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FINAL REPORT

INVESTIGATION OF THRUSTERS FOR CRYOGENIC REACTION CONTROL SYSTEMS

CASE FILE COPY

APPENDICES

VOLUME II

By
R. J. JOHNSON

TRW
SYSTEMS GROUP

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
NASA Lewis Research Center
Contract NAS 3-11227

P. N. HERR,
Project Manager

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CONTENTS (Continued)

Page

APPENDICES – VOLUME II

A ENGINEERING IGNITION AND FLAMMABILITY DESIGN DATA FOR H_2/O_2 . . . . . A-1

B CATALYTIC REACTOR THERMAL RESPONSE DATA. B-1

C CALCULATION OF C* EFFICIENCY C-1

D COMPUTER PROGRAMS D-1

APPENDIX A

ENGINEERING IGNITION AND FLAMMABILITY DESIGN DATA FOR H_2/O_2

The thermodynamics of H_2/O_2 ignition and combustion requirements can be readily determined by considering some rudimentary chemistry and energy balances. It is well known that the reaction

$$aH_2 + bO_2 = cH_2O + dH_2 \tag{A1}$$

is spontaneously unstable, if a sufficient "energy" addition can be introduced to the mixed gases. Energy addition requirements are a carry over from early investigation ignition work with spark devices where the amount of Joulean heating as measured by Joulean dissipation in a spark plug was used as a measure of energy requirement. Subsequent thermodynamic reasoning somewhat tempers this concept. The thermal reaction mechanism is not totally understood; however, it is feasible that the reaction steps to sustain the reaction are of the following types:

$$\begin{aligned} OH + H_2 &= H_2O + H \\ H + O_2 &= OH + O \\ O + H_2 &= OH + H \end{aligned} \tag{A2}$$

The second reaction above is endothermic by 17 kcal. As such, at ambient temperatures to some threshold temperature a mixture of H_2 and O_2 is very stable. Above some temperatures the second reaction becomes sufficiently frequent to sustain the overall reaction, and continued deflagration occurs. The exact point of this occurrence is pressure dependent and is not generally a priori predictable. At reduced pressures the initial deflagration levels to explosion. As the pressure is increased, a second explosion limit is reached wherein steady-state reaction with no explosion can occur. This is theoretically explained by inclusion of the reaction

$$H + O_2 + M = HO_2 + M \tag{A3}$$

HO_2 becomes a chain breaking radical which is relatively unreactive and diffuses from the overall reaction to the reacter walls. Exact predictions of reactive behavior are dependent upon the vessels and its wall coatings. Finally, the reaction

$$Z\ HO_2 \xrightarrow{\text{wall}} H_2O_2 + O_2 \tag{A4}$$

is introduced to substantiate the no explosion reaction regions, although experimentally it is found to be entirely dependent upon vessel geometry.

Because of the highly dissipative nature of the H_2/O_2 gas mixture with respect to energy additon, it is perhaps more instructive from an engineering ignition device standpoint to examine the overall threshold temperatures at which auto-ignition can take place and sustain a reaction. Data are limited from which such a curve can be established; however, available empirical data are given in Figure A-1. The threshold temperature is affected by diluent H_2, O_2 or a third body such as He or N_2. The auto-ignition temperature is nominally given as 1100°F, but it is seen to be strongly influenced by off-stoichiometric mixture rations. The addition of H_2 or He (presumed to be similar to N_2) rapidly raises the autoignition temperature, whereas the addition of O_2 markedly decreases the autoignition temperature.

Engineering flammability design limits for H_2/O_2 mixtures nominally run from 4 to 96 percent (volume measure) for H_2 in O_2. Data from the Bureau of Mines at the fuel rich limit are shown in Figure A-2 for several O_2 temperatures and H_2 temperatures. A two-dimensional flammability region plot is given in Figure A-3 for the H_2/O_2/He system. The data points represent all known available data. Reactions will quench in well stirred mixtures of H_2/O_2/diluents which are outside the limits. At the fuel rich side there is little effect of temperature. For a reaction to sustain itself independent of any other mechanism, the local mixture compositions must satisfy the criteria shown graphically in Figure A-2 and A-3. Further, in order for a thermal source such as a hot gas jet to ignite the mixture, it must be at a total temperature level of at least the auto-ignition temperature given in Figure A-1.

The theoretical temperatures versus MR for H_2/O_2 and H_2/O_2/He are given in Figures A-4 to A-10 along with theoretical characteristic velocity data. Mixture ratio is defined as

$$MR = \frac{\text{weight of } O_2 + \text{weight of diluent}}{\text{weight of } H_2 + \text{weight of diluent}} \tag{A5}$$

A comparison of the results in Figures A-4 to A10 leads to the following conclusions:

1) The T versus MR at fuel rich mixture ratios is quite steep, pointing out a possible need for reasonably accurate MR control to prevent burnout of the igniter device.

2) The effect of helium in oxygen is to replace an active species and thus reduce combustion temperature.

3) The effect of helium in hydrogen is to replace H_2 diluent which has a high heat capacity with helium which has a lower heat capacity. Therefore, the combustion temperature increases slightly.

4) The MR for optimum performance is shifted toward higher mixture ratios—especially for mixtures produced from an oxidizer stream 25 wt percent He or more.

The theoretical performance and combustion temperature charts also define a locus for maximum C^* and temperature. It is noted that a theoretical performance variation of less than ±5 percent can be achieved as up to 50 wt percent He is added to the oxidizer stream. The optimum theoretical combustion temperature remains relatively constant also, although the mixture ratio goes through an order-of-magnitude change. Theoretical combustion temperatures as a function of mixture ratio are presented in Figures A-11 and A-12 for chamber pressures of 10 and 100 psia, respectively.

Flame speeds for H_2/O_2 mixtures near stoichiometry used in the igniter designs of this program are given in Figures A-13 and A-14 for both turbulent and laminar flames. To prevent flashback in premix systems the throughput velocities must be greater than the flame speeds across the entire flow passage. This means that particular attention must be paid to the boundary flow where the velocity gradients may be such as to support flame propagation.

There exists a finite ignition delay for H_2/O_2 which is brought about by finite reaction rate chemistry, and it is important to recognize this because of the high heat capacity of H_2. The ignition delay chemistry can be characterized by τ_D. If a hot torch of $H_2O + H_2$ is introduced into a mixture to be ignited, particular attention must be given to the possibility of reaction quenching due to H_2 heat absorbtion, if the total mix temperature, in the absence of reaction is lower than the autoignition temperature. If the local mechanical mixing delay time is characterized by τ_M, then

$$\tau_D << \tau_M \text{ for ignition} \tag{A6}$$

The effects of pressure on ignition are not well documented. Laboratory data are shown in Figure A-15. It is well known that geometry changes can affect these results; however, they can serve as a guide.

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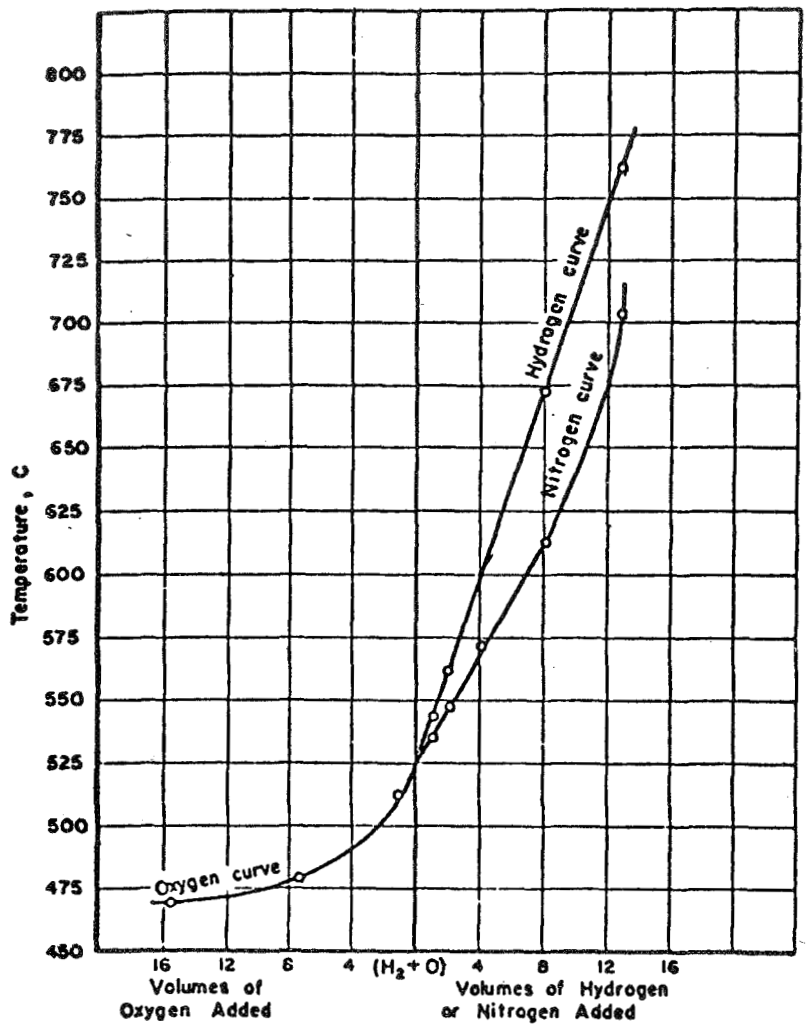


Figure A-1. Change in Autoignition Temperature by Dilution of a Stoichiometric Mixture of Hydrogen and Oxygen

