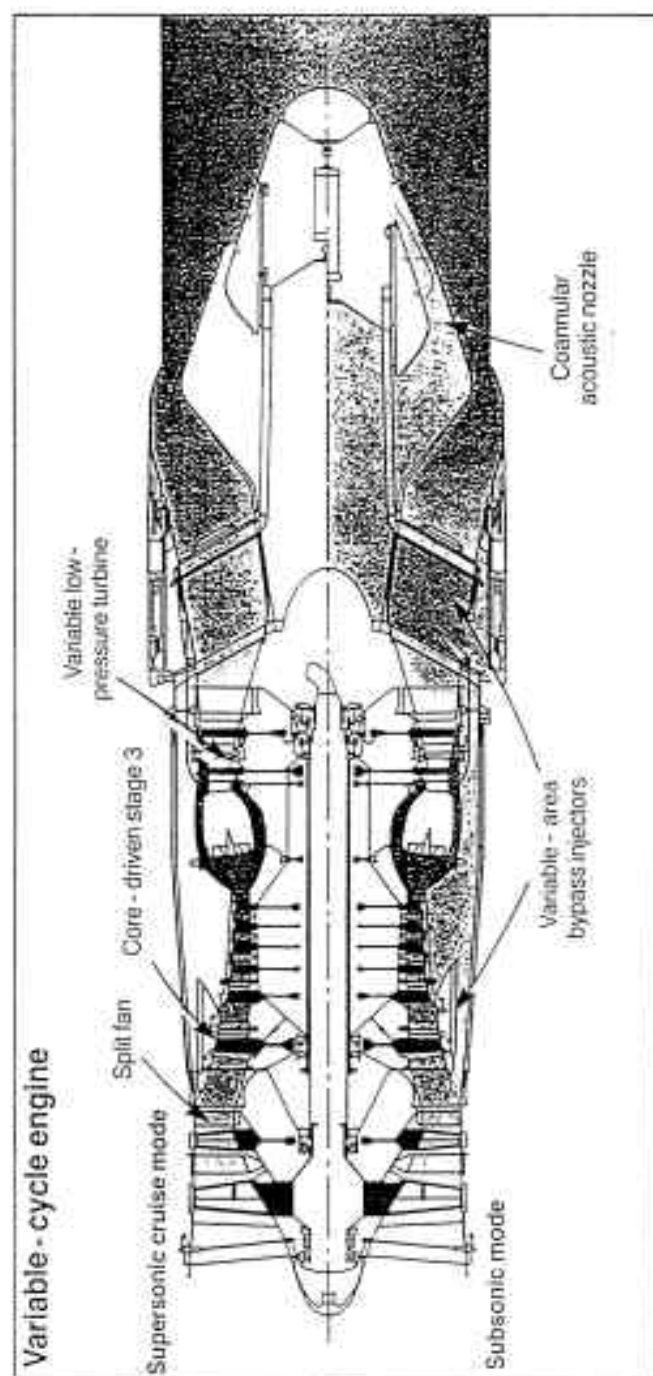


Figure 5.9 G.E. Variable Cycle Engine.

Compressor Characteristics

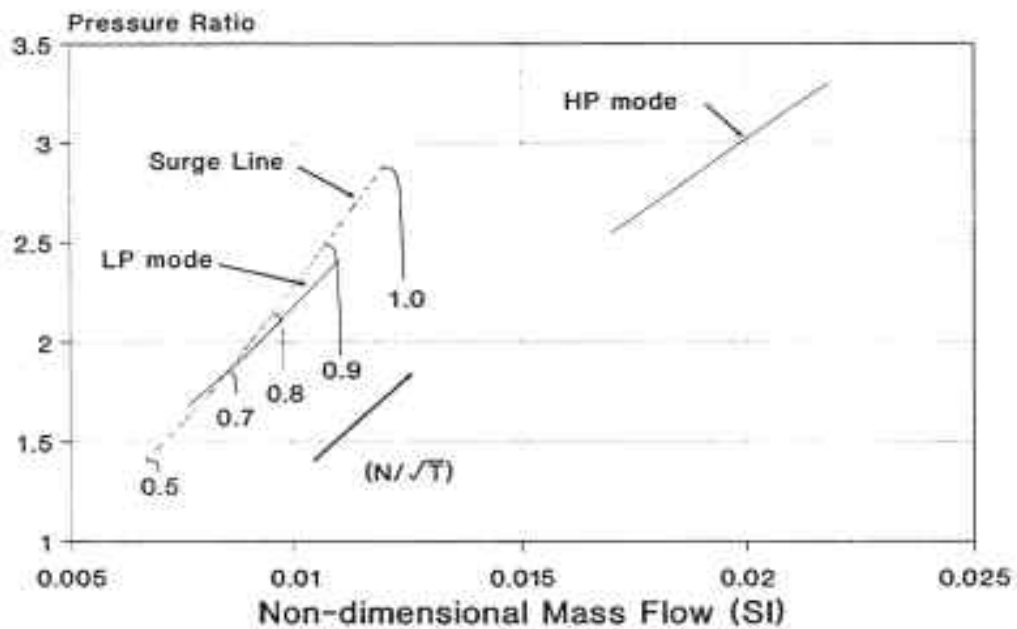
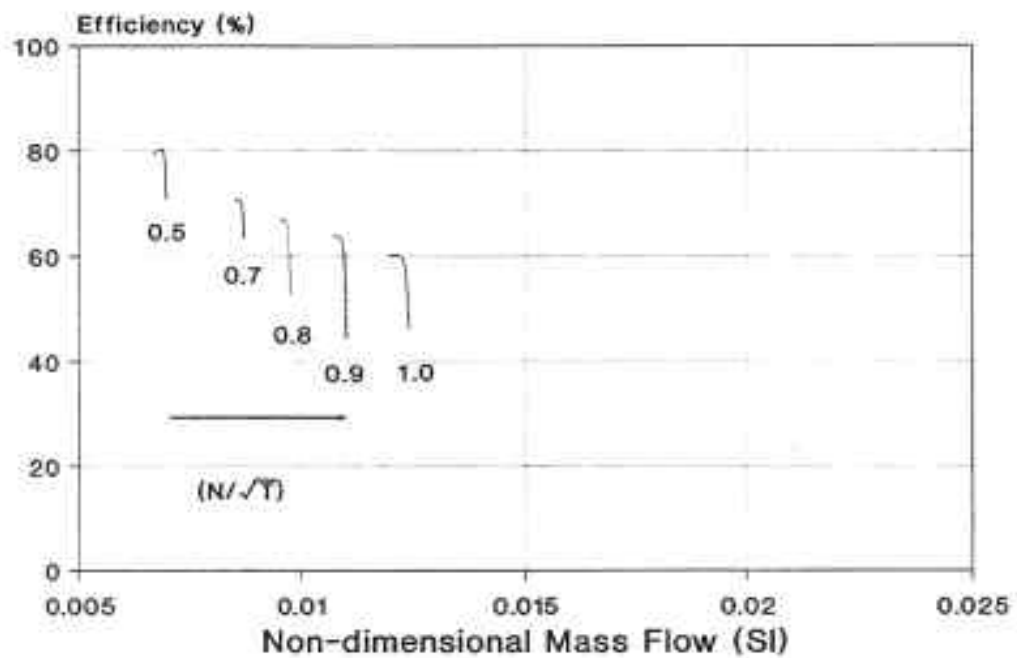


Figure 5.10 IP Compressor Variable Last
Stator by 40 degrees

Compressor Characteristics

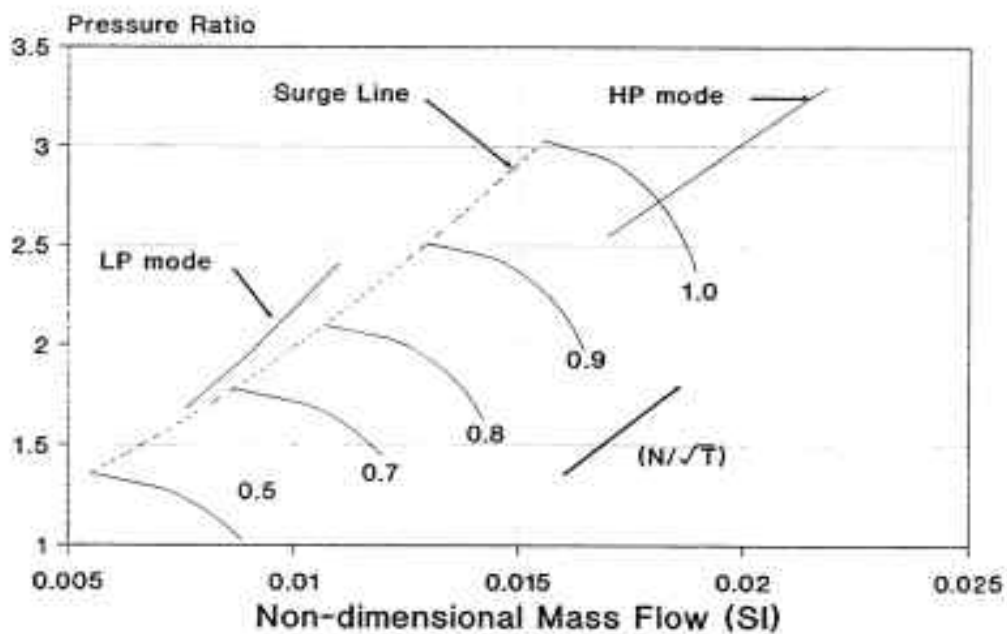
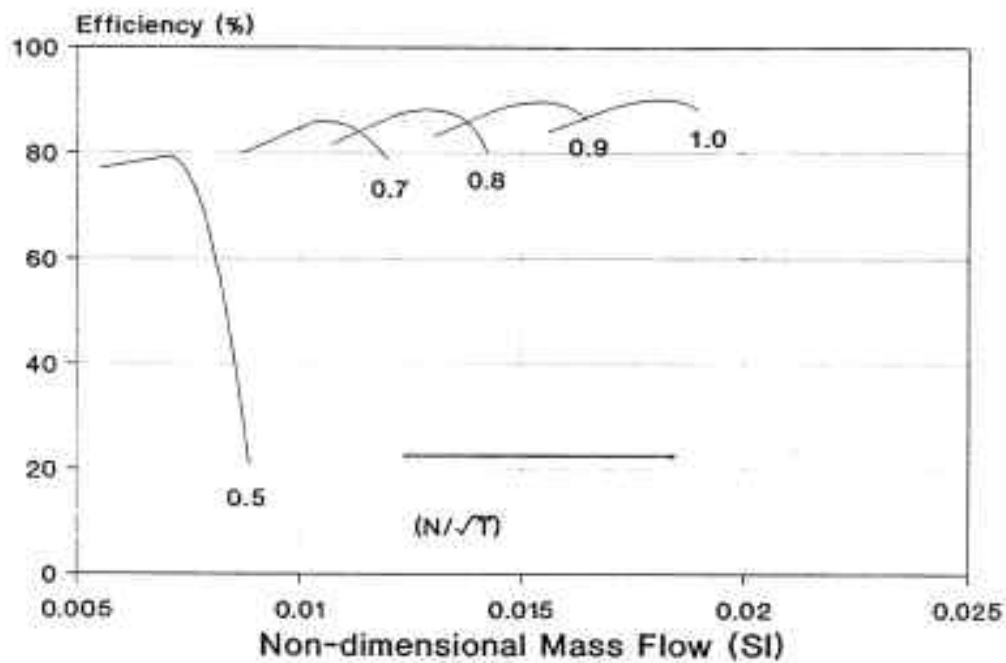


Figure 5.11 IP Compressor Variable All Stator by 10 degrees.

Compressor Characteristics

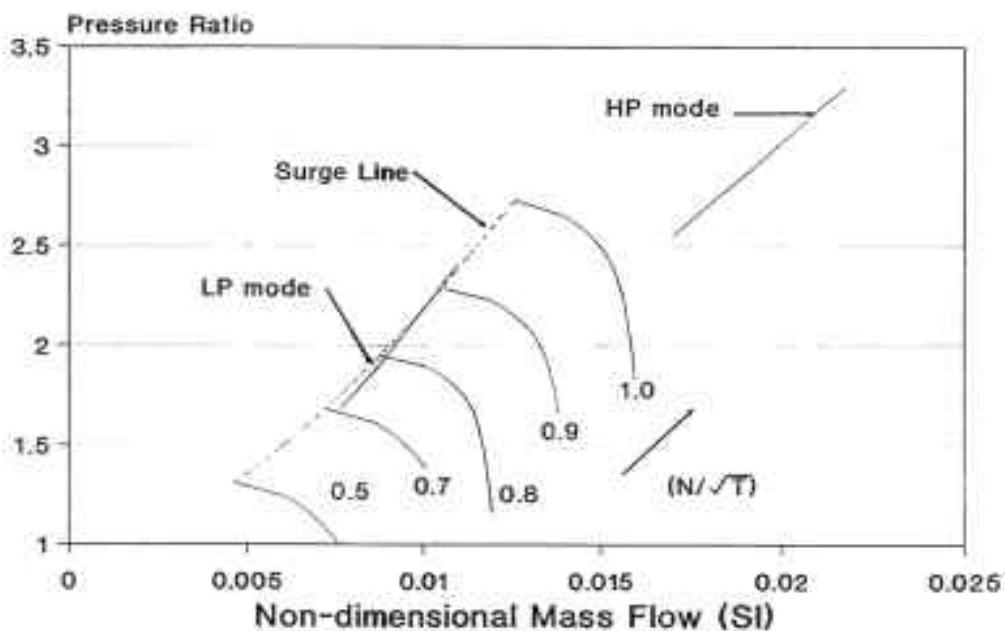
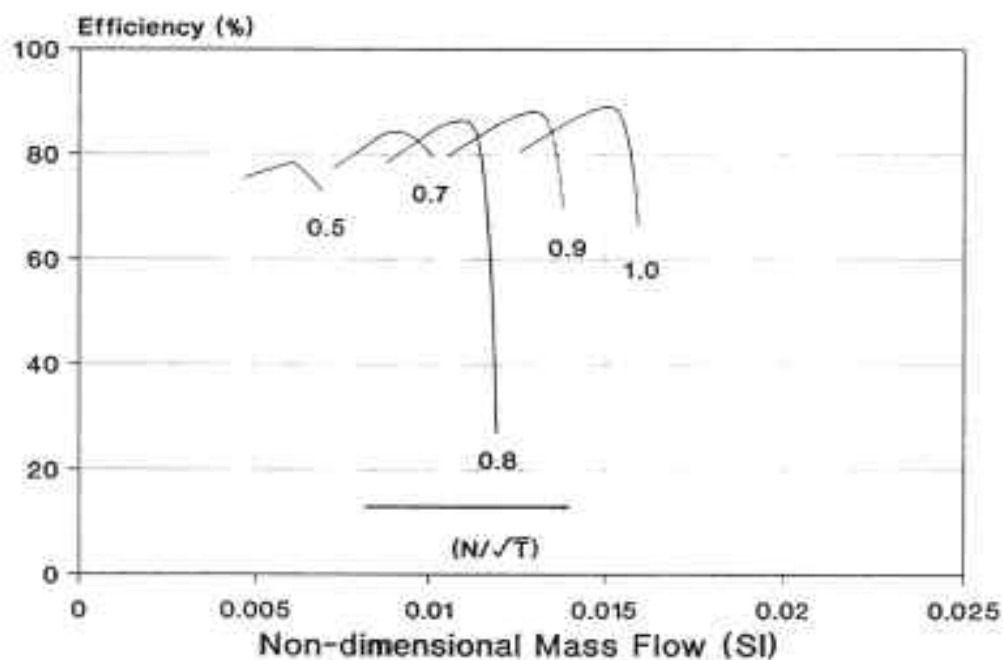


Figure 5.12 IP Compressor Variable All Stators by 20 degrees.

