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COMPUTER SIMULATION
OF ENGINE SYSTEMS

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Abstract

Availability of a suitable propulsion system is generally acknowledged to be a key requirement for the successful development of a new airplane. In order to evaluate the suitability of a propulsion system, it is necessary to be able to calculate both the steady state and transient performance of the engine throughout the flight regime. In addition, installation effects on thrust and specific fuel consumption must be accountef for as well as engine weight, dimensions, and cost. This paper discusses the availability throughout the government and industry of analytical methods for calculating these quantities.

Nomenclature

CET	combustor exit temperature - $^{\circ}R$, $^{\circ}K$
C_f	flow coefficient
CF	corrected flow - lb_m/sec, Kg/sec
E	error
f	function
f^1	first derivative of function
I	moment of inertia - lb_m-ft^2, Kg-m^2
M	Mach number
N	speed - rpm, rad/sec
P	total pressure - lb/in^2, new/m^2
P_r	pressure ratio
R	gas constant - Btu/lb_{mole}-$^{\circ}R$, joules/Kg_{mole}-$^{\circ}K$
R	ray line
T	total temperature - $^{\circ}R$, $^{\circ}K$
t	time - sec
u	internal energy - Btu, joules
V	variable
$\tilde{V}$	total volume - ft^3, m^3
W	airflow - lb_m/sec, Kg/sec
$\dot{w}$	airflow - lb_m/sec, Kg/sec
γ	specific heat ratio
δ	corrected pressure
∂	partial derivative
δV	change in V
Δh	change in enthalpy - Btu/lb-sec, cal/Kg-sec
η	efficiency
θ	corrected temperature

Subscripts:

a	adiabatic
c	compressor
ext	external
P	polytropic
R	recovery
t	turbine

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Introduction

The availability of a suitable propulsion system is generally acknowledged to be a key requirement for the successful development of a new airplane. In order to evaluate the suitability of a propulsion system, it is necessary to be able to calculate the steady state thrust and specific fuel consumption throughout the flight regime. It is also imperative to evaluate the transient behavior of the engine. The path from one steady state operating point to another is fraught with danger. Will an engine overspeed? overtemperature? surge? stall?

Once the analyst is satisfied that the engine will operate satisfactorily, he must then account for installation effects on the engine performance. Inlet/engine/nozzle/afterbody interactions must be determined. The next step in determining suitability might be the determination of the engine weight and dimensions and finally the engine cost.

A great deal of work has been expended by industry and/or government personnel to develop analytical tools capable of evaluating the aforementioned factors in determining engine suitability. This paper will trace the historical development of some of these tools and discuss the status as of today. The computer codes discussed will be far from all-inclusive since probably no one has a complete list of those that exist. We would appreciate information being forwarded to us about any codes that do exist that are not mentioned especially if they are not restricted to the internal use only of the originating organization.

Steady State Engine Calculations

On the first figure are reviewed some of the reasons we want to simulate turbine engines. For any particular mission application, there is a choice to be made of the best engine cycle from almost an infinite set. Past experience has shown that the faster an airplane flies, from fuel consumption and engine weight considerations, the more the optimum engine cycle approaches a turbojet. At low subsonic speeds, propellers are most optimum, at high subsonic speeds, high bypass ratio turbofans, and at SST speeds, the turbojet. But more than these considerations dictate the optimum choice. We have noise constraints, multisegmented missions, etc. Tradeoffs are made to determine the best choice.

Mission and engine simulation can be used to identify what new technologies such as turbine blade materials or cooling techniques offer the most attractive payoffs. Limited dollars must be budgeted in the most beneficial manner.

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Likewise, dollars can be saved by completely modeling an engine in the computer to study any potential problem areas prior to cutting of hardware.

Computer simulations may also be used to save the airlines money by helping to identify the source(s) of decreased engine performance.

Design Points - Initially, after the invention of the turbine engine, all that could be calculated readily was the design point performance. The thermodynamics of jet engines at the design point are straightforward. The temperature ratio across a component is some function of the pressure ratio. For an isentropic process we have:

$$\frac{T_2}{T_1} = \frac{P_2}{P_1}^{(\gamma-1)/\gamma}$$

If we define a polytropic efficiency to represent inefficiency in a process:

$$\frac{T_2}{T_1} = \frac{P_2}{P_1}^{(\gamma-1)/\gamma\eta_p}$$

We usually, however, talk about adiabatic efficiency:

$$\frac{T_2}{T_{1_{ideal}}} = \frac{P_2}{P_1}^{(\gamma-1)/\gamma}$$

$$\frac{(T_2 - T_1)_{actual}}{(T_2 - T_1)_{ideal}} = \frac{1}{\eta_a}$$

$T_2 > T_1$ compressors; $T_2 < T_1$ turbines

Starting at the inlet, total conditions at the compressor face are determined from the equations relating total to static conditions and the inlet recovery together with the atmospheric properties tables and the flight conditions. The temperature rise across the compressor is calculated from the pressure rise and adiabatic efficiency. The work of compression is then calculated. The temperature rise and pressure drop across the combustor are used to calculate turbine entrance conditions and fuel flow.

The work of compression and the efficiency are used to calculate the pressure drop and temperature drop across the turbine and the conditions entering the nozzle used to calculate the thrust.

Matching - You are probably familiar with what are known as State Properties. Four of these are Temperature, Pressure, Enthalpy, and Entropy. Any two of these are sufficient to determine the other two. This, plus the use of the laws of Conservation of Mass and Energy, along with any physical relationships such as shafts connecting components are sufficient to determine match points in an engine. Let us see how we put them all together to "match" an engine.

The performance of the engine components is usually represented by component maps. Some typi-

cal maps are shown in figure 2 for a compressor, a turbine, an inlet, and a nozzle.

For a compressor, we usually have Pressure Ratio plotted versus Corrected Airflow along lines of constant Corrected Speed. We have a Surge Line, and islands of constant Adiabatic Efficiency. Theta represents the temperature into the compressor divided by the sea level standard static temperature of 518.67° R. Delta is the pressure in, divided by the SLS value of 14.696 psi. For turbines, we usually see Equivalent Flow as a function of Pressure Ratio for lines of constant Corrected Speed. We can see that for turbines we reach a choked condition at which no additional flow can be forced through the turbine by pressure ratio alone. This is a result of the existence of what are essentially nozzles in the turbine itself.

For inlets, a typical map might be the inlet total Pressure Ratio as a function of Mach number, and, for a nozzle, the Flow Coefficient as a function of Pressure Ratio. Any or all of these maps might have a third dimension, i.e., be "stacked" to account for variable component geometry such as area.

Having such maps, it is now possible to hand match the engines. As an example, by appropriate transformations, the compressor map and turbine map could be transformed into maps which could be over-layed. Points could then be determined where the physical speeds were the same, the airflows (allowing for fuel burned) are the same and where the change in enthalpy across both was the same. This procedure is thoroughly discussed in reference 1. It uses simplifying assumptions, is very difficult to use on complex engines, and is very slow and laborious taking man-days to determine many operating points.

As previously mentioned, the need for computerized matching became apparent. The approximate history of the development of today's matching codes is shown in figure 3.

Some, but not all of the steady state matching codes are shown in figure 4. These can be broken down into three sub-types: specific decks for a particular engine such as the customer decks; generalized decks with pre-configured engine possibilities such as SMOTE, SMITE, SUPERSMITE, TECSON, and GENENG; and modularized decks that either internally calculate a flowpath from component to component or punch out a main routine to do the same. These latter decks can thus be used to simulate almost any engine. Example of these are SOAPP, SYNTHA, ALLISON'S, GSA, NEPCOMP, and NNEP.

Let us now briefly go through a computerized match of a One-Spool Turbojet where only the compressor and turbine have component maps.

Figure 5 shows a schematic of the turbojet and its maps. One problem with compressor maps is trying to identify where you are on the map. If the speed lines are almost horizontal, pressure ratio as a search parameter is almost meaningless as is airflow if the speed lines are vertical. Past experience has shown that introduction of a dummy field parameter simplifies the problem. This dummy parameter is "R" or "ray" lines which are shown on the map. We transform the compressor map into three separate maps as shown. "R" and speed then become

the independent variables. For the turbine, speed and pressure ratio will uniquely determine airflow and efficiency.

Let us now employ conservation of mass and energy and mechanical speed to match the engine. In figure 6 the turbojet engine has been "broken up" in a particular manner as follows:

At any given flight condition, the Mach number, altitude, and airflow into the inlet will determine the corrected flow at the compressor face. But, based upon the R and speed that the compressor is operating at, it may not be capable of passing that exact corrected airflow amount. This has been indicated by showing a flow imbalance. Going now to the burner, knowing the burner exit temperature, pressure, airflow and fuel, we can calculate the corrected flow at the turbine face. Likewise, the turbine may not pass this flow. Now looking at the nozzle, the throat area may not pass all the flow being delivered by the turbine.

We know from conservation of energy that (assuming no external loads) the work of the turbine has to equal the work of compression, and, that there is a shaft connecting the compressor and turbine (which may or may not be geared). We see in figure 7 a summary of what has just been discussed indicating that any imbalances are treated as "errors".