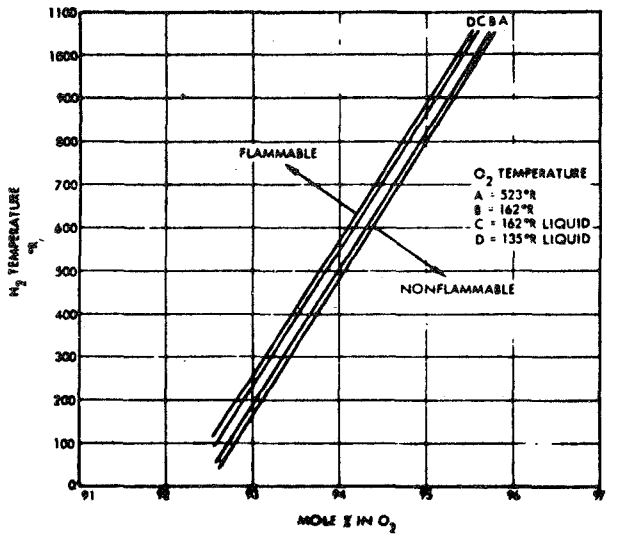


Figure A-2. Flammability Limits for H_2/O_2 at Fuel Rich MR

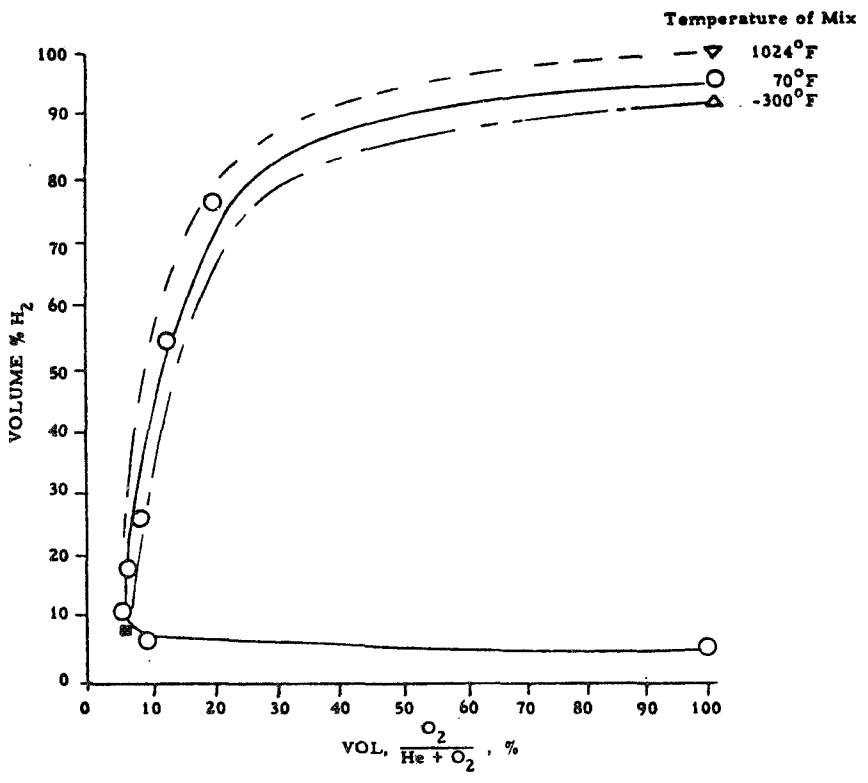


Figure A-2. Flammability Limits for H_2/O_2 at Fuel Rich MR

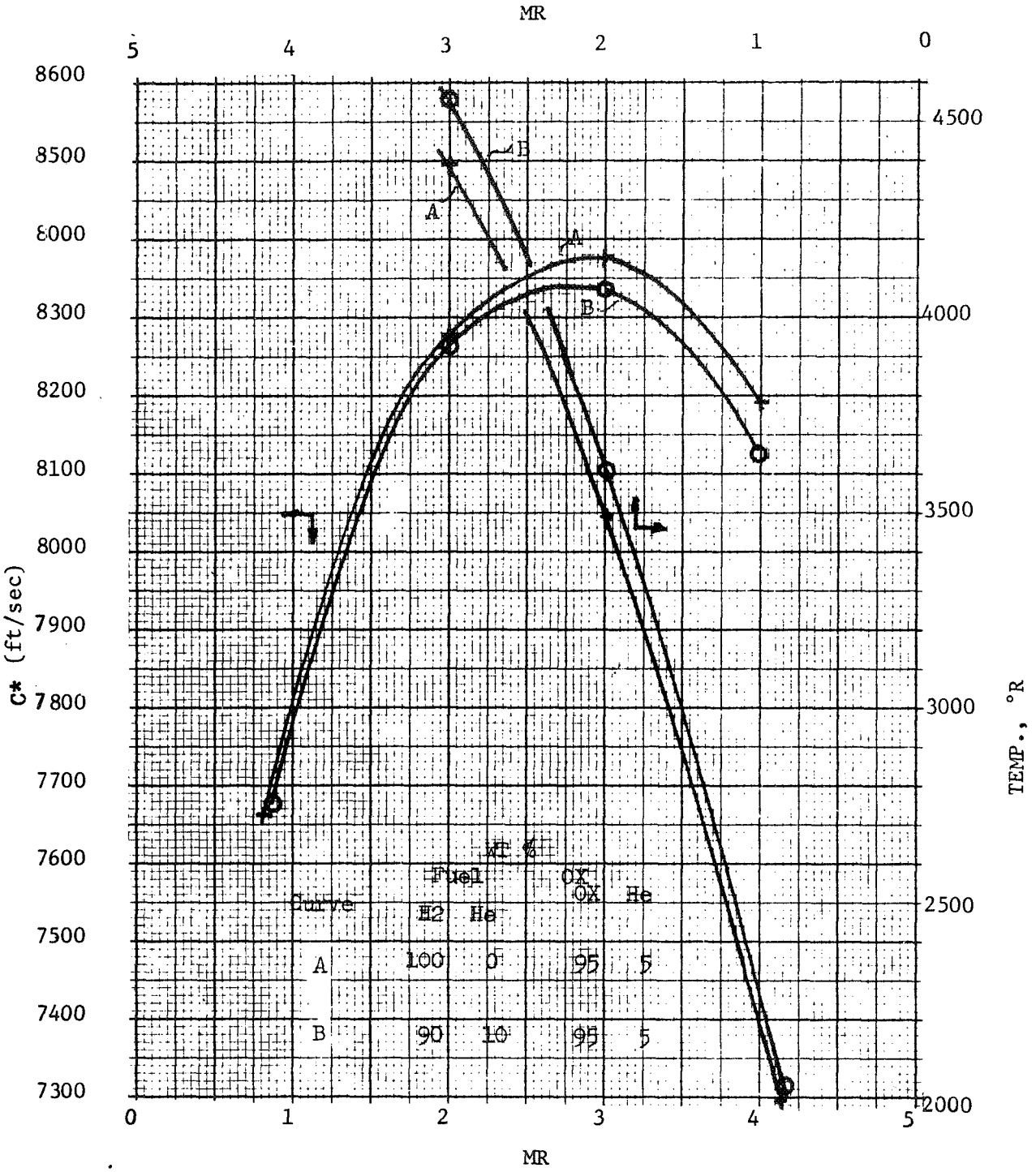


Figure A-4. Theoretical Performance and Combustion Temperature for 5 WT Percent He in O_2

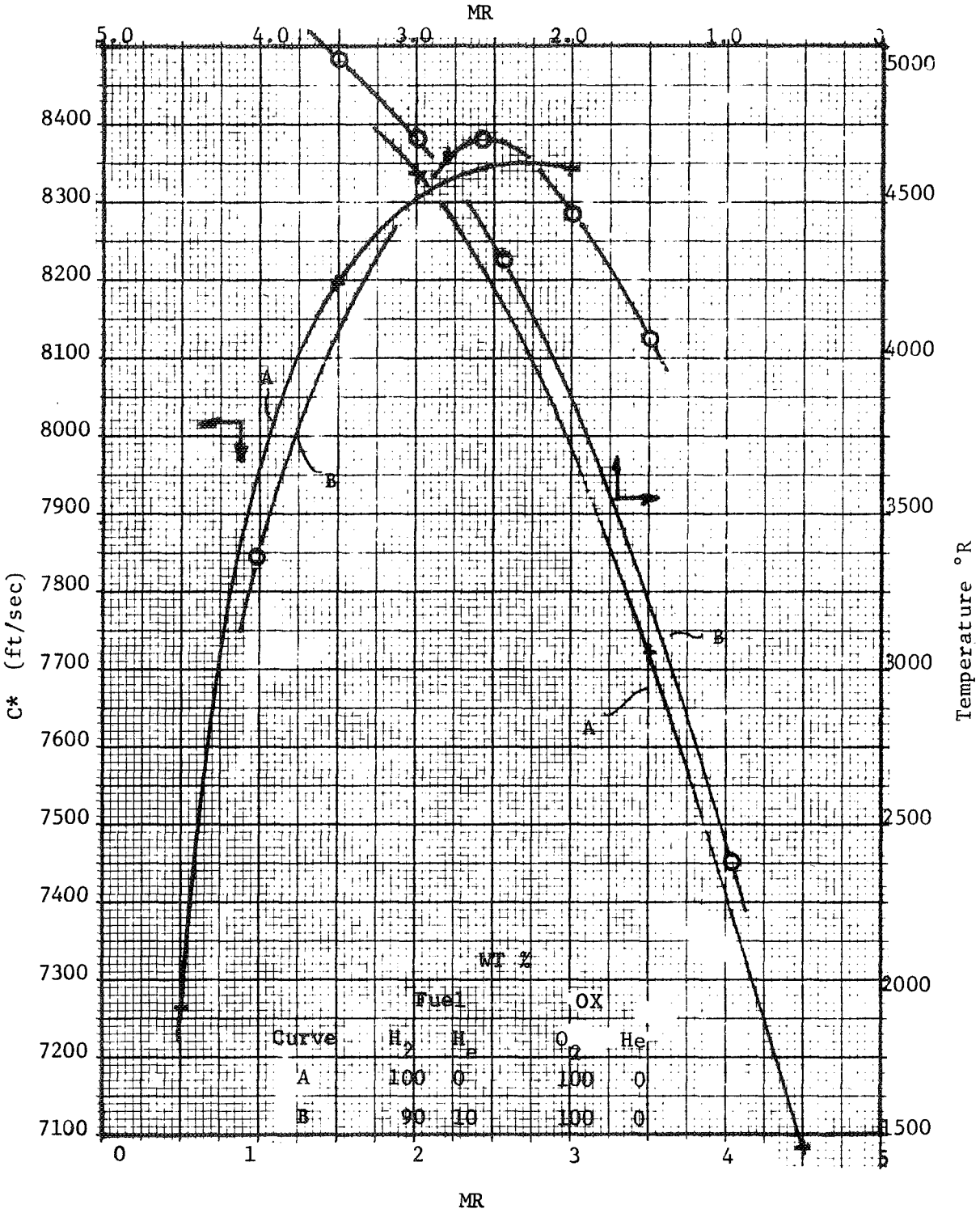


Figure A-5. Theoretical Performance and Combustion Temperature for 10 WT Percent He in H_2

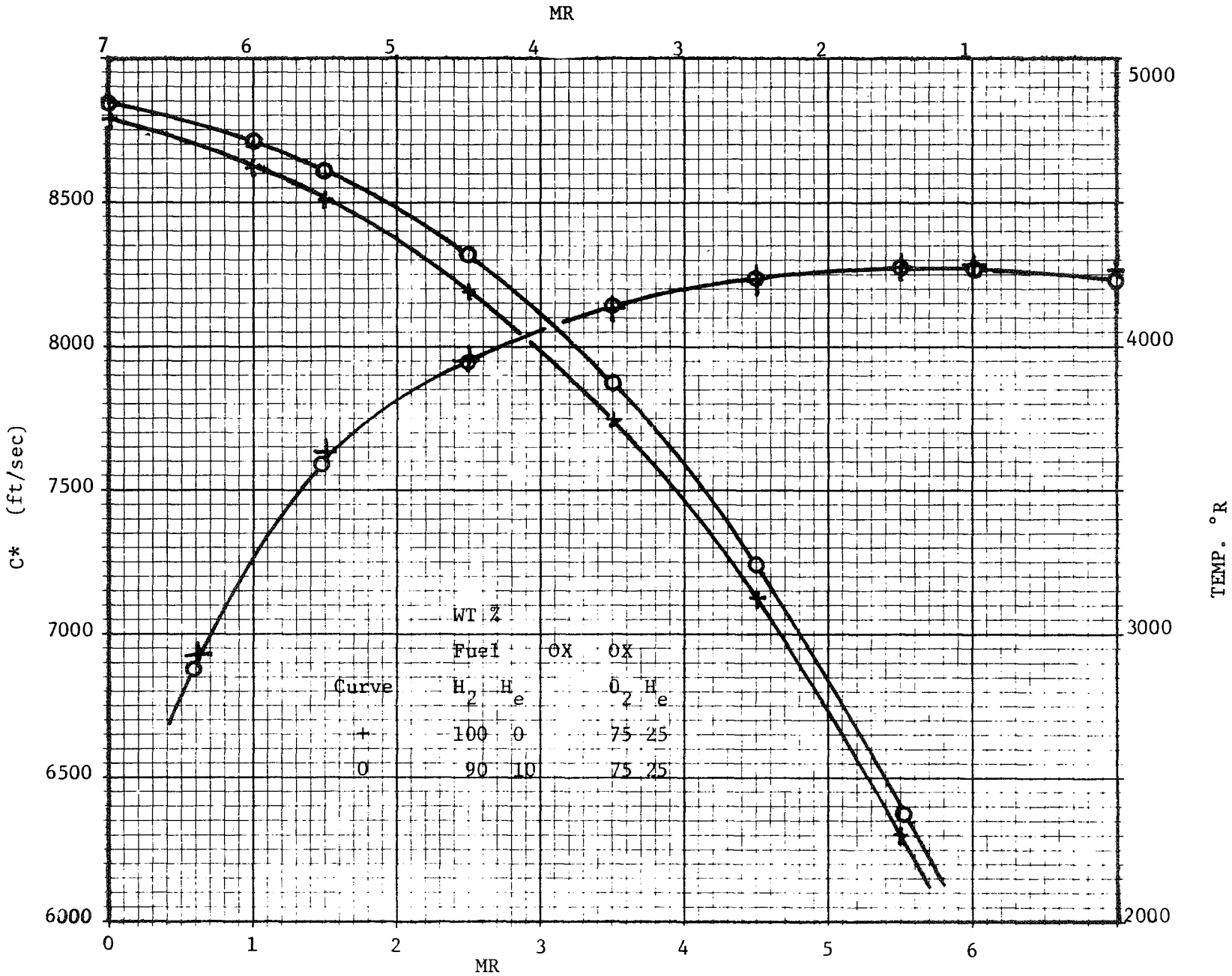


Figure A-6. Theoretical Performance and Combustion Temperature for 25 WT Percent He in O_2

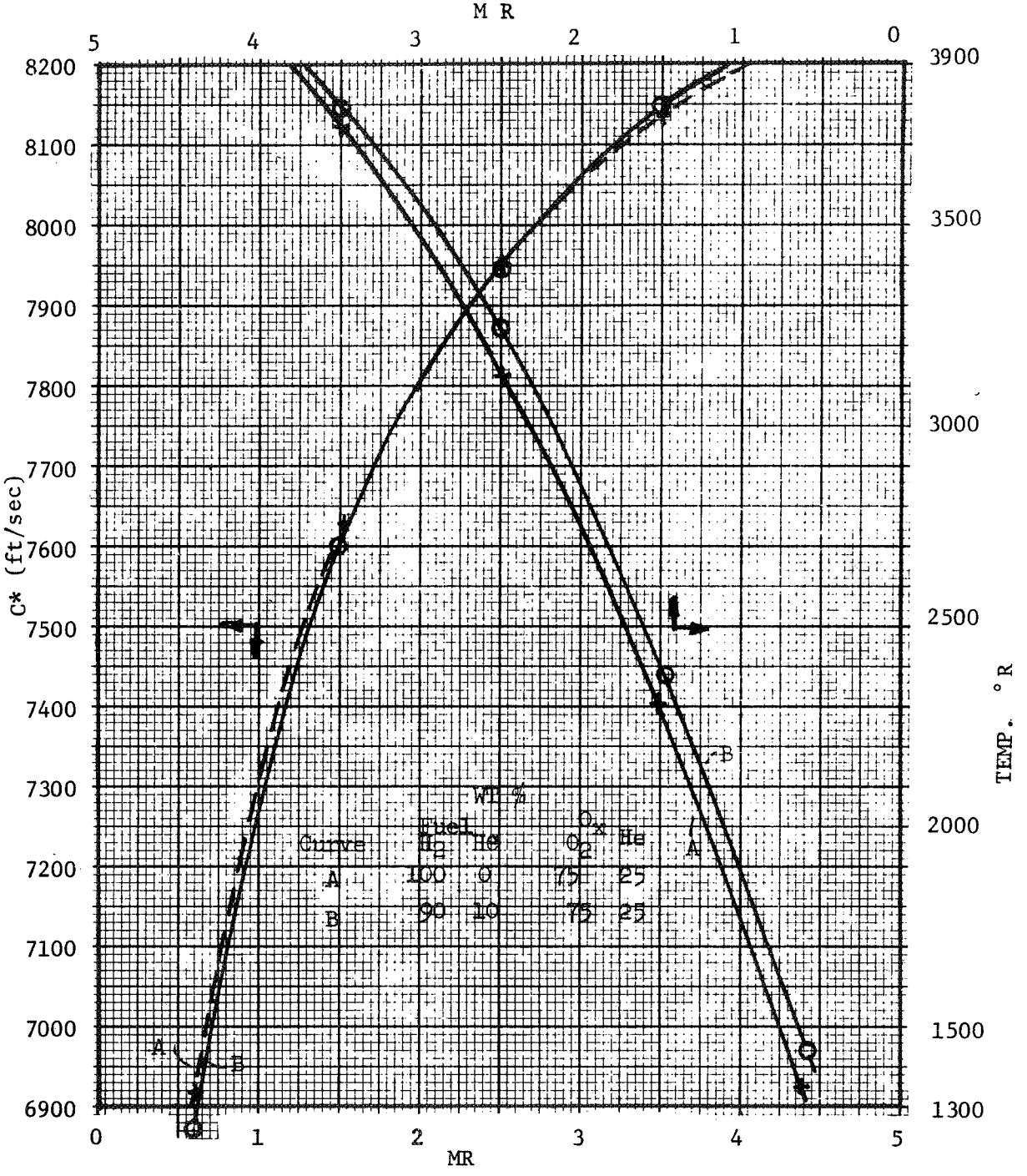


Figure A-7. Theoretical Performance and Combustion Temperature for 25 WT Percent He in O_2

