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NASA Technical Memorandum 105193

Supersonic Propulsion Simulation by Incorporating Component Models in the Large Perturbation Inlet (LAPIN) Computer Code

(NASA-TM-105193) SUPERSONIC PROPULSION
SIMULATION BY INCORPORATING COMPONENT MODELS
IN THE LARGE PERTURBATION INLET (LAPIN)
COMPUTER CODE (NASA) 33 p C5CL 21E

N92-15993

Unclas
G3/07 0062348

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December 1991



SUPERSONIC PROPULSION SIMULATION BY INCORPORATING COMPONENT MODELS IN THE LARGE PERTURBATION INLET

(LAPIN) COMPUTER CODE

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SUMMARY

An approach to simulating the internal flows of supersonic propulsion systems is presented. The approach is based on a fairly simple modification of the publicly-available Large Perturbation Inlet (LAPIN) computer code. LAPIN uses a quasi-one-dimensional, inviscid, unsteady formulation of the continuity, momentum, and energy equations. The equations are solved using a shock-capturing, finite-difference algorithm. The original code, developed for simulating supersonic inlets, includes engineering models of unstart/restart, bleed, bypass, and variable duct geometry, by means of source terms in the equations. The source terms also provide a mechanism for incorporating, with the inlet, propulsion system components such as compressor stages, combustors, and turbine stages. This requires each component to be distributed axially over a number of grid points. Because of the distributed nature of such components, this representation should be more accurate than a lumped-parameter model. Components can be modeled by performance map(s), which in turn are used to compute the source terms. The report describes the general approach first. Then, simulation of a compressor/fan stage is discussed to demonstrate the approach in detail. Generalized source terms, valid for either subsonic or supersonic flow through the compressor, are derived for the momentum and energy equations. A starting transient is presented as a demonstration case for a single-stage, supersonic fan in a mixed-compression inlet. Discussion of the compressor stage implementation in LAPIN provides guidance for the addition of other components.

INTRODUCTION

A computer simulation can be a valuable tool during a propulsion system development cycle. In particular, dynamic simulations provide a means for investigating potential interactions between individual components. For example, supersonic propulsion systems may exhibit an inlet unstart (expulsion of the normal shock) that can induce engine stall (refs. 1 to 3) and possibly surge. Conversely, engine stall will cause inlet unstart (ref. 2). Simulations can also be used to guide the design and testing of propulsion control systems prior to wind tunnel and flight tests (refs. 4 and 5).

Typically, lumped-parameter models (e.g., refs. 6 and 7) have been used to represent turbine type engines. More recently, a distributed turbojet engine model, using a one-dimensional finite-difference approach, was proposed in reference 8. Distributed models are generally more accurate, while lumped models have the advantage of faster execution times. Lumped models (e.g., ref. 9) have also been used to represent the gas dynamics of supersonic inlets. However, the ability to represent normal shocks with such models is quite limited. A more accurate model, using a linearized wave-equation approach, is described in references 10 and 11. That

model is able to predict inlet resonance conditions with reasonable accuracy, but is limited to small perturbations or excursions in normal-shock position.

The Large Perturbation Inlet (LAPIN) computer code was developed (ref. 12) to deal with large flow-field perturbations in supersonic inlets, such as hammer shock and unstart/restart transients. The approach is based upon a quasi-one-dimensional, inviscid, unsteady formulation of the continuity, momentum, and energy equations. The equations are solved using one of several user-selectable shock-capturing, finite-difference algorithms. LAPIN is publicly available from the NASA Computer Software Management and Information Center (COSMIC).

In many cases, propulsion system simulation can be adequately accomplished by coupling a distributed inlet model with a lumped engine model. One exception is a Supersonic Through-Flow Fan (STFF) system. The STFF system exhibits a so-called "starting transient"—the passage of a normal shock through the supersonic fan stage (from upstream to downstream) during the acceleration phase of the flight path. Unstarting, the reverse transient, occurs during deceleration. A STFF system configuration proposed for commercial supersonic transports and its advantages relative to a conventional mixed-flow turbofan engine are described in reference 13. The simulation of a STFF system and its starting/unstarting transients requires the use of distributed inlet/engine models like LAPIN and the engine model reported in reference 8. The simulation effort described in this report was motivated by NASA Lewis' interest in the STFF concept. However, the approach is applicable to conventional supersonic propulsion systems as well.

The simulation approach, described in this report, is based on a modified version of the LAPIN code. LAPIN was originally developed for the simulation of mixed-compression, supersonic inlets. The effects of an engine at the diffuser exit had to be approximated by appropriate variations of the downstream boundary conditions. Engineering models of unstart/restart, bleed, bypass, and variable duct geometry effects were implemented by means of source terms in the governing equations. Subsequent studies have shown that the entire compression system (inlet and fan/compressor stages) can be simulated by adding source terms in the LAPIN model and by using component performance maps to furnish the information needed to calculate the additional source terms. This approach is generic and can be extended to incorporate other propulsion components with the inlet, including combustors and turbine stages. The turbine-engine model described in reference 8 uses a similar approach.

This report describes the general approach for adding propulsion-component source terms to LAPIN and the terms specific to fan (compression) stages. A starting transient is presented as a demonstration case for a single-stage, supersonic fan in a mixed-compression inlet. The actual implementation and modifications to LAPIN subroutines is discussed, which provides guidance for the addition of other components.

GENERAL APPROACH

The general approach for adding propulsion-component source terms to the LAPIN equations is illustrated in figure 1. The entire flow field is represented by a one-dimensional grid with supersonic boundary conditions (BCs) upstream and subsonic boundary conditions downstream. LAPIN solves the unsteady, inviscid continuity, momentum, and energy equations (fig. 1) in finite-difference form at each grid point. The equations are written in weak conservation law form, with source/sink terms on the right-hand side. The pressure times duct-area

axial gradient and pressure times duct-area time-rate-of-change terms are included with the momentum and energy source terms, respectively. The quantities M_g , F_g , and Q_g represent the remaining cumulative source terms for the continuity, momentum, and energy equations, respectively. They are used to incorporate: terms originally included for mass bleed/bypass and unstart/restart phenomena; terms added later (ref. 14) for mass injection, duct pressure recovery losses, and heat addition; and the new terms proposed in this report for incorporating propulsion components.

In general, a component can be located anywhere on the grid and can be modeled by some form of performance map. (A component model in equation form could also be used.) The source terms must be derived in terms of information that can be furnished by the map(s) or equation(s). In LAPIN, the source terms are calculated at each time-step using the following procedure. First, inputs to the map are derived from a combination of flow field information and "externally" supplied information. Examples of external information are mechanical speed for a compressor/turbine component and fuel flow for a combustor. The external information is supplied either by means of LAPIN input statements (NAMELIST's) or an auxiliary subroutine (e.g., a control system algorithm). Outputs from the performance map, needed for the source terms, are then computed. Finally, component source terms required for the continuity, energy, and momentum equations are calculated and distributed to the grid points encompassed by the component.

To simulate a particular "engine," the associated components (e.g., compressor, combustor, turbine) are located in tandem on the grid. It is conceivable that component source terms will overlap other source terms. An example might be a compressor with a bleed region. The next section discusses some of the details of adding source terms to LAPIN that are specific to compression-stage components.

COMPRESSION STAGE SOURCE TERMS

The performance of a compressor or fan component is typically represented by a multi-dimensional map like the one illustrated in figure 2. In the illustration, component total pressure ratio and adiabatic efficiency are plotted as functions of corrected mass flow rate and corrected rotor speed. Such a map typically represents the overall performance of the component. However, particular applications and/or desired levels of simulation fidelity may require separate maps for each stage (i.e., rotor-stator combination) or even separate maps for each rotor and stator. Another approach might represent the rotor by a map and the stator by a simple drag source term. For the map shown in figure 2, the only external input required is compressor (or fan) mechanical speed. Other variables needed to compute the map inputs are mass flow rate and the total temperature and total pressure at the stage inlet. The map outputs are the stage total pressure ratio and adiabatic efficiency. As an alternative, the map inputs could be defined as corrected speed and stage total pressure ratio. In this case, the map outputs would be the stage inlet corrected mass flow rate and the adiabatic efficiency. In some cases, the shape of the corrected speed lines will dictate which procedure is used.

For component maps, such as the one shown in figure 2, it is possible to have certain combinations of inputs (i.e., corrected mass flow rate and corrected speed) that extrapolate to a map output that lies outside the normal operating range defined by the map. A computational strategy must be devised to handle such cases. If an output point should occur in the stall region, two possible options would be to (1) output a message that stall has occurred and terminate the

simulation or (2) use an estimated stall behavior (i.e., reduced total pressure ratio and adiabatic efficiency) and continue the simulation. In most cases, the second choice is preferable. That approach was adopted in reference 8 for stage characteristics like those in figure 3. The curves drawn to the left of point b in figure 3 are assumed to represent the in-stall performance. References 15 and 16 present the so-called ϕ versus ψ characteristics, which are similar to the pressure ratio versus corrected mass flow rate curve of figure 3(a). The ϕ and ψ parameters are normalized with respect to the mean rotor speed. Both ψ and corrected mass flow rate are indicators of stage inlet axial velocity or Mach number. Reference 15 explains that the ϕ versus ψ curves remain the same, with varying rotor speed, if the compressibility effects are small. Hence, the single-curve characteristic is valid at low Mach numbers and may provide a useful first approximation at moderately high Mach numbers. This would not apply to a supersonic stage and it is obvious that the approach and type of performance map(s) selected to simulate compression stages will be highly dependent on the particular application. Additionally, a time lag may be imposed on map inputs/outputs, as suggested in reference 8.

For purposes of illustrating the source terms that are specific to compression stages, it is assumed that the inputs to the performance map are corrected speed and stage inlet corrected mass flow rate. The corresponding performance map outputs are the stage total pressure ratio and adiabatic efficiency, which can be combined to calculate the stage total temperature ratio.