Let us digress for a moment. Shown in figure 8 is a procedure known as a Newton-Raphson Iteration for One-Dimension. If we have some dependent variable "E" and we want its value as a function of an independent variable "V" equal to zero, we can take a first-order Taylor Expansion (assumes linearity) about a basepoint Eo. We then rearrange solving for a new guess value Vo and the value of the function Eo and its derivative with respect to V. As an example, let us solve iteratively for the square root of 9 and let our first guess be 10!

$$f(V) = E = V^2 - 9$$

$$dE/dV = 2V$$

$$V_1 = Vo - (Vo^2 - 9)/2Vo$$

$$f(v) = 91; \quad dE/dV = 20; \quad V_1 = 10 - (91/20) = 5.45 \rightarrow Vo$$

Repeating, we will get successively 3.55, 3.043, and 3.000 a fairly rapid convergence.

In figure 9, we see a multi-dimension Newton-Raphson iterator. There are N variables and N "E" values or errors. Each E is assumed to be some base E plus the sum of the partials of this E with respect to all of the variables times the change in the variable. This is written in Matrix Notation as the error vector E is some base error Eo + a matrix of partial derivatives times the vector of variable changes. Since we want the E's to go to zero, set the LHS to zero, subtract the Eo vector from both sides, and multiply both sides by the inverse of the matrix of partials. This gives us the step changes to take in each of the variables. We usually set a maximum limit on the step changes, going in the indicated directions but perhaps cutting the size of the step.

We will now apply MDNR to our turbojet case.

E_1 through E_3 are the corrected flow errors and E_4 is the work error. We have four errors and therefore need four independent variables. Some possible choices are shown in figure 10.

In figure 11, we see what happens if we choose the first four. These four choices enable us to determine all four errors and therefore the problem is solvable. We make a first guess on all the variables to form Eo. Then we perturb each variable one at a time to generate the partial derivative matrix. After calculating the inverse, we generate steps in each of the variables, run a new Eo and repeat. Clever techniques such as those used to SUPERSMITE, NEPCOMP, and NNEP can be employed to avoid calculating some partials that we know a priori to be zero and to continue to use the same partial derivative matrix as long as the sum of the square of the errors is reducing.

Different Computer Types - Let us now touch briefly on the three types of computers used; Analog, Digital, and Hybrid. Analog computers are able to integrate essentially instantaneously through the use of amplifiers. Digital integration is much slower. There is a forward difference method which yields an explicit solution for X(t) and a backward difference method which uses the value calculated in an iterative solution. The second method is much slower than the first for each integration step but makes up for the difference by being able to take much larger time steps hence much fewer integrations. Analog computers are very poor when it comes to evaluating multivariate functions and for doing iterative calculations. Therefore, the best features of both computers have been utilized in the Hybrid. Shown in figure 12 are which calculations are performed by which part of the Hybrid. This type of computer does the best job of real time simulation, i.e., changes occur at the same rate as in the real engine. This finds its biggest benefit in Dynamic Simulations.

Transient (Dynamic) Simulations

As opposed to steady state performance, in dynamic simulations we look at the engine as it goes from one steady state point to another, that is what happens in between. The question that might be addressed is "Can we get there from here, and if so, how long does it take to get there?". The two end points might be quite well behaved but surge, overspeed, failure to settle out, any number of undesirable things may occur. When the 727's first started in service there was a series of accidents where the airplane crashed short of the runway. This was a direct result of the spool-up time to go from part throttle to full throttle on the engines. By lowering the allowable flap setting on landing, more power was used and the spool-up time was shortened enough to allow recovery when landing short. Thus we can see that the dynamic performance of an engine may be even more critical than the steady state performance.

In figure 13 we compare the equations for steady state versus dynamic. Conservation of mass must be modified to include the accumulation of mass in finite volume. This is represented by the dP/dt term where mass varies as the rate of change of the pressure.

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For conservation of energy, the work of the turbine is no longer just equal to the work of compression plus external loads. An excess of turbine energy (or an insufficiency) will result in a speeding up (slowing down) of the spool. This is represented by dN/dt. Notice that the moment of inertia enters into this term.

Also note that the airflow out of the turbine does not equal the airflow in. We, therefore, also have to account for the buildup of internal energy due to mass storage and also for the rate of change of specific internal energy du/dt.

With these two modifications, steady state codes become dynamic codes. Some of these are listed on figure 14. DYNGEN is a dynamic version of GENENG and HYDES is a hybrid.

The matching codes that have been mentioned and the numerical methods used are described in references 1 through 17 and can be categorized as follows:

Hand matching	Ref. 1
Generalized pre-configured codes	Refs. 2, 3, 4, 5, 6, 7, 8, and 9
Modularized any configuration	Refs. 10, 11, 12, 13, and 17
Numerical methods	Refs. 14, 15, and 16

We have talked about steady state and dynamic codes. The modularized codes can be used to study very complicated engines such as the variable cycle en-engines. At least one of these codes NNEP is capable of eliminating components, bringing in new ones, and running multiple flowpaths with stacked component maps. An example of a mixed/ unmixed turbine engine is shown in figure 15. Both "modes" are identical up to the turbine exit. The separate flow mode eliminates the mixer, closes down on the main nozzle area, and introduces another nozzle on the bypass stream.

A typical throttle curve for this engine is shown in figure 16 where the dashed line is for mixed flow, and the solid for separate. Looking first at the fixed geometry curves (FG) we see that when operating at part power, SFC increases for installed engines. By installed we mean we have accounted for inlet spillage drag and boattail drag. Variable geometry (VG), when used correctly, can reduce both the SFC and change the shape of the curves. Which brings us to the subject of Optimization.

Optimization

There are two basic approaches to optimization in engine simulation: inside the loop; and outside the loop. By outside the loop we mean that the engine is first matched; then the free variables are changed and the engine rematched. This procedure is continued until the optimum is reached.

By inside the loop, we mean that at the same time the engine is being matched, the free variables are being changed. When the match point is finally achieved, the free variables are at their optimum values. This is theoeretically a much faster, better way of doing it.

At the present time, at least two operating codes have outside the loop optimization: Allison's and NNEP. A discussion of methods tried for both types of optimization is found in reference 12. The General Electric Co. has indicated that they have achieved the first working inside the loop method (ref. 18).

Engine Weight Prediction

The cycle analysts are in the position of being able to do a pretty thorough job of predicting optimum engine performance though admittedly at great costs in computer time. Performance, however, is not the only thing affecting engine selection. The airplane has to carry that engine over the entire mission so its weight is also very important.

Historically, preliminary mission analysis studies used engine weight prediction based upon historical trends such as reference 19. Many of the new engine concepts of today such as variable cycle engines, cruise missile engines, etc., are not similar enough to the data base of existing engines to use those methods. Of course, the engine companies have complex design codes to calculate the engine weight.

In order to calculate more realistic values of engine weight and dimensions for new engines, NASA Lewis awarded a contract to the Boeing Military Airplane Development Division of the Boeing Company to develop an engine weight estimation code.

The first version of this code WATE-1 (ref. 20) was completed in 1977. It used a preliminary design approach where stress level, maximum temperature, material, geometry, stage loading, hub-tip ratio, and shaft mechanical overspeed are used to determine individual component weights. The total engine weight was then calculated as the sum of the individual components. The contract required that the code predict both individual component and total engine weight within ±10 percent accuracy.

A relatively high level of detail was found necessary in order to obtain the required accuracy. Component weight data for 29 different engines were used as a data base. This data base is shown in table I. The list of engines includes military and commercial, turbofans and turbojets, augmented and dry, hardware engines and proposed engines, and supersonic and subsonic engines.

WATE-1 was constructed to operate as an adjunct to NNEP. After running a cycle point on NNEP the thermodynamic properties were fed to the WATE-1 set of subroutines along with inputs representing the design features of the components. The engine weight, length, and dimensions were then calculated. At the same time, parts counts are generated for the engine such as number of blades, size of discs, etc.

In 1978, NASA Lewis awarded a follow-on contract to the Boeing Company to extend the capabilities of WATE-1. This new version, WATE-2 (ref. 21) was completed in 1979 and has added many desirable features. Weight determination is done for each component at its critical operating point as follows: NNEP is now used to "fly" the engine through-

out the flight envelope of the aircraft and the maximum values of the flow, temperature, pressure, and engine speed stored for use in sizing the components. Based upon these critical conditions, the weight is determined. The capability to calculate the weight of radial flow components and of small engines was added in conjunction with a subcontract to the Garrett Division of AiResearch Manufacturing Company of Arizona. The engine center of gravity and moments of inertia are also now calculated.

The accuracy of the code is shown in figure 17. As can be seen, all of the engines fall within the ± 10 percent band and, in most cases, approach ± 5 percent or better especially in terms of engine weight.

WATE has built-in default values for most of the inputs. If the user does not enter values, these default values are automatically used. Many of these were used in the calculation of these weights. If more information was available to us, especially in terms of geometry inputs of the rotating components, these already small errors could probable be reduced even further.

To illustrate how preliminary design techniques and the data base have been combined to give more realistic engine weights, consider the procedure used to calculate the weight of a compressor. Figure 18 shows how the compressor is separated into its component parts. The weight of each of these parts is calculated separately. First, the tip velocity of the first stage of the rotor is calculated from the pressure ratio of the first stage based upon a curve generated from the data base as shown in figure 19.

The first stage flow area is calculated from an input Mach number and the corrected airflow from the cycle data. The inner and outer radius of the flowpath are determined from the area and hub-to-tip ratio and the compressor rpm determined from the tip velocity and outer radius. Dimensions of succeeding stages are determined based on the design mode selected (constant mean, tip, or hub design).

Volume of metal in the blades is determined from blade height, aspect ratio, and a constant which accounts for fir-tree mount volume, taper ratio, and thickness to chord variation in typical blades. From the data base, values of the constant were determined based on compressor type. Weights of the blades are determined from the blade density and volume, and the number of blades from the solidity and chord length. The blade pull stress is then calculated from tip speed, blade height and density, and the thickness from figure 20, which again is a correlation of data base information. From this the disc weight is calculated.