Compressor Characteristics

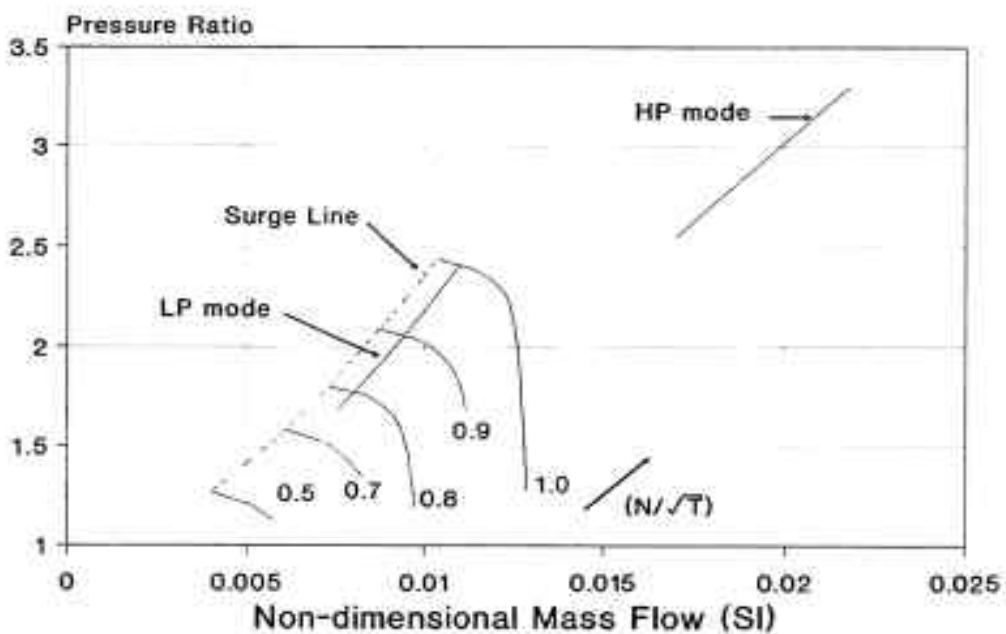
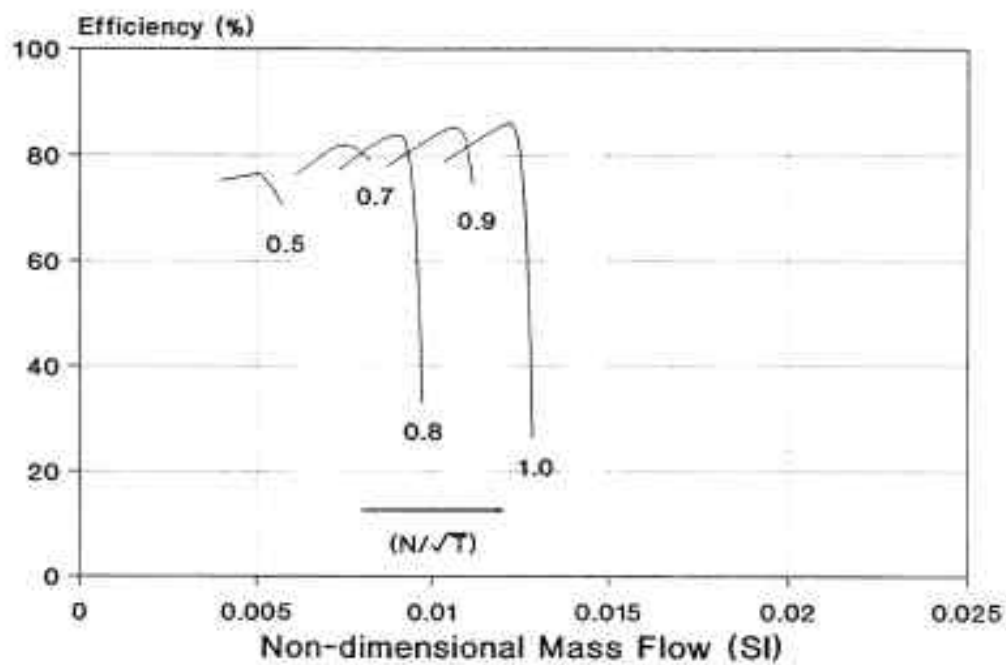


Figure 5.13 IP Compressor Variable All Stators by 30 degrees.

Compressor Characteristics

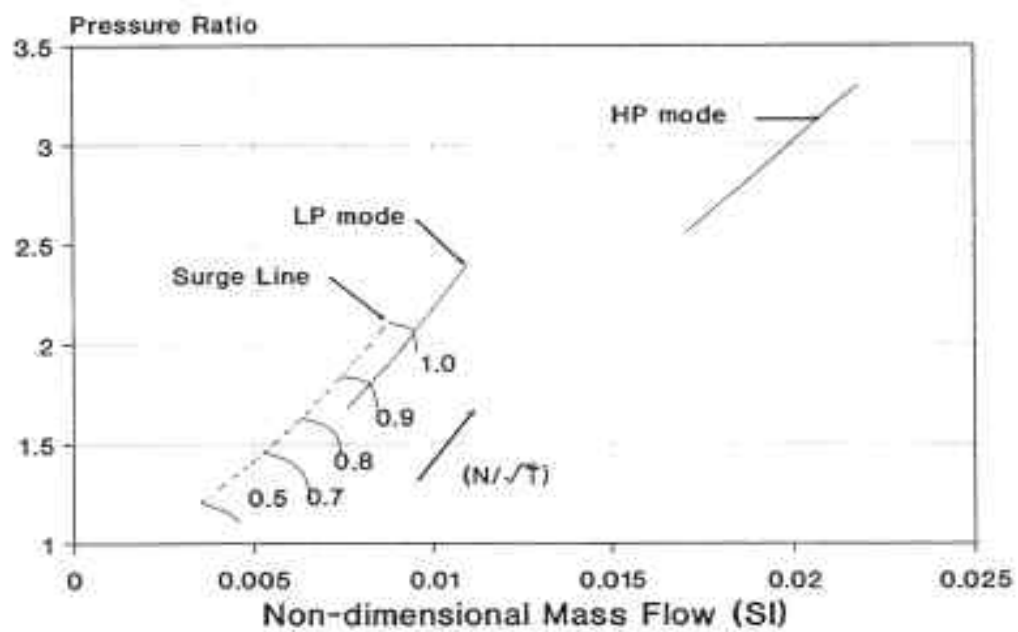
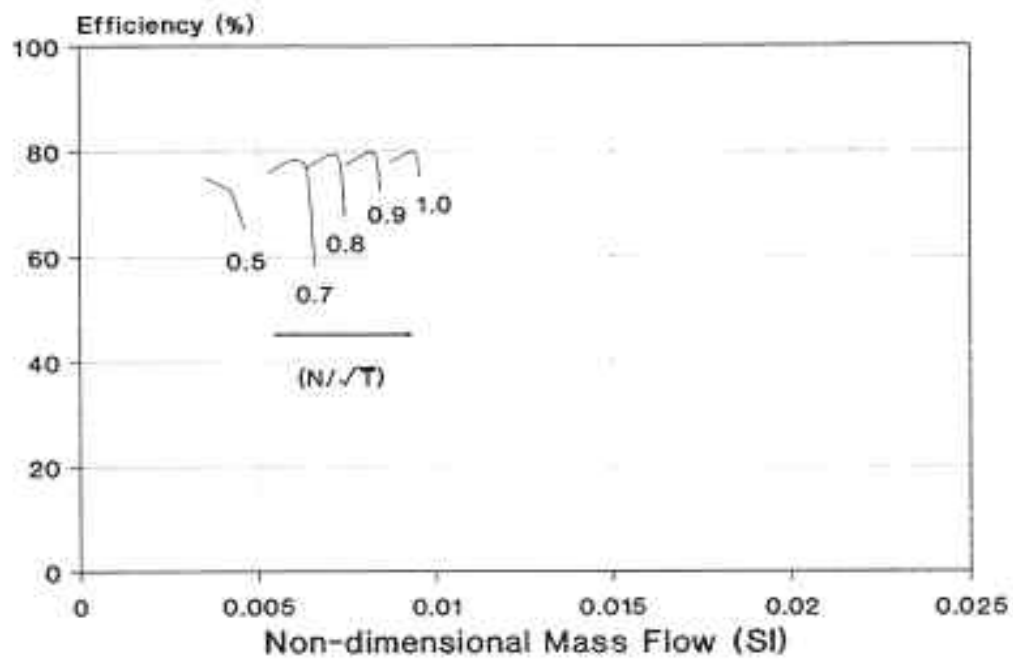


Figure 5.14 IP Compressor Variable All
Stators by 40 degrees.

Compressor Characteristics

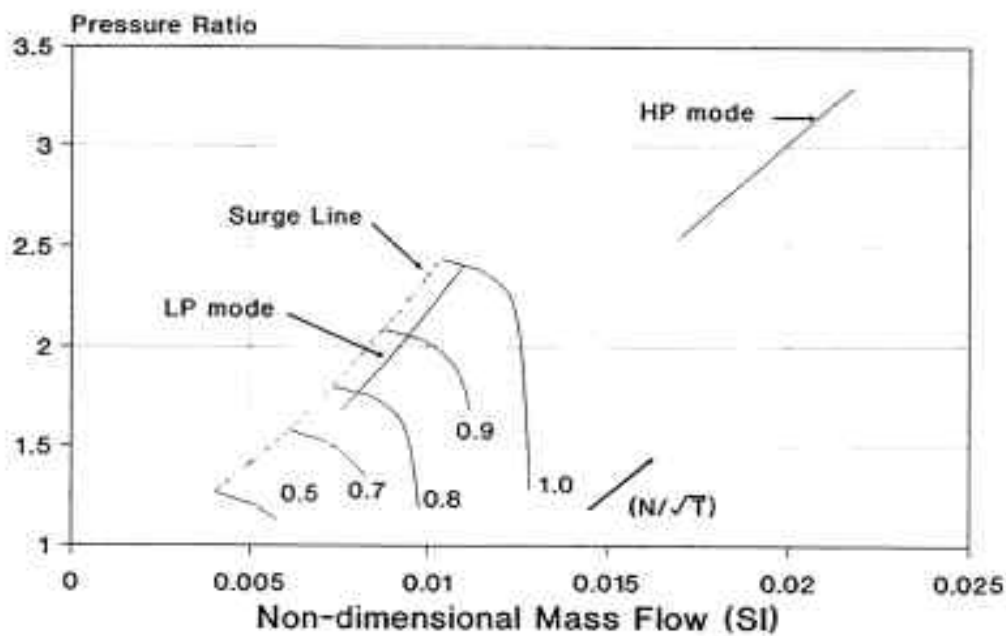
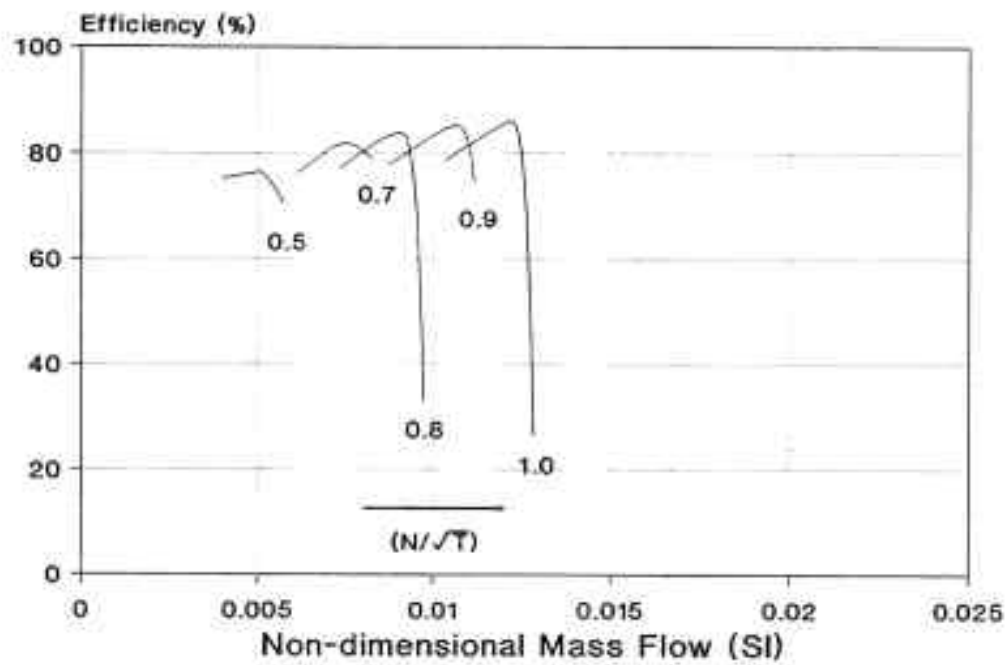


Figure 5.13 IP Compressor Variable All Stators by 30 degrees.