Since compression stages add energy and momentum to the flow, the corresponding source terms must be derived for the energy and momentum equations. Details of the derivation are presented in appendix B. The sketch in appendix B shows a representative compression stage and the associated nomenclature. The equations for the source terms are as follows:

Energy Equation

The total work per pound done by the stage is equal to the change in enthalpy across the stage (ref. 17).

$$\Delta h = c_p (T_{t3} - T_{t2})$$

Knowing the stage inlet total temperature, pressure, and mass flow rate and the stage mechanical speed, the stage total pressure ratio and adiabatic efficiency can be obtained from a performance map (e.g., fig. 2). The change in total temperature across the stage (ref. 17) is then calculated from:

$$T_{t3} - T_{t2} = T_{t2} \frac{\left[\left(\frac{P_{t3}}{P_{t2}} \right)^{(\gamma-1)/\gamma} - 1 \right]}{\eta}$$

The stage total rate of energy addition, written in terms of known flow field information and information derived from the component performance map, is

$$Q_{st} = m_2 c_p (T_{t3} - T_{t2}) = (\rho A u)_2 c_p (T_{t3} - T_{t2})$$

Momentum Equation

Between two stations, i and $i+1$, within the compression stage region, the net force due to the fan in the axial direction can be expressed as:

$$F_{st} = m_{i+1} u_{i+1} - m_i u_i + A_{i+1} p_{i+1} - A_i p_i - p_m (A_{i+1} - A_i)$$

This is a general expression that accounts for flow-area variation through the compression stage region. As shown in appendix B, the force can be rewritten as

$$F_{st} = \rho_i u_i^2 A_i FAC_1 + p_i A_i FAC_2 \frac{(A_{i+1} + A_i)}{2A_i}$$

which is in terms of known flow field variables at the cell entrance (station i) and the information derived from the component performance map (total temperature and pressure ratios across the cell contained in FAC_1 and FAC_2). This term represents only the incremental force between stations i and $i+1$. The total force term for the entire stage cannot be represented by a single equation, when flow area varies. If the stage flow area is constant, then the previous equation reduces to:

$$F_{st} = \rho_2 u_2^2 A_2 FAC_1 + p_2 A_2 FAC_2$$

which is the total force between the stage endpoints (stations 2 and 3). FAC_1 and FAC_2 are the same terms from the previous equation, evaluated at the stage endpoints (i.e., $i+1$ is the stage exit 3, and i is the stage inlet 2).

The governing equations (fig. 1), as solved by LAPIN at each grid point, are normalized and in finite difference form. In general, the source term contribution at a grid point covers a cell region that extends between stations (i) and ($i+1$) that are midway between the grid point (j) and the adjacent upstream ($j-1$) and downstream ($j+1$) grid points. Cell lengths depend on the grid spacing and grid points at the component end points are handled specially. The normalized source terms per unit length, specific to compression stages, are derived in appendix B and shown to be as follows:

Energy Equation

$$\hat{Q}_{st} = \hat{\rho}_2 \hat{u}_2 \left(3 \frac{T_{t3} - T_{t2}}{T_{t0}} \right) \left(\frac{L_{ref}}{L_{stage}} \right)$$

which is the total energy source term to be distributed to the compression stage grid points.

Momentum Equation (Variable Area Through Stage)

$$\hat{F}_{st} = \left[\frac{(\hat{\rho}_i \hat{u}_i)^2}{\hat{\rho}_i} FAC_1 + \hat{\rho}_i FAC_2 \left(\frac{A_{i+1} + A_i}{2A_i} \right) \right] \frac{L_{ref}}{x_{i+1} - x_i}$$

which is the incremental momentum source term between stations i and $i+1$.

Momentum Equation (Constant Area Through Stage)

$$\hat{F}_{st} = \left[\frac{(\hat{\rho}_2 \hat{u}_2)^2}{\hat{\rho}_2} FAC_1 + \hat{\rho}_2 FAC_2 \right] \frac{L_{ref}}{L_{stage}}$$

which is the total momentum source term to be distributed to the compression stage grid points. FAC_1 and FAC_2 are the same terms from the previous equation, evaluated at the stage end-points (i.e., $i+1$ is the stage exit 3, and i is the stage inlet 2), and $L_{stage} = x_3 - x_2$.

DEMONSTRATION CASE

In order to demonstrate the actual implementation of compression stage source terms, an example case was contrived and run using LAPIN. The example case corresponds to the starting transient for a single-stage supersonic fan in a mixed-compression inlet. The compression stage was implemented by modifying the heat addition section of the LAPIN subroutine USS. The implementation of that source term is explained in reference 14. Details of the required FORTRAN modifications, including the addition of a new NAMELIST and changes to subroutines INPUT, BBINIT, and USS are addressed in appendix C.

Description

This section describes how the demonstration transient was simulated and the required LAPIN input. A listing of the input and results for selected time steps is given in figure 4. LAPIN has seven NAMELIST inputs, each beginning with &NAME and ending with &END. NAMELIST CONTRL is used to specify the finite-difference algorithm (split characteristics for this case, ref. 12), the time step size, and the time steps to output results. Boundary condition

initial values and time histories are defined by NAMELIST BC. The free-stream Mach number, static pressure, and static temperature initial conditions are MHUSIC (Mach UpStream Initial Condition), PRUSIC, and TMUSIC respectively. The initial exit Mach number MHDSIC is 0.164 and is varied with time, as will be explained later. NAMELIST GEOM is used to define the duct geometry. A geometry, without a centerbody, was chosen for simplicity. In an actual case a centerbody is required for the fan and would be used to control duct Mach number distribution. The duct axial-area distribution is shown in figure 5. Distances are normalized by the cowl radius, COWLR. The duct throat is located at $x = 0.7$ and has a Mach number of about 1.5 for the specified free-stream conditions. As shown in figure 5, the fan location (specified in NAMELIST HEAT) has been set in a region where the duct area varies, to demonstrate that the source terms are not restricted to constant area sections. The duct was configured to give a fan face Mach number of about 2.0, when the fan is operating supersonically. Although not used in the demonstration case, fan-stage blockage could be specified in NAMELIST GEOM by using the strut area and location arrays, ASTRUT and XSTRUT respectively. Bypass, bleed and injection mass flows are not used in the demonstration case, but could be defined in NAMELIST BYBL and NAMELIST INJECT if required. NAMELIST HEAT is ordinarily used to specify location and rate of heat addition to the flow. As mentioned earlier, the heat addition section of LAPIN subroutine USS was modified to simulate the effects of a compression stage. Values of stage adiabatic efficiency and total pressure ratio were input by means of NAMELIST HEAT, rather than using a map subroutine to determine component performance. A single compression stage (NHEAT = 1) is used, with normalized starting location XHEAT(1) of 1.56 and length LHEAT(1) of 0.7 (fig. 5). The stage adiabatic efficiency was assumed to have a constant value of 0.65 for all values of pressure ratio and was specified by using XHEAT(2). The HEATIN array defines linear variations of stage total pressure ratio between the time points specified by the TIMHT array. NAMELIST LOSS is used to define drag and skin friction terms, not used by the demonstration case.

Discussion of Results

The duct exit Mach number and stage total pressure ratio were varied in a manner intended to simulate a supersonic fan starting transient. Their time histories are plotted in figure 6. Axial distributions of Mach number, static pressure, total pressure, and total temperature are shown in figure 7 at several instants of time during the transient. Figure 8 shows time histories of the stage total pressure ratio and temperature ratio during the transient. The ratios, initially 1.0, were calculated from downstream ($x = 2.4$) and upstream ($x = 1.5$) values.

During the first 0.03 sec of the transient, the simulation is brought to a steady-state condition. Curve A of figure 7 shows the initial operating condition with the normal shock located upstream of the fan stage and just downstream of the inlet throat ($x = 0.7$). The normal shock location is evidenced by the sharp decrease in Mach number (fig. 7(a)) and increase in static pressure (fig. 7(b)). The normal shock also causes a drop in total pressure that is evident in curve A of figure 7(c) for x greater than about 0.8. It should be noted that the same curve also exhibits a total pressure increase in the vicinity of the normal shock. This increase is the result of a numerical instability that is often produced by finite-difference algorithms at grid points adjacent to discontinuities like normal shocks. The severity of the instability depends on the grid spacing and the Mach number upstream of the shock. A more severe case is observed in curve E of figure 7(d) at about $x = 3.1$.

With the fan initially operating at a subsonic Mach number, the fan total pressure ratio is ramped from 1.0 to 1.4 during the timespan from 0.03 to 0.07 sec (fig. 6(b)). The normal shock moves downstream, to satisfy the duct exit Mach condition, as the fan pressure ratio rises. As a result, the fan inlet Mach number varies. The shock position, for the pressure ratio of 1.4, is just upstream of the fan (fig. 7, curve B). The expected total temperature ratio is 1.155 based on a total pressure ratio of 1.4 and an efficiency of 0.65. Figure 8 verifies that the LAPIN energy and momentum source terms do produce the correct stage total pressure and temperature ratios.

From 0.09 to 0.12 sec, the fan pressure ratio remains constant while the duct exit Mach number is ramped up to "pop" the shock through the fan. Once the shock enters the fan, the fan inlet Mach number jumps to a supersonic Mach number of about 2.0 and remains constant. At 0.1 sec, the shock is approximately halfway through the fan (fig. 7, curve C). The fan overall total pressure ratio momentarily drops below 1.4 with the shock in the fan, due to total pressure losses associated with the normal shock. As the shock moves downstream, the shock Mach number increases and the associated shock total-pressure loss increases. The resulting decrease of the fan overall pressure ratio is illustrated in figure 8(a). The spikes in pressure ratio, at values above 1.4 and below 0.85 in figure 8(a), are due to the aforementioned numerical instability. After the shock exits downstream of the fan, the pressure ratio jumps back to a value of 1.4 (fig. 8(a)). Curve D of figure 7 shows conditions at $t = 0.11$ sec, after the shock exits the fan but before completion of the ramp in duct exit Mach number.

The final phase of the transient results from an increase in fan pressure ratio from 1.4 to an assumed operating value of 2.4. The final steady-state conditions are shown by curve E of figure 7. Again, figure 8 shows that the LAPIN energy and momentum source terms do produce the correct stage total pressure and temperature ratios. The expected total temperature ratio, corresponding to a total pressure ratio of 2.4 and an efficiency of 0.65, is 1.437.

The authors recognize that the demonstration transient does depart from a realistic starting transient in some respects. However, the main objective of the transient was to demonstrate that the source-term approach for representing components does work. In particular, the compression stage source terms were shown to be valid for both subsonic and supersonic conditions at the stage inlet.

CONCLUDING REMARKS

An approach for simulating supersonic propulsion systems, using the publicly-available Large Perturbation Inlet (LAPIN) computer code, was described. In general, source terms in the governing continuity, energy, and momentum equations can be used to incorporate component models with a supersonic inlet. Simulation of a compression/fan stage was discussed to demonstrate the approach in detail. The source terms, required for the momentum and energy equations, were derived in terms of quantities that can be obtained from a compression stage performance map. The source terms are generic and are valid for stages with either subsonic or supersonic inlet conditions and with variable flow area through the stage. Results were presented to demonstrate the approach for the starting transient of a supersonic fan in a mixed-compression inlet. The results verified that the energy and momentum source terms do produce the correct stage total pressure and temperature ratios. Details concerning the LAPIN implementation of the source terms were presented, providing guidance for incorporating other components such as combustors and turbines. This will allow LAPIN to be used to simulate a variety of supersonic propulsion configurations.

