N87-11207

COMPONENT SPECIFIC MODELING

M. L. Roberts
General Electric Company
Aircraft Engine Business Group

1. INTRODUCTION

Modern jet engine design imposes extremely high loadings and temperatures on hot section components. Fuel costs dictate that minimum weight components be used wherever possible. In order to satisfy these two criteria, designers are turning toward improved materials and innovative designs. Along with these approaches, however, they must also have more accurate, more economical, and more comprehensive analytical methods.

Numerous analytical methods are available which can, in principle, handle any problem which might arise. However, the time and expense required to produce acceptable solutions is often excessive. This program addresses this problem by developing specialized software packages, which will provide the necessary answers in an efficient, user-oriented, streamlined fashion. Separate component-specific models will be created for burner liners, turbine blades, and turbine vanes using fundamental data from many technical areas.

2. OBJECTIVE

The overall objective of this program is to develop and verify a series of interdisciplinary modeling and analysis techniques which have been specialized to address three specific hot section components: combustor burner liners, hollow air-cooled turbine blades, and air-cooled turbine vanes. These techniques will incorporate data as well as theoretical methods from many diverse areas, including cycle and performance analysis, heat transfer analysis, linear and nonlinear stress analysis, and mission analysis. Building on the proven techniques already available in these fields, the new methods developed through this contract will be integrated to predict temperature, deformation, stress, and strain histories throughout a complete flight mission.

3. APPROACH

The work breakdown structure and tasks were discussed in detail at the Second HOST Workshop last year, and will not be repeated here. Three major development activities make up the Base Program. These are:

1. The Thermodynamic Engine Model,
2. The Thermomechanical Loads Model, and

3. The Three Component Specific Models:

   - Combustor liner,
   - Turbine blade, and
   - Turbine vane.

Thermodynamic Engine Model

The Thermodynamic Engine Model provides a decomposition/synthesis approach to compute, at any point in a mission, the engine rotor speeds and the gas path dynamic variables (velocity, temperature, pressure, density and Mach number) at any station along the combustor liner and high pressure turbine flow path. With this capability it is possible to synthesize the gas path variable history at any station of the burner liner, blade or vane for a complete mission, without computing the complete engine cycle for all mission points.

To develop the model, the engine cycle deck was run at 148 cycle points covering the complete engine operating range shown in Figure 1. From this data an engine performance cycle map and an interpolation scheme were developed to compute the gas path parameters at a chosen engine station, given the engine operating conditions specified in terms of altitude, free stream Mach number and engine power level. This also was discussed in more detail at last year's Workshop. Results for a typical commercial airline flight: altitude, Mach number, thrust, core rotor speed, compressor discharge temperature and turbine inlet temperature, are shown in Figures 2 through 7. It is not necessary to run complete missions. Segments can be run separately and joined to synthesize alternate missions.

Thermomechanical Loads Model

The Thermomechanical Loads Model accepts as input variables the rotor speeds and the gas path temperatures and pressures at one or more engine stations and computes component metal temperatures and surface pressures. Again, a decomposition/synthesis approach is used.

A typical commercial engine combustor liner, for which the loads model was developed, is shown in Figure 8. The correlation for the burner liner temperature, T_{liner}, was developed in terms of the cooling effectiveness factor, η_c:

$$\eta_c = \frac{T_4 - T_{liner}}{T_4 - T_3}$$

where

T_3 = compressor discharge temperature, and

T_4 = combustor exit temperature.

Using both engine test data and analytic results, a correlation of cooling effectiveness with combustor length was developed for a single combustor panel (Figure 9). For a realistic picture of liner temperatures, both average temperature and the hot streak temperature associated with the fuel nozzles were correlated. The effects of pressure and of altitude on cooling effectiveness were examined separately. Both were small enough to be neglected. To account for the temperature gradient through the metal thickness, an expression was derived from cooling effectiveness, compressor discharge temperature and pressure, and combustor exit temperature. By correlating test data against combustor pressure, a representative expression for temperature gradient through the metal thickness was obtained at five typical points along the length of the combustor liner. This completed the work on the combustor liner Thermomechanical Loads Model.

A common approach was taken for the cooled high pressure turbine blade and vane. Cooling effectiveness was again used as the correlating factor. The initial step was to correlate cooling effectiveness as a function of engine operating conditions at the 50% span station. Cross sections at 50% span of the typical blade and vane configurations are shown in Figures 10 and 11. A typical correlation of cooling effectiveness with gas path temperature and power level at the 50% span station on the turbine vane is shown in Figure 12. Both test data and analytical predictions are shown. The trend of the data is quite similar to the combustor data; however, there is some disagreement between the analytical and test values of η_c. This was not unexpected because of: (1) differences between nominal values of design variables (T_{gas}, $T_{coolant}$, wall thicknesses and thermal properties) and actual values; (2) uncertainties in calculating the gas side and coolant side heat transfer coefficients, the radiation heat flux, film cooling effectiveness, etc., (3) measured temperature errors due to uncertainties in thermocouple measurements, flow checking measurements, etc. As a result the test data was used to validate the trend lines, and the analytic predictions were used for the correlation.

From data of the type shown in Figure 12, a correlating equation was developed to predict cooling effectiveness at the 50% span station as a function of operating conditions:

$$\frac{1-\eta_c}{1-\eta_{c,Ref}} = \left(\frac{T_3}{T_{3,Ref}}\right)^{0.04} \left(\frac{T_{4.1,Ref}}{T_{4.1}}\right)^{0.04}$$

where $\eta_{c,Ref}$, $T_{3,Ref}$ and $T_{4.1,Ref}$ denote the reference values and conditions of Figures 10 and 11.