Compressor Characteristics

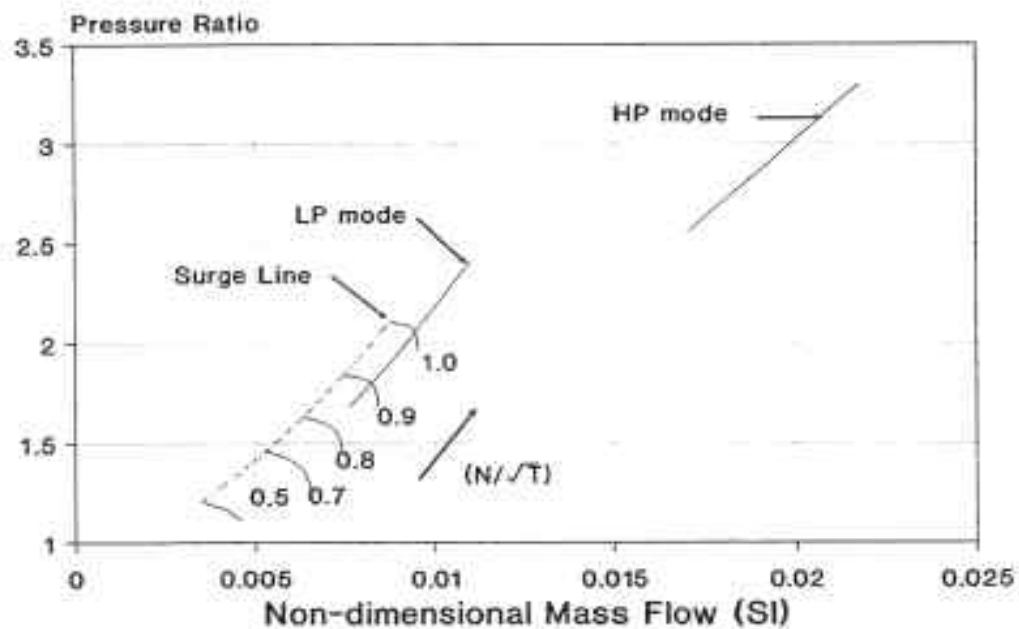
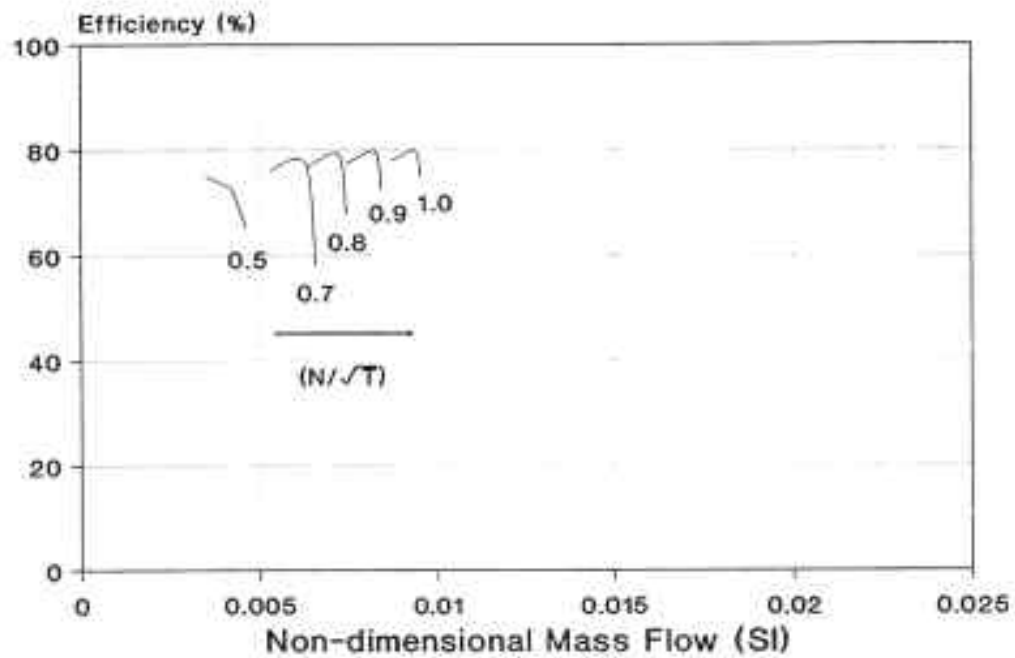


Figure 5.14 IP Compressor Variable All
Stators by 40 degrees.

Compressor Characteristics

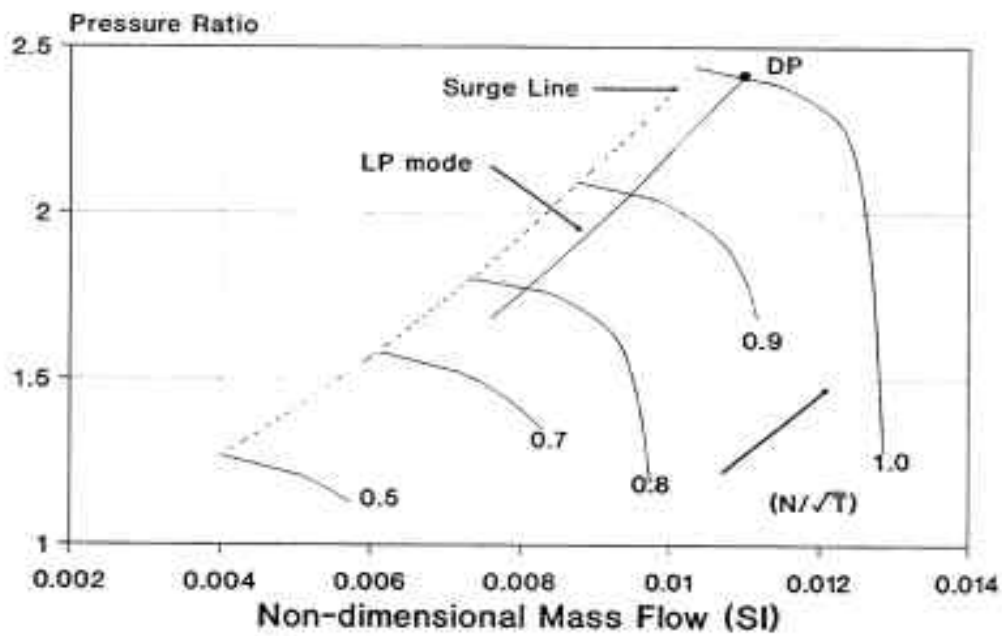
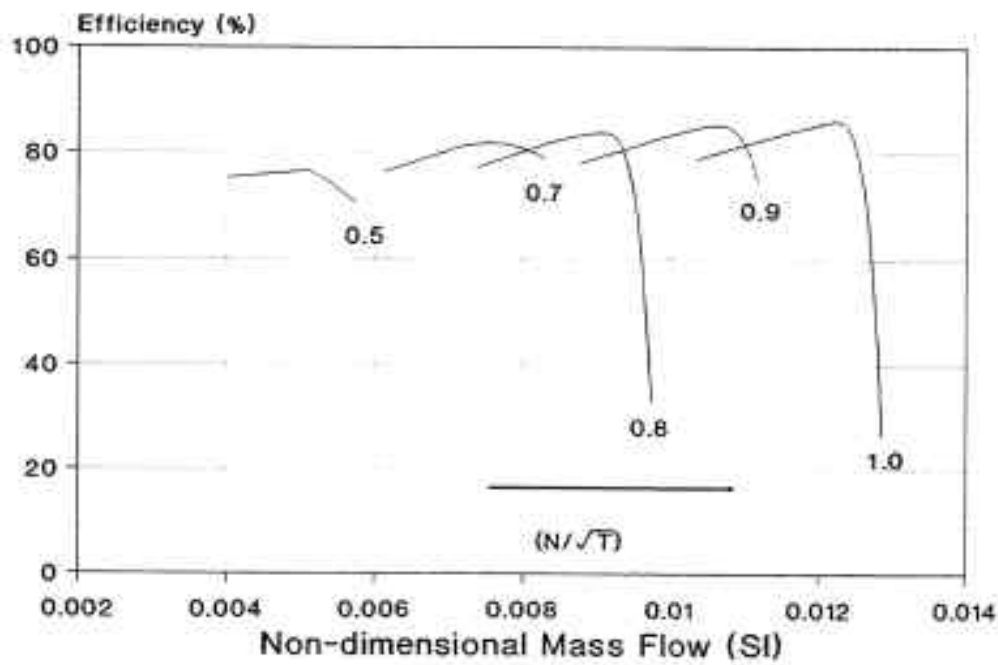


Figure 5.15 IP Compressor Variable All Stators by 30 degrees (only LP mode).

Compressor Characteristics

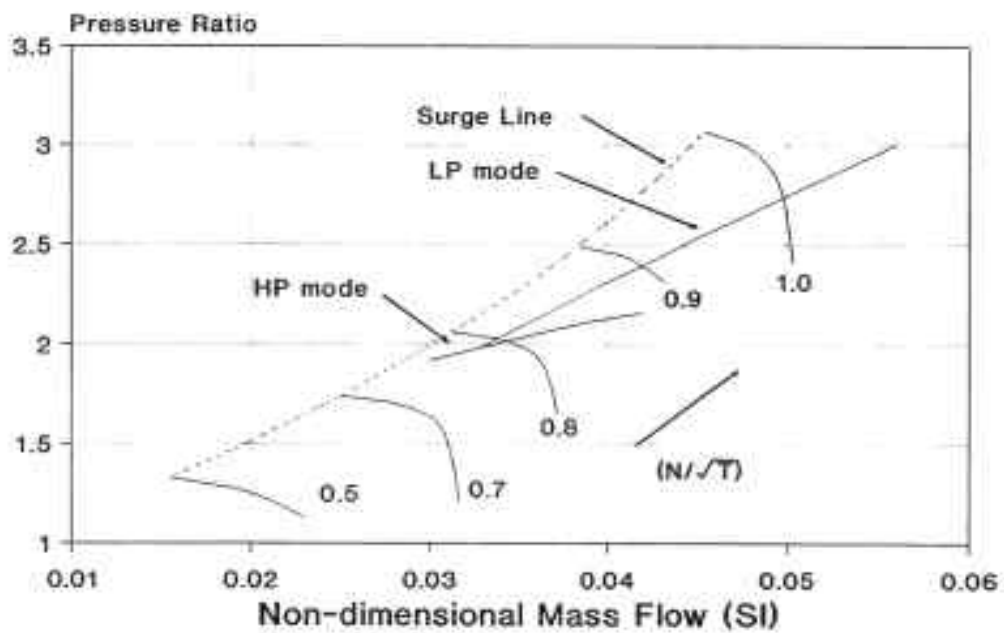
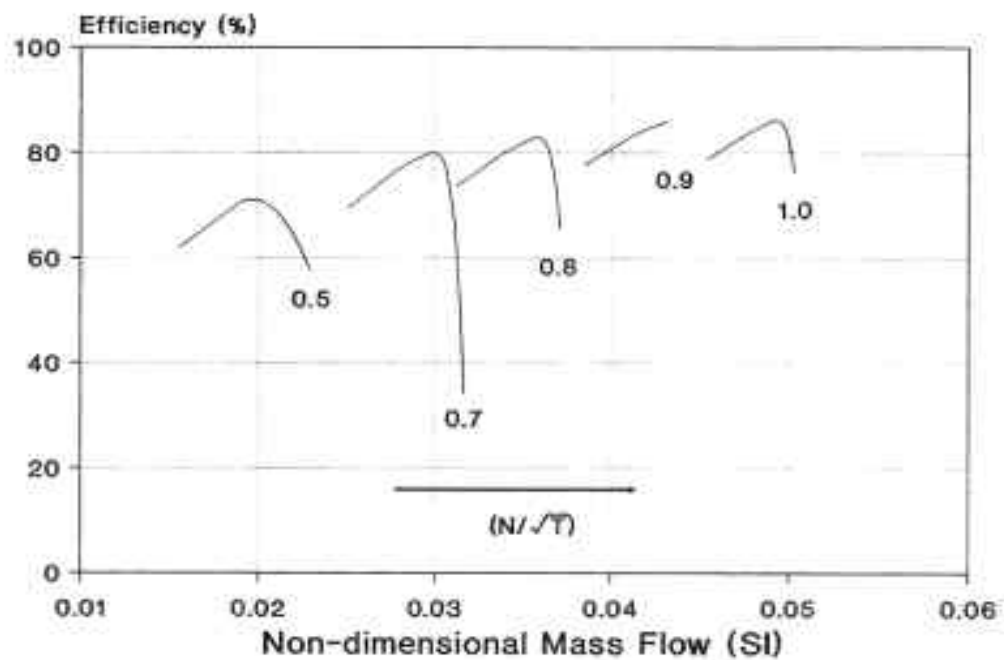


Figure 5.16 LP Compressor Variable All Stators by 10 degrees.