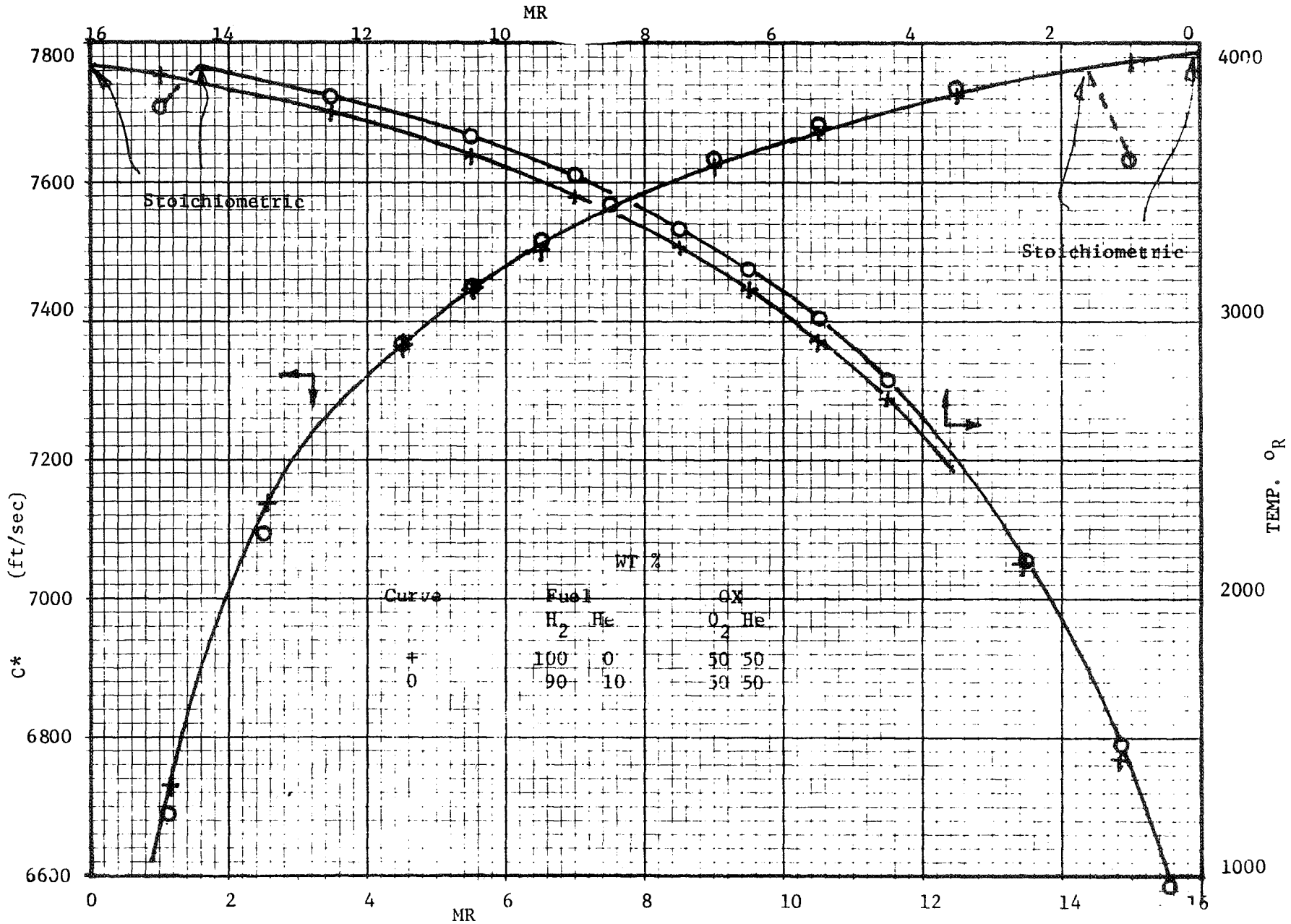


Figure A-8. Theoretical Performance and Combustion Temperature for 50 WT Percent He in O_2

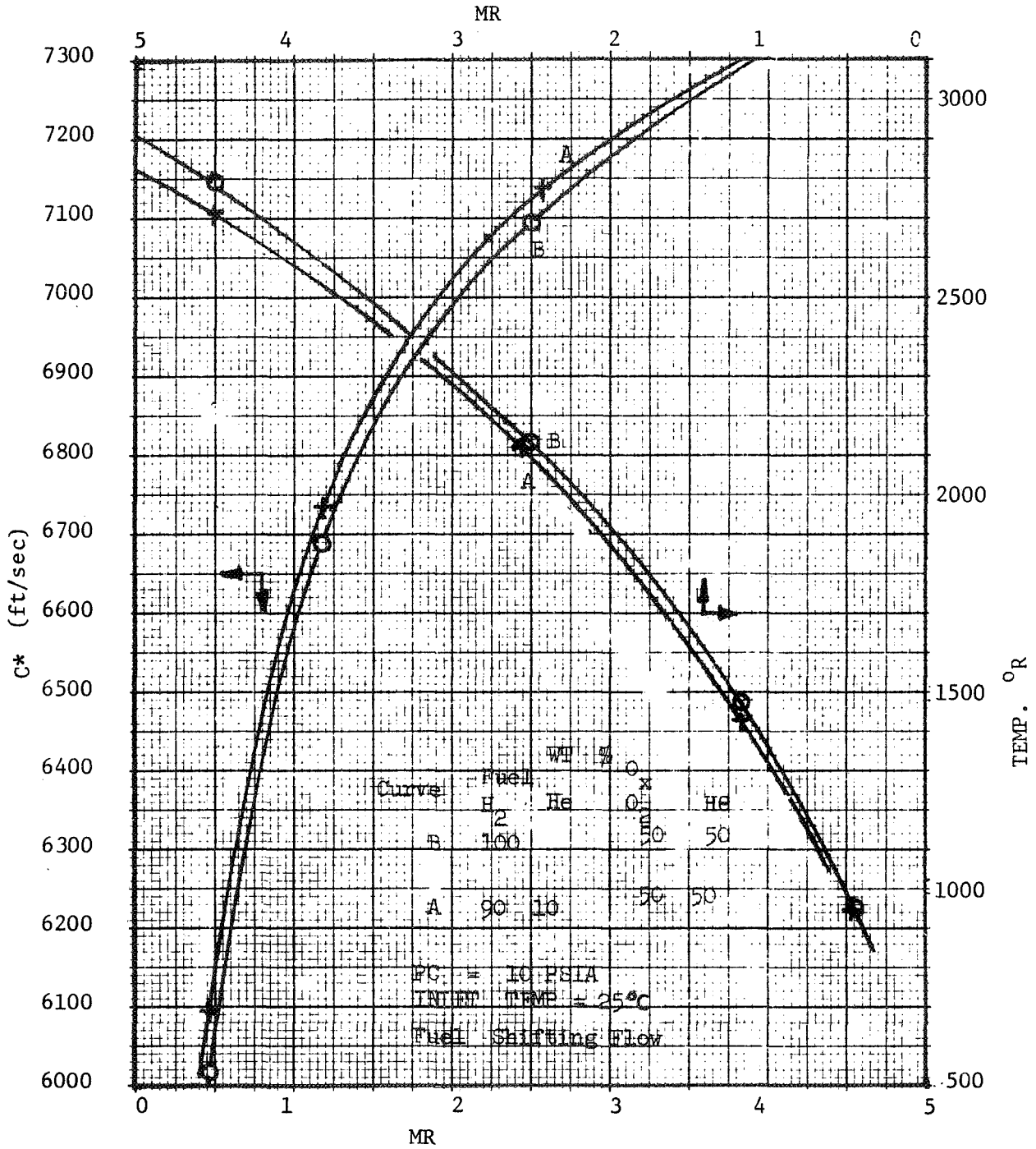


Figure A-9. Theoretical Performance and Combustion Temperature for 50 WT Percent He in O_2

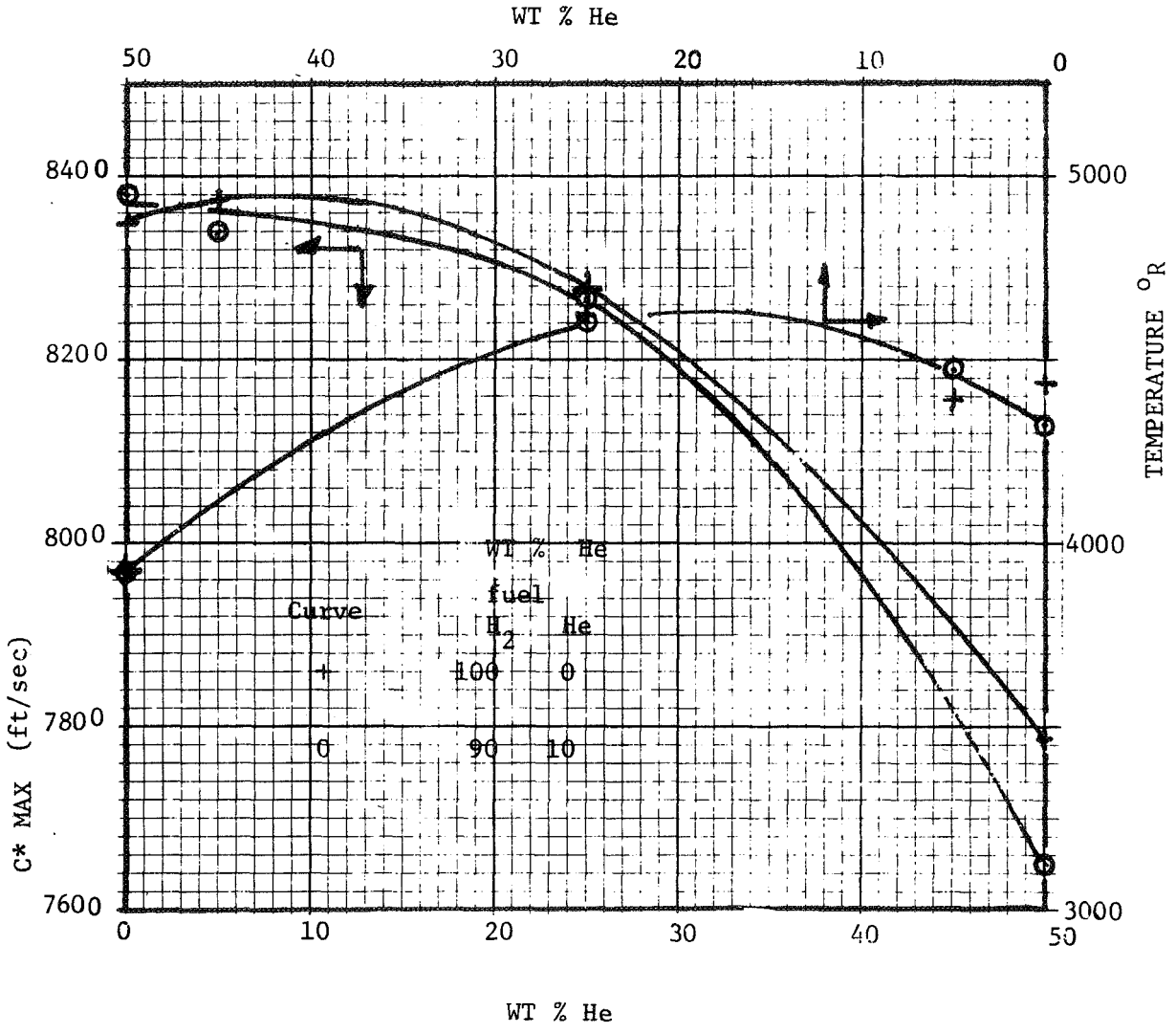


Figure A-10. Locus of Optimum Performance and Combustion Temperature Versus WT Percent He in O_2

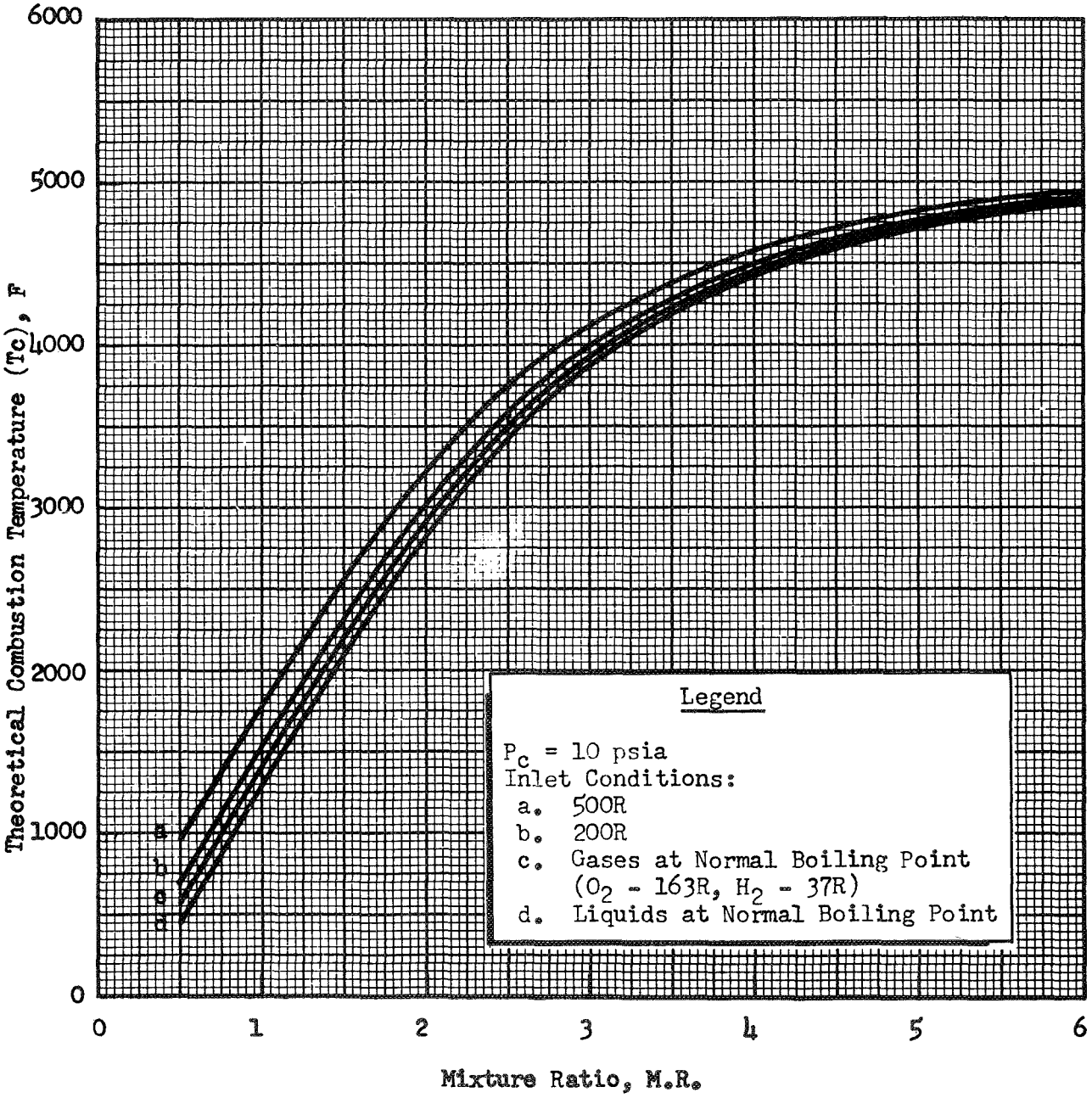


Figure A-11. Theoretical Combustion Temperature as a Function of Mixture Ratio for Four Propellant Conditions (P_c = 10 psia) (Rocketdyne NASA-CR-72176)