To predict the metal temperatures at other points on the blades, the cooling effectiveness factors at other spanwise stations were correlated against the values at the 50% span station, $_{50\%}$. This was done at eight chordwise stations on both the suction and pressure surfaces. Thus, using the gas path temperatures T_3 and $T_{4.1}$ for any engine operating condition, the reference data shown in Figures 10 and 11, and the correlating equation for η_c, the Thermomechanical Loads Model can be used to predict the metal temperatures over the complete blade and vane surfaces.

To establish the general procedure for predicting the static gas pressure distributions along the airfoil surfaces of the blade and vane, typical design gas pressure distributions were collected and normalized. Figure 13 shows the typical turbine normalized vane gas static pressure distribution.

With the completion of the Thermomechanical Loads Model, the capability exists for synthesizing a mission, determining the hot section flow path gas properties for the complete mission, and, from these, the metal temperature and pressure histories for the combustor and turbine blade and vane.

Component Specific Model Development - Geometric Modeling

The approach taken to model the three components is to select typical components, identify the key input parameters and develop master regions from these parameters. The finite element mesh can then be overlaid on each master region. Figure 14 shows a typical combustor nugget, some of the physical input parameters, and the master region definition based on those parameters. Figure 15 shows representative 2D and 3D models generated from these master regions. Modeling of the other two components, the blade and vane, is currently progressing.

4. CONCLUSION

When completed, this program will provide a non-linear stress analysis system for hot section engine parts that will allow the component designer to evaluate quickly the effects of mission variations on component life. It will be easy to use, cost effective, and make a significant contribution to assessing hot section durability.