APPENDIX A. - SYMBOLS

A	local area
c_p	specific heat at constant pressure
E	total internal energy
F_s	cumulative momentum source term
F_{st}	compression-stage momentum source term
Δh	change in enthalpy
L	length
M	Mach number
M_s	cumulative continuity source term
m	mass flow rate
p	pressure
Q_s	cumulative energy source term
Q_{st}	compression-stage energy source term
T	temperature
t	time
u	axial fluid velocity
x	axial coordinate
γ	ratio of specific heats
η	stage adiabatic efficiency
ρ	fluid density

Subscripts:

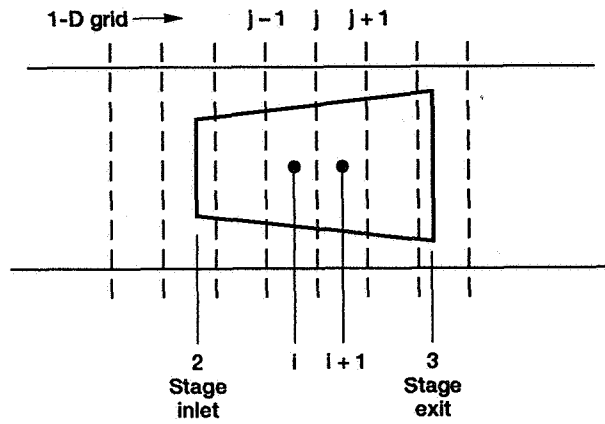
i	axial section within component region
j	grid point
m	mean value
ref	reference value
t	total or stagnation condition
0	free-stream condition
2	compression stage inlet
3	compression stage exit

Superscripts:

$\hat{\quad}$	normalized variable
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APPENDIX B - DERIVATION OF COMPRESSION-STAGE

ENERGY/MOMENTUM SOURCE TERMS



Energy Equation

The total work per pound done by the stage is equal to the change in enthalpy across the stage (ref. 17).

$$\Delta h = c_p (T_{t3} - T_{t2})$$

The total rate of energy addition of the stage per unit length is

$$Q_{st} = m_2 \Delta h = \frac{\rho_2 A_2 u_2 c_p (T_{t3} - T_{t2})}{L_{stage}}$$

In LAPIN the energy source terms are normalized, as follows

$$\hat{Q}_{st} = \frac{Q_{st} L_{ref}}{(\rho u^3 A)_{ref}} = \frac{\rho_2 A_2 u_2}{(\rho A u)_{ref}} \frac{c_p (T_{t3} - T_{t2})}{u_{ref}^2} \frac{L_{ref}}{L_{stage}}$$

where u_{ref} is the free-stream sonic velocity.

The LAPIN primitive state variables are

$$\hat{\rho} = \frac{\rho A}{(\rho A)_{ref}}, \quad \hat{u} = \frac{u}{u_{ref}}, \quad \text{and} \quad \hat{p} = \frac{p A}{(\rho u^2 A)_{ref}}$$

so

$$\hat{Q}_{st} = \hat{\rho}_2 \hat{u}_2 c_p \frac{(T_{t3} - T_{t2})}{u_{ref}^2} \frac{L_{ref}}{L_{stage}}$$

Assuming adiabatic and thermally perfect conditions, free-stream velocity is related to free-stream sonic velocity by (ref. 18):

$$\left(\frac{u_0}{u_{ref}} \right)^2 = \frac{6M_0^2}{5} \left(\frac{5}{5 + M_0^2} \right) = \frac{6}{5 + M_0^2} M_0^2$$

Rearranging, substituting $u_0^2/\gamma RT_0$, and multiplying by T_{t0}/T_{t0}

$$u_{\text{ref}}^2 = \frac{5 + M_0^2}{6} \frac{\gamma RT_0}{T_{t0}} T_{t0}$$

since $c_p = \gamma R/(\gamma - 1)$ and $T_0/T_{t0} = 5 / (5 + M_0^2)$

$$u_{\text{ref}}^2 = \frac{5 + M_0^2}{6} (\gamma - 1) c_p \frac{5}{5 + M_0^2} T_{t0}$$

which, for $\gamma = 1.4$, reduces to

$$u_{\text{ref}}^2 = c_p \frac{T_{t0}}{3}$$

substituting for u_{ref}^2 , \hat{Q}_{st} becomes

$$\hat{Q}_{\text{st}} = \hat{\rho}_2 \hat{u}_2 \frac{3(T_{t3} - T_{t2})}{T_{t0}} \frac{L_{\text{ref}}}{L_{\text{stage}}}$$

which is the total energy source term to be distributed over the compression stage grid points.

Momentum Equation

Between two stations, i and $i+1$, within the compression stage region, the net force due to the fan can be expressed as:

$$F_{\text{st}} = m_{i+1}u_{i+1} - m_i u_i + A_{i+1}p_{i+1} - A_i p_i - p_m (A_{i+1} - A_i)$$

This is a general expression that accounts for area variation through the compression region.

Assuming the mass flow rate is constant (i.e., $m_{i+1} = m_i = \rho_i u_i A_i$) and $p_m = (p_{i+1} + p_i)/2$

$$F_{st} = \rho_1 u_i^2 A_i \left(\frac{u_{i+1}}{u_i} - 1 \right) + \left(\frac{A_{i+1} + A_i}{2A_i} \right) p_i A_i \left(\frac{p_{i+1}}{p_i} - 1 \right)$$

Using relationships from reference 18, the ratio u_{i+1}/u_i can be converted to a function of Mach numbers and total temperatures—

$$\frac{u_{i+1}}{u_i} = \frac{u_{i+1}}{u_i} \sqrt{\frac{\gamma R T_i}{\gamma R T_{i+1}}} \sqrt{\frac{\gamma R T_{i+1}}{\gamma R T_i}} \sqrt{\frac{T_{t_i}}{T_{t_{i+1}}}} \sqrt{\frac{T_{t_{i+1}}}{T_{t_i}}}$$

substituting M for $u/\sqrt{\gamma RT}$ and $(1 + 0.2M^2)^{1/2}$ for $(T_t/T)^{1/2}$

$$\frac{u_{i+1}}{u_i} = \frac{M_{i+1}}{M_i} \left(\frac{1 + 0.2M_i^2}{1 + 0.2M_{i+1}^2} \right)^{1/2} \sqrt{\frac{T_{t_{i+1}}}{T_{t_i}}}$$

Likewise, the ratio p_{i+1}/p_i can be converted to a function of Mach numbers and total pressures—

$$\frac{p_{i+1}}{p_i} = \frac{p_{i+1}}{p_i} \frac{p_{t_i}}{p_{t_{i+1}}} \frac{p_{t_{i+1}}}{p_{t_i}} = \left(\frac{1 + 0.2M_i^2}{1 + 0.2M_{i+1}^2} \right)^{3.5} \frac{p_{t_{i+1}}}{p_{t_i}}$$

Letting $MR = \left(\frac{1 + 0.2M_i^2}{1 + 0.2M_{i+1}^2} \right)$, F_{st} becomes

$$F_{st} = \rho_1 u_i^2 A_i \left[\overbrace{\frac{M_{i+1}}{M_i} MR^{1/2} \sqrt{\frac{T_{t_{i+1}}}{T_{t_i}}} - 1}^{FAC_1} \right] + \left(\frac{A_{i+1} + A_i}{2A_i} \right) p_i A_i \left[\underbrace{MR^{3.5} \frac{p_{t_{i+1}}}{p_{t_i}} - 1}_{FAC_2} \right]$$

The normalized force term per unit length is

$$\hat{F}_{st} = \frac{F_{st}}{x_{i+1} - x_i} \frac{L_{ref}}{(\rho u^2 A)_{ref}}$$

In terms of LAPIN primitive variables

$$\hat{F}_{st} = \left[\frac{(\hat{\rho}_1 \hat{u}_1)^2}{\hat{\rho}_1} FAC_1 + \left(\frac{A_{i+1} + A_i}{2A_i} \right) \hat{p}_i FAC_2 \right] \frac{L_{ref}}{x_{i+1} - x_i}$$

which is the incremental momentum source term between stations i and $i+1$, for the variable area case. If area through the fan stage is constant, the expression for \hat{F}_{st} becomes

$$\hat{F}_{st} = \left[\frac{(\hat{\rho}_2 \hat{u}_2)^2}{\hat{\rho}_2} FAC_1 + \hat{p}_2 FAC_2 \right] \frac{L_{ref}}{L_{stage}}$$

which is the total momentum source term to be distributed over the compression stage grid points and where FAC_1 and FAC_2 are evaluated at the stage end points (i.e., station $i+1$ is the stage exit 3 and station i is the stage inlet 2) and $L_{stage} = x_3 - x_2$.

APPENDIX C. - LAPIN IMPLEMENTATION OF SOURCE TERMS

The continuity, momentum, and energy equation source terms provide the hooks for incorporating propulsion component models with LAPIN. This appendix begins with a general discussion of the procedure used to implement additional source terms, and modifications required to some LAPIN computer-program subroutines. Then, implementation of the compression stage source terms in subroutine USS is discussed in some detail, in order to provide guidance for adding other components. The discussion assumes that the reader has some familiarity with LAPIN. Additional information concerning the program, its subroutines, and the source terms can be found in references 12, 14, and 19. (To the authors' knowledge, there are no references that discuss the LAPIN FORTRAN source code.)

GENERAL PROCEDURE

The following steps outline the general procedure for implementing additional source terms:

1. Represent component(s) by means of performance map(s) (e.g., compression stage pressure ratio and efficiency as functions of corrected airflow and corrected speed). Three ways to include component performance map data in LAPIN are: calls to a subroutine, interpolation of block data arrays, or interpolation of arrays input by means of NAMELIST(s).
2. Derive the normalized source terms for the components in terms of quantities that can be obtained from the performance maps (e.g., total pressure ratio and temperature ratio for compression stages). Source terms for the compression stage example presented in this report are derived in appendix B.
3. Modify LAPIN to reflect the addition of the new source terms to the continuity, momentum, and energy equations. In general, this requires modification of LAPIN subroutines INPUT, BBINIT, and USS, as described below.

General FORTRAN Modifications and Additions

Subroutine INPUT.—Subroutine INPUT reads the input data for LAPIN by means of NAMELISTs and sets up a number of variables that are used as flags throughout the program. New NAMELIST(s) must be added to LAPIN to provide for definition of additional component(s) and external input time histories (e.g., mechanical speed as a function of time for compression stages). The new NAMELIST(s) can be modeled after the existing ones used for other user-supplied source terms—BYBL for bypass and bleed mass flows; INJECT for injection mass flow; HEAT for heat addition; and LOSS for drag and skin friction. Subroutine INPUT must be modified to include the NAMELIST(s) declaration(s) and the corresponding READ statement(s). COMMON block(s) must also be added to pass the necessary information to subroutines BBINIT and USS.