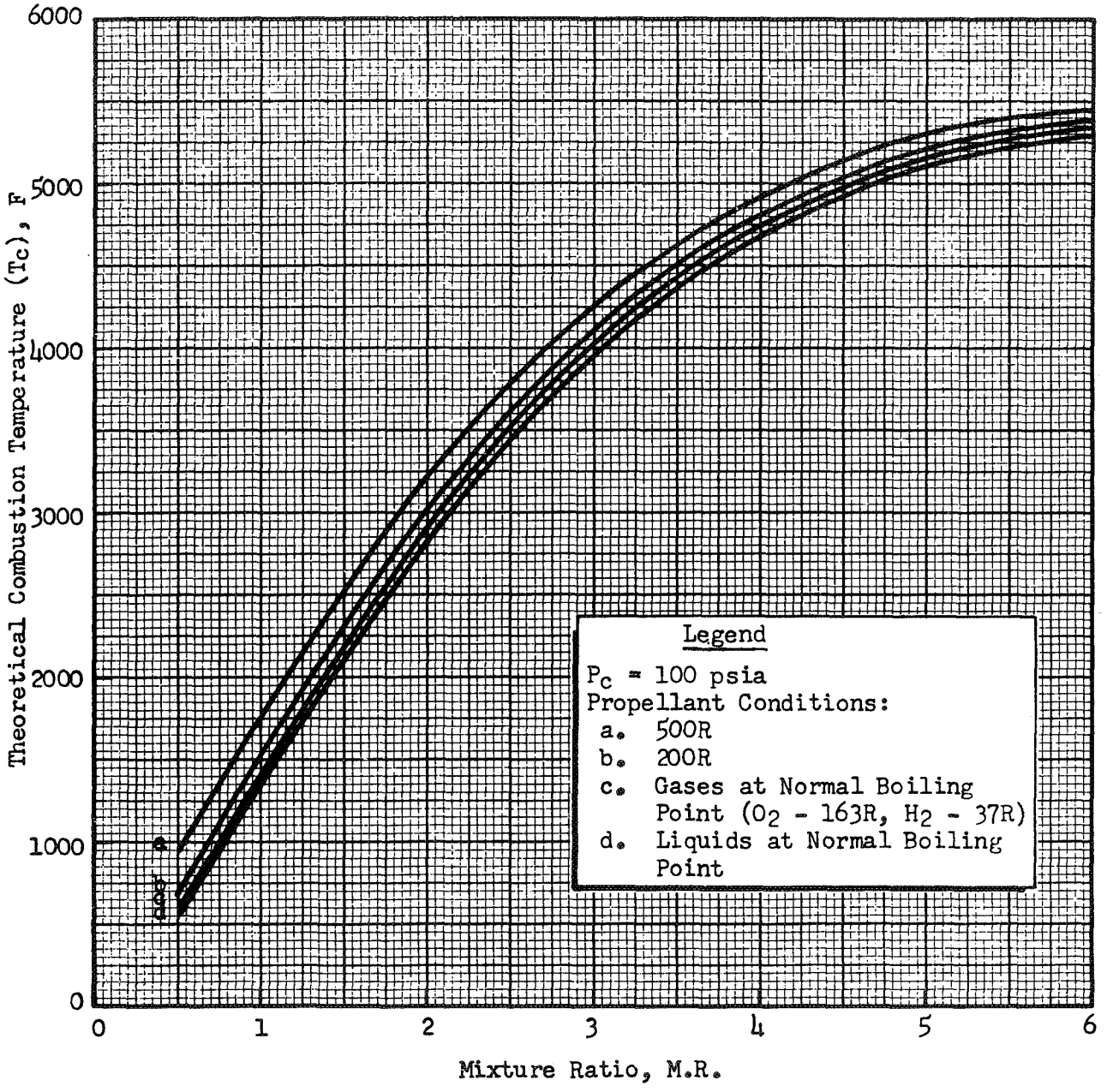


Figure A-12. Theoretical Combustion Temperature as a Function of Mixture Ratio for Four Propellant Conditions (P_c = 100 psia) (Rocketdyne NASA-CR-72176)

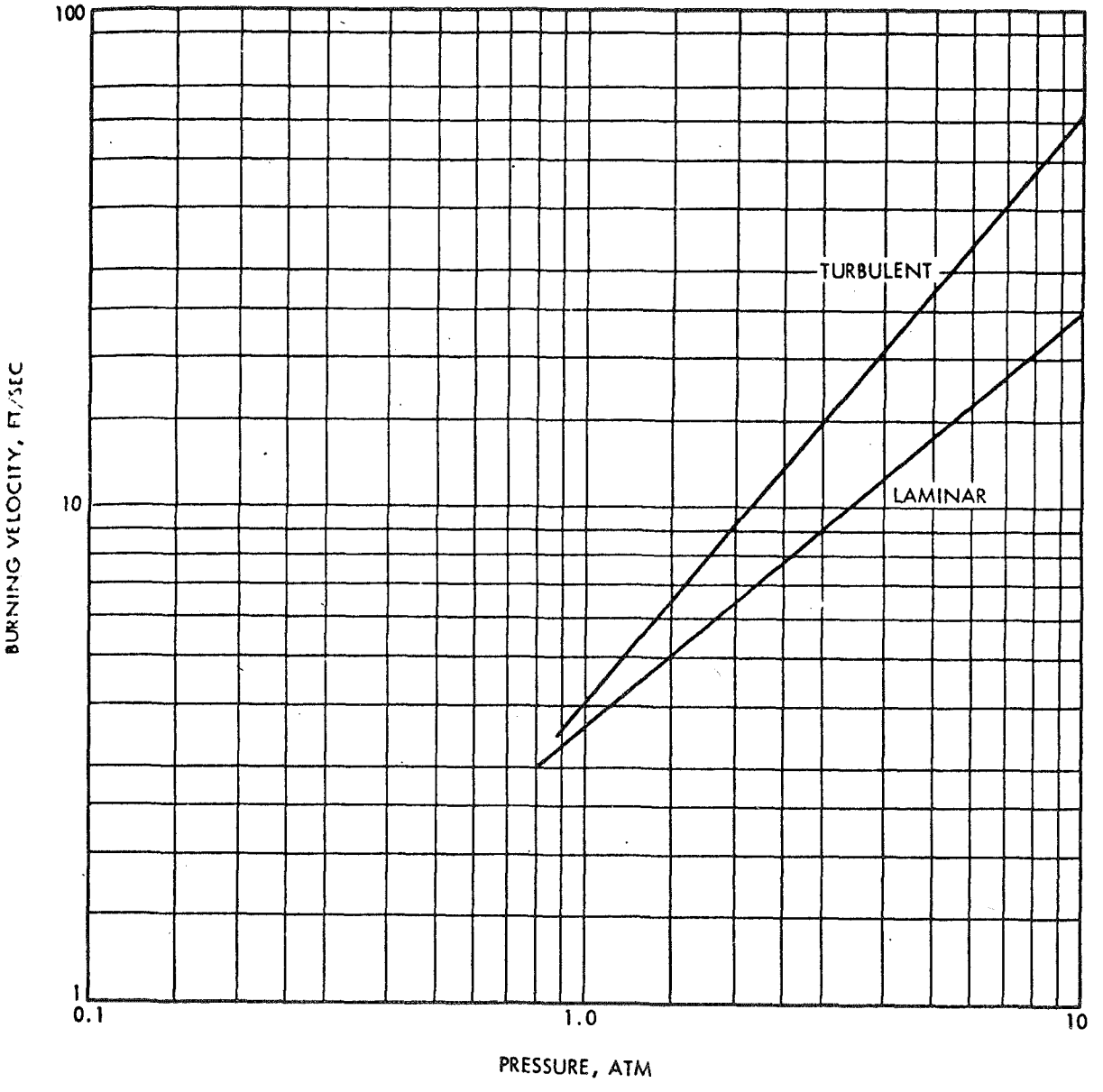


Figure A-13. H_2/O_2 Flame Velocities at Ambient Temperature Obtained from Literature (Laminar curve is well defined. Turbulent curve based on four data points).