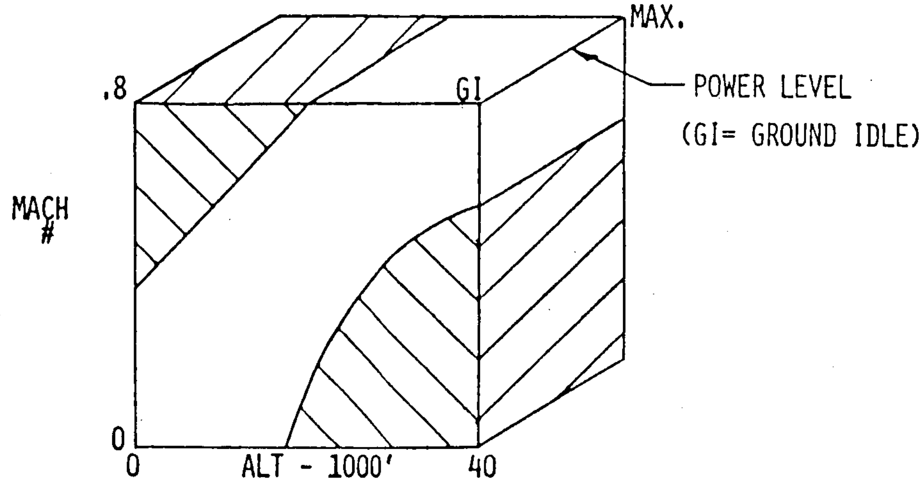


FIGURE 1. ENGINE OPERATING RANGE

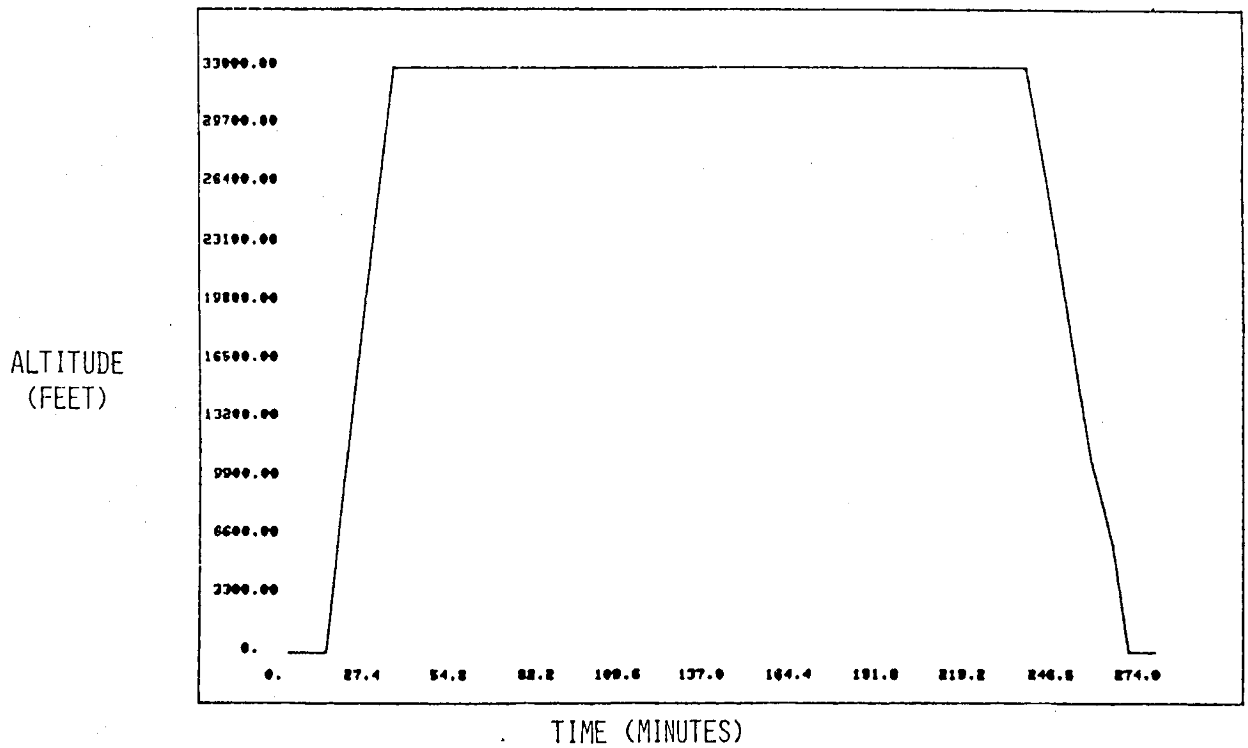
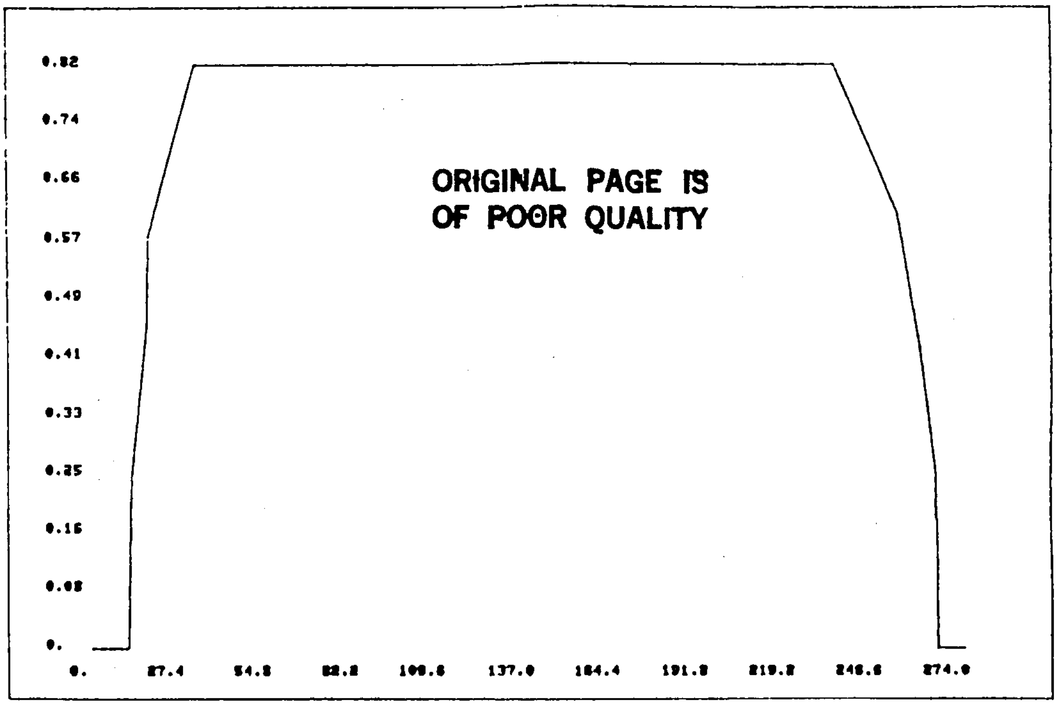


FIGURE 2. ALTITUDE VS. TIME

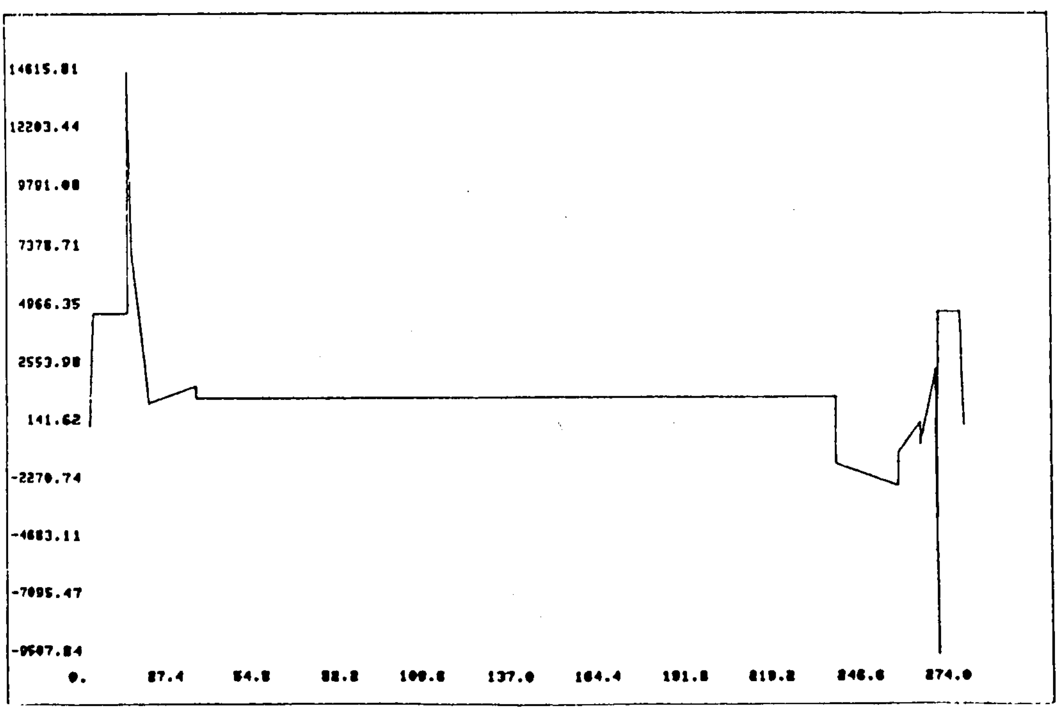
MACH NO.