The combination of WATE and NNEP is a very powerful analytical tool. As an example, a recent study considered the question of optimum cycle parameters for a duct burning turbofan for a supersonic cruise airplane (ref. 22). Some of the results of this study are duplicated here. The fuel mass and bare engine mass for 88 950 newton (20 000 lb) thrust engines flying 6440 kilometers (4000 mi) operating at Mach 2.4, $\epsilon 6$ 460 meters (54 000 ft) initial attitude are shown in figure 21. These masses are shown as functions of Bypass Ratio and Overall Pressure Ratio (OPR) with and without duct burning. The cycle analyst looking only at the fuel mass

in figure 21 would conclude that the optimum engine would operate dry and have an OPR of about 16 at a Bypass Ratio of 1.8 or more. However, when the mission analyst adds the fuel and engine masses as shown in figure 22, the optimum engine operates with the ductburner on, an OPR of 12 and a BPR of 0.8.

Installation Effects

The previous example (fig. 16) of varying engine flow by the use of variable geometry to reduce installation effects showed the importance of inlet and nozzle component performance, external as well as internal. That figure was generated with a simplified model for inlet and boattail drag that is built into the NNEP program.

The Air Force contracted with Boeing in 1977 to develop a method for the "Rapid Evaluation of Propulsion System Effects" (ref. 22). The purpose was to develop computerized preliminary analysis procedures for inlet and nozzle internal and external losses for subsonic and supersonic aircraft. Two computer programs were developed: (1) a propulsion installation effects program that calculates installed performance using input maps of inlet and nozzle/afterbody characteristics for specific configurations (PIPSI), and (2) a "derivative" program that allows the user to generate new sets of input maps by perturbations to the geometrics of the basic input maps (DERIVP).

The types of performance maps to be generated are shown in figures 23 and 24 for inlets and nozzles, respectively. The necessary maps are obtained from either a data base or theoretical calculations. The data base contains performance data (usually experimental) for a spectrum of inlet (axisymmetric, 2D, pitot, mixed compression) or nozzle (axisymmetric, 2D, twin, etc.) types. The inlets in the data base are depicted in figure 25 and listed in table II. The derivative procedure (ref. 23) can be used to adjust the data base for changes in design Mach number, sideplate shape, subsonic diffuser loss, cowl lip bluntness, takeoff door area, external cowl initial angle, bleed system design, and bypass system exit design.

Items not included in the data are determined analytically (ref. 24). Nozzle/afterbody data are treated in a similar manner. The data base, being primarily experimental, offers increased confidence in areas that are difficult to treat theoretically such as viscous effects.

Boeing, under contract to NASA Lewis, has expanded the capability of these codes to make them interactive with NNEP, calculate inlet weight, and generate any missing data analytically (ref. 25).

Engine Cost

The question of cost is entering more and more into the selection process for optimum engines. Engine acquisition cost while significant is only a fraction of the total life cycle cost for an engine. Maintenance, spares, operating costs, etc., frequently are much larger pieces of the pie than just the manufacturing costs along. This is shown on figure 26.

The government and industry are investing much

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effort in developing computational methods for calculating the individual pieces of the life cycle cost pie. NASA, the Navy, and Boeing have joined in an effort to develop a computer code to calculate the acquisition cost of an engine (ref. 26).

A life cycle costing model was developed over a period of 2 years by the Aircraft Engine Life Cycle Cost Methods Improvement Working Group which met at Wright-Patterson Air Force Base (ref. 27) and was followed by contracted efforts between the Air Force and industry to expand upon, improve, and demonstrate the use of the model (refs. 28 to 31).

American Airlines developed a computer program for predicting maintenance costs for the total propulsion system following standard airline practice for large subsonic commercial aircraft (ref. 32). This model, an improvement over the old ATA method, calculates direct and indirect operating costs accounting for the number of spare engines, parts, reversers, etc. Two forms of prediction methodology were developed. The first is very detailed and requires a large number of inputs detailing the engine including pressures, temperatures, diameters, tip speeds, stages, and prices. The second form is suggested as a simple replacement for the ATA engine maintenance cost equations and is a function of flight time, sea level combustor exit temperature, engine price, and labor rate.

The area of life cycle costing is justly receiving more and more attention. Significant breakthroughs in modeling can be expected.

Concluding Remarks

As we have seen, fairly sophisticated analytical tools are available for calculation of engine performance, weight, dimensions, installation effects, and we are moving forward in developing the capability to determine life cycle costs. What are the top priority items for the future?

Analytical Models - We believe that analytic models of component performance is one of the largest payoff improvements. If quick, accurate mathematical models can be developed for compressors, turbines, etc., we may be able to eliminate table lookups from within the codes. Inaccuracies occur as a result of transcribing map input to digital data. Large amounts of computer time are spent interpolating within tabular data. Numerical derivatives would be calculatable, "noise" would be reduced. There is, of course, a restriction as to the allowable core requirements for these models.

Optimization - It was already mentioned that optimization is important. Anything that can be done to quicken the process would be of enormous benefit especially with the advent of engines with variable geometry. Here is an area we feel that might best be addressed by mathematicians rather than by engineers, but the forming of teams to attack the problem must be sold to management as being beneficial.

Guesses - We previously alluded to guessing initial values of the variables to form the Eo vector and did not mention how. Most of your preconfigured codes have built into them Guess routines that, based on past use, predict as a function of flight conditions, CET, etc., good first

guesses to the variables. Your "model anything" codes, however, do not have good routines. Here again is an area that mathematicians may be able to help out in.

Perhaps the solution is to keep track of the free variables as more and more cases are run for a particular engine. Regression analysis may then be useful in predicting how the variables are changing, constantly updating the "guess" routine being internally generated, which, of course, then leads us into the fourth item - Internal Smartness.

Internal Smartness - The using of what has been happening to predict what is going to happen. Methods may be developed to generate good first guesses, to guess how the optimization variables should be first set, to use techniques such as holding the matrix of partials constant. Anything to speed up the convergence process, to make as few calculations as possible. This could greatly shorten computation times.

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TABLE I. - DATA BASE ENGINES

Engine	Manufacturer	Manufacturing[1] status	Type of cycle[2]	Augmentation[3]	Primary use[4]
GE4/J4C	GE	P	TJ	AB	C
GE4/J5	GE	X	TJ	AB	C
GE9/F2B	GE	X	TF	AB	M
JT11F	P&WA	P	TJ	AB	M
TJ70	WE	S	TJ	AB	C
GE4/J6G	GE	S	TJ	AB	C
GE4/J5H2	GE	S	TJ	--	C
JT8D-15	P&WA	P	TF	--	C
JT9D	P&WA	P	TF	--	C
TF34	GE	P	TF	--	M
VSCE-502	P&WA	S	TF	DH	C
VCE-201A	P&WA	S	VCE	--	C
VCE-201B	P&WA	S	VCE	--	C
VCE302A	P&WA	S	VCE	--	C
VSCE-502B	P&WA	S	TF	DH	C
VCE-112B	P&WA	S	VCE	--	C
VSCE-501	P&WA	S	TF	DH	C
VCE-110B	P&WA	S	VCE	--	C
A/B TF-2	P&WA	S	TF	AB	C
D/H TF-2	P&WA	S	TF	DH	C
D/H TF-12	P&WA	S	TF	DH	C
JT10D	P&WA	X	TF	--	C
CFM56	GE/SNECMA	P	TF	--	C
CF6-50	GE	P	TF	--	C
CF6	GE	P	TF	--	C
JT8D	P&WA	P	TF	--	C
CJ805-23	GE	P	TF	--	C
YJ93	GE	P	TJ	AB	M
JT3D	P&WA	P	TF	--	C

[1]Manufacturing status: P = production, S = study proposal,
   and X = experimental.
[2]Type of cycle: TJ = turbojet, TF = turbofan, and VCE =
   variable cycle engine.
[3]Augmentation: AB = afterburner and DH = ductheater.
[4]Primary use: C = commercial and M = military.

TABLE II. - SOURCES OF DATA FOR MATRIX OF INLET MAPS

Inlet number	Inlet configurations and sources of data used to develop the inlet maps
1	A-7 type inlet; developed from published A-7 data and engineering analysis
2	F-8 type inlet; developed from published F-8 inlet data and analysis
3	Subsonic inlet type; based on data and methods from Boeing subsonic inlet (i.e., 707, 727, etc.)
4	Subsonic inlet type; based on data and methods used to develop Boeing 747-type inlets
5	Normal shock inlet; based on data from Rockwell tests of F-100 airplane inlet
6	Normal shock-type inlet; based on data from Rockwell F-100 inlet, Boeing LWF inlet tests, and GD LWF inlet data
7	Fixed-geometry, two-shock inlet; based on data from Boeing LWF inlet tests
8	Four-shock, variable-ramp inlet; theoretical design based on analysis, optimized for $Mo = 2.0$
9	Four-shock, variable-ramp inlet; based on data from NR inlet tests of IPS model
10	Fixed-geometry, single-cone inlet; based on analytical design for a $Mo = 1.6$ VTOL
11	Three-shock, half-round inlet with variable-diameter centerbody; analytical design for a supersonic Navy VTOL configuration
12	Three-shock, half-round inlet with variable second-cone angle; GD tailor-mate tests
13	Mixed-compression; analytical design documented in AFFDL-TR-72-147-vol IV
14	Mixed-compression; based on XB-70-type configuration and data
15	Mixed-compression; based on NASA AMES configuration and tests of a Mach 3.5, 2-D inlet
16	Mixed-compression axisymmetric; based on Boeing analytical study of an AST inlet for NASA AMES
17	Mixed-compression axisymmetric; based on data from NASA AMES tests of $M_\infty = 3.0$ inlet
18	Mixed-compression axisymmetric; based on results of Boeing analytical studies for a NASA AMES Mach 3.5 inlet

BEST ENGINE
TO EXAMINE, FROM AN ALMOST INFINITE SET, CYCLES WHICH
APPEAR ATTRACTIVE FOR A PARTICULAR APPLICATION.