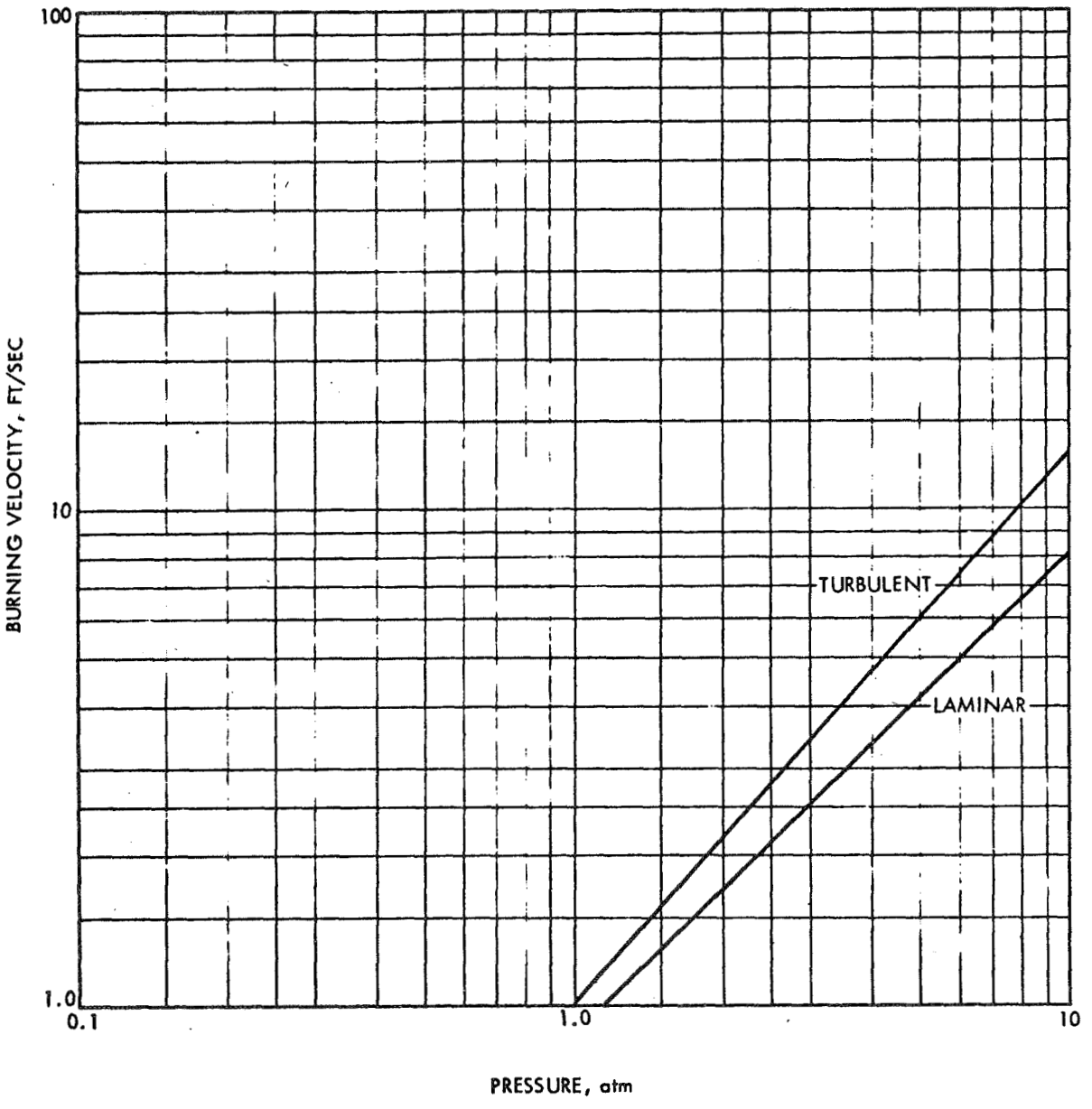


Figure A-14. Flame Velocity Data Extrapolated to 210°F for H_2/O_2

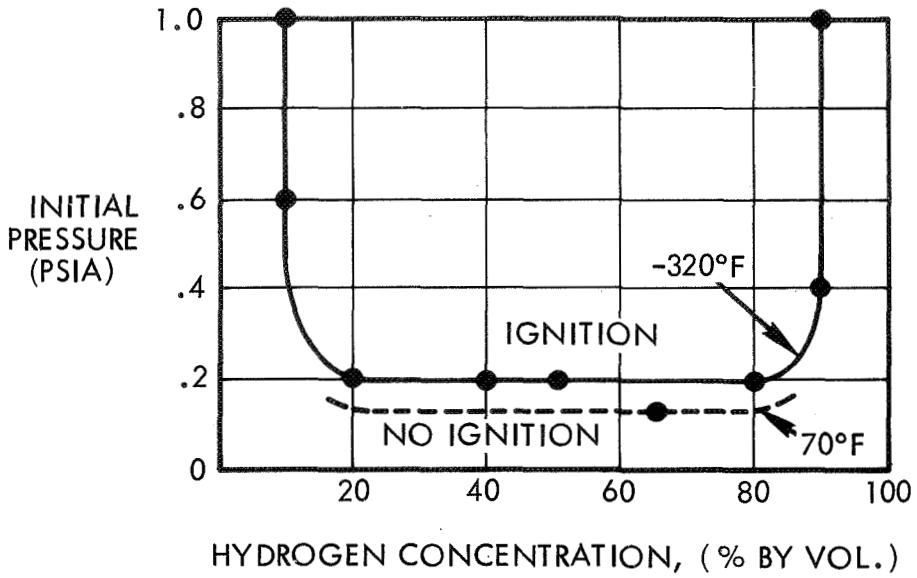


Figure A-15. Flammability Limits

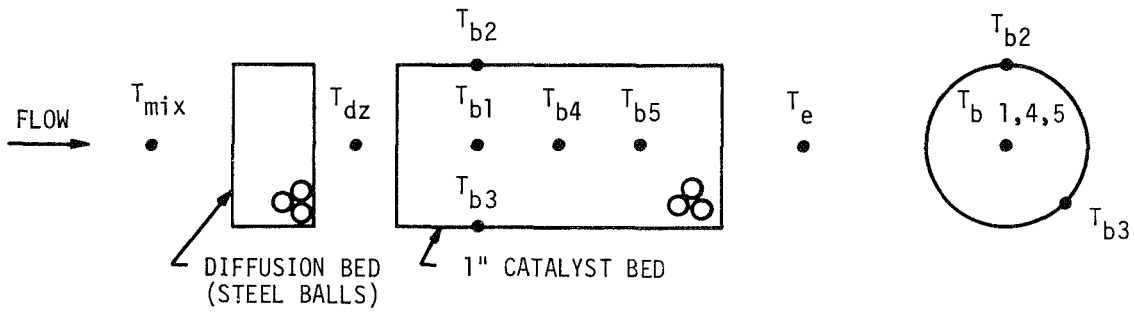
APPENDIX B

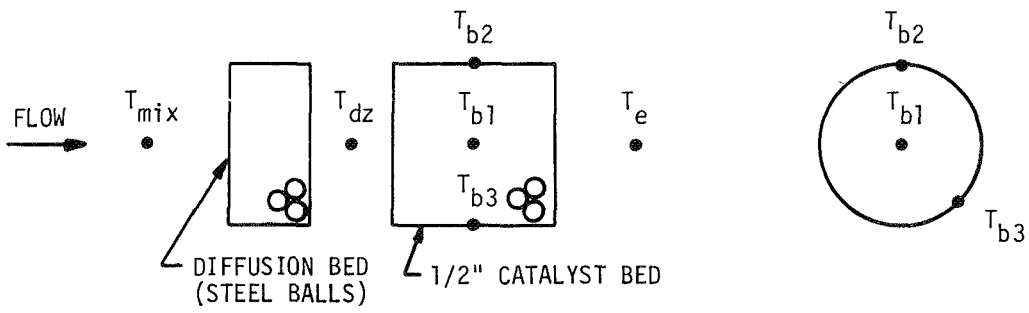
CATALYTIC REACTOR THERMAL RESPONSE DATA

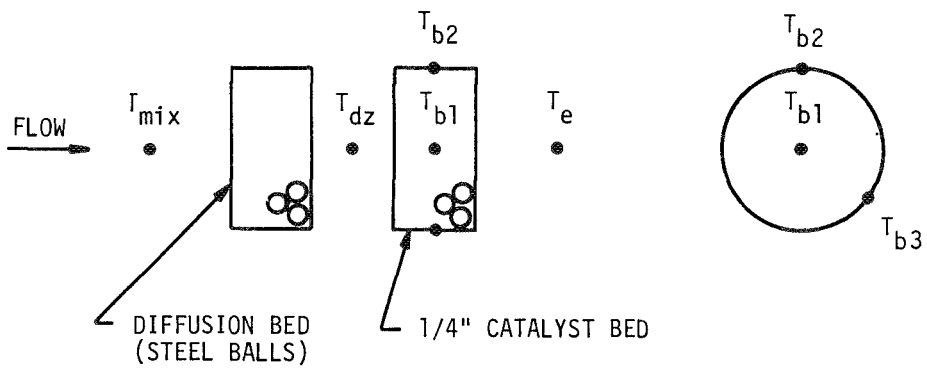
Time-temperature plots of all valid thermal data from the catalytic reactor tests are included in this appendix. Specific operating conditions for each test firing can be found in Tables 14 and 15 of Section 4.3 of Volume 1.

Thermocouple locations for each length catalyst bed were as shown in the following illustration:

THERMOCOUPLE LOCATIONS







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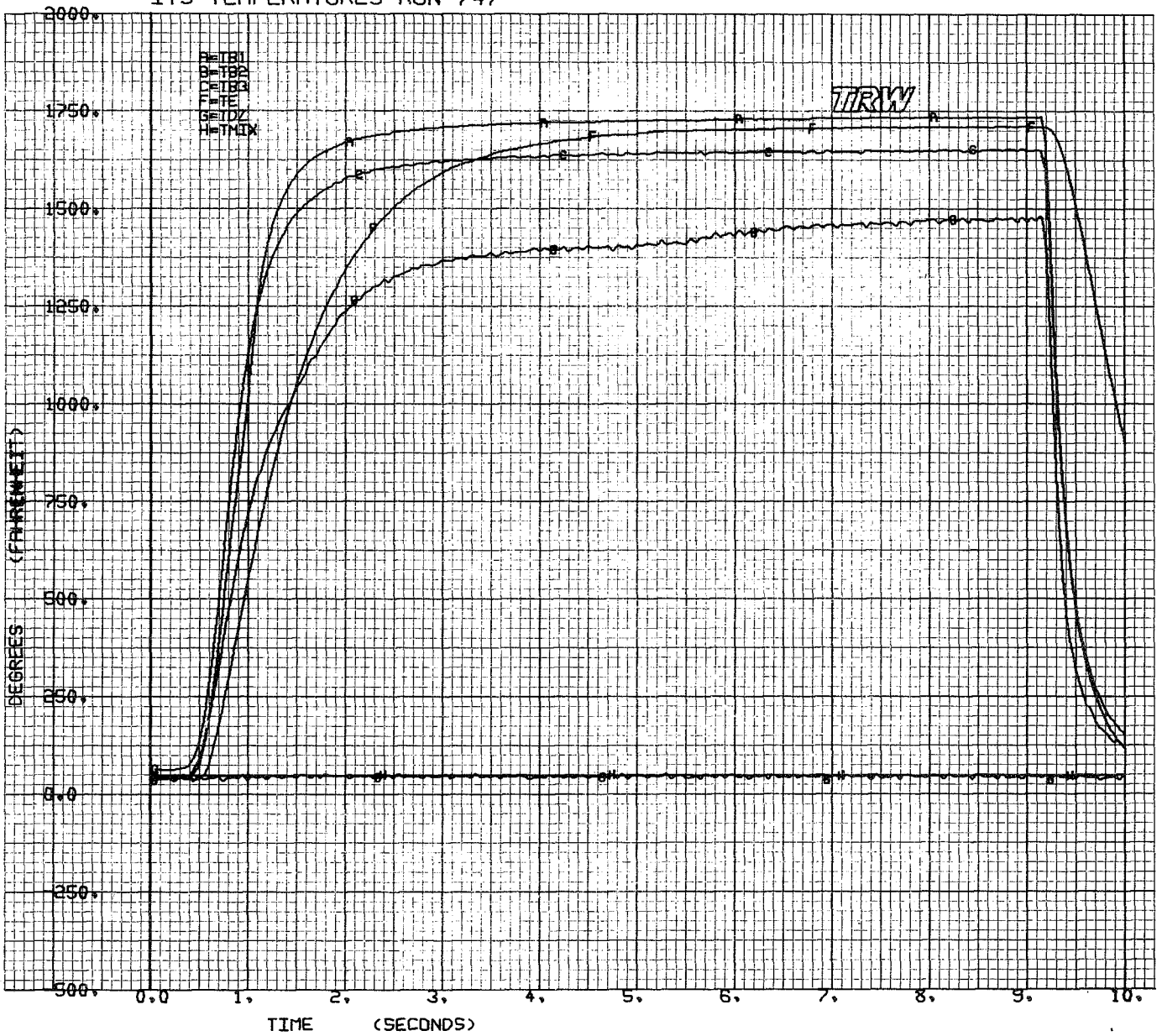

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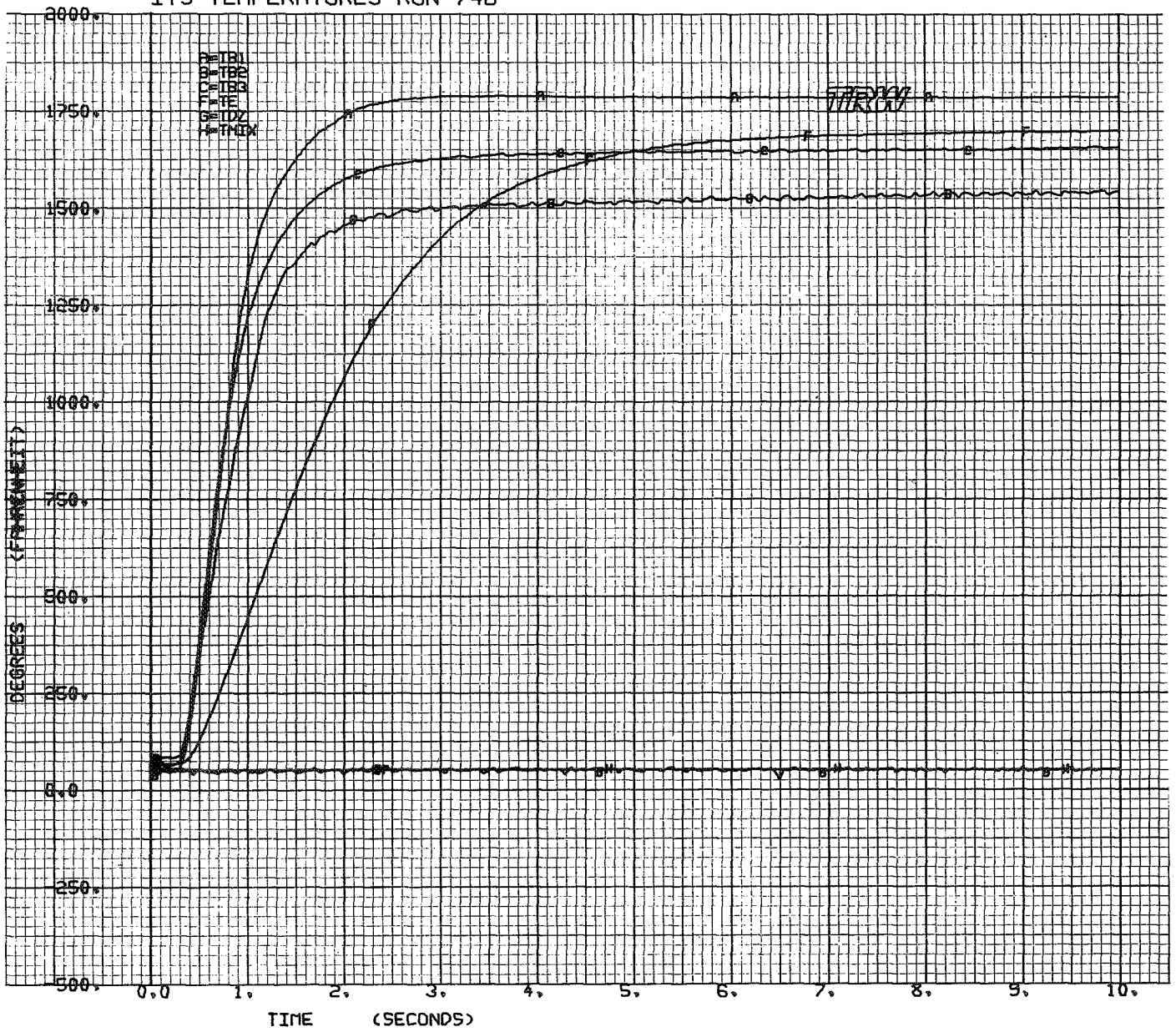
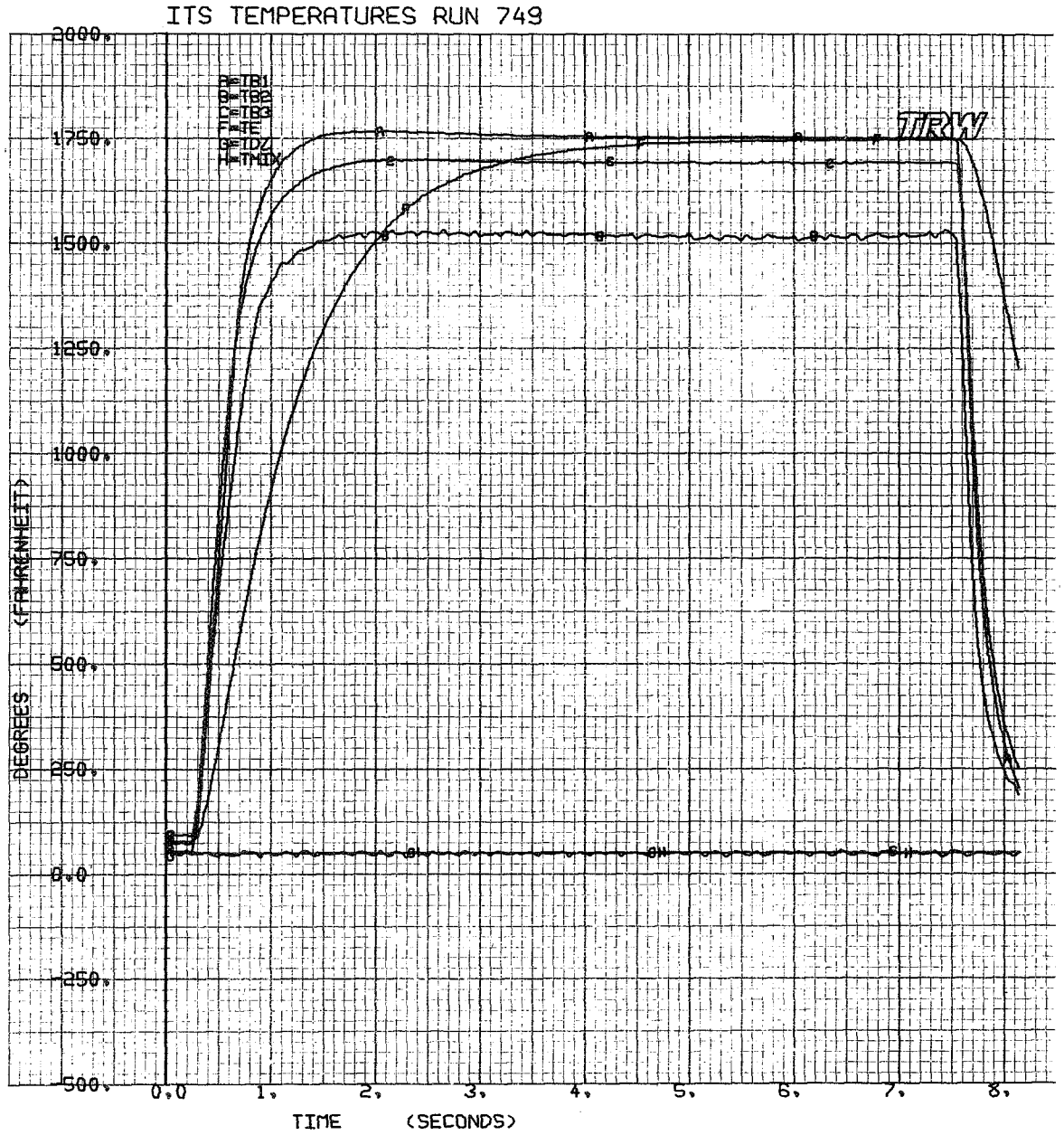
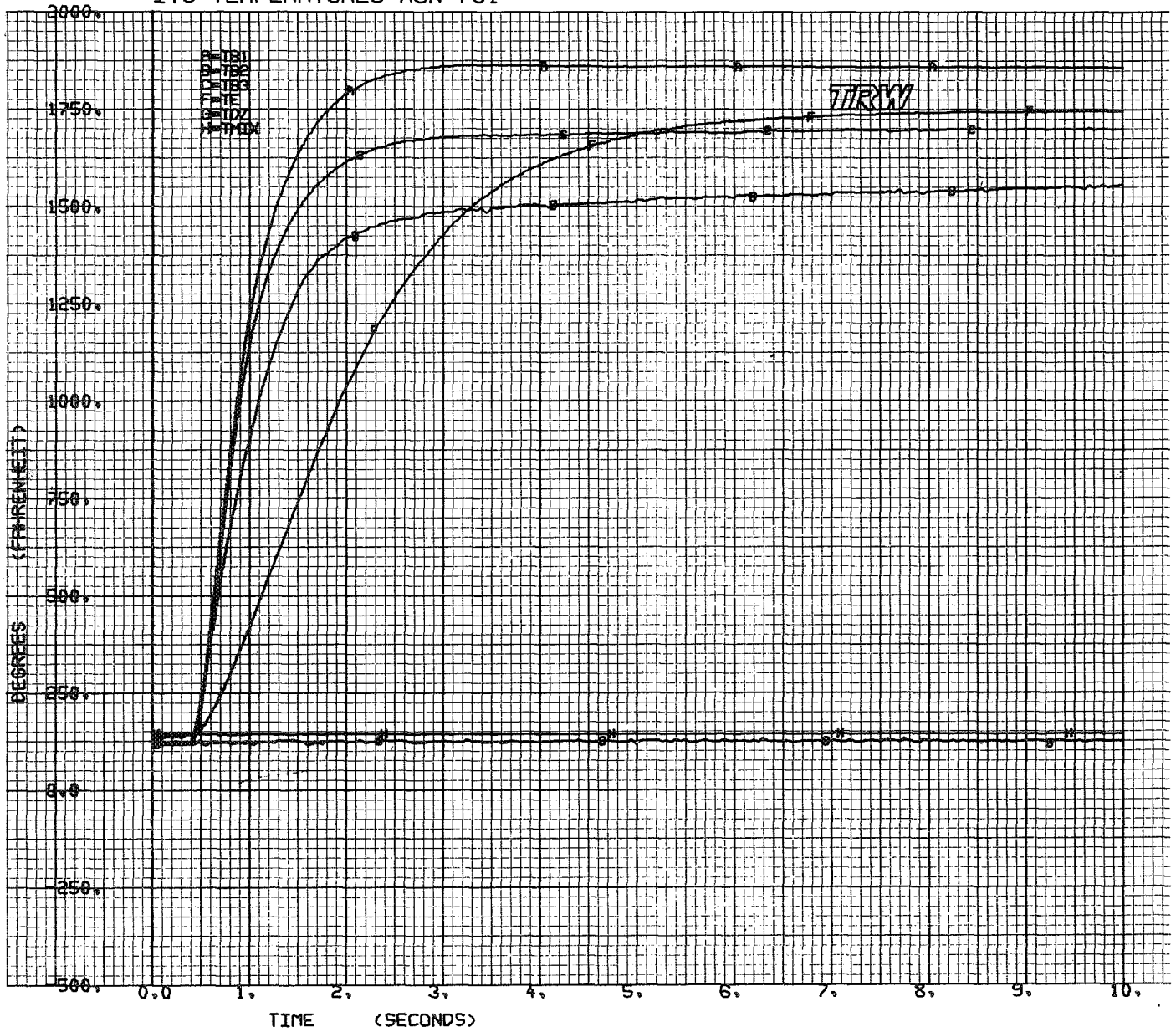