TIME (MINUTES)

FIGURE 3. MACH NO. VS. TIME

THRUST
(POUNDS)



TIME (MINUTES)

FIGURE 4. THRUST VS. TIME

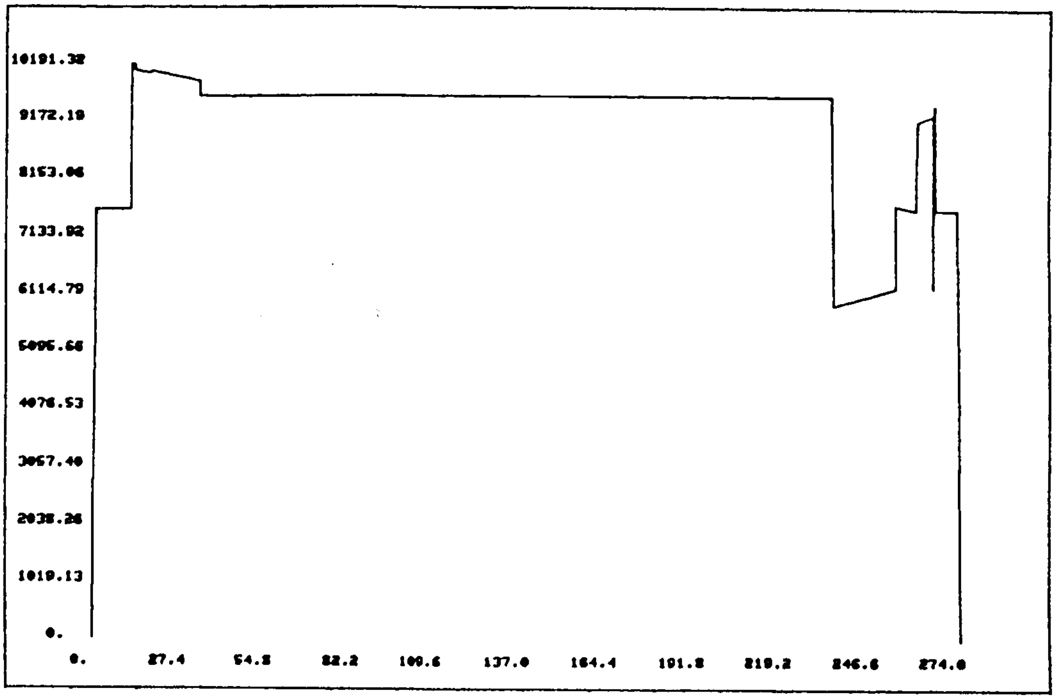


FIGURE 5. CORE ROTOR SPEED VS. TIME

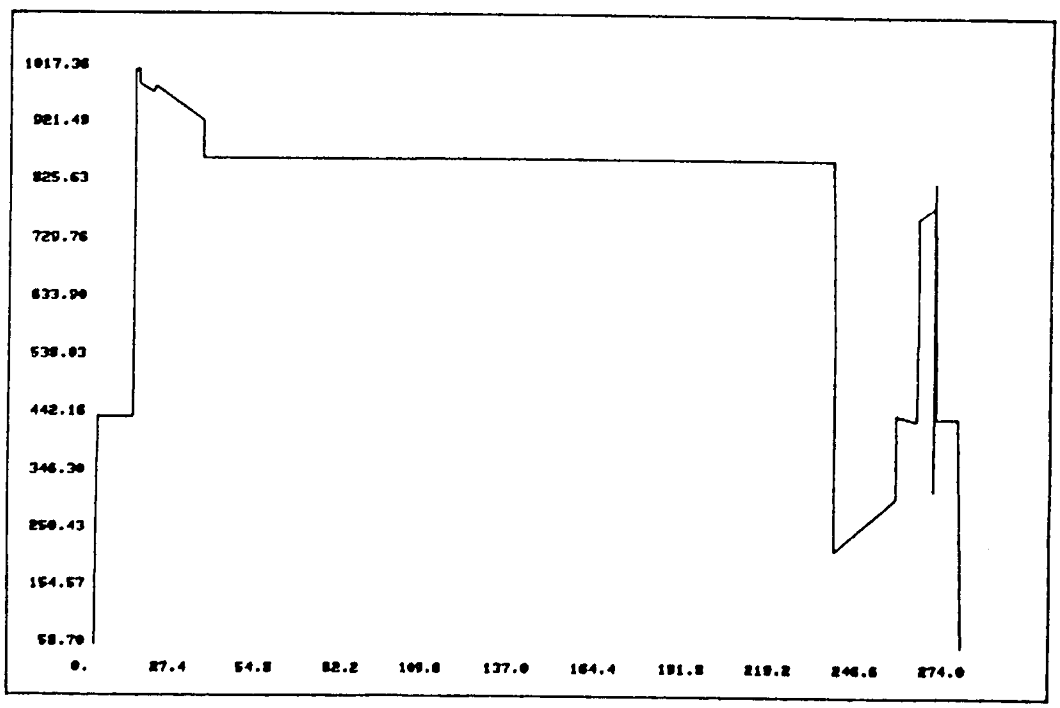
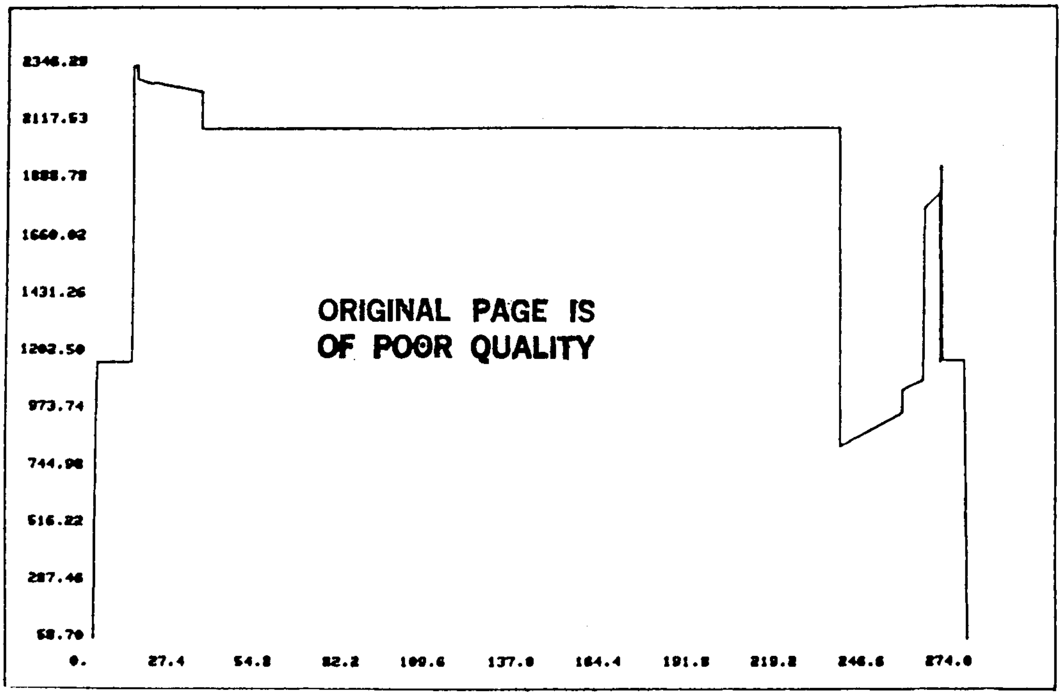