TECHNOLOGY
TO GUIDE COMPONENT DEVELOPMENT PEOPLE IN WHICH AREAS OF
THE ENGINE APPEAR TO BE MOST CRITICAL OR SHOW PROMISE OF
HIGHEST PAYOFF FROM ADVANCING STATE OF THE ART

COST SAVING
TO SIMULATE A COMPLEX NONLINEAR SYSTEM PRIOR TO IT BEING
BUILT TO INSURE THAT IT WILL WORK PROPERLY.

TROUBLESHOOTING
TO IDENTIFY SOURCE(S) OF EXISTING ENGINE PERFORMANCE
DEGRADATION.

Figure 1. – Why simulate turbine engines?
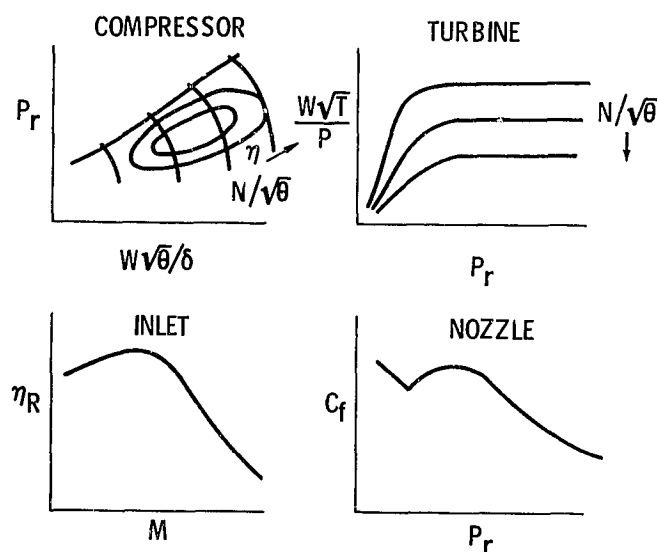
Figure 2. – Typical component maps.

HAND MATCHING
● 1 SPOOL GAS GENERATOR

● TURBOPROP
● 2 SPOOL GAS GENERATORS
● DYNAMICS 1 SPOOL GAS GENERATOR

● 2 SPOOL TURBOFAN (BYPASS ENGINE)

COMPUTERIZED MATCHING
● TURBOJETS
● 2 SPOOL TURBOFANS
● 2/3 SPOOLS TURBOJETS, TURBOFANS, AFTFANS, ETC.

● ARBITRARY ENGINES

1940
1950
1960
1970
1980
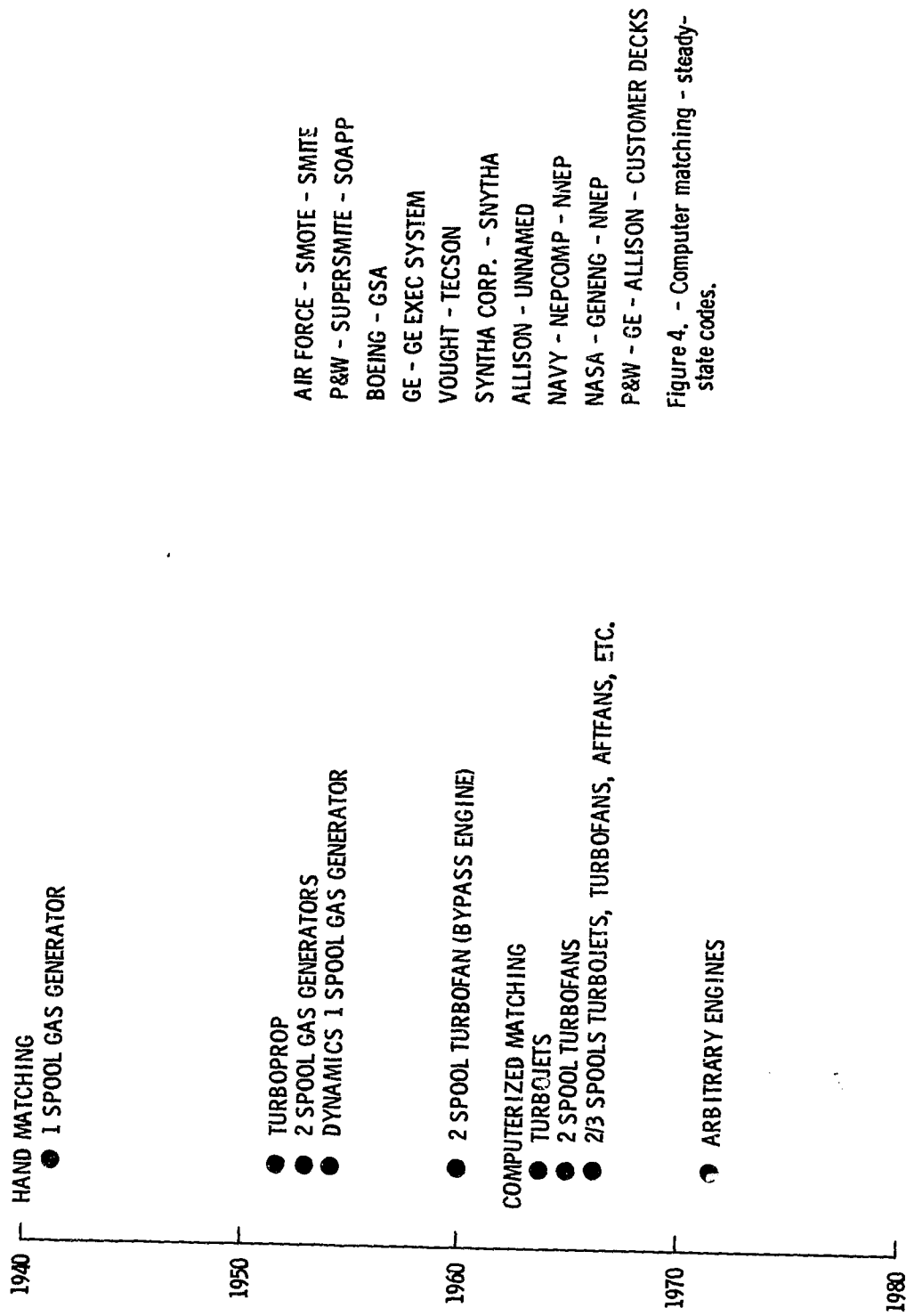
Figure 3. – Approximate history of methods of matching turbine engines.

AIR FORCE – SMOTE – SMITE
P&W – SUPERSMITE – SOAPP
BOEING – GSA
GE – GE EXEC SYSTEM
VOUGHT – TECSON
SYNTHA CORP. – SNYTHA
ALLISON – UNNAMED
NAVY – NEPCOMP – NNEP
NASA – GENENG – NNEP
P&W – GE – ALLISON – CUSTOMER DECKS

Figure 4. – Computer matching – steady-state codes.