ITS TEMPERATURES RUN 749
A=TB1
B=TB2
C=TB3
E=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
TIME (SECONDS)

ITS TEMPERATURES RUN 751



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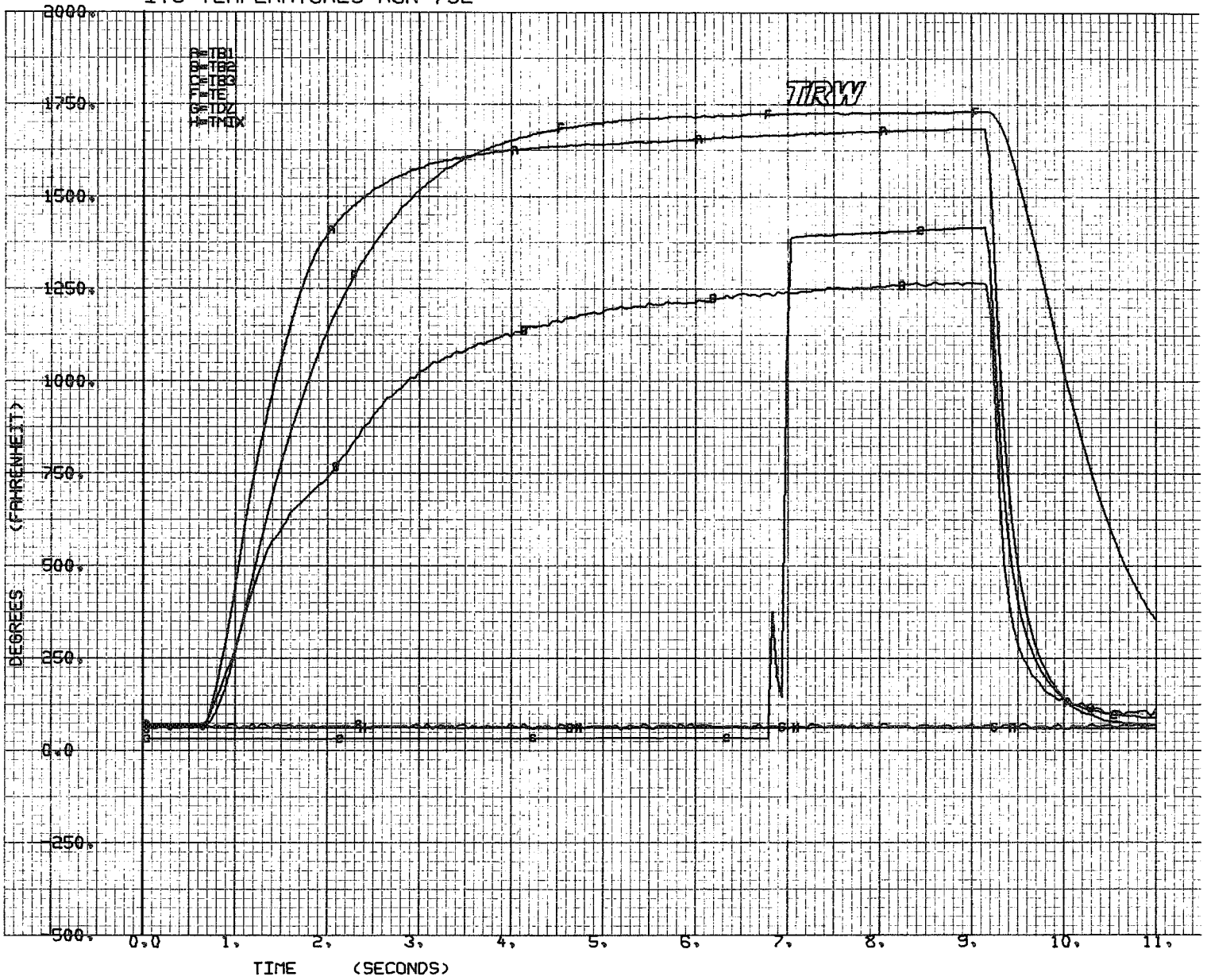

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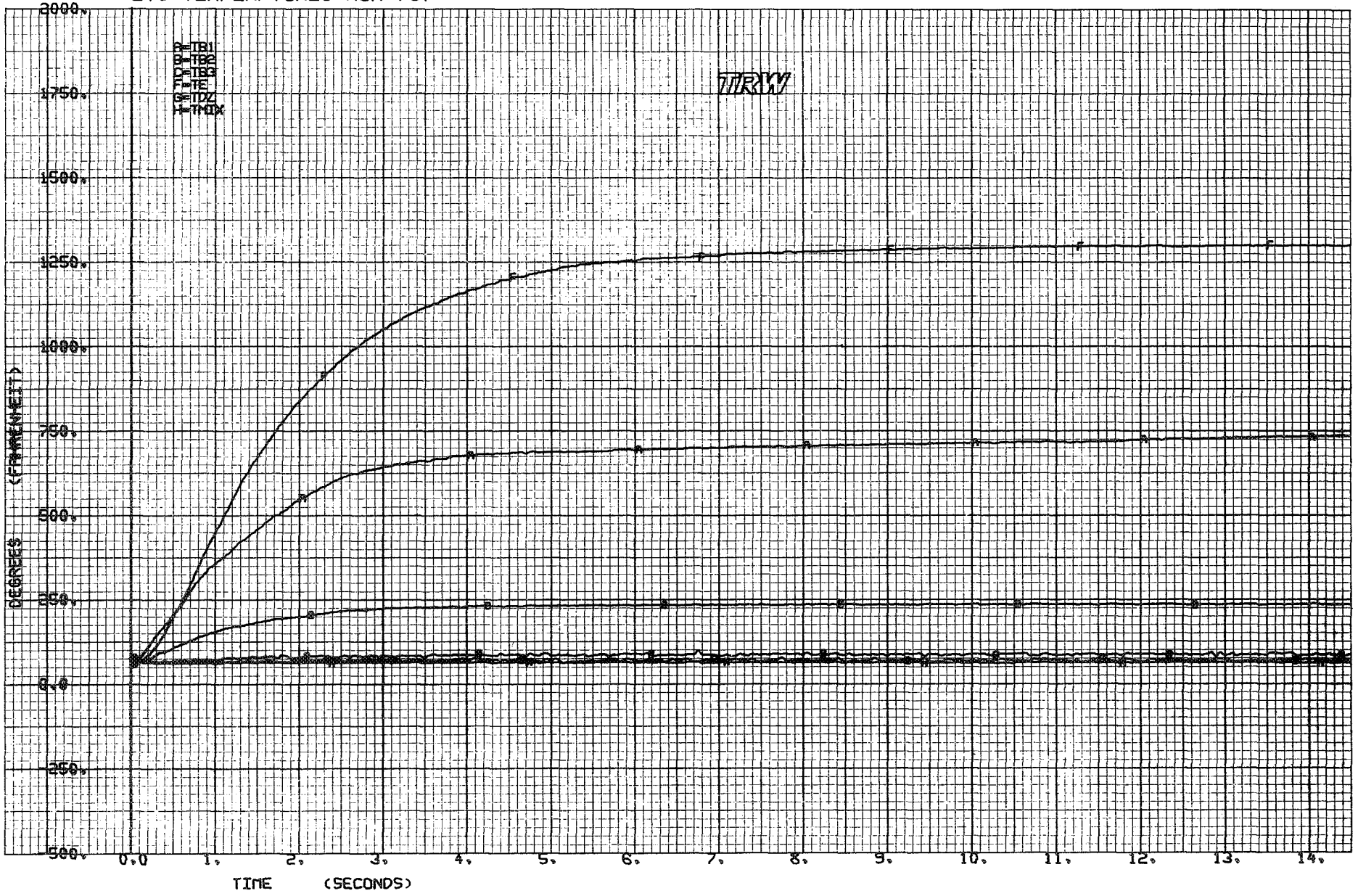