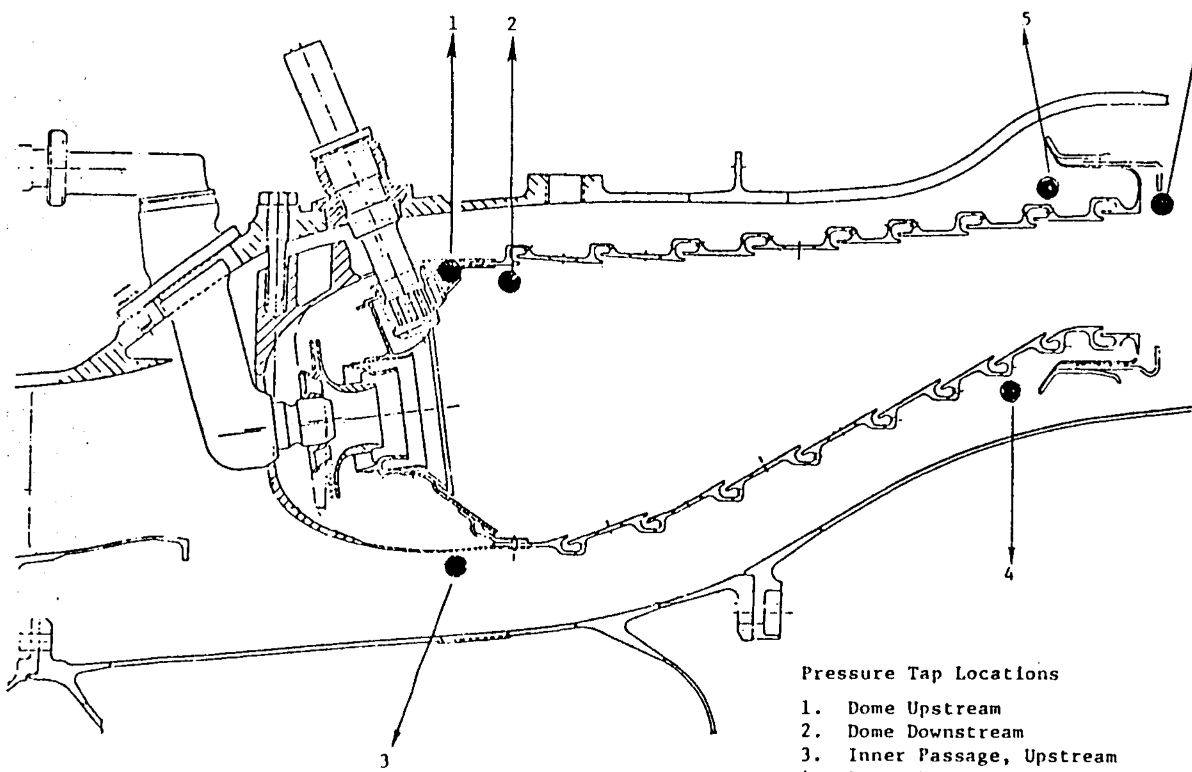
FIGURE 6. COMPRESSOR DISCHARGE TEMPERATURE VS. TIME

TURBINE INLET TEMPERATURE (^{o}F)



TIME (MINUTES)

FIGURE 7. TURBINE INLET TEMPERATURE VS. TIME



Pressure Tap Locations

1. Dome Upstream
2. Dome Downstream
3. Inner Passage, Upstream
4. Inner Passage, Downstream
5. Outer Passage, Downstream
6. Combustor Downstream