ONE SPOOL TURBOJET

INLET — COMPRESSOR — BURNER — TURBINE — NOZZLE

R_1 R_2 R_3

P_r vs $\frac{W\sqrt{\theta}}{\delta}$, $N/\sqrt{\theta}$

P_r vs R , $N/\sqrt{\theta}$

$\frac{W\sqrt{\theta}}{\delta}$ vs R , $N/\sqrt{\theta}$

η vs R

$\frac{W\sqrt{T}}{P}$ vs P_r , $N/\sqrt{\theta}$

η vs P_r
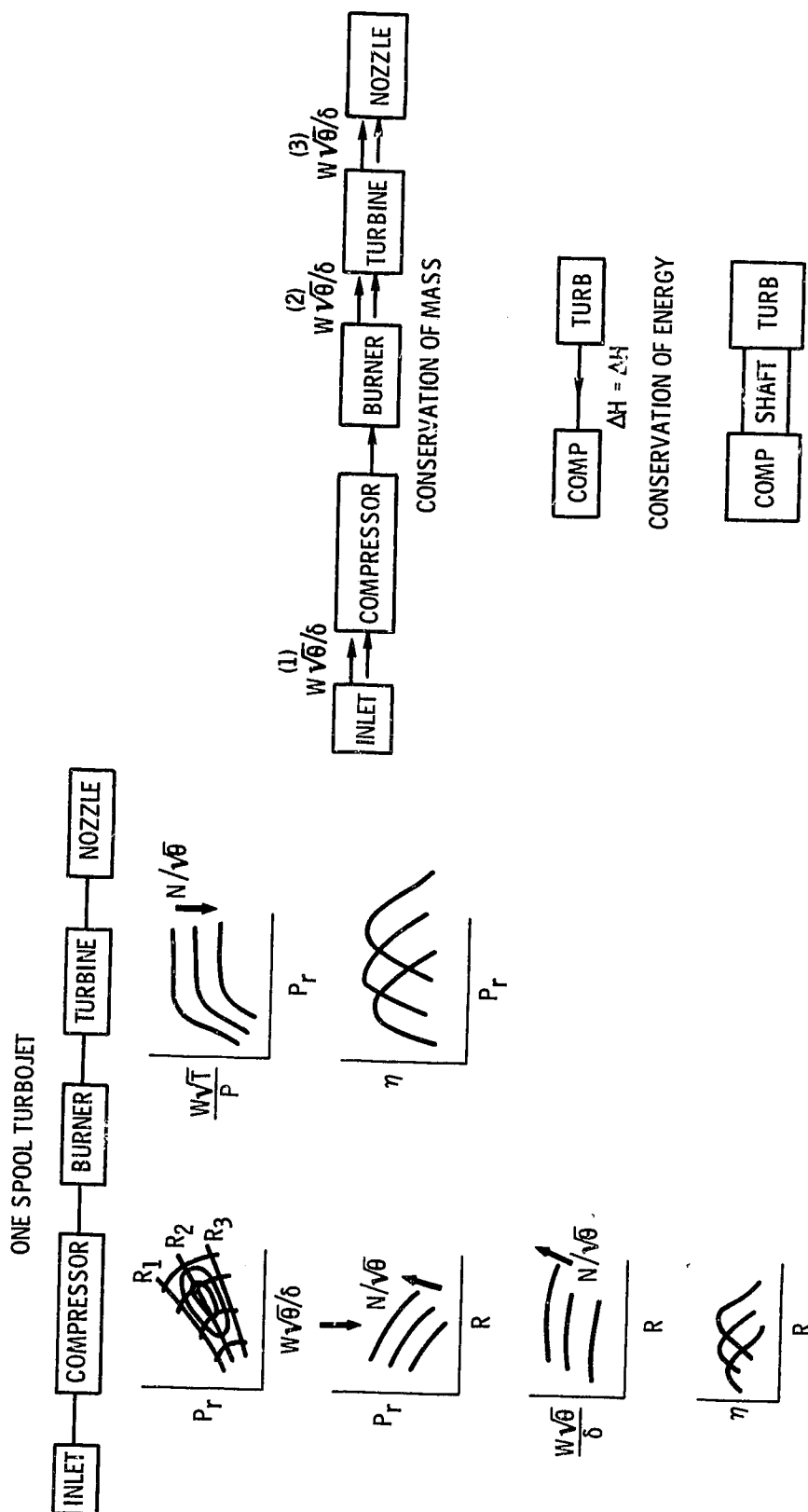
Figure 5. – Matching in steady state.

INLET — COMPRESSOR — BURNER — TURBINE — NOZZLE

(1) $W\sqrt{\theta}/\delta$

(2) $W\sqrt{\theta}/\delta$

(3) $W\sqrt{\theta}/\delta$

CONSERVATION OF MASS

COMP → TURB

$\Delta H = \Delta H$

CONSERVATION OF ENERGY

COMP — SHAFT — TURB

MECHANICAL SPEED

Figure 6. – One spool turbojet as set up for matching.

MECHANICAL SPEED
    COMPRESSOR SPEED SET BY SHAFT SPEED
    TURBINE SPEED SET BY SHAFT SPEED

CONSERVATION OF MASS
    ERROR (1) = CORRECTED FLOW (CF) AT INLET EXIT - CF AT
      COMPRESSOR
    ERROR (2) = CF AT BURNER EXIT - CF AT TURBINE INLET
    ERROR (3) = CF AT TURBINE EXIT - CF AT NOZZLE ENTRANCE

CONSERVATION OF ENERGY
    ERROR (4) = WORK OF TURBINE AND WORK OF COMPRESSION

Figure 7. - Balancing "ERRORs" of one-spool turbojet.

ASSUME TAYLOR EXPANSION $E = f(V) = 0$

$$E = E_0 + \left(\frac{\partial E}{\partial V}\right) \delta V = E_0 + \left(\frac{\partial E}{\partial V}\right)(V_1 - V_0)$$

$$V_1 - V_0 = \frac{E - E_0}{(\partial E/\partial V)}; \text{ but } E = 0 \text{ and } E_0 = f(V_0)$$

THEREFORE

$$V_1 = V_0 - \frac{f(V_0)}{f'(V_0)}$$

Figure 8. - Newton-Raphson iteration - 1D.

$$E_i = E_{0_i} + \sum_{j=1}^{n} \left(\frac{\partial E_i}{\partial V_j}\right) \delta V_j$$

MATRIX NOTATION

$$
\begin{pmatrix} E_1 \\ E_2 \\ \cdot \\ \cdot \\ \cdot \\ E_n \end{pmatrix}
=
\begin{pmatrix} E_1 \\ E_2 \\ \cdot \\ \cdot \\ \cdot \\ E_n \end{pmatrix}_0
+
\begin{pmatrix} \frac{\partial E_1}{\partial V_1} \cdots \frac{\partial E_1}{\partial V_n} \\ \cdot \quad\quad \cdot \\ \cdot \quad\quad \cdot \\ \cdot \quad\quad \cdot \\ \frac{\partial E_n}{\partial V_1} \quad \frac{\partial E_n}{\partial V_n} \end{pmatrix}
\begin{pmatrix} \delta V_1 \\ \cdot \\ \cdot \\ \cdot \\ \cdot \\ \delta V_n \end{pmatrix}
$$

WANT $E_1 \rightarrow E_N = 0$

$$
\begin{pmatrix} \frac{\partial E_1}{\partial V_1} \cdots \frac{\partial E_1}{\partial V_n} \\ \cdot \quad\quad \cdot \\ \cdot \quad\quad \cdot \\ \cdot \quad\quad \cdot \\ \frac{\partial E_n}{\partial V_1} \quad \frac{\partial E_n}{\partial V_n} \end{pmatrix}^{-1}
\begin{pmatrix} -E_1 \\ -E_2 \\ \cdot \\ \cdot \\ \cdot \\ -E_n \end{pmatrix}_0
=
\begin{pmatrix} \delta V_1 \\ \delta V_2 \\ \cdot \\ \cdot \\ \cdot \\ \delta V_n \end{pmatrix}
$$

Figure 9. – Multidimension Newton Raphson.

E_1, E_2, AND E_3 ARE CORRECTED FLOW ERRORS
E_4 IS WORK ERROR
HAVE 4 ERRORS NEED 4 VARIABLES
SOME POSSIBLE CHOICES
    INLET AIRFLOW
    "R" OF COMPRESSOR
    SHAFT SPEED
    TURBINE PRESSURE RATIO
    VARIABLE GEOMETRY SETTINGS (NOZZLE AREA, PITCH, . . .)
    COMBUSTOR OUTLET TEMPERATURE

Figure 10. – MDNR applied to TJ.

(1) INLET AIRFLOW + η_R + ATMOS → CF AT INLET EXIT

(2) "R" + (3) SHAFT SPEED → CF AT COMPRESSOR INLET $\Bigg\} E_1$

    CF AT COMPRESSOR EXIT + FUEL AND CET AND ΔP → CF ────→ E_2
      BURNER EXIT

(4) PR + (3) SHAFT SPEED → CF TURBINE INLET

    CF AT TURBINE EXIT → CORRECTED FLOW AT NOZZLE INLET $\Bigg\} E_3$

    NOZZLE AREA + P + T → CORRECTED FLOW AT THROAT

    CYCLE CONDITIONS → WORK VALUES E_4
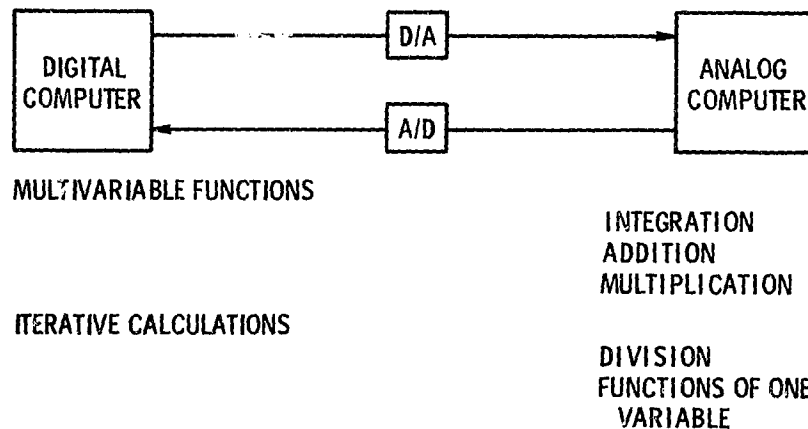
THEREFORE SOLVABLE

Figure 11. - If first 4 chosen.

DIGITAL COMPUTER → D/A → ANALOG COMPUTER

ANALOG COMPUTER → A/D → DIGITAL COMPUTER

MULTIVARIABLE FUNCTIONS

          INTEGRATION
          ADDITION
          MULTIPLICATION

ITERATIVE CALCULATIONS

          DIVISION
          FUNCTIONS OF ONE
            VARIABLE

Figure 12. - Hybrid computer simulation.

CONSERVATION OF MASS

$$\dot{W}_{out} = \dot{W}_{in} \quad \text{VERSUS} \quad \dot{W}_{out} = \dot{W}_{in} - \frac{\tilde{V}}{\gamma RT}\frac{dP}{dt}$$

CONSERVATION OF ENERGY

$$\dot{W}_t\,\Delta h_t = \dot{W}_c\,\Delta h_c + (HP)_{ext} \quad \text{VERSUS} \quad \dot{W}_t\,\Delta h_t = \dot{W}_c\,\Delta h_c + \left(\frac{2\pi}{60}\right)^2 IN\frac{dN}{dt} + (HP)_{ext}$$

WHERE FOR DYNAMIC

$$\dot{W}_{out}\,h_{out} = \dot{W}_{in}\,h_{in} - (\dot{W}_{in} - \dot{W}_{out})u - \frac{P\tilde{V}}{RT}\frac{du}{dt}$$

Figure 13. – Steady state versus dynamic equations.