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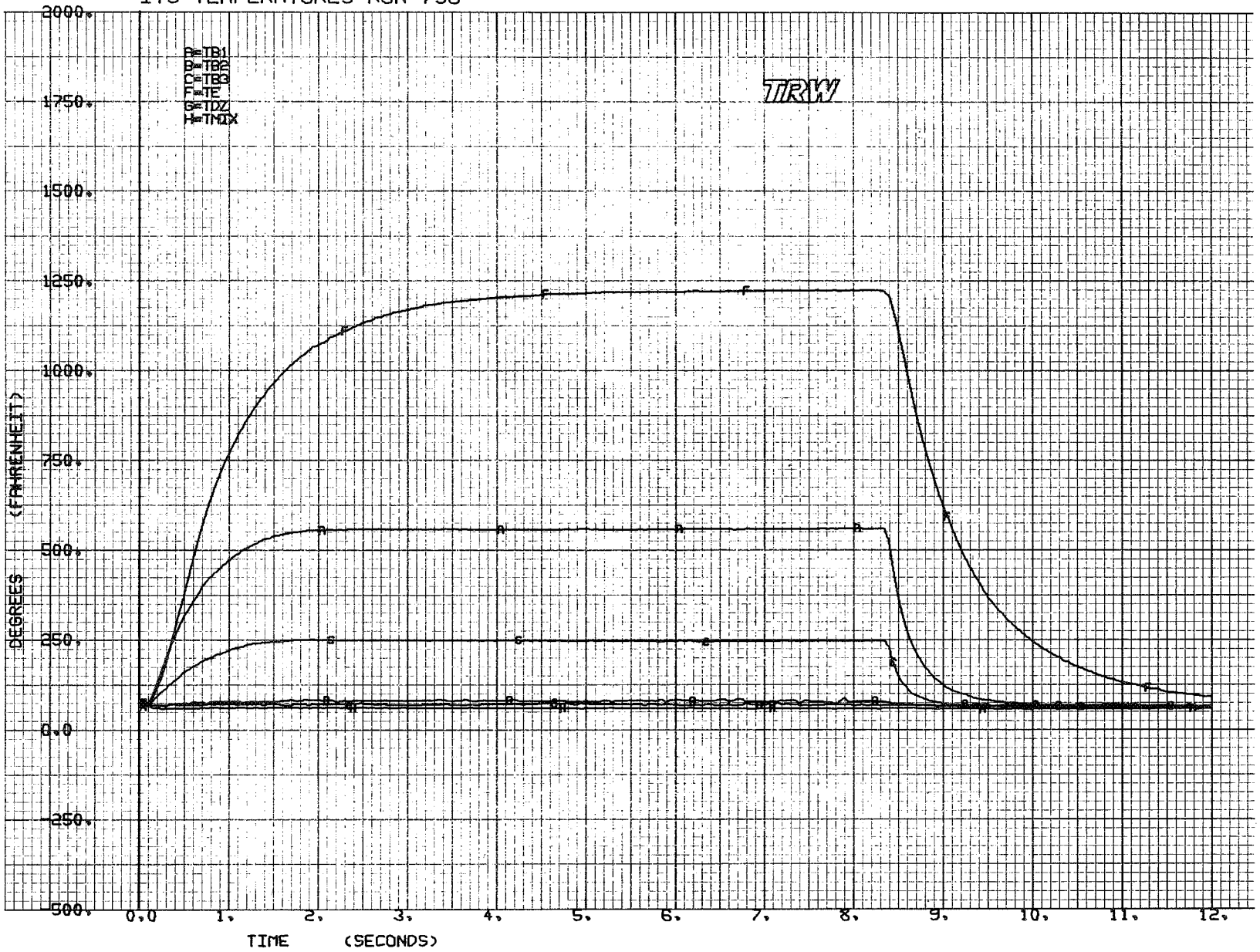
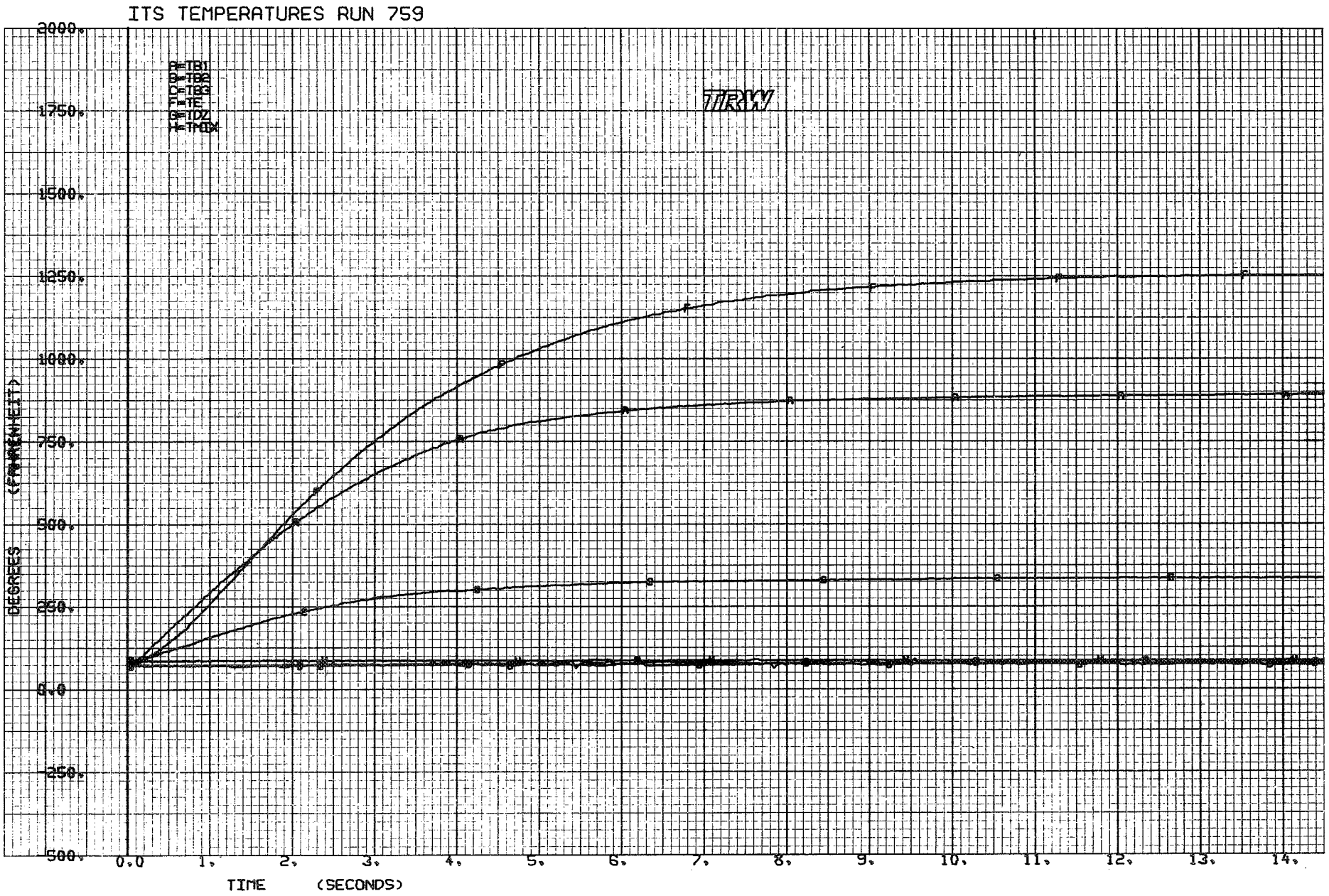
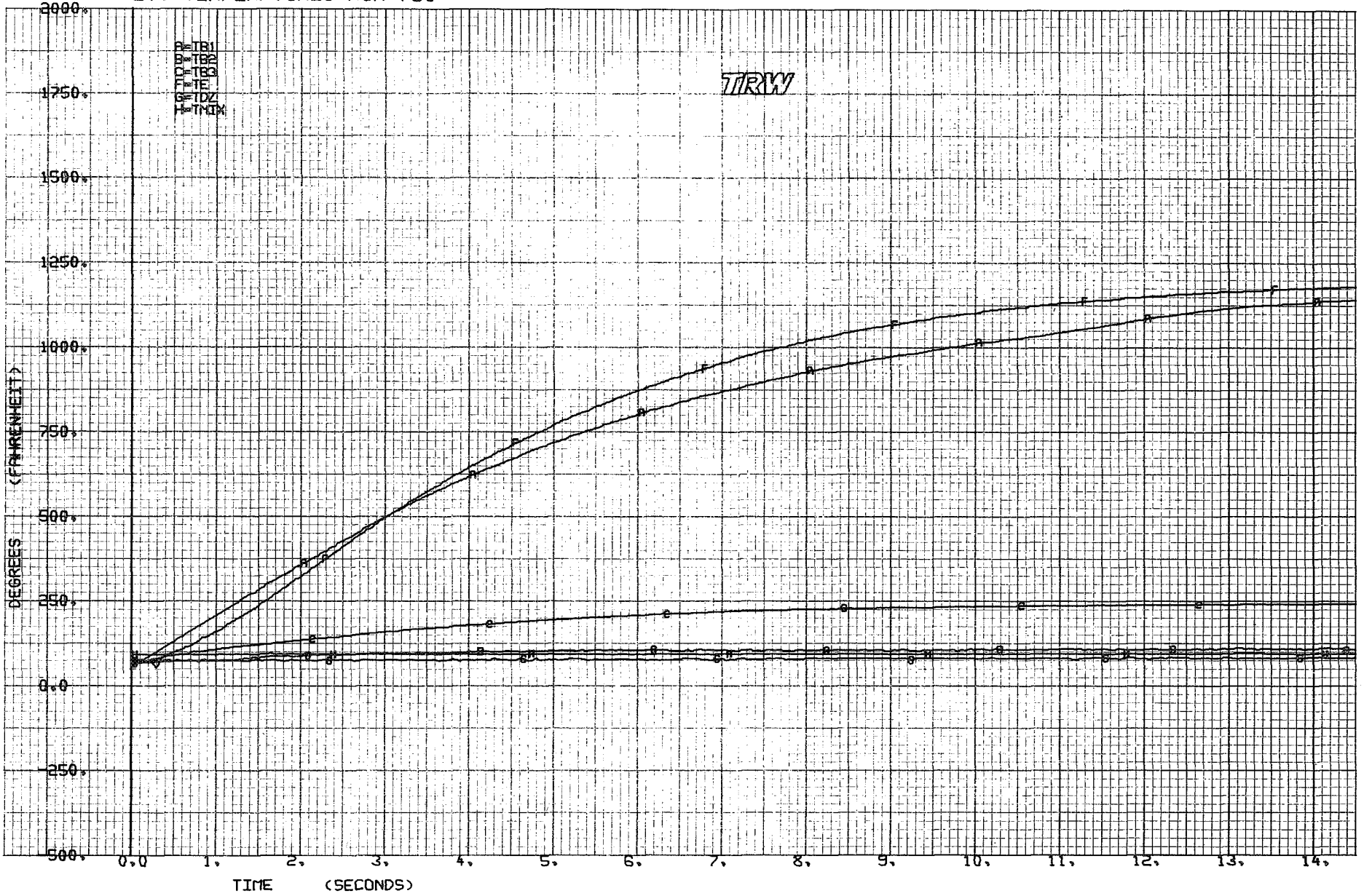



ITS TEMPERATURES RUN 759
A=TB1
B=TB2
C=TB3
F=TF
G=TDZ
H=TMOX
TIRAW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
TIME (SECONDS)

ITS TEMPERATURES RUN 760



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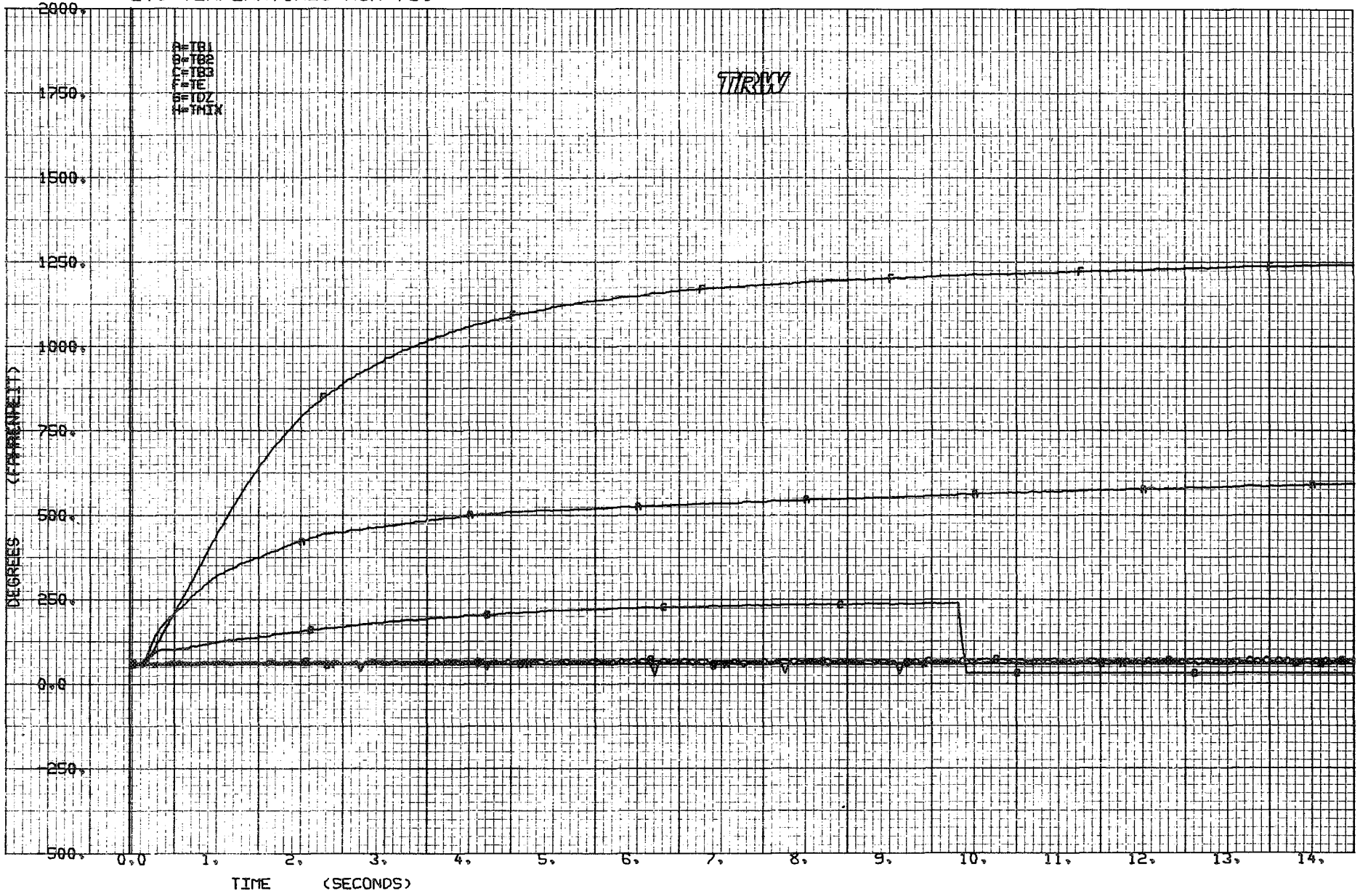