Compressor Characteristics

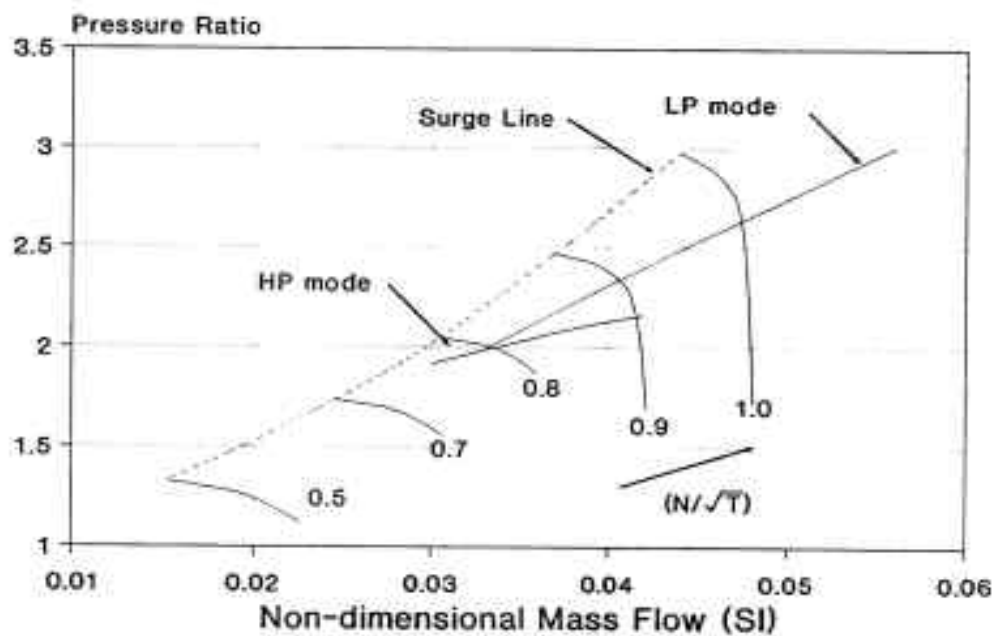
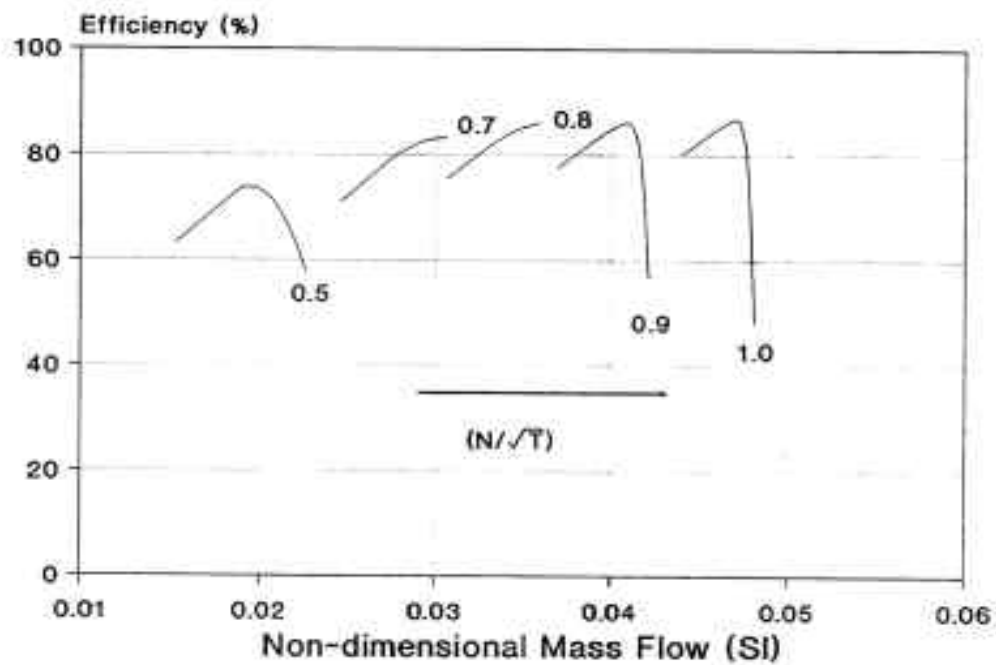


Figure 5.17 LP Compressor Variable IGV by 20 degrees•All Stators by 10 degrees

Compressor Characteristics

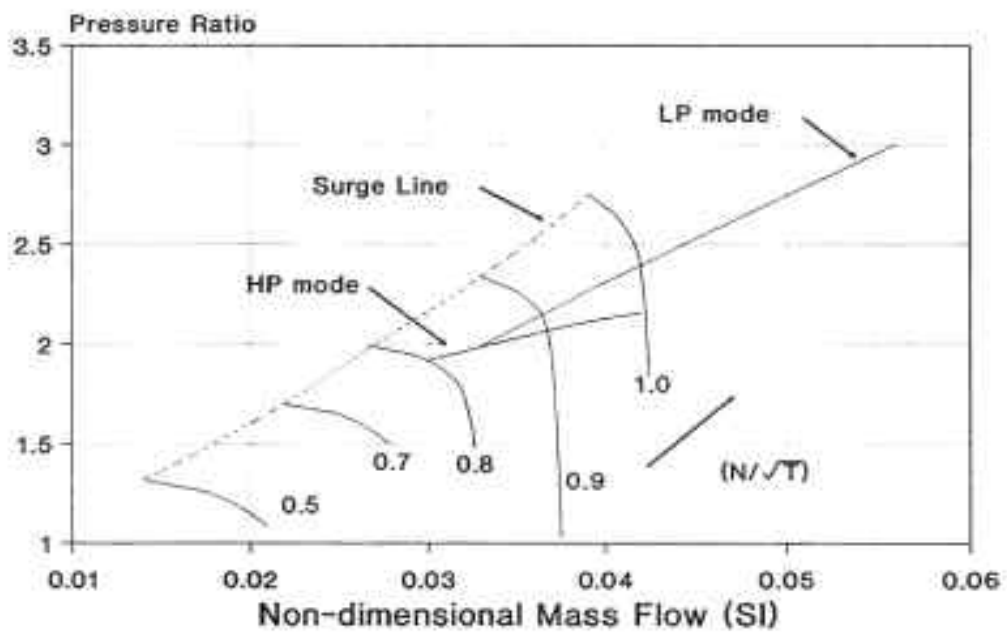
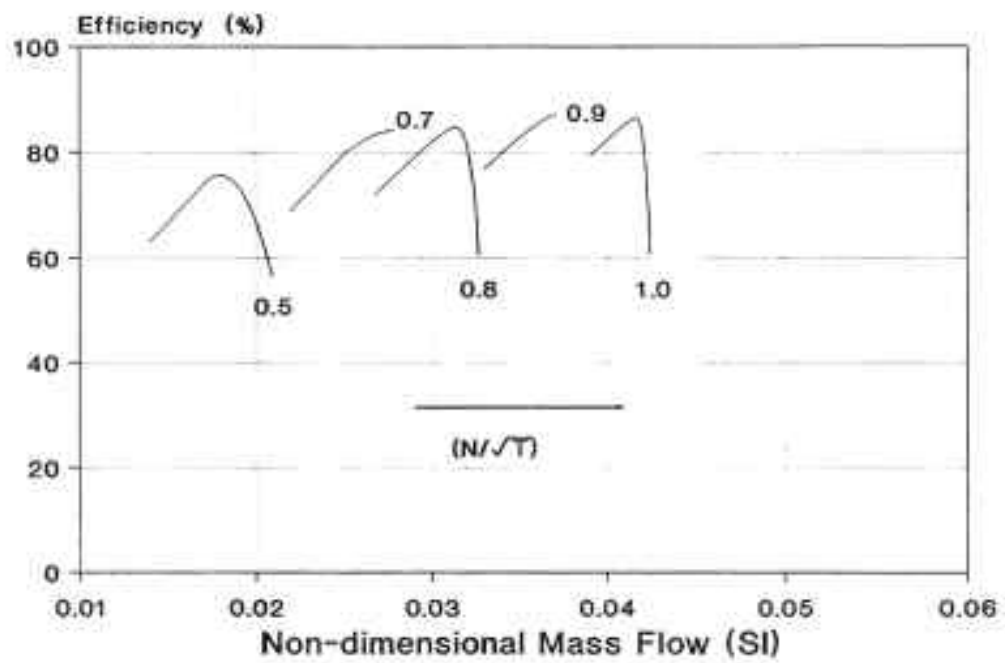


Figure 5.21 LP Compressor Variable IGV by 35 degrees•All Stators by 15 degrees.

Compressor Characteristics

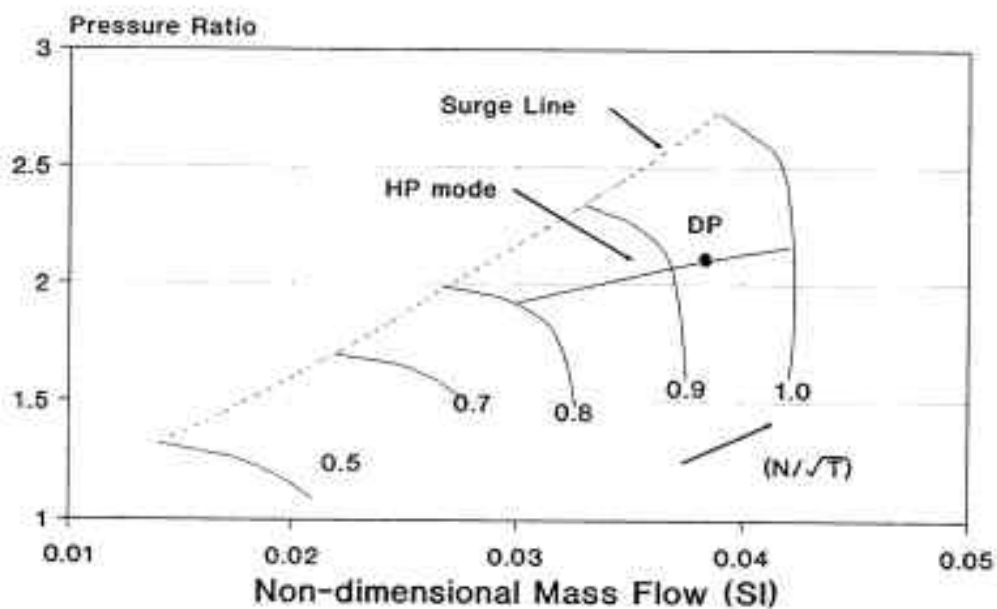
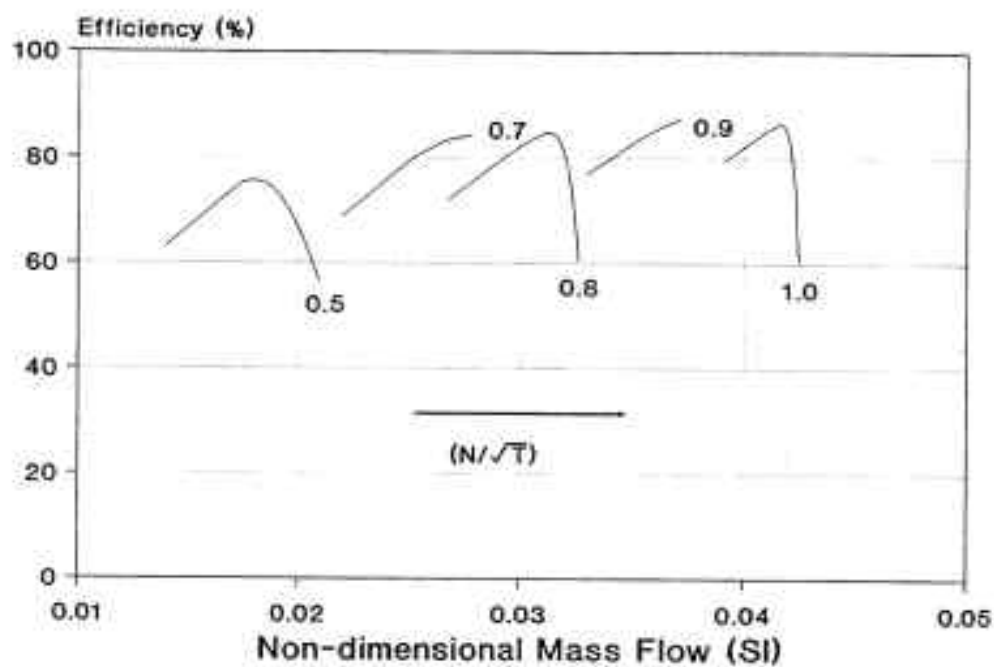


Figure 5.22 LP Compressor Variable IGV
by 35 degrees+All Stators by 15 degrees
(only HP mode).

COMPRESSOR CHARACTERISTICS

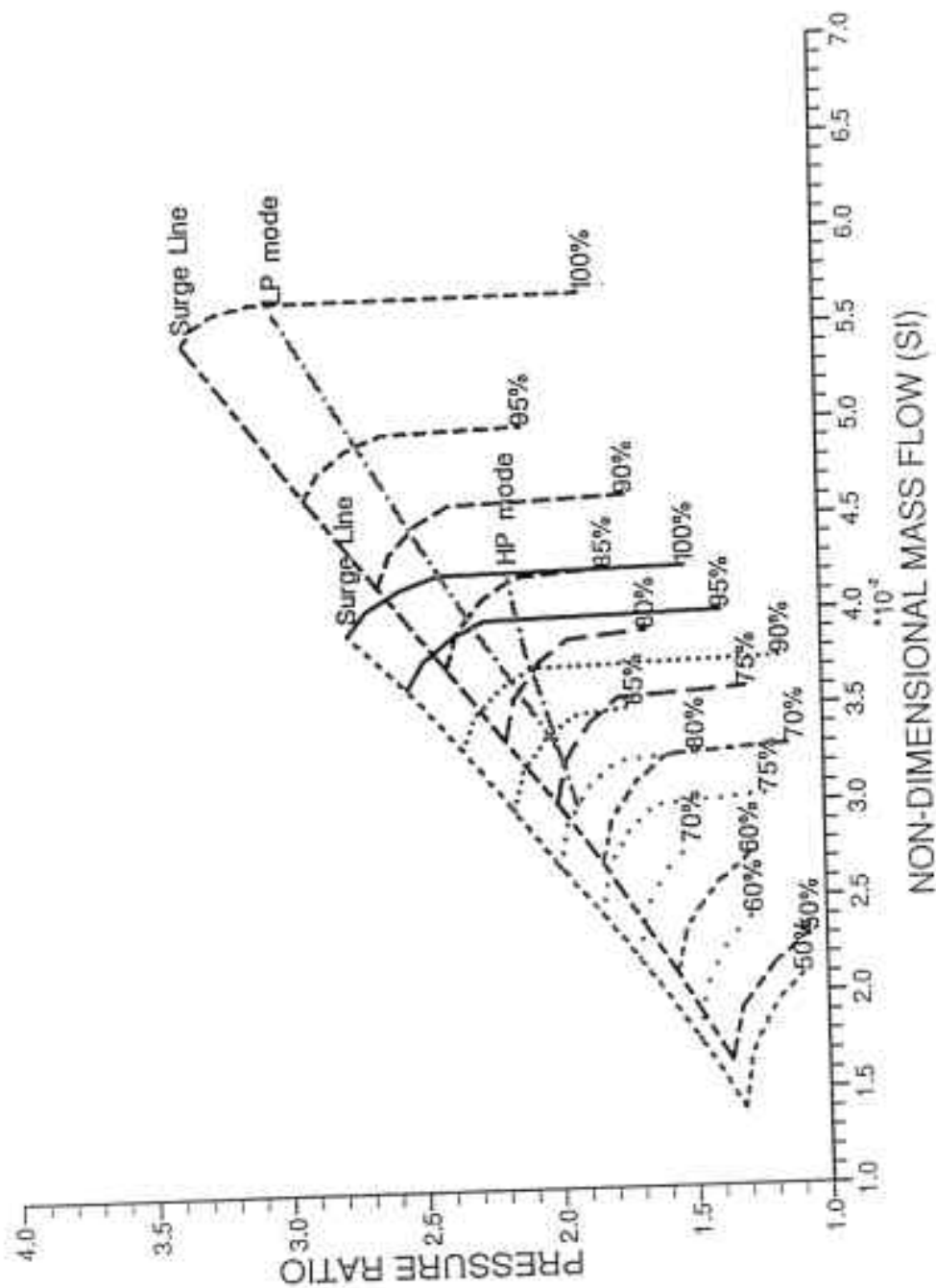


Figure 5.23 LP Compressor Variable Geometry.