Subroutine BBINIT.—This subroutine is used to determine the starting and ending grid indices associated with each user-defined source-term zone. Subroutine BBINIT must be modified to include the COMMON blocks added to subroutine INPUT, and statements that determine starting and ending grid indices for the new source terms.

Subroutine USS.—Subroutine USS is used to calculate the values of the source terms at each time step for the continuity, energy, and momentum equations. A separate section must be added to subroutine USS to calculate the source terms for each new component type. Components of the same type are calculated in the same section by using arrays of dimensioned variables. In general, the calculation procedure is as follows, for each component of the same type:

1. Set the starting and ending grid indices for the component.
2. Calculate flow field conditions required as inputs to the component performance map. (For the compression stage these are the conditions at the grid point just upstream of the stage, as will be explained in more detail below.)
3. Calculate the external variables required as inputs to the component performance map. (For the compression stage this would be a function of the mechanical speed obtained from NAMELIST input or from an auxiliary subroutine.)
4. Calculate the component performance map output variables, needed to calculate the source term values. This could be done by calling a subroutine.
5. Calculate the source term value(s) at each grid point associated with the component.
6. Add the source term value(s) to the right hand side of the continuity, energy, and momentum equations.

MODIFICATIONS TO SUBROUTINE USS FOR COMPRESSION STAGE SOURCE TERMS

Implementation of the compression stage source terms is illustrated in the FORTRAN listings of USS provided at the end of this appendix. A listing for the variable flow area case is given first, followed by a listing for the constant area case. The heat addition section of USS was modified to implement and test the compression stage source terms derived in appendix B. For the following discussion, the reader should refer to the first listing. Throughout the discussion, FORTRAN program variables are expressed in all capital letters. Corresponding quantities from appendix B are given in square brackets.

At line 5, if there are no stages (i.e., $NHEAT \leq 0$) this section of code is bypassed by jumping to statement 800. Most of the lines from 6 to 17 are standard for all source term sections of USS. JBEG and JEND (lines 7 and 8) are the beginning and ending grid indices for the stage. The variable JSTAGE was added to identify the grid point just upstream of the stage that will not have a contribution from the stage source term. Flow variables at JSTAGE are needed to compute inputs to the compression stage map. As mentioned in the discussion of the demonstration transient, a stage map was not used. Instead, the pressure ratio was input as a function of time by means of the NAMELIST HEAT arrays HEATIN and TIMHT. The instantaneous stage pressure ratio PR $[P_{t3}/P_{t2}]$ is determined by interpolation using subroutine LININT (lines 14 to 19). The adiabatic efficiency ETA $[\eta]$ is input by means of NAMELIST HEAT variable XHEAT(2) on line 20. When a compression stage performance map is used the following steps should be taken:

1. Calculate corrected airflow at grid point JSTAGE.
2. Determine mechanical speed (e.g., by interpolation of external input via NAMELIST) and compute the associated corrected speed.
3. Call the performance map subroutine to obtain stage total pressure ratio and adiabatic efficiency.

Once PR and ETA are known, the ratio DTOVRT $[(T_{t3} - T_{t2})/T_{t2}]$ is calculated on line 30. The variables Q(J,1) $[\hat{\rho}]$, Q(J,2) $[\hat{\rho}\hat{u}]$, and Q(J,3) $[\hat{E}]$ are the LAPIN primitive variables at grid point J (grid point JSTAGE on lines 31 and 32). They are used to calculate TTJSTAGE $[T_{t2}/T_{t0}]$ on line 34. The variable CELHRT3 $[3(T_{t3} - T_{t2})/T_{t0}]$, needed for the energy source term, is calculated on line 36. The variable DPOVRP $[(P_{t3} - P_{t2})/P_{t0}]$, needed for the momentum source term calculation, is calculated in a similar manner (lines 37 to 42).

The next step is to distribute the source terms to the grid points encompassed by the stage. In general, the source term contribution at a grid point covers a cell region that extends between stations (i) and (i+1) that are midway between the grid point (j) and the adjacent upstream (j-1) and downstream (j+1) grid points (see sketch in appendix B). Cell lengths depend on the grid spacing. Grid points at the component end points are handled specially.

The loop for calculating the source term contributions at each component grid point begins on line 44. The cell length DXCEL $[(x_{i+1} - x_i)/L_{ref}]$ for the grid point is calculated first in the section from line 45 to 69. This section is common to most of the LAPIN source terms. The portion of the total temperature change for the cell is calculated on line 73, based on cell length. Finally, the energy source term due to the stage is calculated and added to the cumulative energy source term for grid point J, G(J,3), on line 74. PSX(J) is a coordinate transformation term.

Calculation of the momentum equation source term for the grid point begins on line 76. In order to calculate this term it is necessary to determine the total pressure ratio and total temperature ratio and the exit Mach number for each grid cell. The cell inlet total pressure PTXX $[P_{t_i}]$, total temperature TTX $[T_{t_i}]$, and Mach number XXM $[M_i]$ are based on information from the previous time step. These values are obtained from average flow variables, as shown on lines 85 to 95. A slightly different procedure is used (lines 99 to 110) for the first grid point (JBEG). For this implementation of the compression stage source terms, it is assumed that the change in total pressure and temperature across each cell is proportional to the cell length. That fact, along with the previously determined values of DTOVRT and DPOVRP, is used to calculate the cell total temperature and pressure ratios, TTR $[T_{t_{i+1}}/T_{t_i}]$ and PTR $[P_{t_{i+1}}/P_{t_i}]$, on lines 115 and 116. The cell exit Mach number XXM2 $[M_{i+1}]$ still remains to be calculated. This is done using an iterative procedure on lines 117 to 140, where the inlet and exit flow areas for the cell were obtained on lines 80 to 83. Note that the iterative procedure depends on whether the cell inlet Mach number is subsonic or supersonic. All variables necessary to calculate the momentum source term for the compression stage are now available. The term is already proportioned to the length of the cell by virtue of the method used to calculate the cell total temperature and pressure ratios. The factors FAC1 $[FAC_1]$ and FAC2 $[FAC_2]$ for the momentum source term, are calculated on lines 142 and 143, respectively. Finally, the source term itself is calculated on lines 144 to 146 and added to the cumulative momentum source term for grid

point J, G(J,2), on line 147. The terms Q2AVG, Q1AVG, and GM1*(Q3AVG - 0.5 *Q2AVG*Q2AVG/Q1AVG) represent $\hat{\rho}_i \hat{u}_i$, $\hat{\rho}_i$, and \hat{p}_i , respectively. PSX(J) is again a coordinate transformation term.

Calculation of the momentum term at each grid point is simplified, when the flow area through the compression stage is constant. The Fortran listing for the constant area case is given at the end of the appendix. The total energy term for the stage is calculated the same as before (lines 20 to 31). As shown in appendix B, a total momentum source term for the stage can also be calculated. The actual procedure is given on lines 33 to 65. The loop for calculating the source terms at each grid point (lines 67 to 98) is reduced to calculating the grid cell length (lines 69 to 94) and distributing the total source terms to the grid point in proportion to the cell length (lines 95 to 97). The simplification due to constant area is evident from the much shorter DO 740 loop, compared to the loop for the variable area version.

```

1  C **> HEAT RELEASE section of SUBROUTINE USS as modified for
2  C **> testing of COMPRESSION (FAN) STAGE SOURCE TERMS
3  C **> (Version used for variable flow area through stage)
4  C **>
5      IF(NHEAT .LE. 0) GOTO 800
6      DO 710 I=1,NHEAT
7          JBEG=IHTST(I)
8          JEND=IHTEND(I)
9          XHCELFF = 0.5E+0*(X(JBEG)+X(JBEG+1))
10         IF (XHEAT(I) .GT. XHCELFF) JSTAGE=JBEG
11         IF (XHEAT(I) .LE. XHCELFF) JSTAGE=JBEG-1
12         NPTS=JEND - JBEG + 3
13         DO 720 LC=1,20
14 720    HEATN(LC)=HEATIN(LC,I)
15         TIME=TAU/TIMFAC
16    C Get the current total pressure ratio HEATT(I)
17        CALL LININT(20,TIME,HEATT(I),TIMHT,HEATN,IDUM)
18        IF (HEATT(I) .LE. 1.0) GO TO 710
19        PR=HEATT(I)
20        ETA=XHEAT(2)
21    C
22    C If using a compressor map
23    C (1) calculate corrected airflow at JSTAGE
24    C (2) interpolate for mechanical speed (external input) and calculate
25    C corrected speed
26    C (3) call map subroutine to get total pressure ratio (PR) and efficiency
27    C (ETA) from performance map
28    C
29    C Calculate the overall temperature delta (TT3-TT2)/TT0
30        DTOVRT=(PR**(GM1/GAM)-1.0)/ETA
31        UXX=Q(JSTAGE,2)/Q(JSTAGE,1)
32        EXX=Q(JSTAGE,3)/Q(JSTAGE,1)
33    C Constant in TTXX (1.2) & PTXX (1.35209) calcs assumes GAMA=1.4
34        TTJSTAGE=GM1*(GAM*EXX - 0.5*GM1*UXX*UXX)/1.2
35        DTOVRT=DTOVRT*TTJSTAGE
36        CELHRT3=3.0*DTOVRT
37    C Calculate the overall pressure delta (PT3-PT2)/PT0
38        PXX=GM1*(Q(JSTAGE,3)-.5*Q(JSTAGE,2)*UXX)
39        TXX=GAM*PXX/Q(JSTAGE,1)
40        PXX=PXX*PSX(JSTAGE)/AR(JSTAGE)
41        PTJSTAGE=PXX*((1.2*TTJSTAGE/TXX)**(GAM/GM1))/1.35209
42        DPOVRP=(PR - 1.0)*PTJSTAGE
43    C Begin loop for calculating source terms at each grid point
44        DO 740 J=JBEG,JEND
45        DXCEL=0.5E+0*(X(J+1)-X(J-1))
46        IF(J .NE. JBEG) GO TO 732