AIR FORCE – SMOTE – SMITE
P&W – SOAPP
BOEING – GSA
GE – GE EXEC SYSTEM
ALLISON – UNNAMED
NASA – DYNGEN, HYDES, UNNAMED ANALOG
P&W, GE – ALLISON – CUSTOMER DECKS

Figure 14. – Dynamic codes.

MIXED FLOW

SEPARATE FLOW
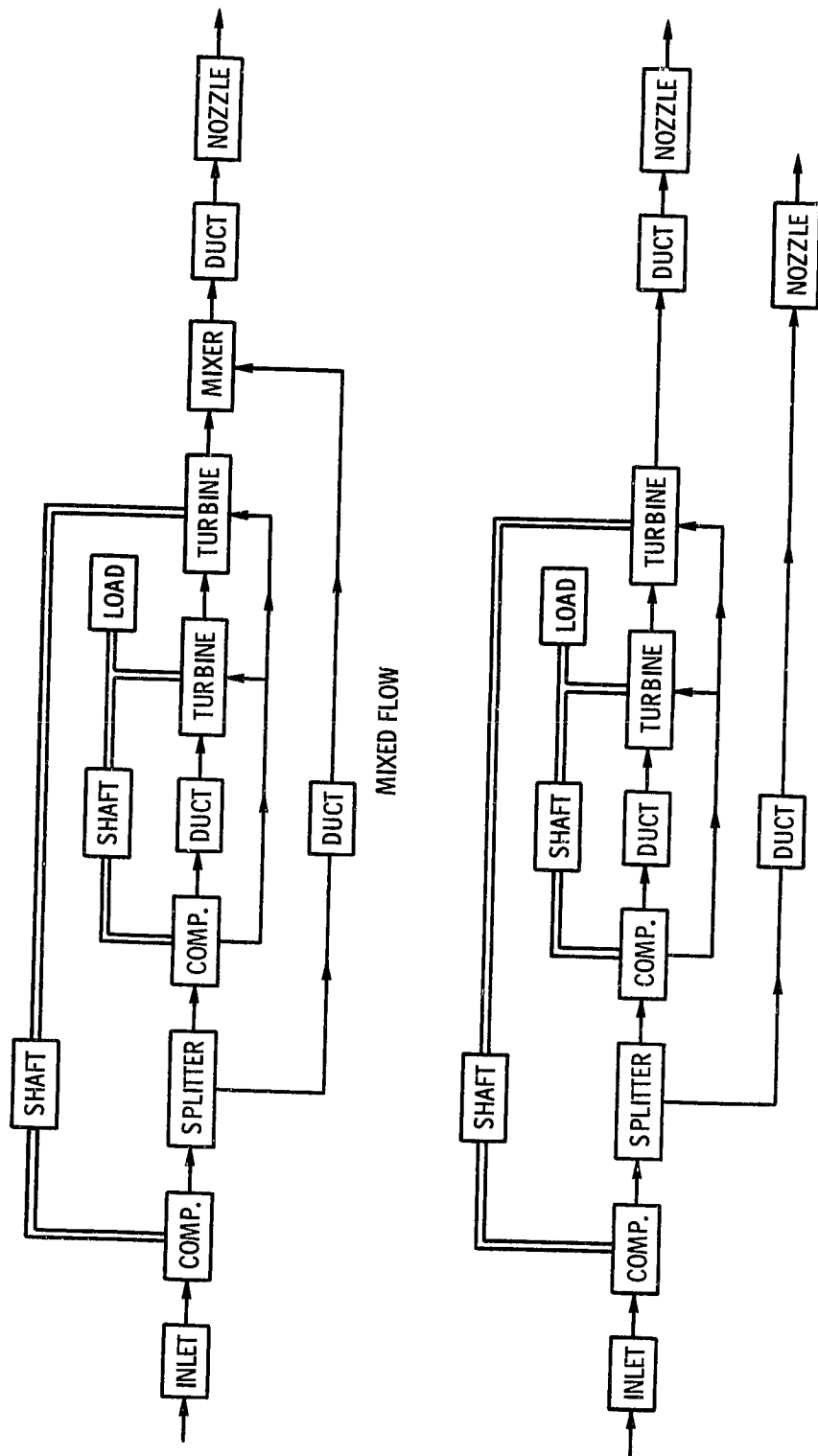
Figure 15. – Two modes of a variable cycle engine.
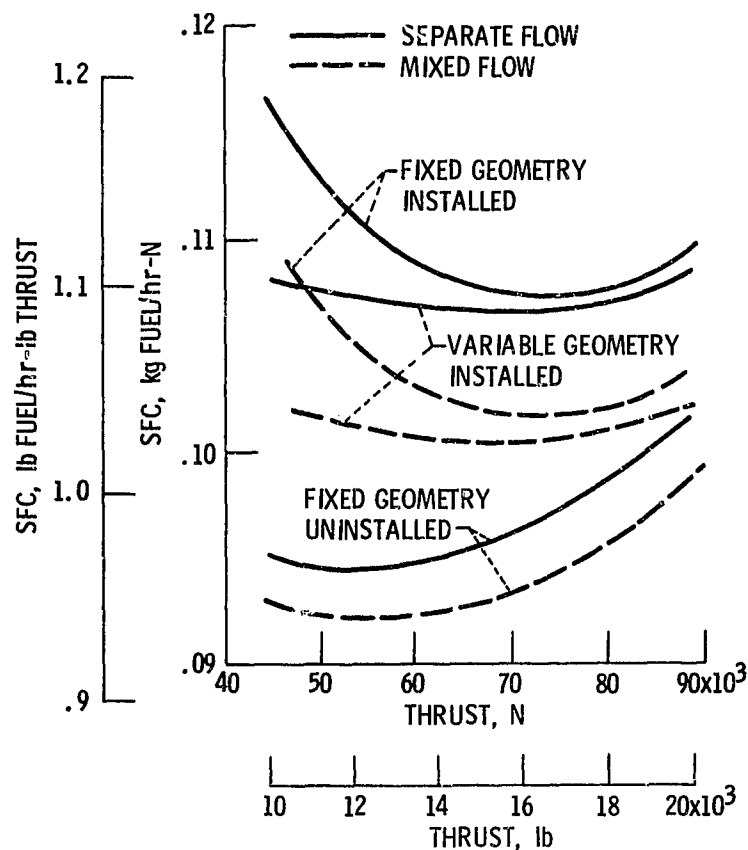
Figure 16. – Engine specific fuel consumption as a func-
tion of engine thrust. Mach 0.9, 11 000 m (36 089 ft).