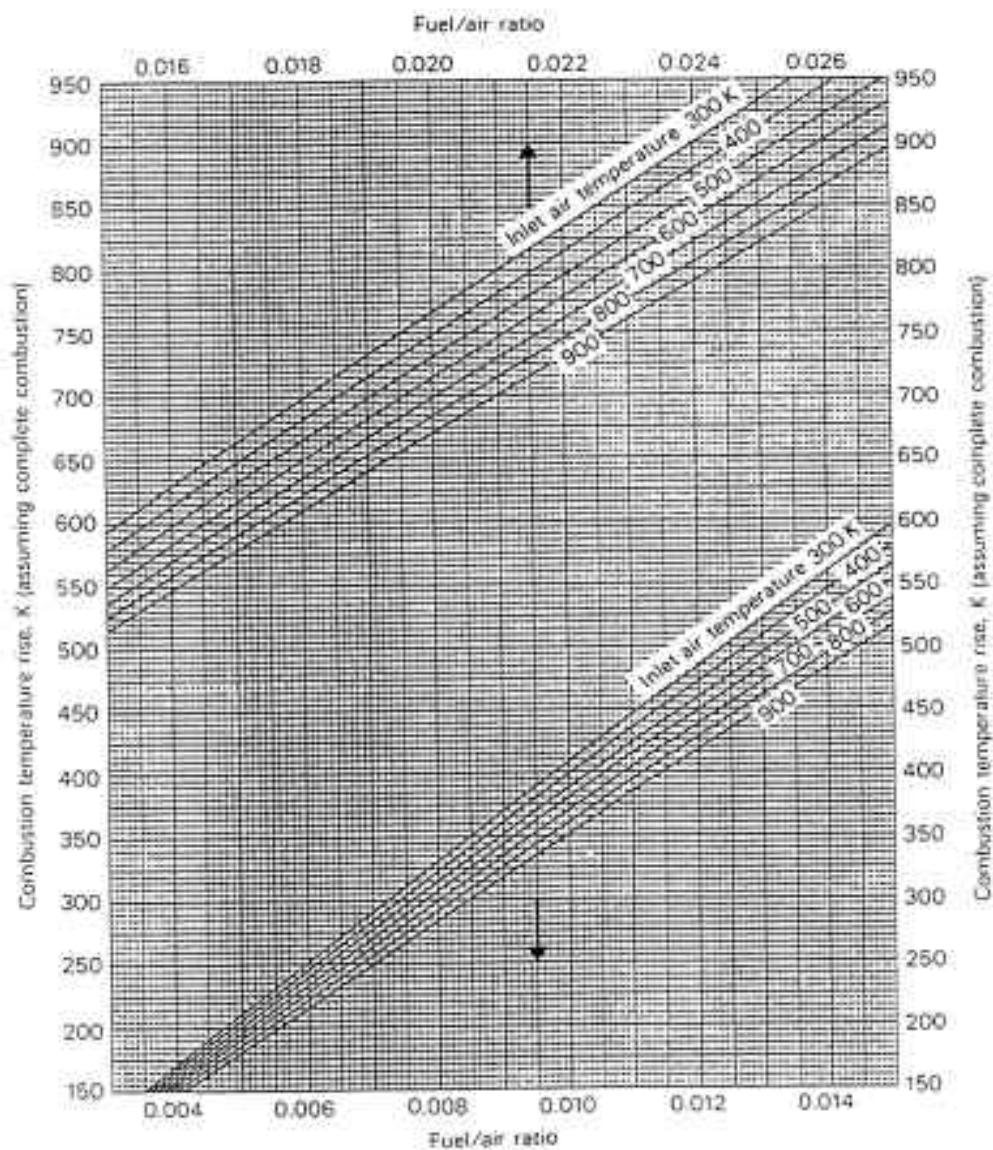


Figure 6.1 Combustion Temperature Rise versus Fuel/air Ratio.

DESIGN AND PERFORMANCE PREDICTION OF AXIAL-FLOW TURBINES

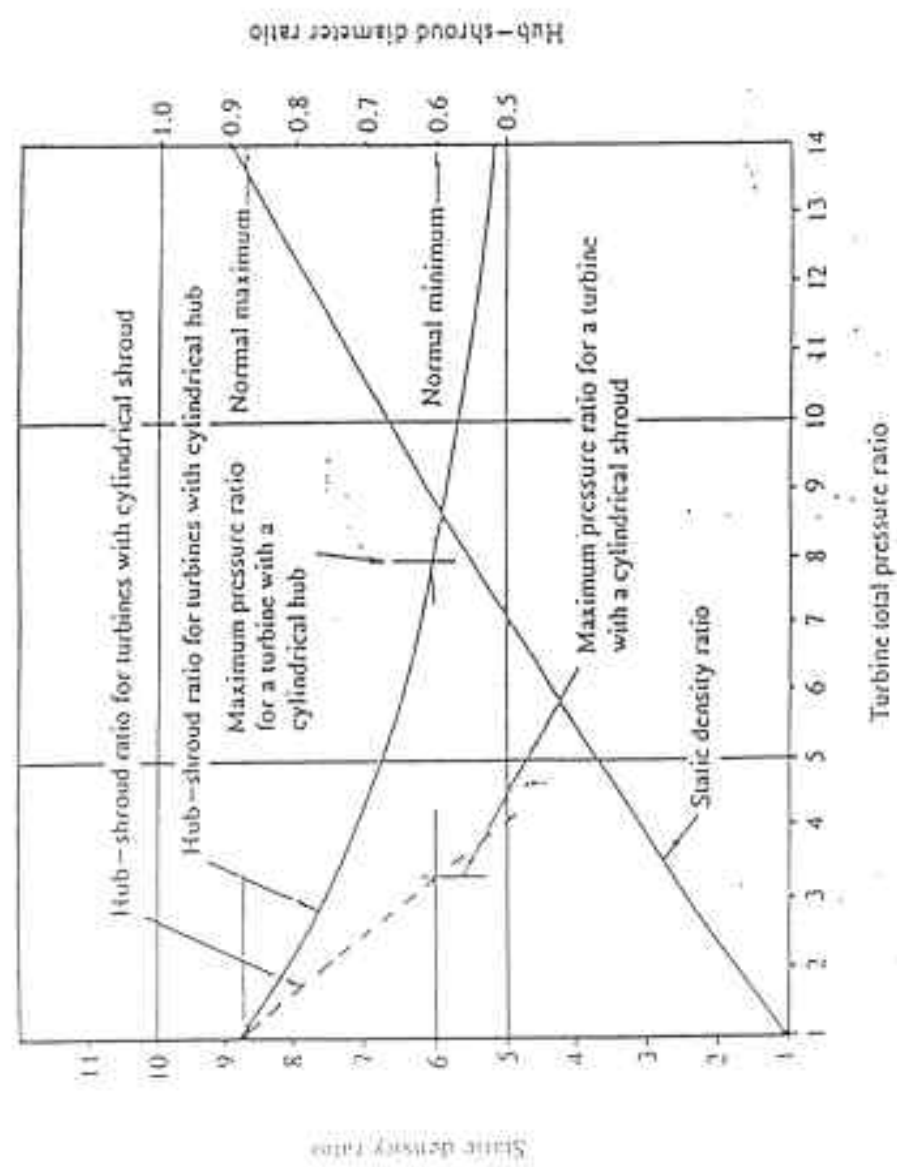


Figure 6.3 Variation of Turbine Static Density Ratio and Hub-shroud Diameter Ratio with Design total Ratio.

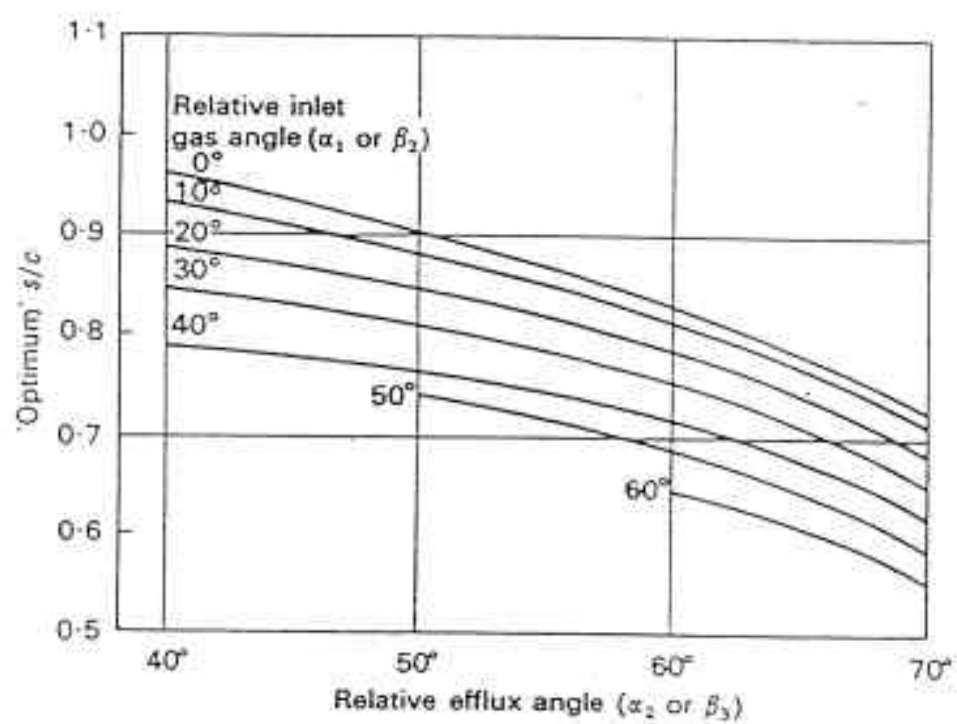


Figure 6.4 'Optimum' Space/chord Ratio.

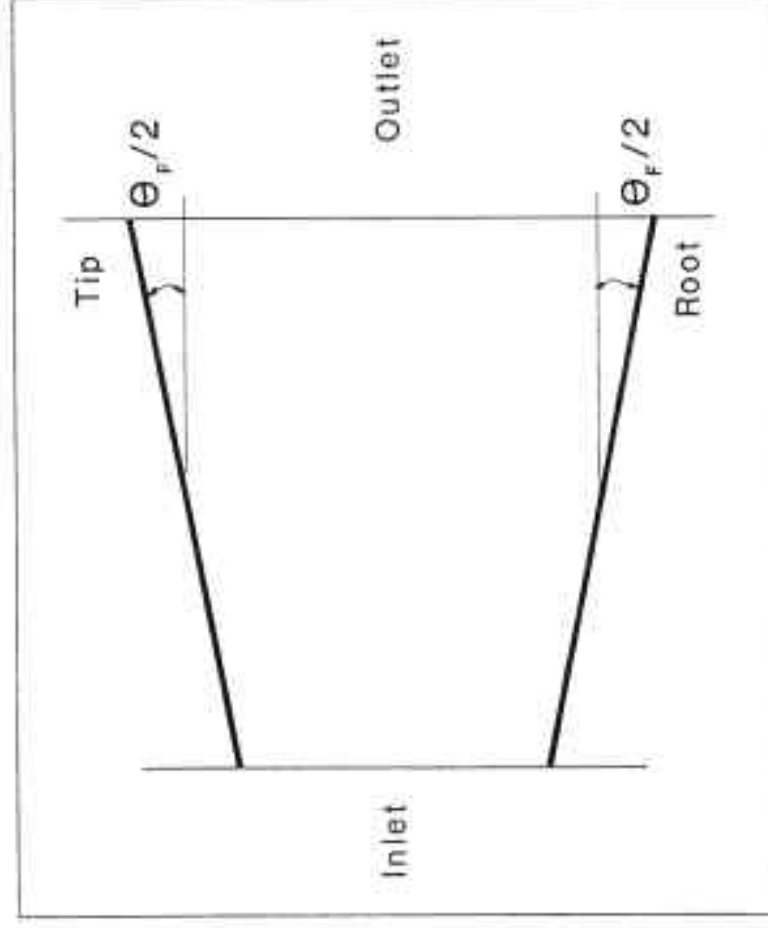
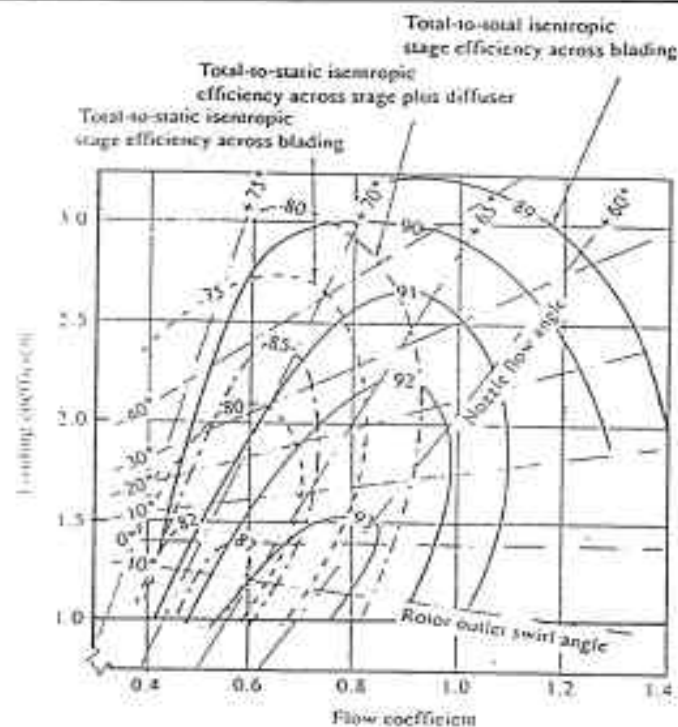
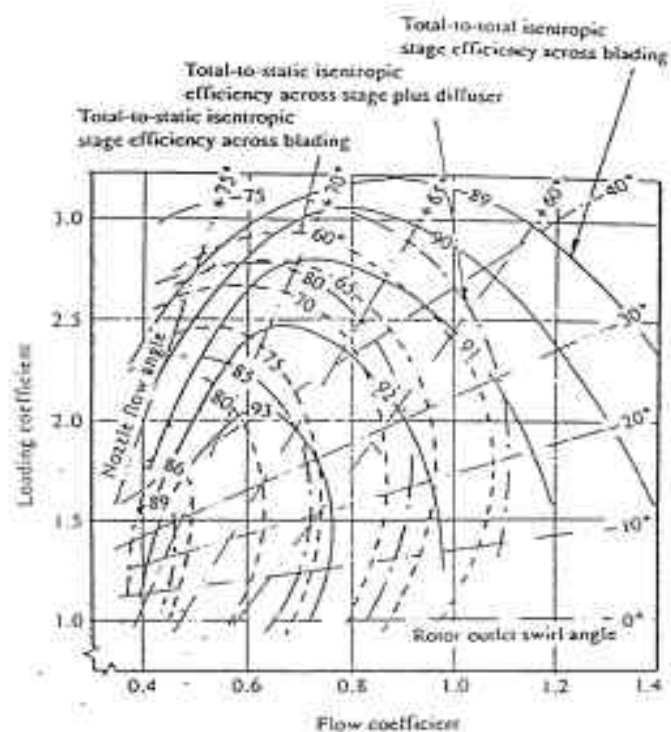