```

```

47     XHCELFB = 0.5E+0*(X(J)+X(J+1))
48     IF(XHEAT(I) .GT. XHCELFB) DXCEL=0.0E+0
49     C**>
50     C**> IF DXCEL=0 the source terms will be 0 ==> go to end of DO LOOP
51     C**>
52     IF(DXCEL .EQ. 0.) GO TO 740
53     IF(XHEAT(I) .LE. XHCELFB) DXCEL=0.5E+0*(X(J)+X(J+1))-XHEAT(I)
54     732 CONTINUE
55     IF(J .NE. JBEG+1) GO TO 734
56     XHCELB = 0.5E+0*(X(J)+X(J-1))
57     IF(XHEAT(I) .GT. XHCELB) DXCEL=0.5E+0*(X(J)+X(J+1))-XHEAT(I)
58     734 CONTINUE
59     IF(J .NE. JEND-1) GO TO 736
60     XHCELFB = 0.5E+0*(X(J)+X(J+1))
61     IF(ENDHT(I) .LT. XHCELFB) DXCEL=ENDHT(I)-
62     1           0.5E+0*(X(J)+X(J-1))
63     736 CONTINUE
64     IF(J .NE. JEND) GO TO 738
65     XHCELB = 0.5E+0*(X(J)+X(J-1))
66     IF(ENDHT(I) .LE. XHCELB) DXCEL=0.0E+0
67     IF(ENDHT(I) .GT. XHCELB) DXCEL=ENDHT(I)-
68     1           0.5E+0*(X(J)+X(J-1))
69     738 CONTINUE
70     C
71     C Calculate the cell work term for the energy equation
72     C
73     CELHR=(DXCEL/LHEAT(I))*CELHRT3
74     G(J,3)=G(J,3)+CELHR*Q(J,2)*PSX(J)
75     C
76     C Calculate the cell force term for the momentum equation
77     C
78
79     C START local avg (J) variables
80     AR1=(AR(J) + AR(J-1))/2.0
81     AR2=(AR(J) + AR(J+1))/2.0
82     if (j .ne. jend) go to 760
83     CALL LININT(NPTS,endht(i),ar2,x(jbegm1),ar(jbegm1),IDUM)
84     760 CONTINUE
85     Q1AVG=(Q(J-1,1) + Q(J,1))/2.0
86     Q2AVG=(Q(J-1,2) + Q(J,2))/2.0
87     Q3AVG=(Q(J-1,3) + Q(J,3))/2.0
88     UXX=Q2AVG/Q1AVG
89     EXX=Q3AVG/Q1AVG
90     TTXX=GM1*(GAM*EXX - 0.5*GM1*UXX*UXX)/1.2
91     PXX=GM1*(Q3AVG-.5*Q2AVG*UXX)
92     TXX=GAM*PXX/Q1AVG

```

```

93      XXM=UXX/SQRT(TXX)
94      PXX=PXX*PSX(J)/AR1
95      PTXX=PXX*((1.2*TTXX/TXX)**(GAM/GM1))/1.35209
96  C   END local avg (J) variables
97
98      if (j .ne. jbeg) go to 759
99      CALL LININT(NPTS,xheat(i),q1avg,x(jbegm1),q(jbegm1,1),IDUM)
100     CALL LININT(NPTS,xheat(i),q2avg,x(jbegm1),q(jbegm1,2),IDUM)
101     CALL LININT(NPTS,xheat(i),q3avg,x(jbegm1),q(jbegm1,3),IDUM)
102     CALL LININT(NPTS,xheat(i),ar1,x(jbegm1),ar(jbegm1),IDUM)
103     UXX=Q2AVG/Q1AVG
104     EXX=Q3AVG/Q1AVG
105     TTXX=GM1*(GAM*EXX - 0.5*GM1*UXX*UXX)/1.2
106     PXX=GM1*(Q3AVG-.5*Q2AVG*UXX)
107     TXX=GAM*PXX/Q1AVG
108     XXM=UXX/SQRT(TXX)
109     PTXX=PTJSTAGE
110     TTXX=TTJSTAGE
111 759   CONTINUE
112
113
114  C   Calculate tot. temp. and press. ratio for cell
115     TTR=DTOVRT*DXCEL/TTXX/LHEAT(I) + 1.0
116     PTR=DPOVRP*DXCEL/PTXX/LHEAT(I) + 1.0
117  C
118  C   Iterate for cell exit Mach no.
119  C   CAUTION! If CF2>CFMAX the iteration will not converge.
120  C   (CFMAX=CF(M=1.0)=1.2**-3=.5787037)
121  C
122     CF1=XXM*(1.0 + 0.2*XXM*XXM)**-3
123     CF2=AR1*CF1*SQRT(TTR)/(PTR*AR2)
124     IF (XXM .GT. 1.0) GO TO 751
125  C   Subsonic cell inlet Mach no.
126     XMGUESS=0.3
127 753   XXM2=CF2*(1.0 + 0.2*XMGUESS*XMGUESS)**3
128     DELM=ABS(XMGUESS-XXM2)
129     IF (DELM .LT. 0.0001) GO TO 752
130     XMGUESS=XXM2
131     GO TO 753
132  C   Supersonic cell inlet Mach no.
133 751   XMGUESS=1.5
134 754   XXM2=5.0*((XMGUESS/CF2)**0.33333 - 1.0)
135     XXM2=SQRT(XXM2)
136     DELM=ABS(XMGUESS-XXM2)
137     IF (DELM .LT. 0.0001) GO TO 752
138     XMGUESS=XXM2

```

```

139      GO TO 754
140      752  CONTINUE
141      XMRAT=(1.0 + 0.2*XXM*XXM)/(1.0 + 0.2*XXM2*XXM2)
142      FAC1=SQRT(XMRAT)*XXM2*SQRT(TTR)/XXM - 1.0
143      FAC2=(AR1 +AR2)*(PTR*XMRAT**3.5 - 1.0)/(2.0*AR1)
144      FFAC1=Q2AVG*Q2AVG*FAC1/Q1AVG
145      FFAC2=GM1*FAC2*(Q3AVG - 0.5*Q2AVG*Q2AVG/Q1AVG)
146      FFAC=(FFAC1 + FFAC2)
147      G(J,2) = G(J,2) + PSX(J)*FFAC
148      740  CONTINUE
149      710  CONTINUE
150      800  CONTINUE

```

```

1  C **> HEAT RELEASE section of SUBROUTINE USS as modified for
2  C **> testing of COMPRESSION (FAN) STAGE SOURCE TERMS
3  C **>   (Version used for constant flow area through stage)
4  C **>
5      IF(NHEAT .LE. 0) GOTO 800
6      DO 710 I=1,NHEAT
7          DO 720 LC=1,20
8 720  HEATN(LC)=HEATIN(LC,I)
9          TIME=TAU/TIMFAC
10         CALL LININT(20,TIME,HEATT(I),TIMHT,HEATN,IDUM)
11         JBEG=IHTST(I)
12         JEND=IHTEND(I)
13
14  C **> Begin additions/changes
15         XHCELF = 0.5E+0*(X(JBEG)+X(JBEG+1))
16         IF (XHEAT(I) .GT. XHCELF) JSTAGE=JBEG
17         IF (XHEAT(I) .LE. XHCELF) JSTAGE=JBEG-1
18         IF (HEATT(I) .LE. 1.0) GO TO 710
19  C
20  C Calculate the work term for the energy equation
21  C
22  C HEATT(I) is fan PR & XHEAT(2) is used as the fan efficiency
23  C
24         DTOVRT=(HEATT(I)**(GM1/GAM)-1.0)/XHEAT(2)
25         TTR=DTOVRT + 1.0
26         UXX=Q(JSTAGE,2)/Q(JSTAGE,1)
27         EXX=Q(JSTAGE,3)/Q(JSTAGE,1)
28  C Constant in TTX (1.2) calc assumes GAMA=1.4
29         TTX=GM1*(GAM*EXX - 0.5*GM1*UXX*UXX)/1.2
30         DTOVRT=DTOVRT*TTX
31         CELHRT=3.0*DTOVRT/LHEAT(I)
32  C
33  C Calculate the total force term for the momentum equation
34  C
35         PXX=GM1*(Q(JSTAGE,3)-.5*Q(JSTAGE,2)*UXX)
36         TXX=GAM*PXX/Q(JSTAGE,1)
37         XXM=UXX/SQRT(TXX)
38         CF1=XXM*(1.0 + 0.2*XXM*XXM)**-3
39         CF2=CF1*SQRT(TTR)/HEATT(I)
40  C**>
41  C**> CFMAX=(1.2)**-3=.5787037
42  C**> IF CF2>CFMAX The following iteration will not converge!
43  C**> (not a likely situation - CF2 should be < CF1)
44  C**>
45         IF (XXM .GT. 1.0) GO TO 751
46         XMGUESS=0.3

```



```

47 753  XXM2=CF2*(1.0 + 0.2*XMGUESS*XMGUESS)**3
48      DELM=ABS(XMGUESS-XXM2)
49      IF (DELM .LT. 0.0001) GO TO 752
50      XMGUESS=XXM2
51      GO TO 753
52 751  XMGUESS=1.5
53 754  XXM2=5.0*((XMGUESS/CF2)**0.33333 - 1.0)
54      XXM2=SQRT(XXM2)
55      DELM=ABS(XMGUESS-XXM2)
56      IF (DELM .LT. 0.0001) GO TO 752
57      XMGUESS=XXM2
58      GO TO 754
59 752  CONTINUE
60      XMRAT=(1.0 + 0.2*XXM*XXM)/(1.0 + 0.2*XXM2*XXM2)
61      FAC1=SQRT(XMRAT)*XXM2*SQRT(TTR)/XXM - 1.0
62      FAC2=HEATT(I)*XMRAT**3.5 - 1.0
63      FFAC1=Q(JSTAGE,2)*Q(JSTAGE,2)*FAC1/Q(JSTAGE,1)
64      FFAC2=PXX*FAC2
65      FFAC=FFAC1 + FFAC2
66  C **> End additions/changes
67  C Begin loop for calculating source terms at each grid point
68      DO 740 J=JBEG,JEND
69      DXCEL=0.5E+0*(X(J+1)-X(J-1))
70      IF(J .NE. JBEG) GO TO 732
71      XHCELF = 0.5E+0*(X(J)+X(J+1))
72  C**>
73      IF(XHEAT(I) .GT. XHCELF) DXCEL=0.0E+0
74  C**>
75  C**> If DXCEL=0 the source terms are 0 ==> go to end of DO LOOP
76  C**>
77      IF(DXCEL .EQ. 0.) GO TO 740
78      IF(XHEAT(I) .LE. XHCELF) DXCEL=0.5E+0*(X(J)+X(J+1))-XHEAT(I)
79 732  CONTINUE
80      IF(J .NE. JBEG+1) GO TO 734
81      XHCELB = 0.5E+0*(X(J)+X(J-1))
82      IF(XHEAT(I) .GT. XHCELB) DXCEL=0.5E+0*(X(J)+X(J+1))-XHEAT(I)
83 734  CONTINUE
84      IF(J .NE. JEND-1) GO TO 736
85      XHCELF = 0.5E+0*(X(J)+X(J+1))
86      IF(ENDHT(I) .LT. XHCELF) DXCEL=ENDHT(I)-
87      1          0.5E+0*(X(J)+X(J-1))
88 736  CONTINUE
89      IF(J .NE. JEND) GO TO 738
90      XHCELB = 0.5E+0*(X(J)+X(J-1))
91      IF(ENDHT(I) .LE. XHCELB) DXCEL=0.0E+0
92      IF(ENDHT(I) .GT. XHCELB) DXCEL=ENDHT(I)-

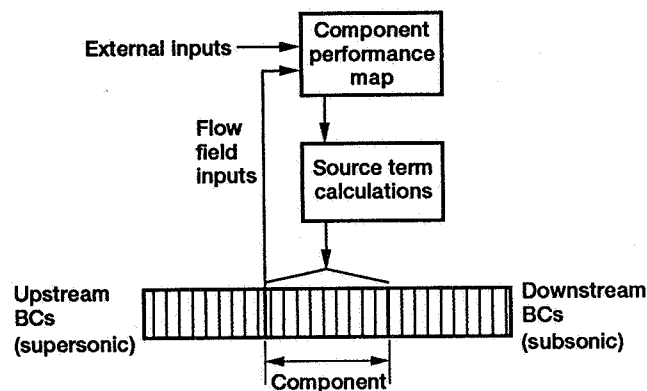
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```
93      1      0.5E+0*(X(J)+X(J-1))
94 738 CONTINUE
95      CELHR=DXCEL*CELHRT
96      G(J,3)=G(J,3)+CELHR*Q(J,2)*PSX(J)
97      G(J,2) = G(J,2) + PSX(J)*FFAC*DXCEL/LHEAT(I)
98 740 CONTINUE
99 710 CONTINUE
100 800 CONTINUE
```