*ENGINE MANUFACTURERS PROPOSAL ENGINES
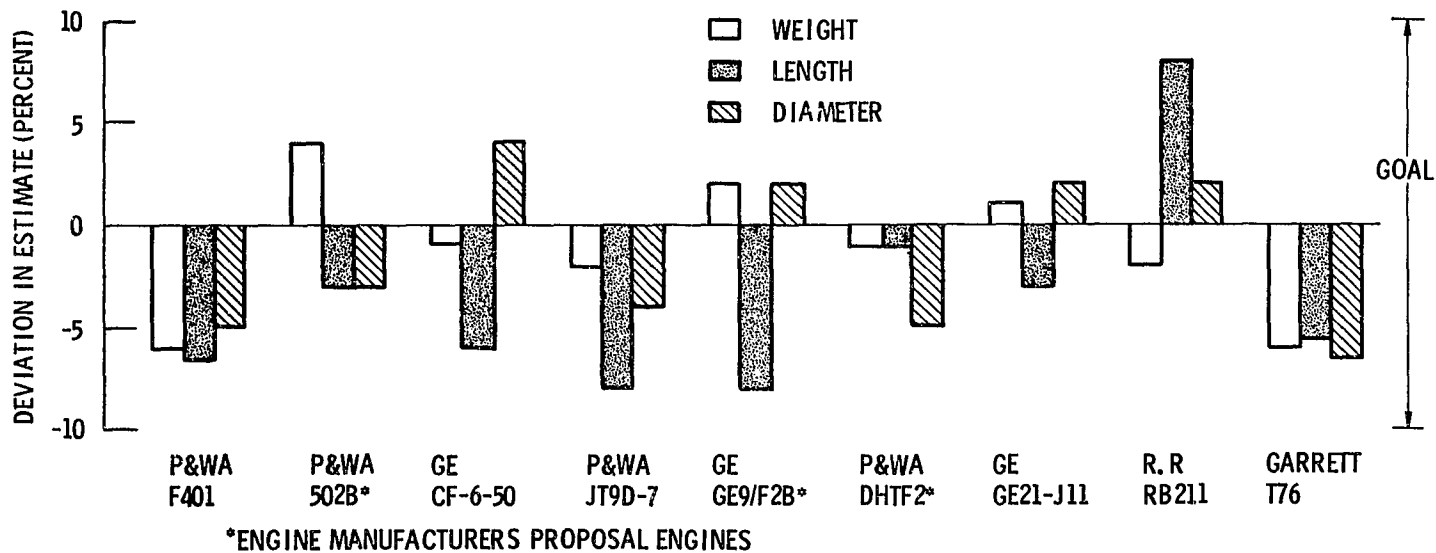
Figure 17. – Program results compared to manufacturers quotations.
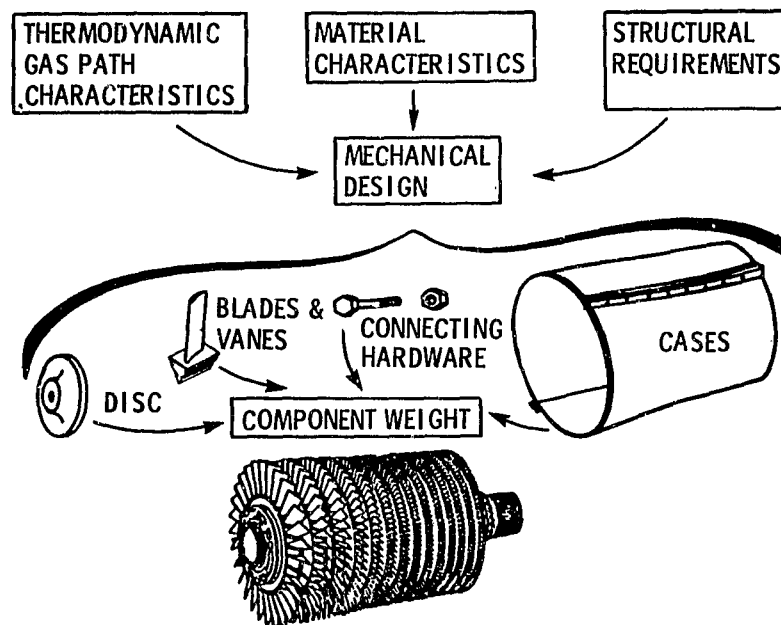
Figure 18. – Compressor weight method.
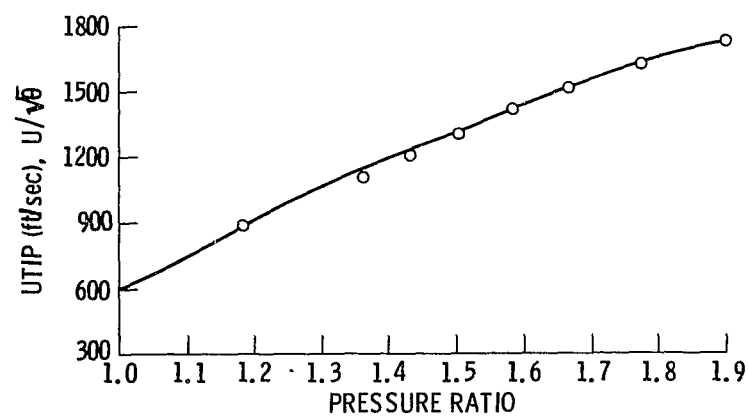
Figure 19. – Compressor tip speed estimate for determina-
tion of shaft speed.
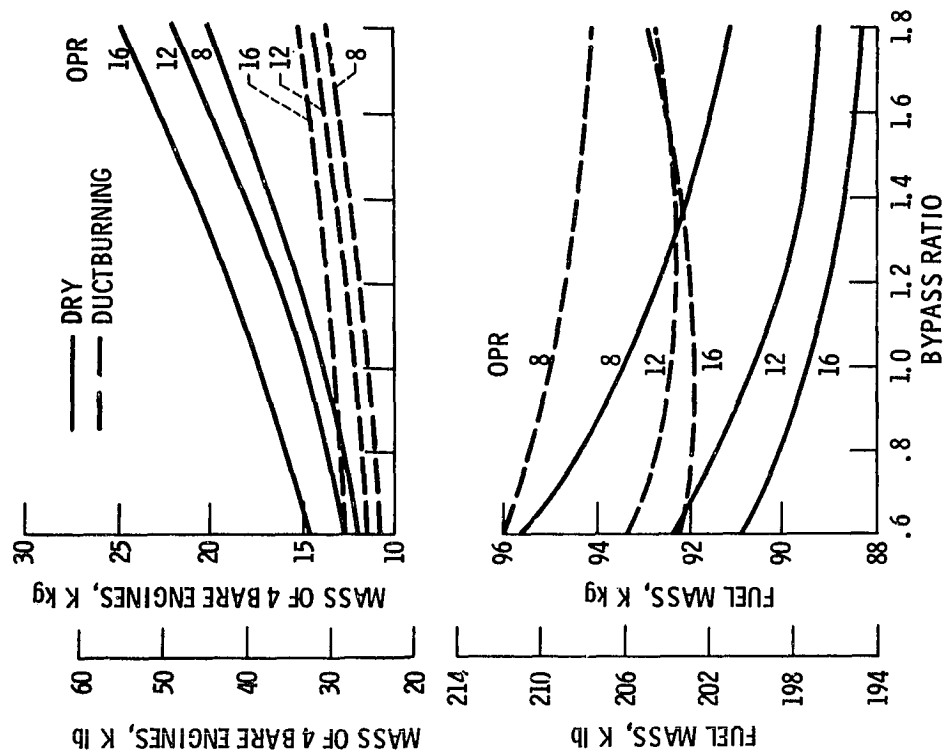
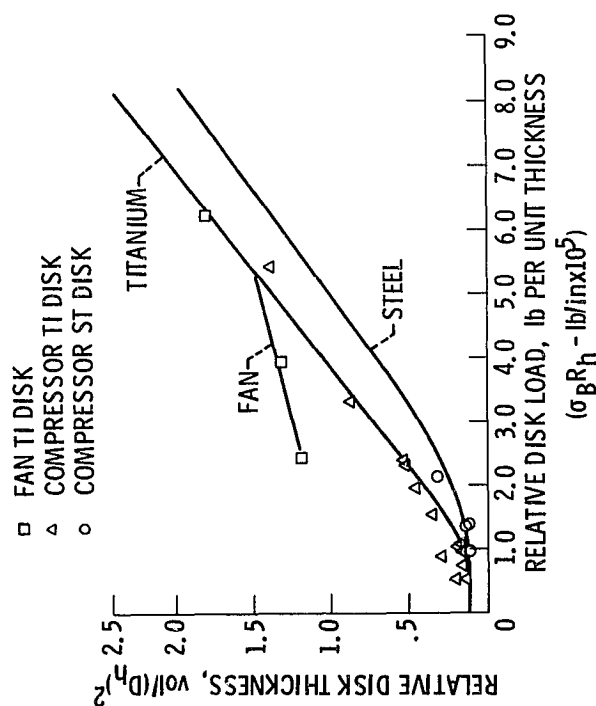Figure 21. – Fuel mass and bare engine mass, 88 950 N (20 000 lb) thrust engines.