FIGURE 8. CF6-50 ROLLED RING COMBUSTOR

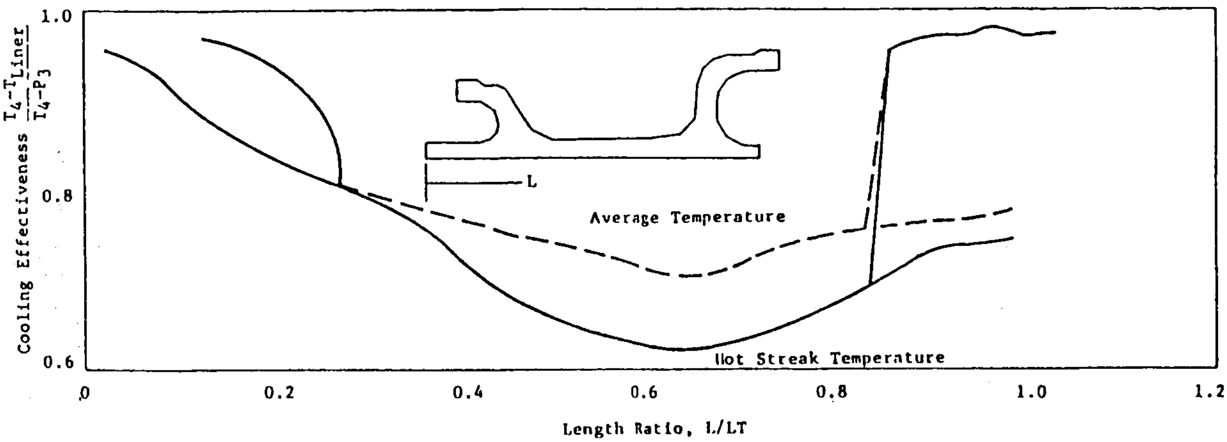


FIGURE 9. COOLING EFFECTIVENESS DISTRIBUTION, PANEL 7 OUTER

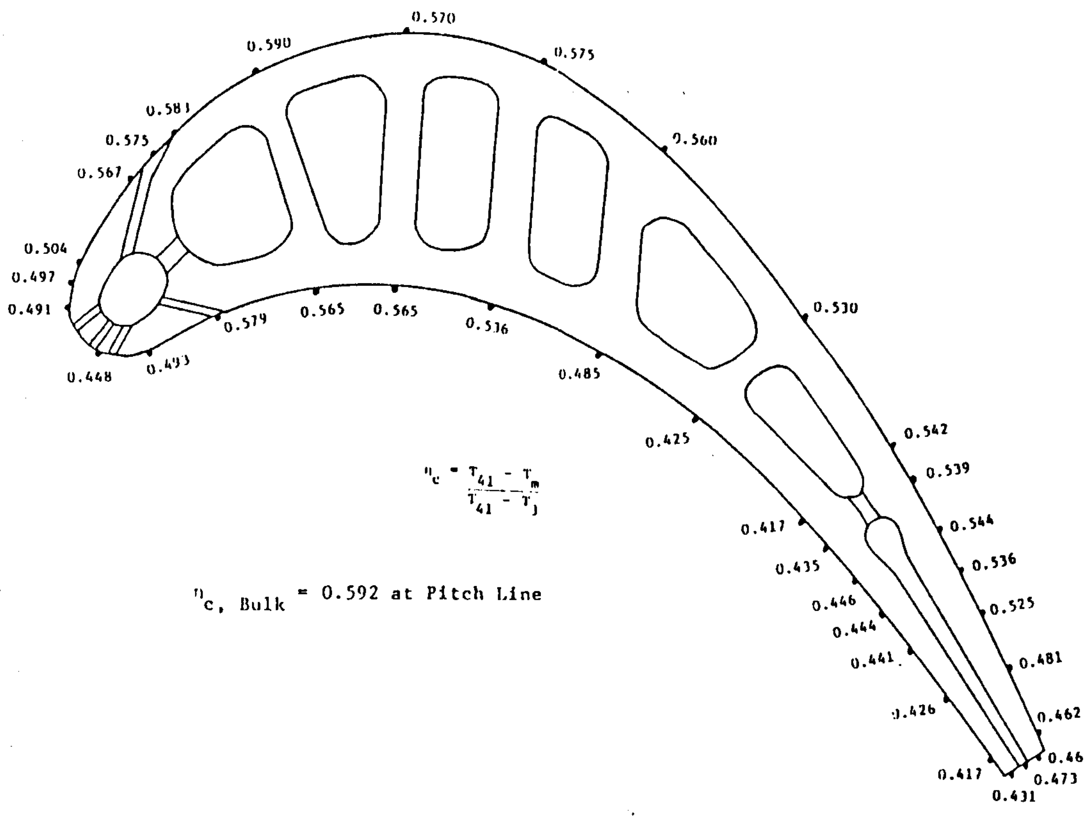


FIGURE 10. TURBINE BLADE SECTION WITH LOCAL GENERALIZED COOLING EFFECTIVENESS (η_c)

FIGURE 11. TURBINE VANE SECTION WITH LOCAL GENERALIZED COOLING EFFECTIVENESS (η_C)