Figure 6.5 Annulus Flare Angle

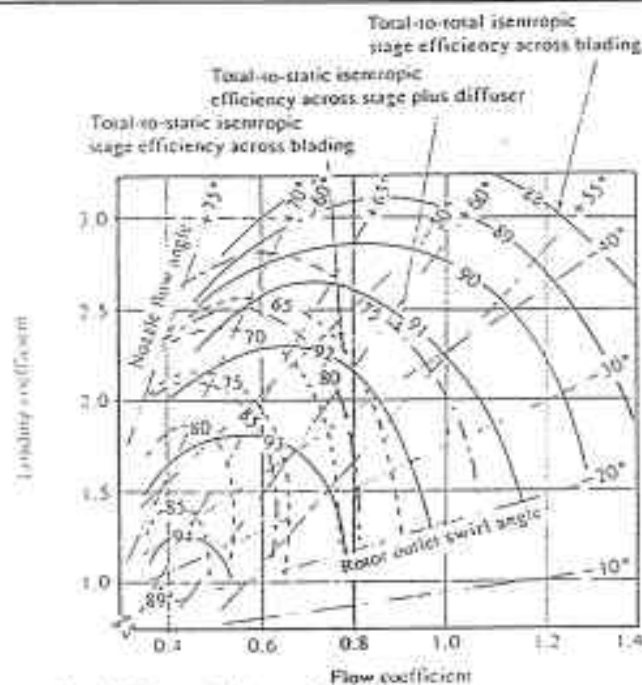


(a) Reaction 0.3



(b) Reaction 0.5

Figure 6.6 Axial-Flow Turbine Stage Efficiency Versus Loading and Flow Coefficients.



(c) Reaction 0.7

Figure 6.6 Contd

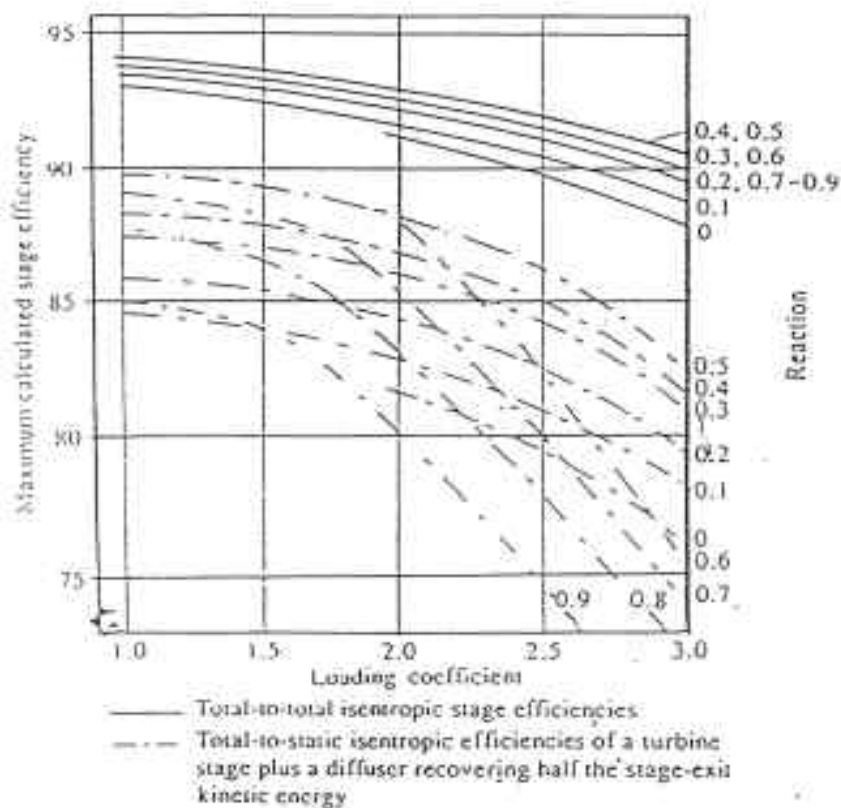


Figure 6.7 (a) Maximum Stage Isentropic Efficiencies versus Loading Coefficient and Reaction.

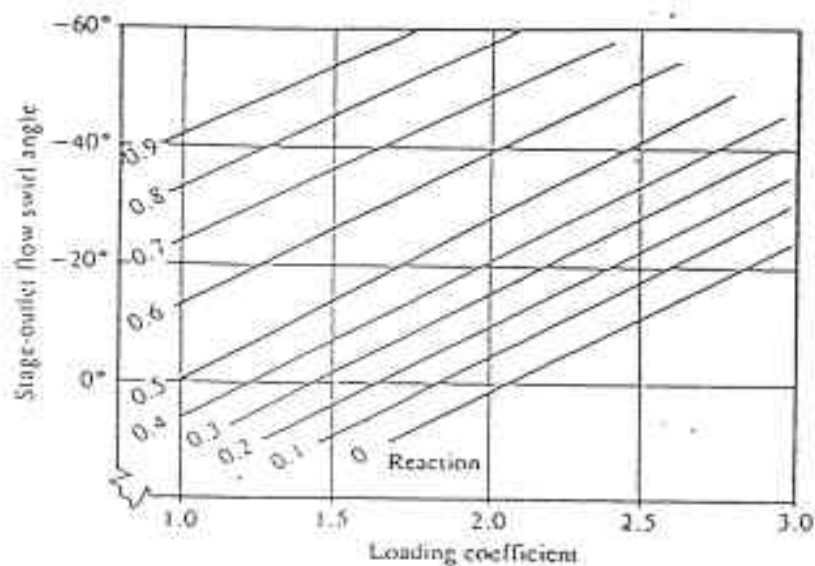


Figure 6.7(b) Stage Outlet Swirl Angles at Maximum Efficiencies versus Loading Coefficient and Reaction.

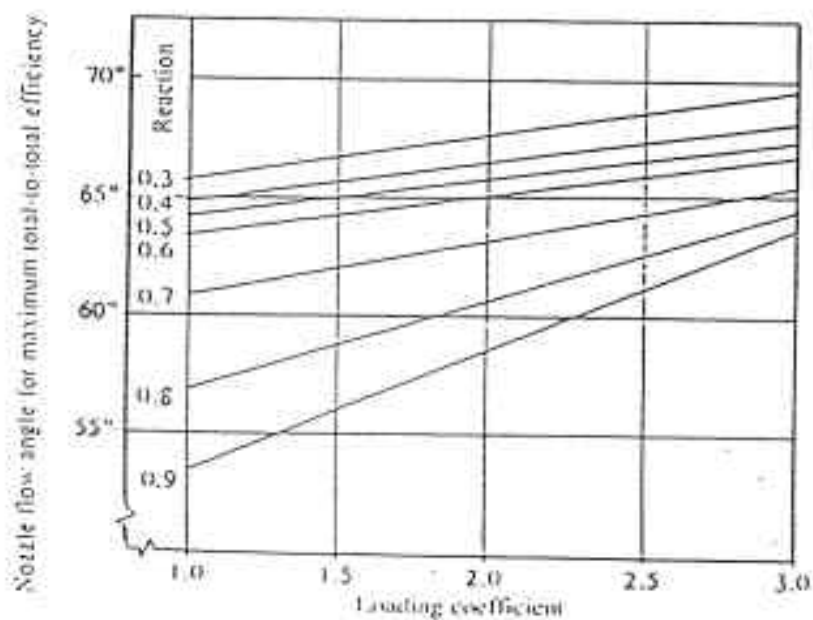


Figure 6.7(c) Optimum Nozzle Flow Angle versus Loading Coefficient and Reaction.