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Continuity: $\frac{\partial(\rho A)}{\partial t} + \frac{\partial(\rho u A)}{\partial x} = M_s$

Momentum: $\frac{\partial(\rho u A)}{\partial t} + \frac{\partial[A(\rho + \rho u^2)]}{\partial x} = p \frac{\partial A}{\partial x} + F_s$

Energy: $\frac{\partial(EA)}{\partial t} + \frac{\partial[Au(E + p)]}{\partial x} = -p \frac{\partial A}{\partial t} + Q_s$

Figure 1.—LAPIN implementation of propulsion component source terms.

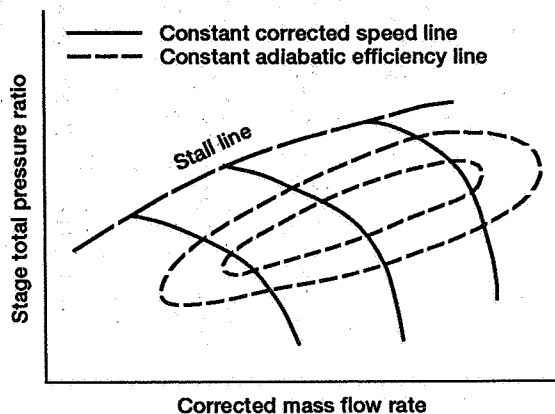
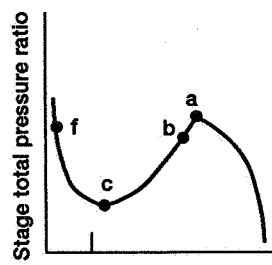
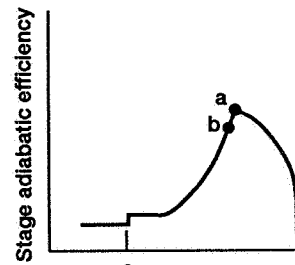


Figure 2.—Typical component performance map for compression stage.



(a) Total pressure ratio.



(b) Adiabatic efficiency

Figure 3.—Compression stage characteristics.

```

**** INPUT DATA MIRROR ****

***** SIMPLE GEOMETRY WITH NO CENTERBODY *****
***** Data for demonstration case *****
&CONTRL SCMC=.TRUE., NMAX=2800, NC=20, TIMMAX=.30,
      TIMDTC=0.00+0, .025, DTIM= .0001, .0001, NDTC=2
&END
&BC MHUSIC=2.0, MHDSBC=.TRUE., MHDSIC=.164,
      PRUSIC= 270.42, TMUSIC=288.36,
      TIMDS=0.00+0, .09, .12, 100.0, MACHDS= 2*1.0, 2*2.0622,
      START=.TRUE., OPCOND=.FALSE.,
&END
&GEOM XC=0.00+0, .7, 1.4, 2.5, 3.5,
      YC=1.0, .81, 1.0, 1.05, 1.5,
      XS=0., 3.6,
      YS=2*0.,
      TWOD=.FALSE., INTYPE=0, XUSLIM=.0,
      XINLET=0., XEXIT=3.5, JMAX=36, JINL=1,
      IPTRAN=8,
      COWLR=.8333333
&END
&BYBL NBL=0 &END
&INJECT &END
&HEAT NHEAT=1,
      XHEAT=1.56, .65, LHEAT=.7,
      TIMHT=0.00+0, 0.03, 0.07, .14, .24, 100.0,
      HEATIN= 2*1.0, 2*1.4, 2*2.4,
&END
&LOSS &END
**** END DATA INPUT ****

***** SIMPLE GEOMETRY WITH NO CENTERBODY *****
***** Data for demonstration case *****