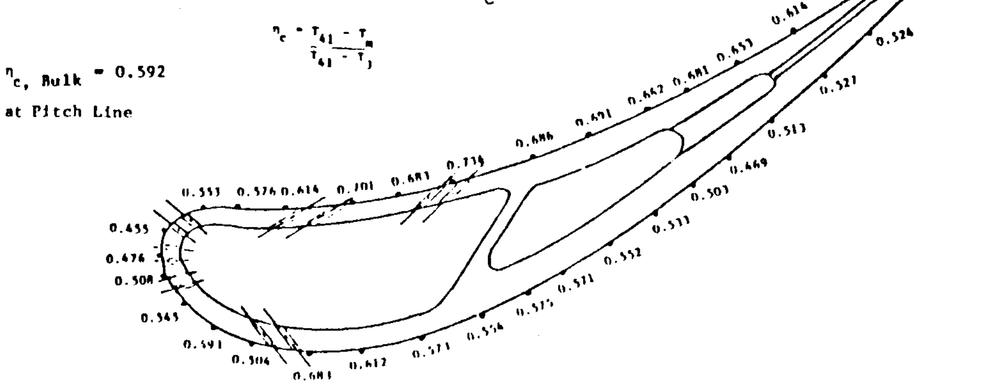
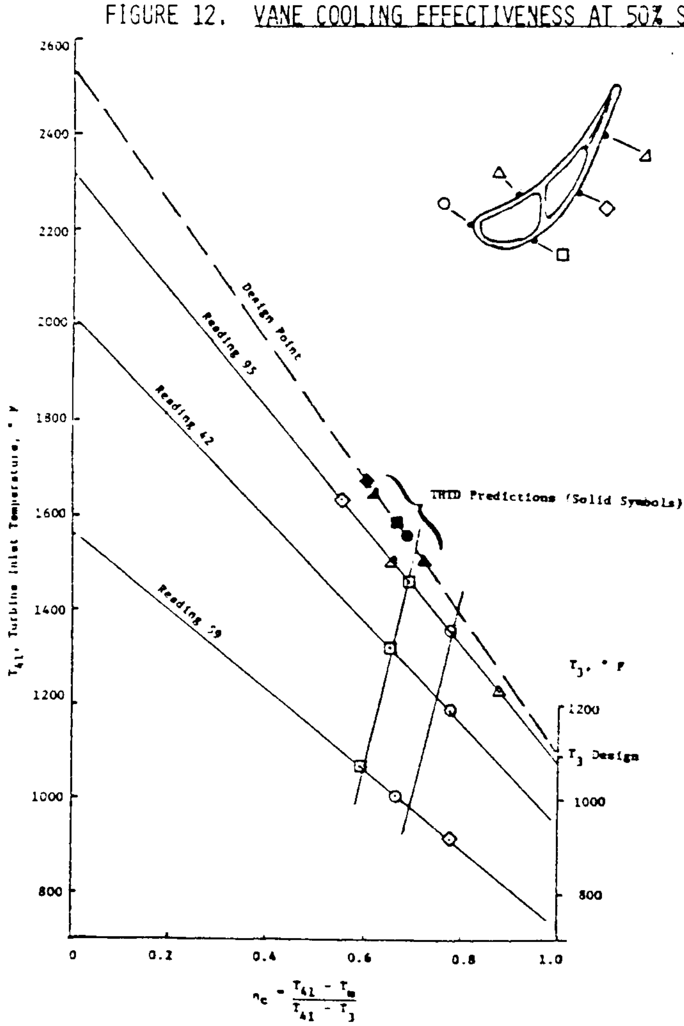