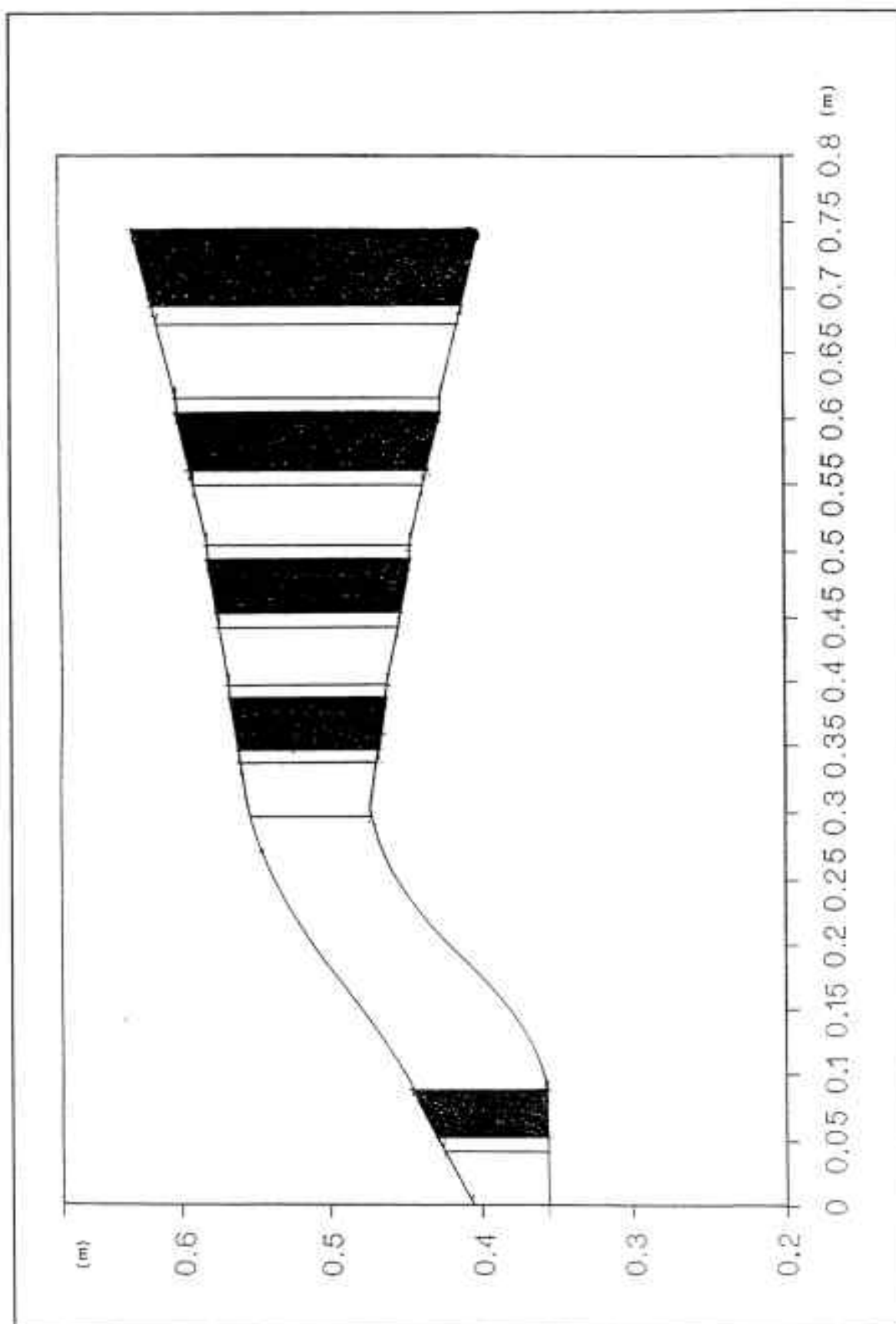


Figure 6.8 HP and LP Turbine Annulus Geometry

Axial-Flow Turbine Characteristics

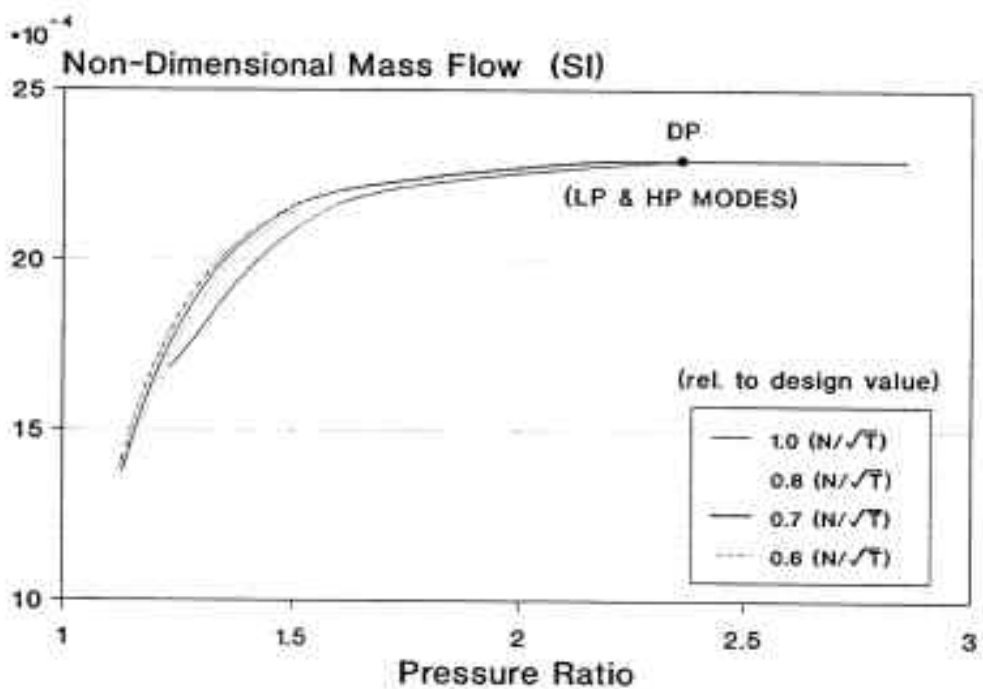
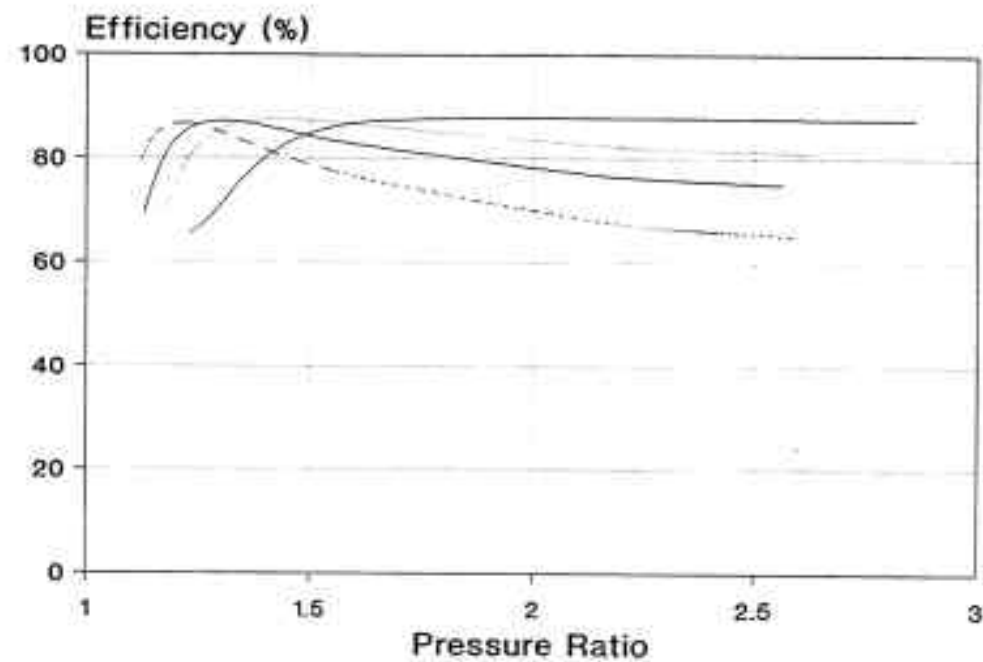


Figure 6.9 Single-Stage HP Turbine.

Axial-Flow Turbine Characteristics

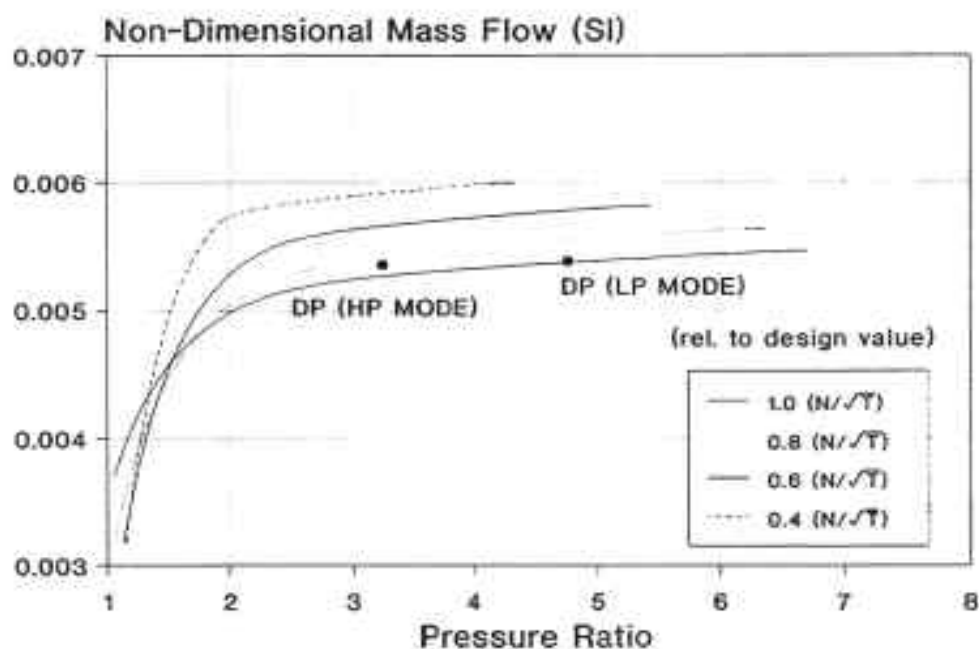
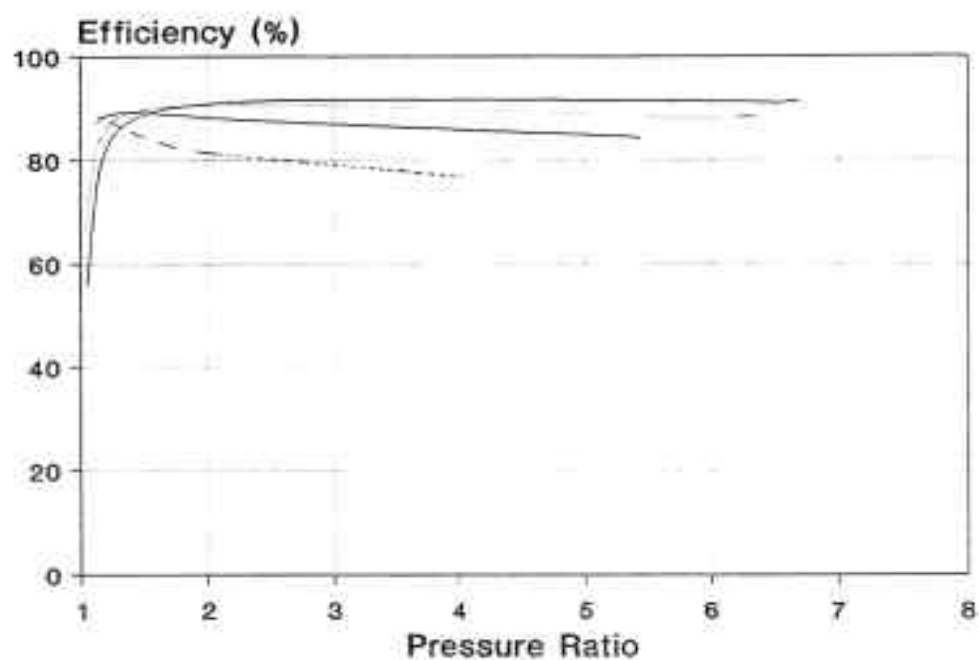


Figure 6.10 Four-Stages LP Turbine.

Compressor Characteristics

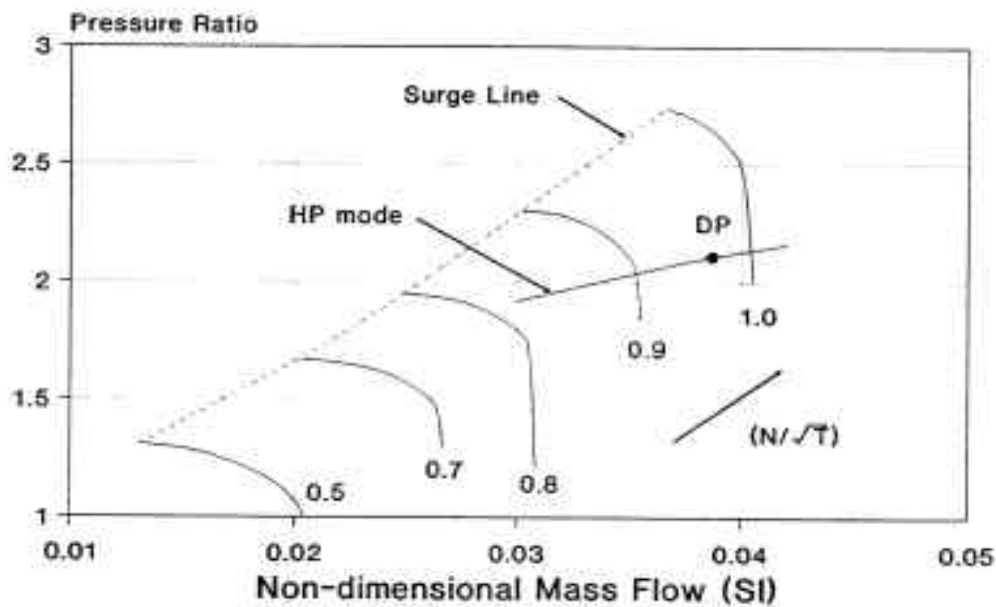
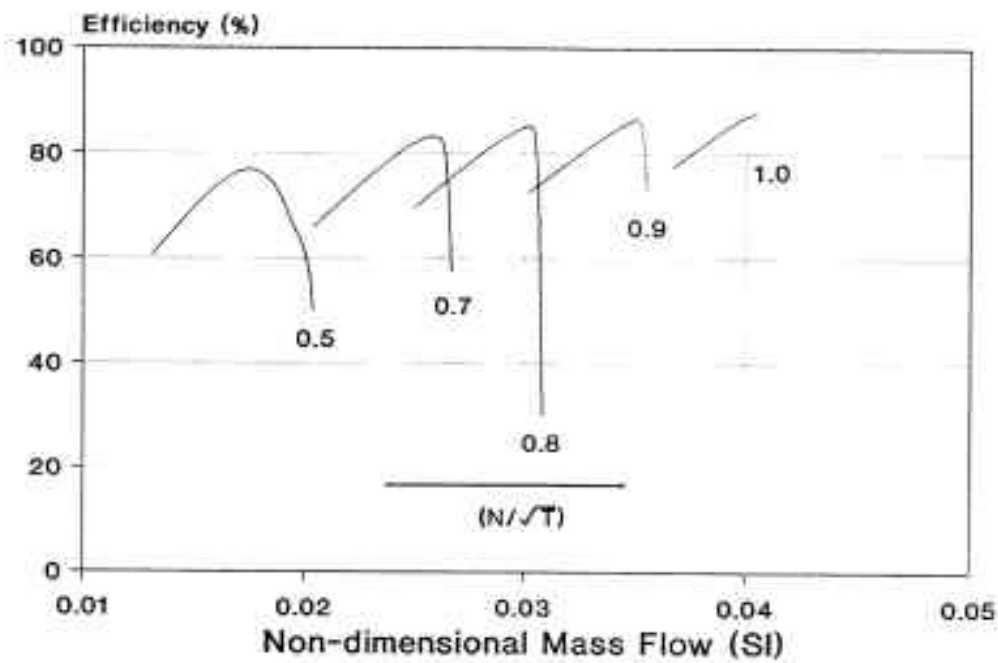


Figure 7.1 LP Compressor Variable IGV
by 35 degrees+All Stators by 20 degrees
(only HP mode)

OFF-DESIGN PERFORMANCE (Second Simulation)