```

```

IMPLICIT SPLIT CHARACTERISTICS
AXISYMMETRIC - VARIABLES NORMALIZED BY INITIAL UPSTREAM STATIC OR TOTAL CONDIONS
UPSTREAM MACH NUMBER BOUNDARY CONDITION, MHUSIC = 2.000000
DOWNSTREAM MACH NO. BOUNDARY CONDITION, MHDSIC = 0.164000
N= 300 TIME= 0.0300 SEC XSHK= 0.0000 XSH2= 0.0000

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X	RHO	U	T	P	M	CFL	MASS	AREA	PT	TT	CMASS
0.0000	1.0000	1.0000	1.0000	1.0000	2.0000	2.9968	1.0000	1.0000	1.0000	1.0000	1.0000
0.1000	1.0771	0.9810	1.0301	1.1095	1.9330	2.9738	1.0000	0.9465	1.0000	1.0000	1.0000
0.2000	1.1649	0.9598	1.0630	1.2382	1.8620	2.9476	1.0000	0.8944	1.0000	1.0000	1.0000
0.3000	1.2661	0.9361	1.0990	1.3914	1.7859	2.9174	1.0000	0.8438	1.0000	1.0000	1.0000
0.4000	1.3844	0.9090	1.1390	1.5768	1.7034	2.8822	1.0000	0.7946	0.9999	1.0000	1.0001
0.5000	1.5259	0.8774	1.1842	1.8069	1.6125	2.8399	1.0000	0.7470	0.9999	1.0000	1.0001
0.6000	1.7005	0.8391	1.2367	2.1031	1.5091	2.7873	1.0000	0.7008	0.9998	1.0000	1.0002
0.7000	1.7773	0.8404	1.2485	2.2190	1.5042	2.7951	0.9799	0.6561	1.0474	1.0075	0.9391
0.8000	3.2229	0.4447	1.6422	5.2927	0.6941	2.1687	1.0045	0.7008	0.9334	1.0003	1.0764
0.9000	3.4609	0.3875	1.6764	5.8019	0.5985	2.0675	1.0017	0.7470	0.9447	0.9981	1.0593
1.0000	3.5584	0.3536	1.7002	6.0500	0.5424	2.0090	0.9999	0.7946	0.9445	1.0001	1.0588
1.1000	3.6403	0.3258	1.7146	6.2415	0.4976	1.9589	1.0007	0.8438	0.9447	0.9997	1.0591
1.2000	3.7017	0.3021	1.7274	6.3944	0.4597	1.9165	1.0002	0.8944	0.9446	1.0002	1.0589
1.3000	3.7542	0.2815	1.7365	6.5193	0.4273	1.8789	1.0004	0.9465	0.9446	1.0000	1.0590
1.4000	3.7964	0.2635	1.7446	6.6231	0.3989	1.8458	1.0002	1.0000	0.9446	1.0001	1.0589
1.5000	3.8028	0.2606	1.7457	6.6386	0.3945	1.8405	1.0001	1.0091	0.9445	1.0000	1.0589
1.6000	3.8089	0.2578	1.7469	6.6538	0.3902	1.8354	1.0000	1.0183	0.9445	1.0000	1.0588
1.7000	3.8150	0.2551	1.7480	6.6685	0.3859	1.8304	1.0000	1.0275	0.9445	1.0000	1.0588
1.8000	3.8209	0.2525	1.7490	6.6829	0.3818	1.8255	1.0000	1.0367	0.9445	1.0000	1.0588
1.9000	3.8267	0.2498	1.7500	6.6969	0.3777	1.8207	1.0000	1.0460	0.9445	1.0000	1.0588
2.0000	3.8323	0.2473	1.7511	6.7106	0.3737	1.8159	1.0000	1.0553	0.9445	1.0000	1.0588
2.1000	3.8378	0.2448	1.7520	6.7239	0.3698	1.8112	1.0000	1.0646	0.9445	1.0000	1.0588
2.2000	3.8430	0.2423	1.7530	6.7369	0.3660	1.8067	1.0000	1.0740	0.9445	1.0000	1.0588
2.3000	3.8482	0.2398	1.7540	6.7495	0.3622	1.8022	1.0000	1.0835	0.9445	1.0000	1.0588
2.4000	3.8531	0.2375	1.7549	6.7619	0.3585	1.7977	1.0000	1.0930	0.9445	1.0000	1.0588
2.5000	3.8580	0.2351	1.7558	6.7738	0.3549	1.7934	1.0000	1.1025	0.9445	1.0000	1.0588
2.6000	3.9001	0.2138	1.7634	6.8775	0.3221	1.7538	1.0000	1.1990	0.9445	1.0000	1.0588
2.7000	3.9331	0.1956	1.7694	6.9592	0.2941	1.7196	1.0000	1.2996	0.9445	1.0000	1.0588
2.8000	3.9595	0.1798	1.7741	7.0247	0.2700	1.6899	1.0000	1.4042	0.9445	1.0000	1.0588
2.9000	3.9809	0.1660	1.7779	7.0778	0.2490	1.6637	0.9999	1.5129	0.9445	1.0000	1.0588
3.0000	3.9984	0.1538	1.7811	7.1213	0.2305	1.6405	0.9999	1.6256	0.9445	1.0000	1.0588
3.1000	4.0128	0.1430	1.7836	7.1574	0.2142	1.6198	0.9999	1.7424	0.9445	1.0000	1.0588

Figure 4.—Listing of input data and results at selected time steps for simulated fan starting transient.

3.2000	4.0249	0.1333	1.7858	7.1874	0.1996	1.6013	1.0000	1.8632	0.9445	1.0000	1.0588
3.3000	4.0350	0.1247	1.7876	7.2127	0.1865	1.5846	1.0000	1.9881	0.9445	1.0000	1.0588
3.4000	4.0435	0.1168	1.7891	7.2341	0.1747	1.5695	1.0000	2.1170	0.9445	1.0000	1.0588
3.5000	4.0508	0.1097	1.7903	7.2524	0.1640	1.5558	1.0000	2.2500	0.9445	1.0000	1.0588
N= 1000 TIME= 0.1000 SEC XSHK= 0.0000 XSH2= 0.0000											
X	RHO	U	T	P	M	CFL	MASS	AREA	PT	TT	CMASS
0.0000	1.0000	1.0000	1.0000	1.0000	2.0000	2.9968	1.0000	1.0000	1.0000	1.0000	1.0000
0.1000	1.0771	0.9810	1.0301	1.1095	1.9330	2.9738	1.0000	0.9465	1.0000	1.0000	1.0000
0.2000	1.1649	0.9598	1.0630	1.2382	1.8620	2.9476	1.0000	0.8944	1.0000	1.0000	1.0000
0.3000	1.2661	0.9361	1.0990	1.3914	1.7859	2.9174	1.0000	0.8438	1.0000	1.0000	1.0000
0.4000	1.3844	0.9090	1.1390	1.5768	1.7034	2.8822	1.0000	0.7946	0.9999	1.0000	1.0001
0.5000	1.5259	0.8774	1.1842	1.8069	1.6125	2.8399	1.0000	0.7470	0.9999	1.0000	1.0001
0.6000	1.7005	0.8391	1.2367	2.1031	1.5091	2.7873	1.0000	0.7008	0.9998	1.0000	1.0002
0.7000	1.9299	0.7898	1.3010	2.5109	1.3848	2.7172	1.0000	0.6561	0.9996	1.0000	1.0004
0.8000	1.7005	0.8391	1.2367	2.1031	1.5091	2.7873	1.0000	0.7008	0.9998	1.0000	1.0002
0.9000	1.5259	0.8774	1.1842	1.8069	1.6125	2.8399	1.0000	0.7470	0.9999	1.0000	1.0001
1.0000	1.3844	0.9090	1.1390	1.5768	1.7034	2.8822	1.0000	0.7946	0.9999	1.0000	1.0001
1.1000	1.2661	0.9361	1.0990	1.3914	1.7859	2.9174	1.0000	0.8438	1.0000	1.0000	1.0000
1.2000	1.1649	0.9598	1.0630	1.2382	1.8620	2.9476	1.0000	0.8944	1.0000	1.0000	1.0000
1.3000	1.0771	0.9810	1.0301	1.1095	1.9330	2.9738	1.0000	0.9465	1.0000	1.0000	1.0000
1.4000	1.0000	1.0000	1.0000	1.0000	2.0000	2.9968	1.0000	1.0000	1.0000	1.0000	1.0000
1.5000	0.9880	1.0030	0.9952	0.9832	2.0109	3.0004	1.0000	1.0091	1.0000	1.0000	1.0000
1.6000	0.9654	1.0173	0.9901	0.9558	2.0448	3.0264	1.0000	1.0183	1.0248	1.0100	0.9807
1.7000	0.9327	1.0435	0.9848	0.9186	2.1030	3.0761	1.0000	1.0275	1.0785	1.0310	0.9415
1.8000	0.9018	1.0696	0.9805	0.8843	2.1604	3.1261	1.0000	1.0367	1.1358	1.0532	0.9036
1.9000	0.8735	1.0945	0.9773	0.8537	2.2143	3.1743	1.0000	1.0460	1.1931	1.0754	0.8692
2.0000	1.9769	0.5512	1.6686	3.2986	0.8535	2.3917	1.1499	1.0553	0.6786	1.0620	1.7464
2.1000	2.7080	0.3561	1.9046	5.1576	0.5160	2.0900	1.0265	1.0646	0.7904	1.1145	1.3710
2.2000	2.8922	0.3317	1.9573	5.6611	0.4741	2.0602	1.0303	1.0740	0.8439	1.1363	1.3014
2.3000	2.9890	0.3194	1.9844	5.9312	0.4535	2.0454	1.0345	1.0835	0.8729	1.1478	1.2697
2.4000	2.9956	0.3174	1.9851	5.9464	0.4506	2.0417	1.0393	1.0930	0.8736	1.1476	1.2745
2.5000	2.9958	0.3165	1.9814	5.9359	0.4497	2.0385	1.0453	1.1025	0.8716	1.1453	1.2836
2.6000	3.0518	0.2878	1.9883	6.0681	0.4082	1.9836	1.0532	1.1190	0.8698	1.1414	1.2936
2.7000	3.1030	0.2637	1.9883	6.1695	0.3740	1.9354	1.0633	1.2996	0.8684	1.1355	1.3047
2.8000	3.1531	0.2430	1.9819	6.2491	0.3453	1.8919	1.0761	1.4042	0.8673	1.1273	1.3173
2.9000	3.2037	0.2251	1.9705	6.3129	0.3208	1.8520	1.0912	1.5129	0.8664	1.1172	1.3312
3.0000	3.2540	0.2095	1.9560	6.3649	0.2995	1.8156	1.1080	1.6256	0.8657	1.1062	1.3461
3.1000	3.3023	0.1956	1.9405	6.4080	0.2808	1.7823	1.1254	1.7424	0.8651	1.0950	1.3614
3.2000	3.3460	0.1833	1.9261	6.4447	0.2641	1.7525	1.1425	1.8632	0.8646	1.0850	1.3764
3.3000	3.3820	0.1722	1.9149	6.4762	0.2489	1.7264	1.1579	1.9881	0.8641	1.0770	1.3906
3.4000	3.4073	0.1623	1.9087	6.5034	0.2349	1.7043	1.1707	2.1170	0.8637	1.0721	1.4034
3.5000	3.4202	0.1533	1.9085	6.5273	0.2220	1.6864	1.1801	2.2500	0.8634	1.0707	1.4143
N= 2800 TIME= 0.2800 SEC XSHK= 0.0000 XSH2= 0.0000											
X	RHO	U	T	P	M	CFL	MASS	AREA	PT	TT	CMASS
0.0000	1.0000	1.0000	1.0000	1.0000	2.0000	2.9968	1.0000	1.0000	1.0000	1.0000	1.0000