FIGURE 12. VANE COOLING EFFECTIVENESS AT 50% SPAN



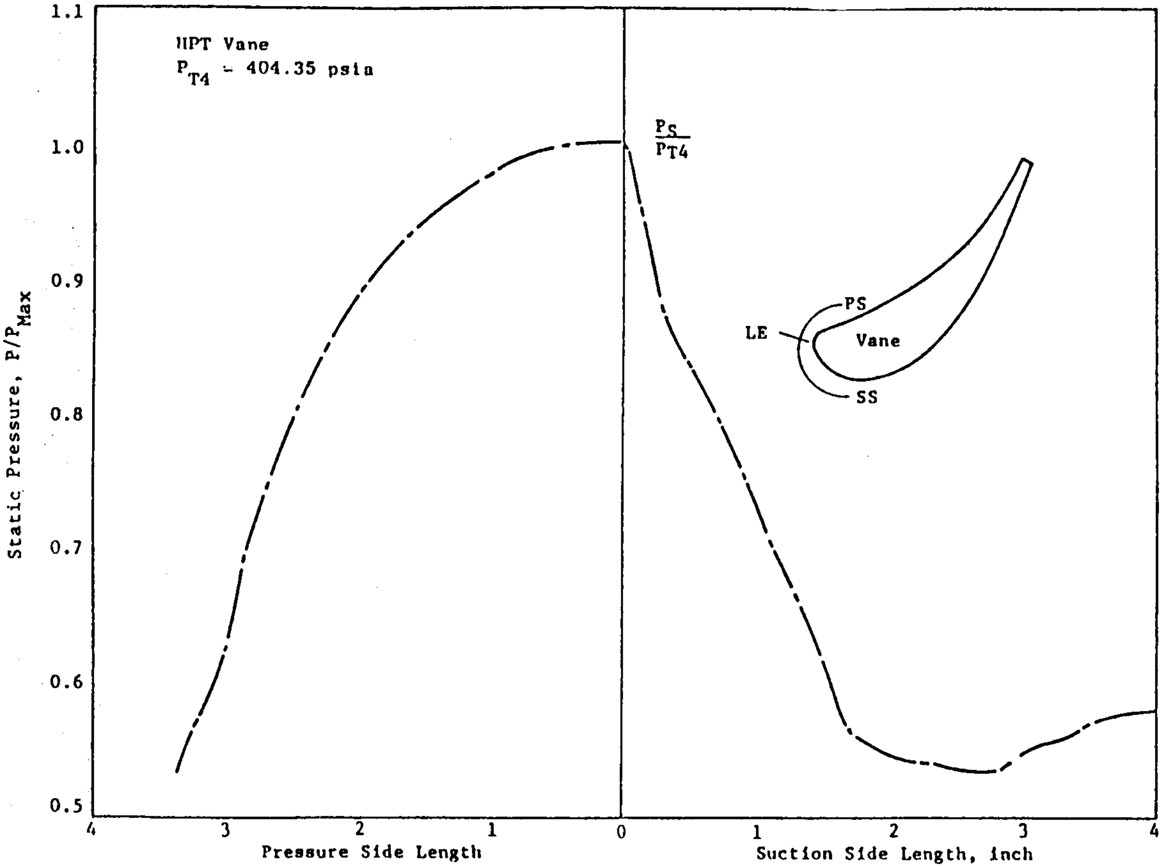


FIGURE 13. GAS STATIC PRESSURE DISTRIBUTION ALONG VANE SURFACE

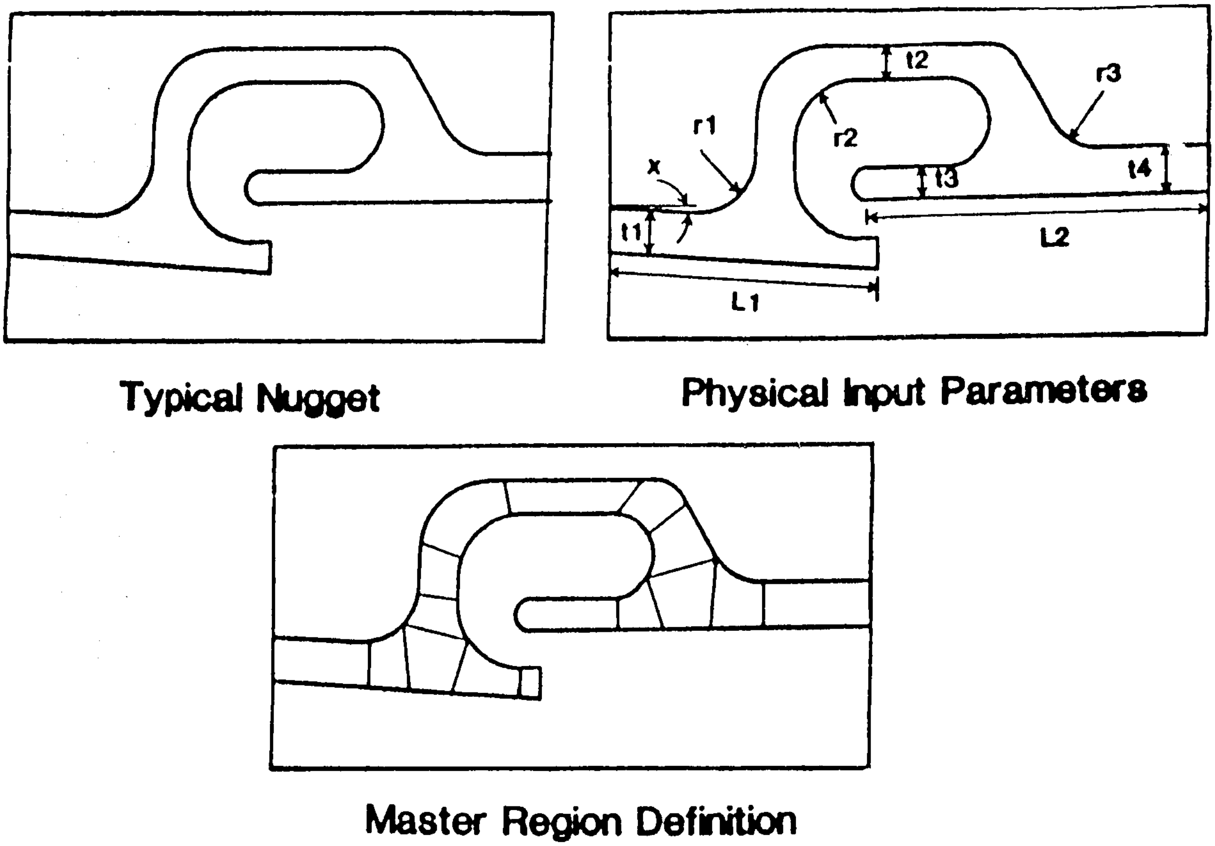


FIGURE 14. COMBUSTOR MODELING PROCESS

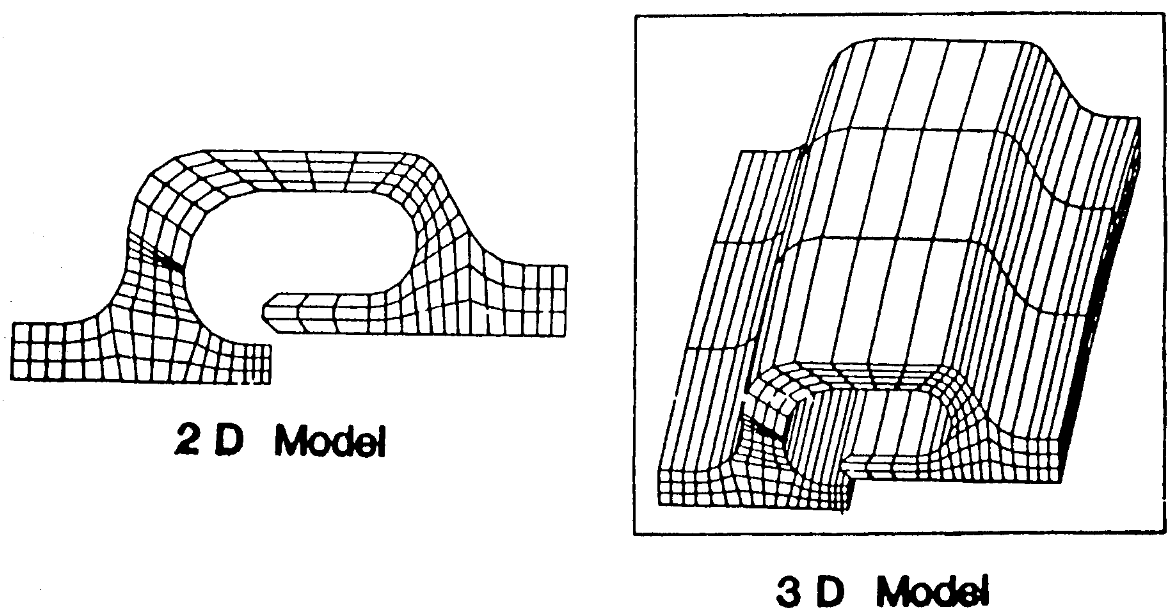


FIGURE 15. COMBUSTOR NUGGET FINITE ELEMENT MODELS