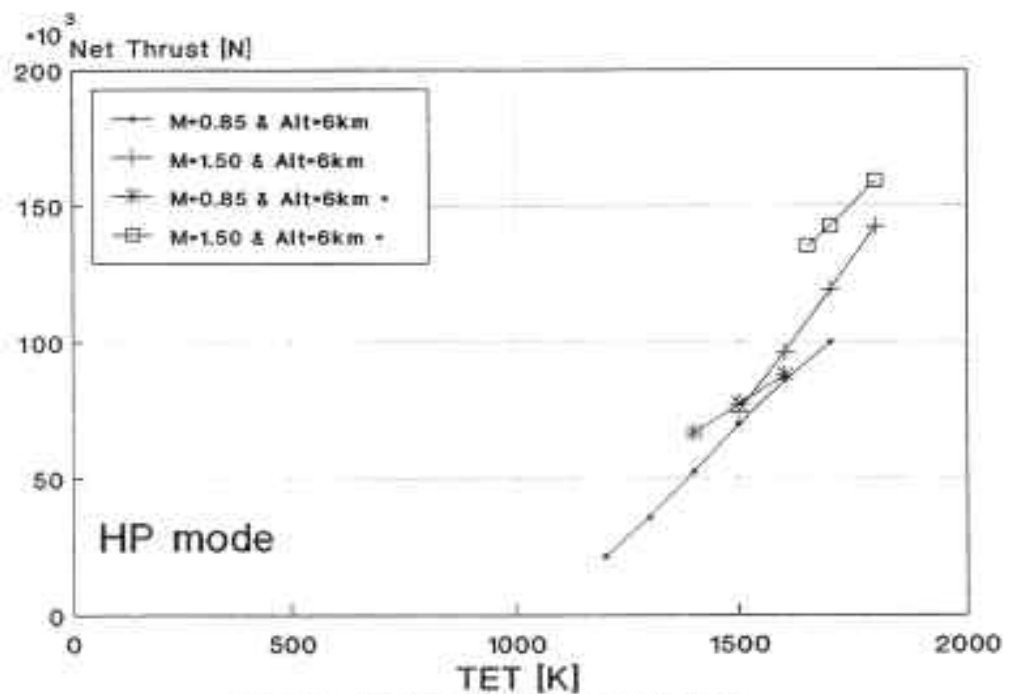
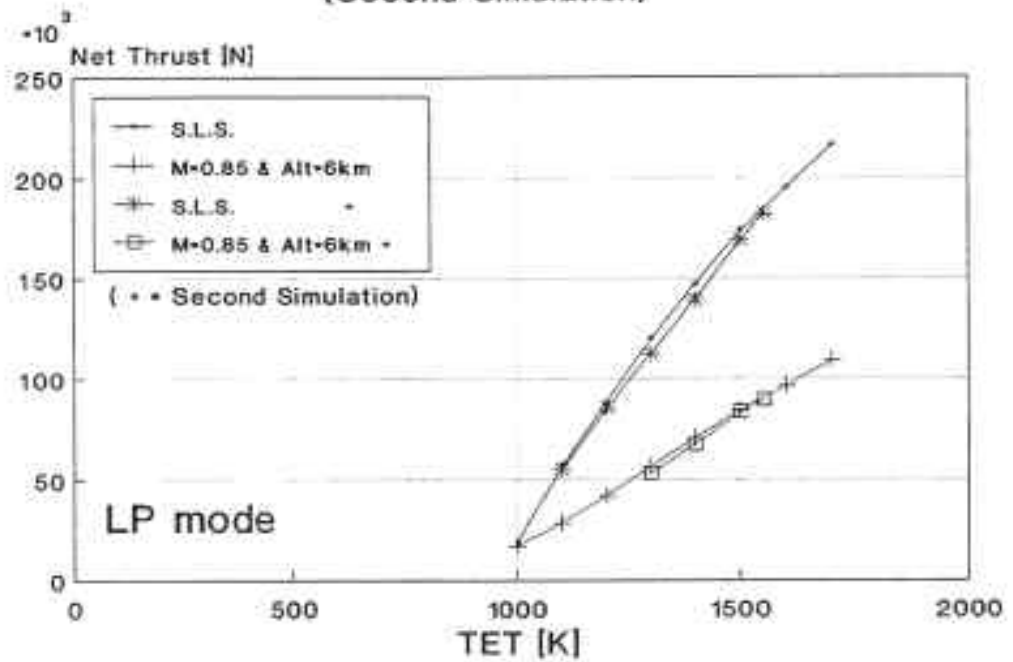


Figure 7.2 Change of Thrust with TET.

HP Compressor Characteristics (LP and HP modes)

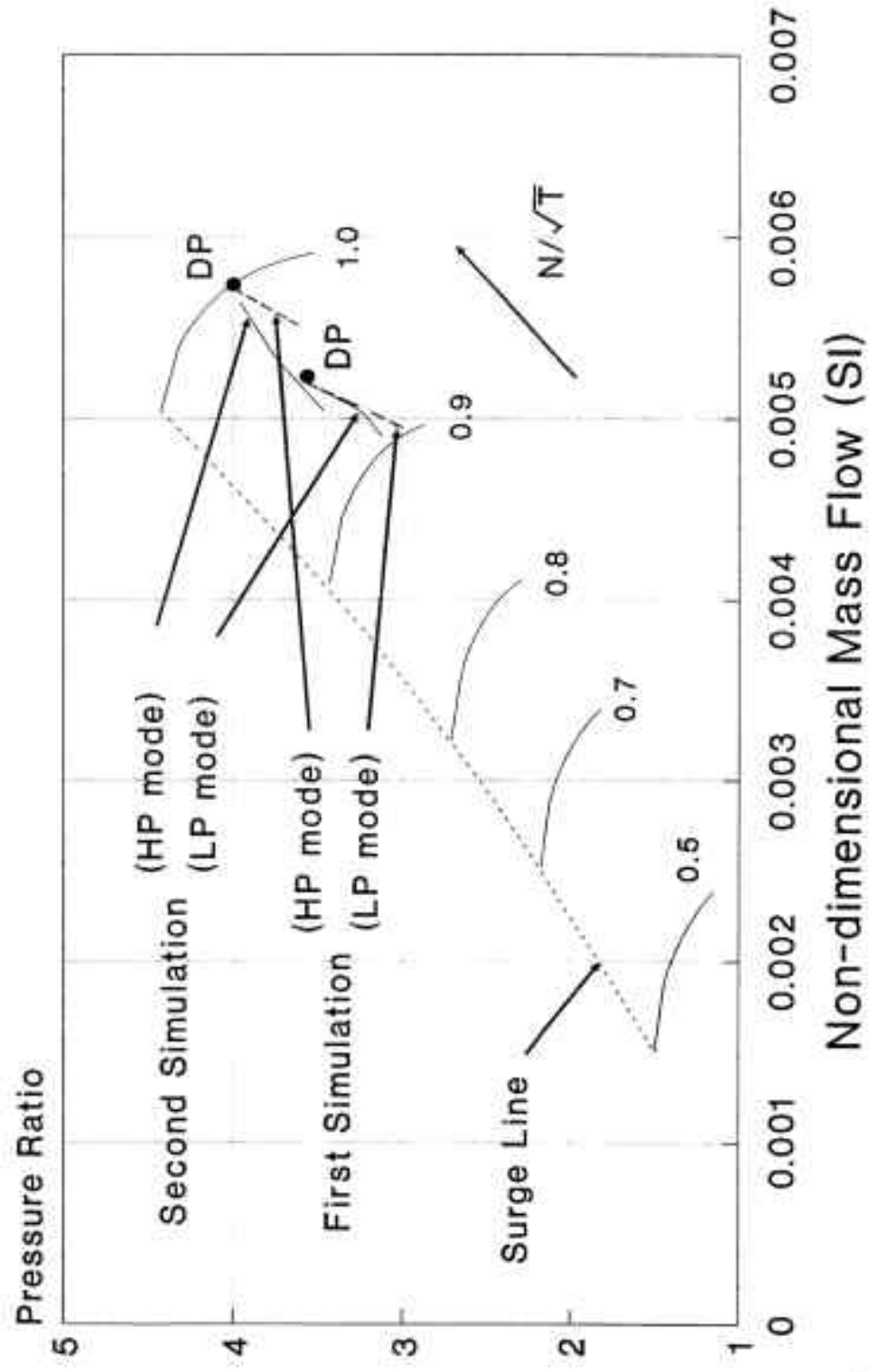


Figure 7.7 HP Compressor Fixed Geometry.

IP Compressor Characteristics (LP mode)

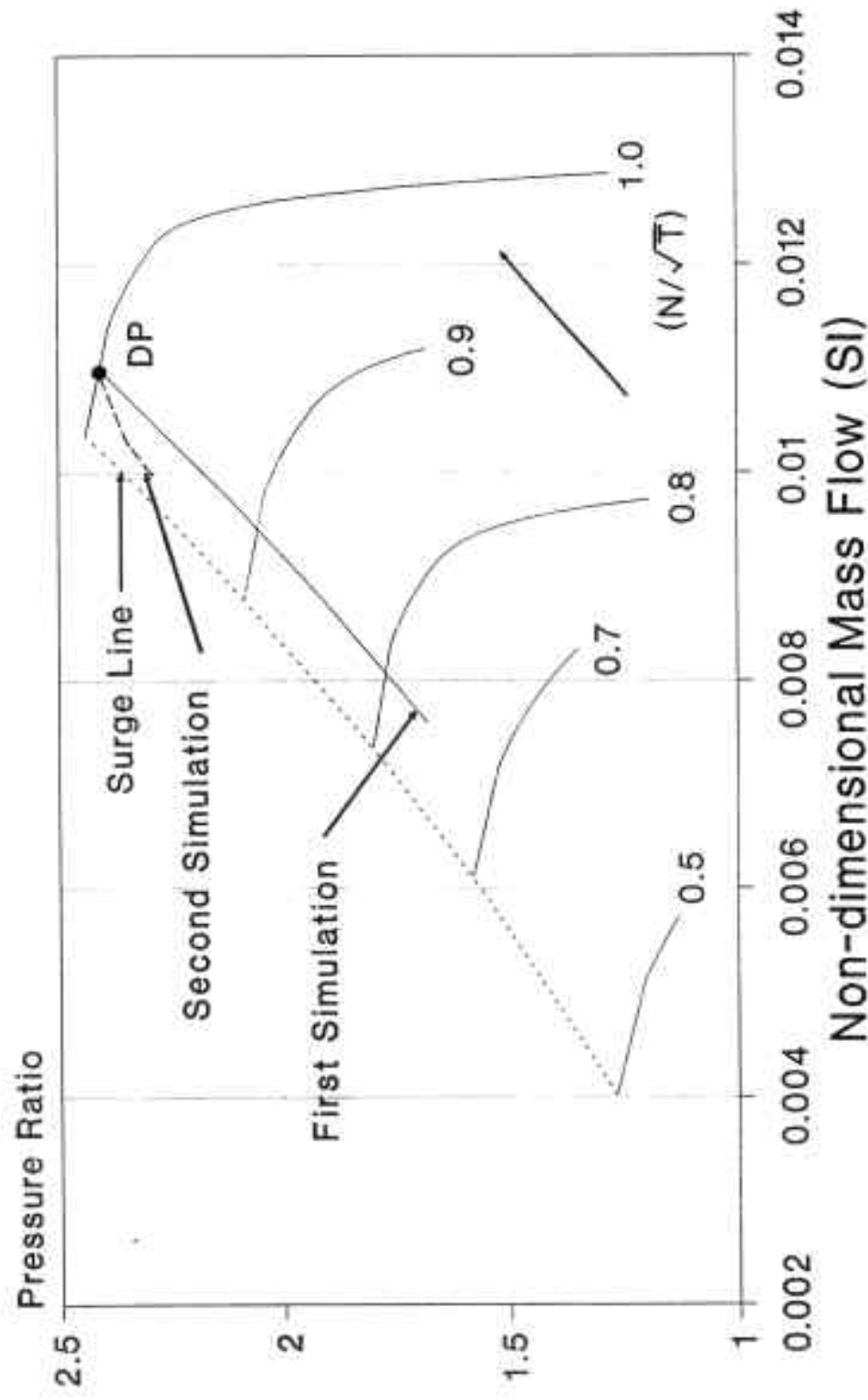


Figure 7.8 IP Compressor Variable All
Stators by 30 degrees.

IP Compressor Characteristics (HP mode)

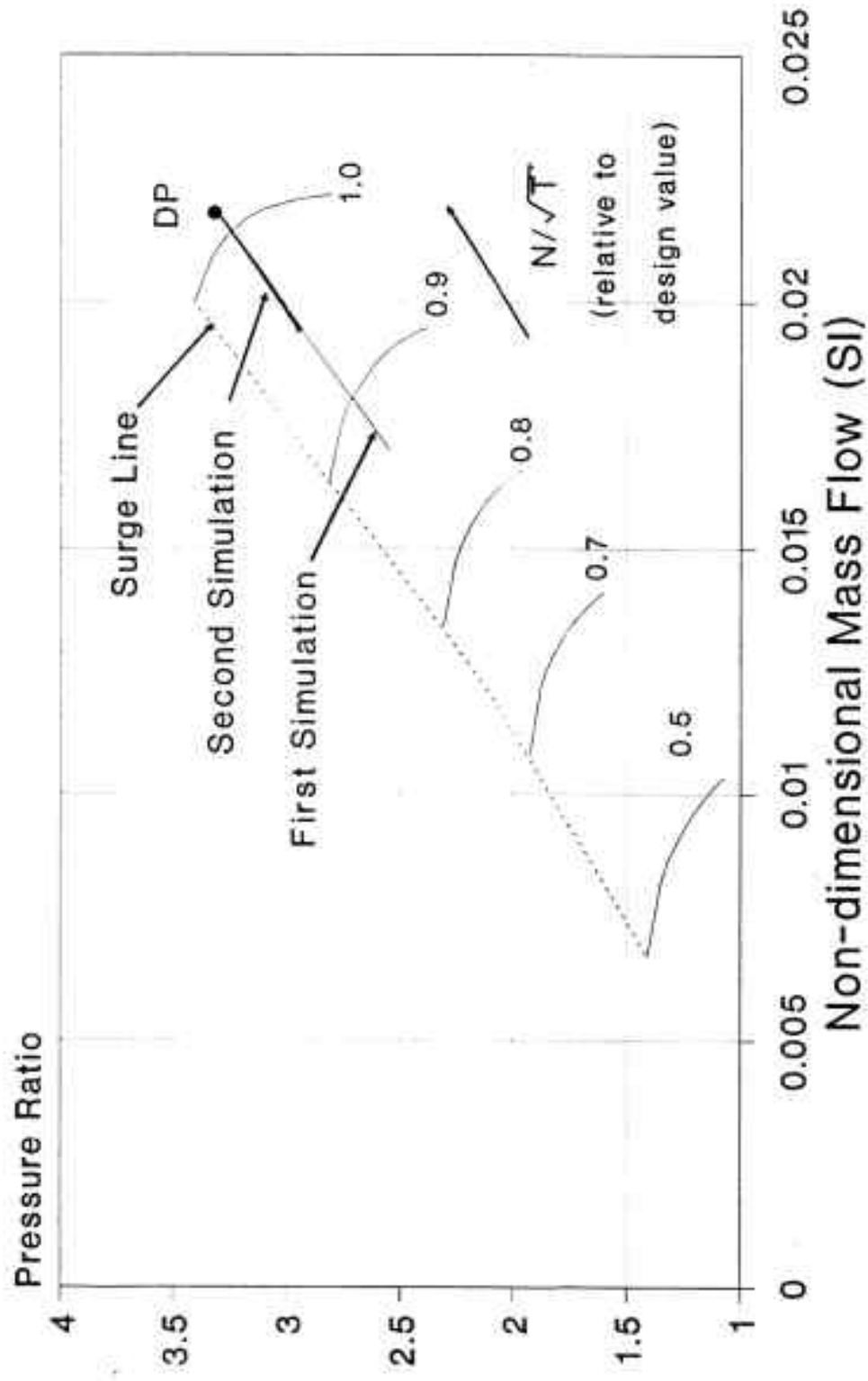


Figure 7.9 IP Compressor 0 degrees.