0.1000	1.0771	0.9810	1.0301	1.1095	1.9330	2.9738	1.0000	0.9465	1.0000	1.0000	1.0000
0.2000	1.1649	0.9598	1.0630	1.2382	1.8620	2.9476	1.0000	0.8944	1.0000	1.0000	1.0000
0.3000	1.2661	0.9361	1.0990	1.3914	1.7859	2.9174	1.0000	0.8438	1.0000	1.0000	1.0000
0.4000	1.3844	0.9090	1.1390	1.5768	1.7034	2.8822	1.0000	0.7946	0.9999	1.0000	1.0001
0.5000	1.5259	0.8774	1.1842	1.8069	1.6125	2.8399	1.0000	0.7470	0.9999	1.0000	1.0001
0.6000	1.7005	0.8391	1.2367	2.1031	1.5091	2.7873	1.0000	0.7008	0.9998	1.0000	1.0002
0.7000	1.9299	0.7898	1.3010	2.5109	1.3848	2.7172	1.0000	0.6561	0.9996	1.0000	1.0004
0.8000	1.7005	0.8391	1.2367	2.1031	1.5091	2.7873	1.0000	0.7008	0.9998	1.0000	1.0002
0.9000	1.5259	0.8774	1.1842	1.8069	1.6125	2.8399	1.0000	0.7470	0.9999	1.0000	1.0001
1.0000	1.3844	0.9090	1.1390	1.5768	1.7034	2.8822	1.0000	0.7946	0.9999	1.0000	1.0001
1.1000	1.2661	0.9361	1.0990	1.3914	1.7859	2.9174	1.0000	0.8438	1.0000	1.0000	1.0000
1.2000	1.1649	0.9598	1.0630	1.2382	1.8620	2.9476	1.0000	0.8944	1.0000	1.0000	1.0000
1.3000	1.0771	0.9810	1.0301	1.1095	1.9330	2.9738	1.0000	0.9465	1.0000	1.0000	1.0000
1.4000	1.0000	1.0000	1.0000	1.0000	2.0000	2.9968	1.0000	1.0000	1.0000	1.0000	1.0000
1.5000	0.9880	1.0030	0.9952	0.9832	2.0109	3.0004	1.0000	1.0091	1.0000	1.0000	1.0000
1.6000	0.9442	1.0402	0.9851	0.9300	2.0960	3.0696	1.0000	1.0183	1.0803	1.0281	0.9386
1.7000	0.8764	1.1105	0.9709	0.8509	2.2540	3.2029	1.0000	1.0275	1.2655	1.0874	0.8240
1.8000	0.8202	1.1761	0.9632	0.7900	2.3967	3.3301	1.0000	1.0367	1.4685	1.1499	0.7302
1.9000	0.7742	1.2348	0.9625	0.7452	2.5173	3.4470	1.0000	1.0460	1.6717	1.2124	0.6587
2.0000	0.7354	1.2886	0.9663	0.7106	2.6217	3.5564	1.0000	1.0553	1.8740	1.2748	0.6025
2.1000	0.7016	1.3387	0.9734	0.6829	2.7138	3.6602	1.0000	1.0646	2.0758	1.3373	0.5571
2.2000	0.6717	1.3861	0.9826	0.6601	2.7965	3.7594	1.0000	1.0740	2.2772	1.3997	0.5195
2.3000	0.6535	1.4122	0.9859	0.6443	2.8445	3.8133	1.0000	1.0835	2.3917	1.4341	0.5007
2.4000	0.6461	1.4161	0.9827	0.6349	2.8571	3.8195	1.0000	1.0930	2.4021	1.4372	0.4991
2.5000	0.6397	1.4178	0.9788	0.6262	2.8662	3.8210	1.0000	1.1025	2.4021	1.4372	0.4991
2.6000	0.5817	1.4339	0.9423	0.5481	2.9543	3.8343	1.0000	1.1990	2.4020	1.4372	0.4991
2.7000	0.5313	1.4484	0.9087	0.4828	3.0388	3.8460	1.0000	1.2996	2.4020	1.4372	0.4991
2.8000	0.4872	1.4617	0.8778	0.4277	3.1202	3.8562	1.0000	1.4042	2.4019	1.4372	0.4991

Figure 4.—Continued.

2.9000	0.4485	1.4739	0.8492	0.3808	3.1988	3.8651	1.0000	1.5129	2.4019	1.4372	0.4991
3.0000	0.4142	1.4851	0.8226	0.3408	3.2747	3.8730	1.0000	1.6256	2.4018	1.4372	0.4991
3.1000	0.8419	0.8519	1.8620	1.5677	1.2486	3.0651	1.2497	1.7424	0.5180	1.3570	2.8103
3.2000	1.6008	0.3353	2.4971	3.9973	0.4243	2.2484	1.0000	1.8632	0.5782	1.4372	2.0733
3.3000	1.6216	0.3102	2.5100	4.0704	0.3916	2.2023	1.0000	1.9881	0.5782	1.4372	2.0734
3.4000	1.6386	0.2883	2.5205	4.1302	0.3631	2.1619	1.0000	2.1170	0.5782	1.4372	2.0734
3.5000	1.6527	0.2689	2.5292	4.1798	0.3382	2.1259	1.0000	2.2500	0.5782	1.4372	2.0734

INITIALIZATION TIME: 0.1000E+01 SECONDS

DO LOOP TIME: 0.1110E+03 SECONDS

TOTAL CPU TIME: 0.1120E+03 SECONDS

Figure 4.—Concluded.

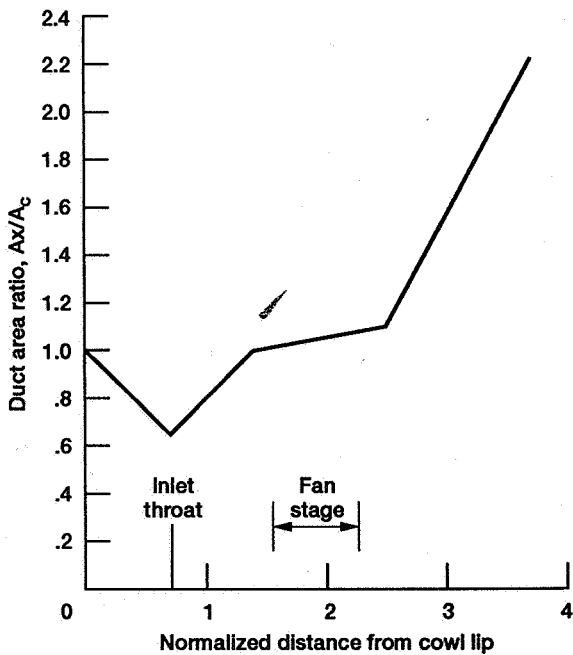
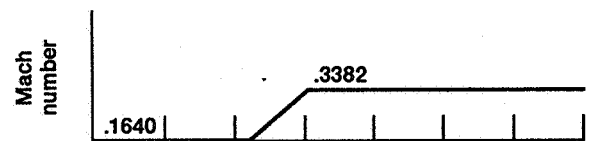
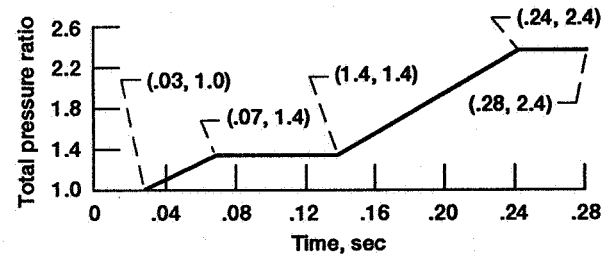


Figure 5.—Duct area distribution for simulated inlet/fan system.



(a) Diffuser exit Mach number.



(b) Fan total pressure ratio.

Figure 6.—Fan stage and diffuser exit boundary condition inputs for simulated fan stage starting transient.

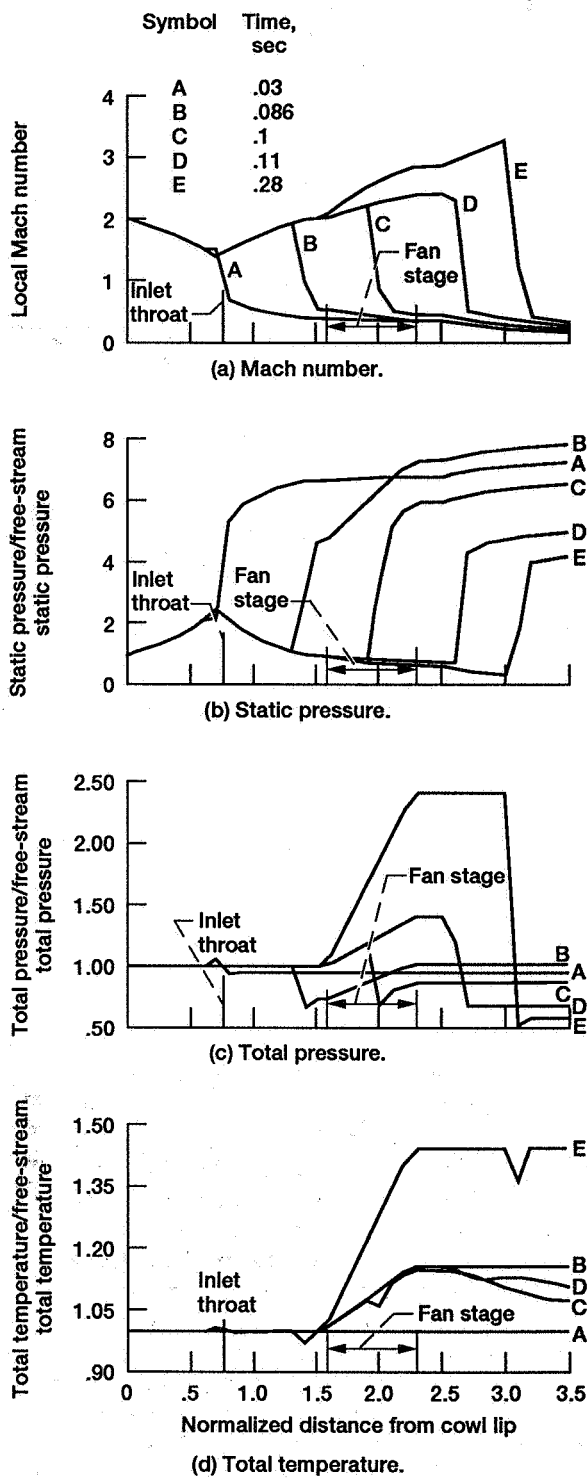


Figure 7.—Axial distributions at various time steps during simulated fan starting transient.

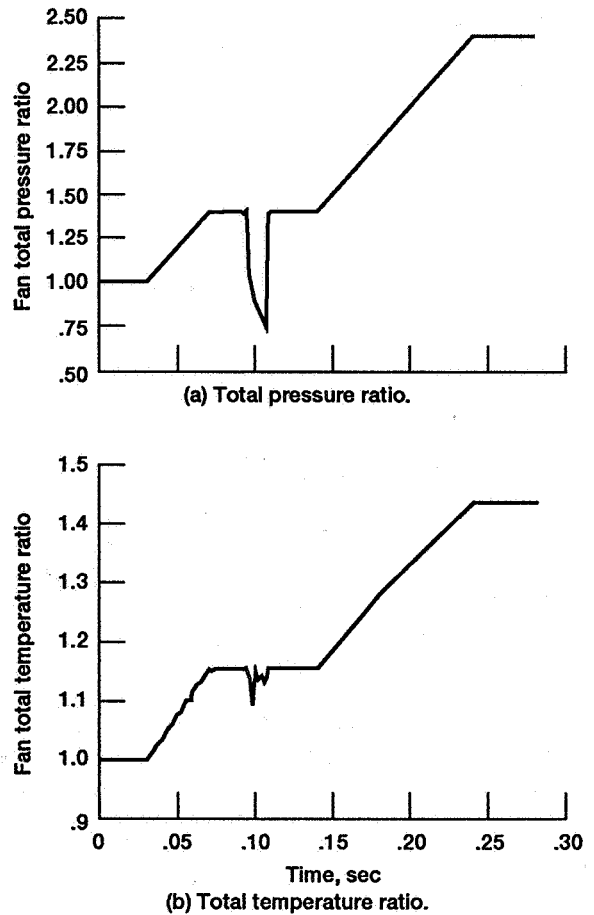


Figure 8.—Calculated fan-ratio time histories during simulated fan starting transient.

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

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1. AGENCY USE ONLY (Leave blank)		2. REPORT DATE December 1991	3. REPORT TYPE AND DATES COVERED Technical Memorandum	
4. TITLE AND SUBTITLE Supersonic Propulsion Simulation by Incorporating Component Models in the Large Perturbation Inlet (LAPIN) Computer Code			5. FUNDING NUMBERS WU-509-10-03	
6. AUTHOR(S) Gary L. Cole and Jacques C. Richard				
7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES) National Aeronautics and Space Administration Lewis Research Center Cleveland, Ohio 44135-3191			8. PERFORMING ORGANIZATION REPORT NUMBER E-6494	
9. SPONSORING/MONITORING AGENCY NAMES(S) AND ADDRESS(ES) National Aeronautics and Space Administration Washington, D.C. 20546-0001			10. SPONSORING/MONITORING AGENCY REPORT NUMBER NASA TM-105193	
11. SUPPLEMENTARY NOTES Responsible person, Gary L. Cole, (216) 433-3655.				
12a. DISTRIBUTION/AVAILABILITY STATEMENT Unclassified - Unlimited Subject Category 07			12b. DISTRIBUTION CODE	
13. ABSTRACT (Maximum 200 words) An approach to simulating the internal flows of supersonic propulsion systems is presented. The approach is based on a fairly simple modification of the publicly-available Large Perturbation Inlet (LAPIN) computer code. LAPIN uses a quasi-one-dimensional, inviscid, unsteady formulation of the continuity, momentum, and energy equations. The equations are solved using a shock-capturing, finite-difference algorithm. The original code, developed for simulating supersonic inlets, includes engineering models of unstart/restart, bleed, bypass, and variable duct geometry, by means of source terms in the equations. The source terms also provide a mechanism for incorporating, with the inlet, propulsion system components such as compressor stages, combustors, and turbine stages. This requires each component to be distributed axially over a number of grid points. Because of the distributed nature of such components, this representation should be more accurate than a lumped-parameter model. Components can be modeled by performance map(s), which in turn are used to compute the source terms. The report describes the general approach first. Then, simulation of a compressor/fan stage is discussed to demonstrate the approach in detail. Generalized source terms, valid for either subsonic or supersonic flow through the compressor, are derived for the momentum and energy equations. A starting transient is presented as a demonstration case for a single-stage, supersonic fan in a mixed-compression inlet. Discussion of the compressor stage implementation in LAPIN provides guidance for the addition of other components.				
14. SUBJECT TERMS Supersonic inlets; Internal compression inlets; Compressors; One dimensional flow; Systems simulation; Supersonic compressors; Propulsion; Difference equations			15. NUMBER OF PAGES 32	
			16. PRICE CODE A03	
17. SECURITY CLASSIFICATION OF REPORT Unclassified	18. SECURITY CLASSIFICATION OF THIS PAGE Unclassified	19. SECURITY CLASSIFICATION OF ABSTRACT Unclassified	20. LIMITATION OF ABSTRACT	