Figure 20. – Data base engine: compressor disk volume volume correlation.
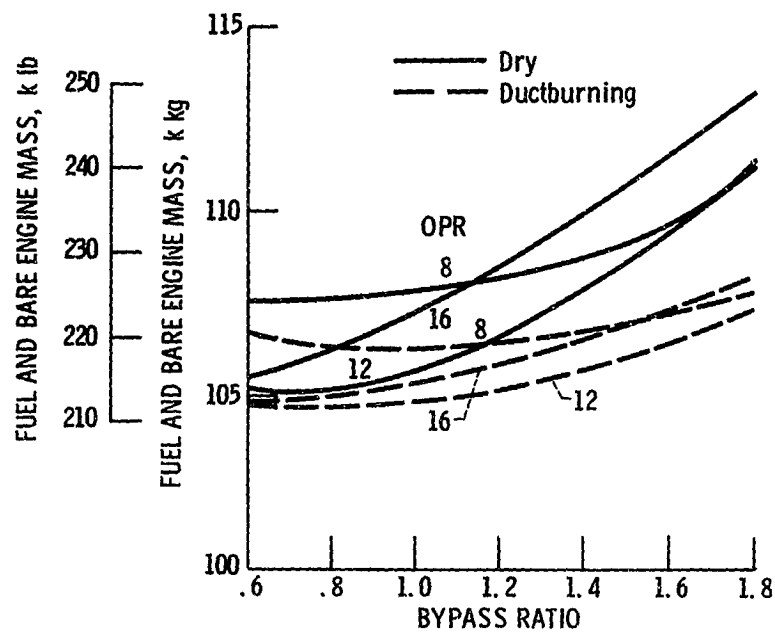
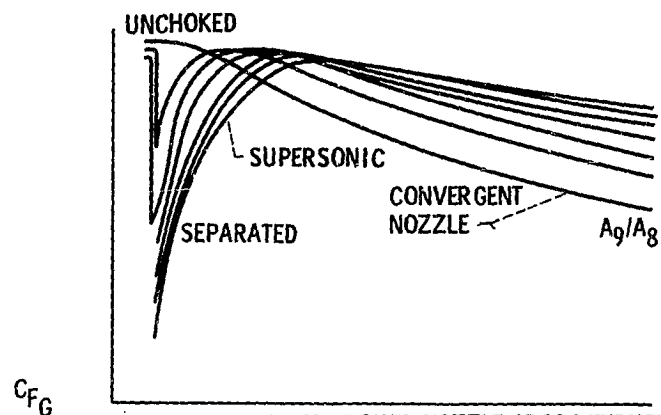Figure 22. - Sum of engine plus fuel mass, 88 950 N (20 000 lb) thrust engines.
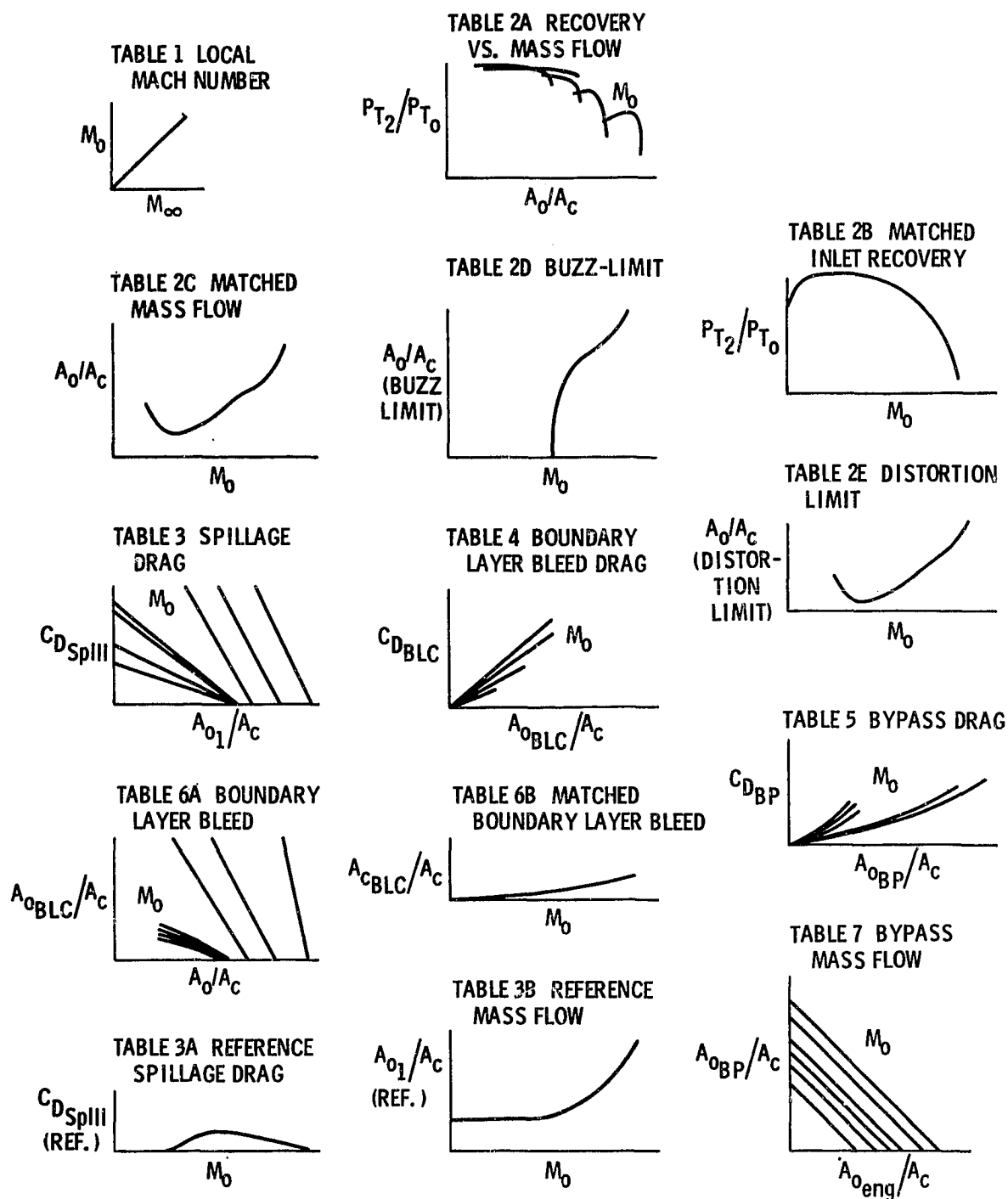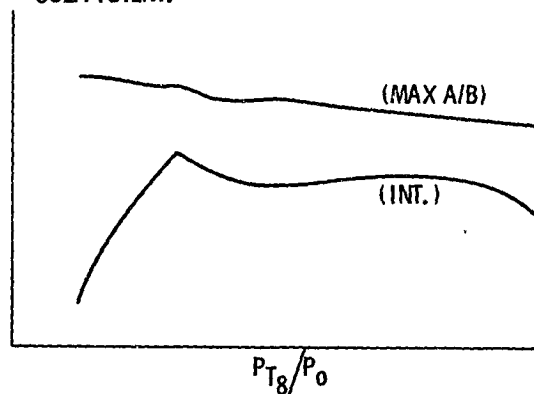
Figure 23. - Format for inlet performance characteristics maps.
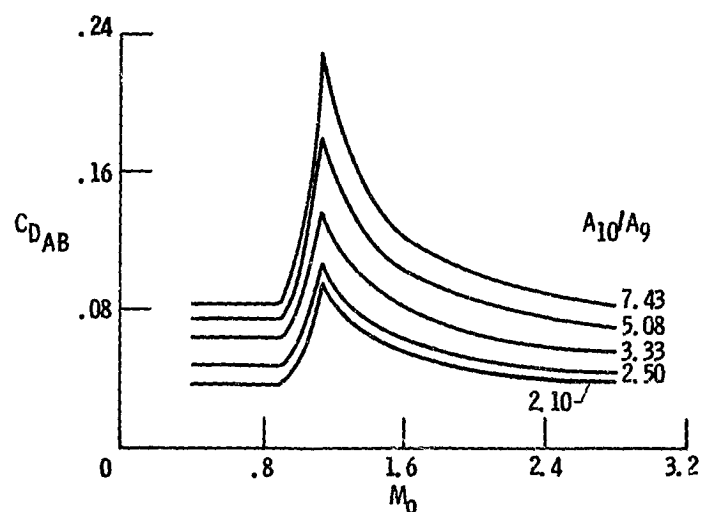
(a) DATA FORMAT FOR NOZZLE/AFTBODY DRAG.

(b) DATA FORMAT FOR ROUND NOZZLE GROSS THRUST COEFFICIENT.

(c) DATA FORMAT FOR NOZZLE GROSS THRUST CO-EFFICIENT FOR TWO-DIMENSIONAL NOZZLES.

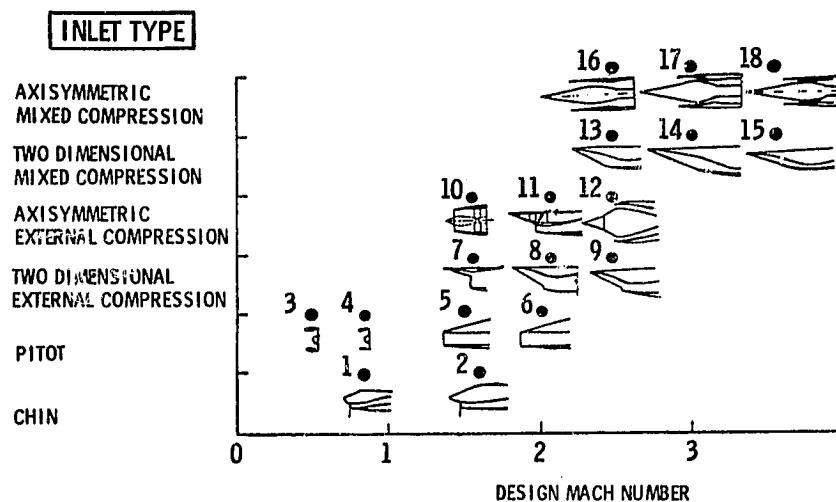Figure 24. - Format for nozzle/aftbody drag and C_{F_G} maps.
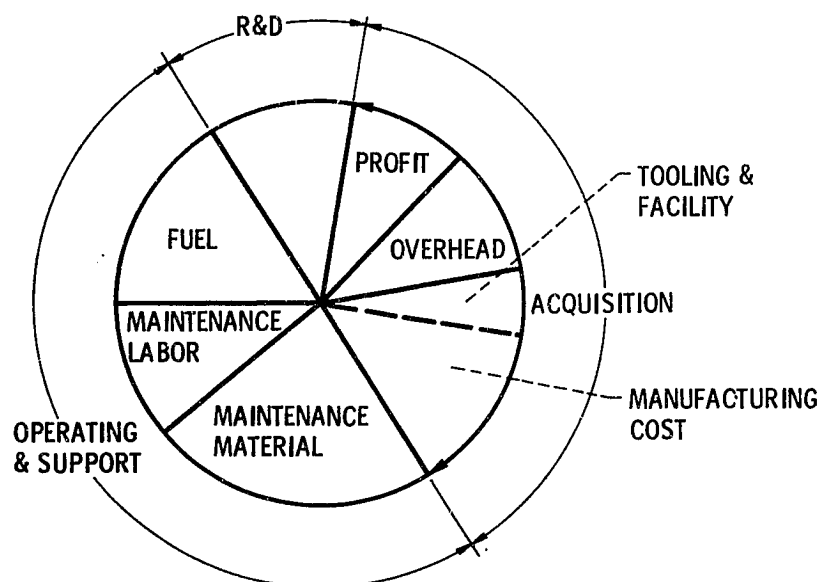
Figure 25. – Matrix of inlet maps.

Figure 26. – Engine life cycle cost.

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15. Supplementary Notes

16. Abstract

Availability of a suitable propulsion system is generally acknowledged to be a key requirement for the successful development of a new airplane. In order to evaluate the suitability of a propulsion system, it is necessary to be able to calculate both the steady state and transient performance of the engine throughout the flight regime. In addition, installation effects on thrust and specific fuel consumption must be accounted for as well as engine weight, dimensions, and cost. This paper discusses the availability throughout the government and industry of analytical methods for calculating these quantities.

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