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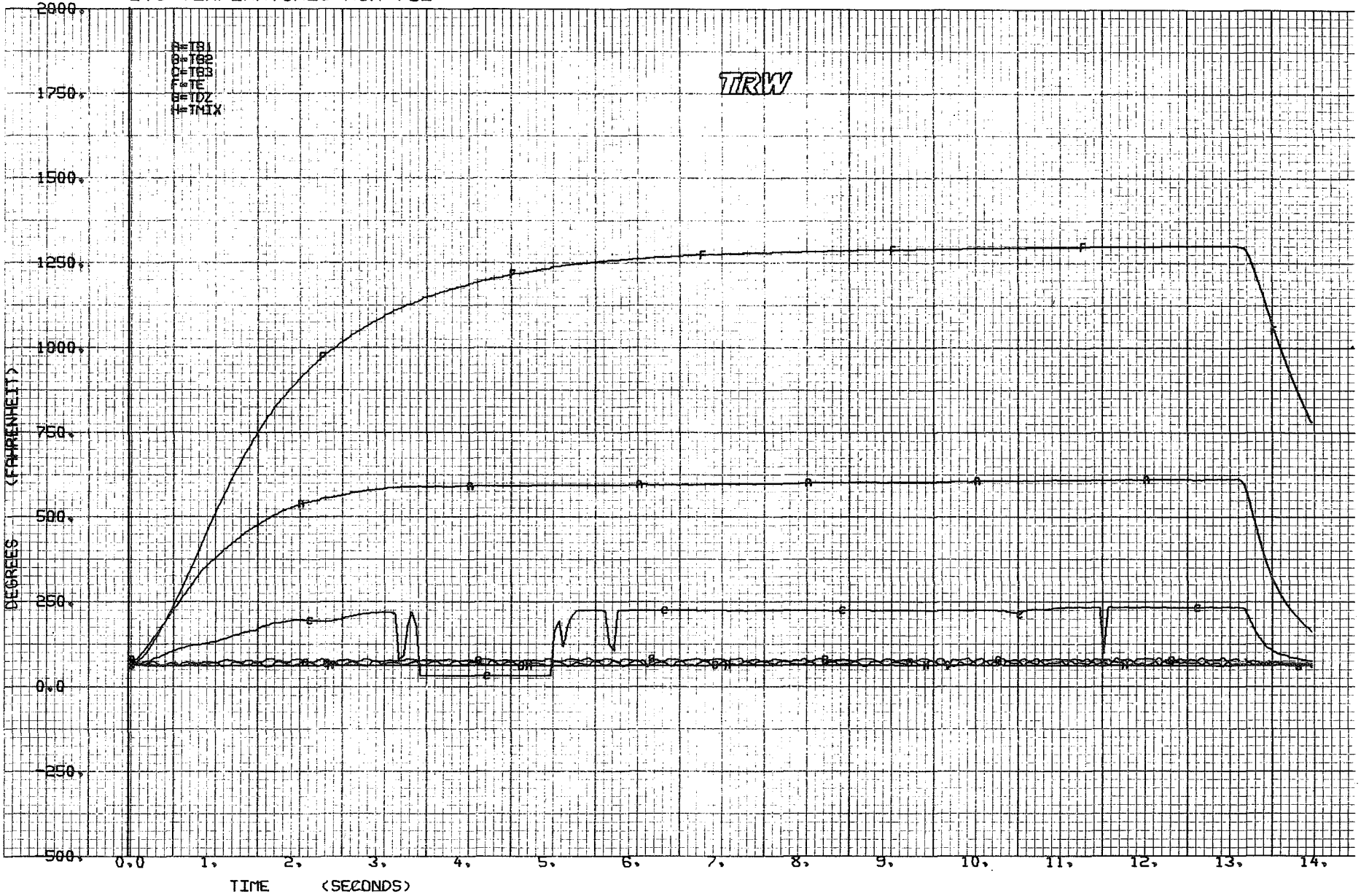

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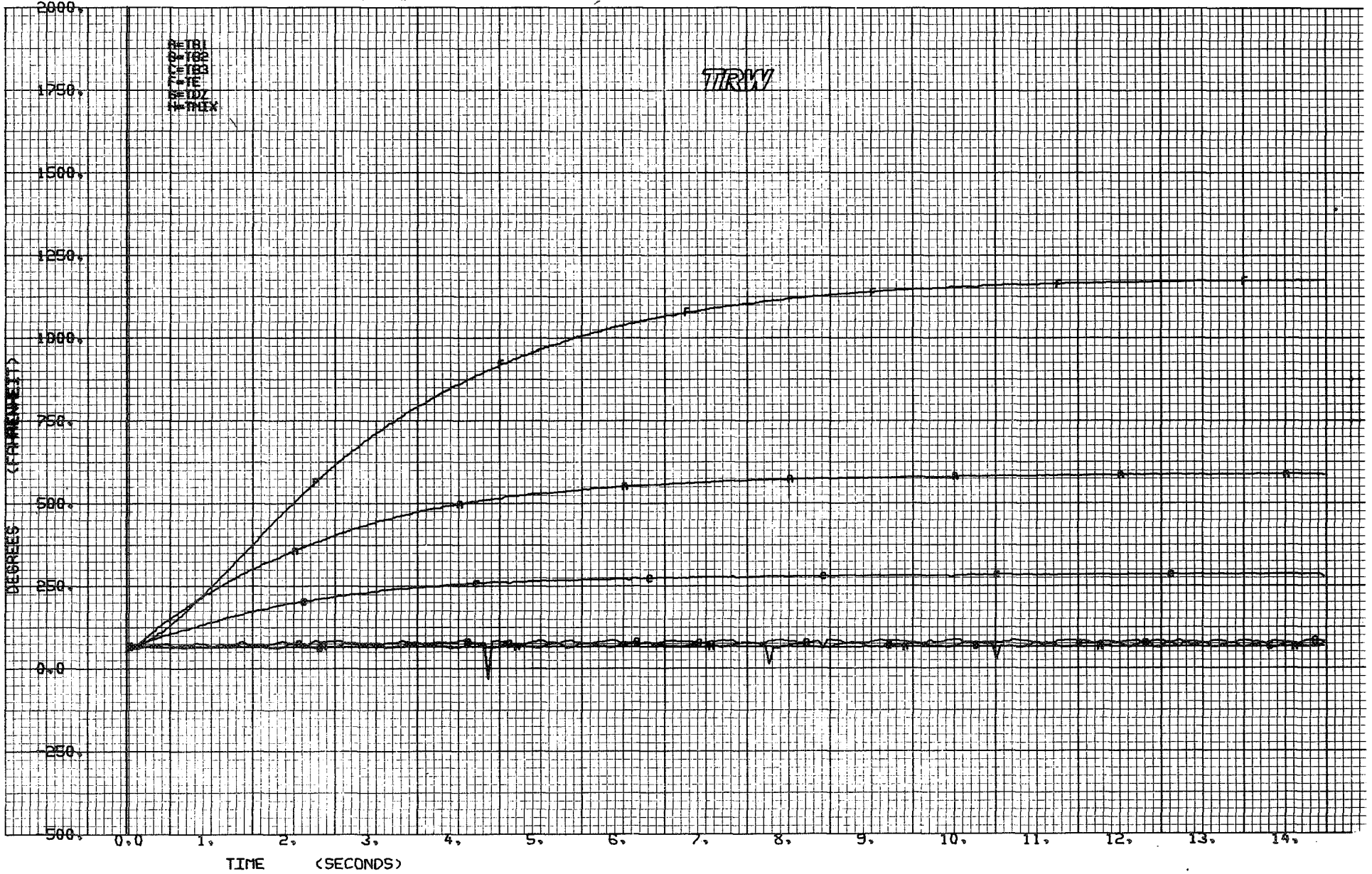

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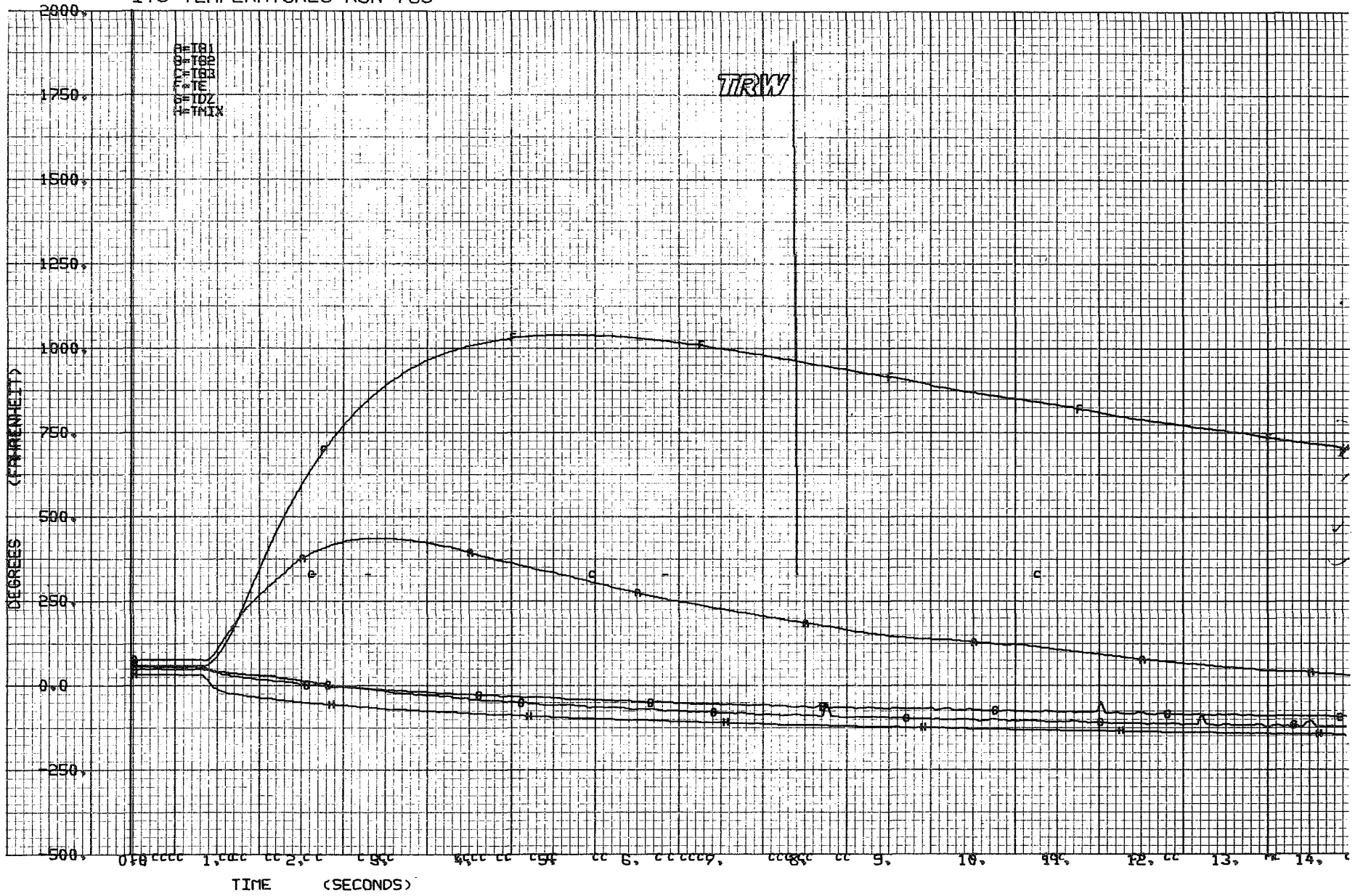

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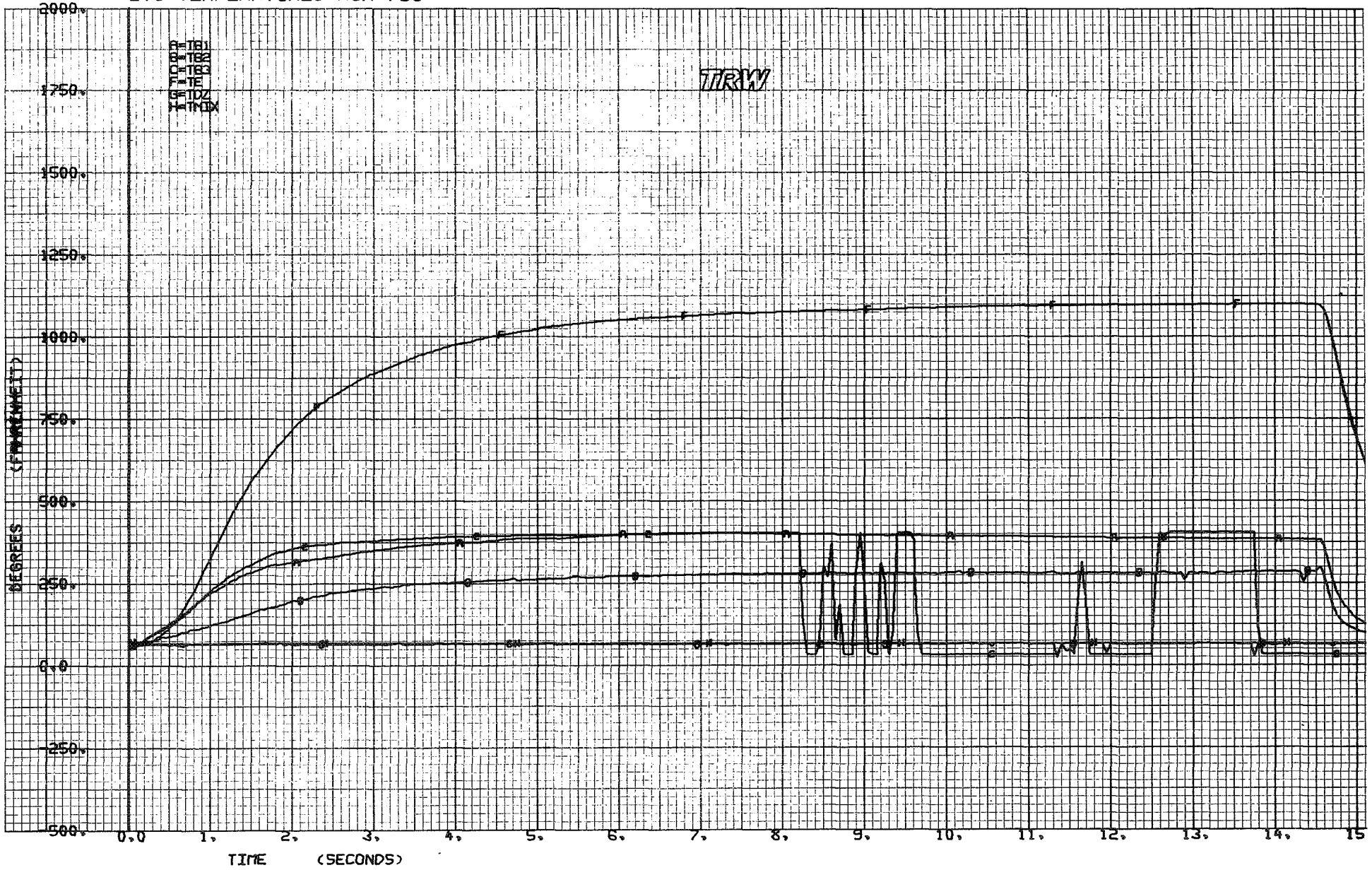

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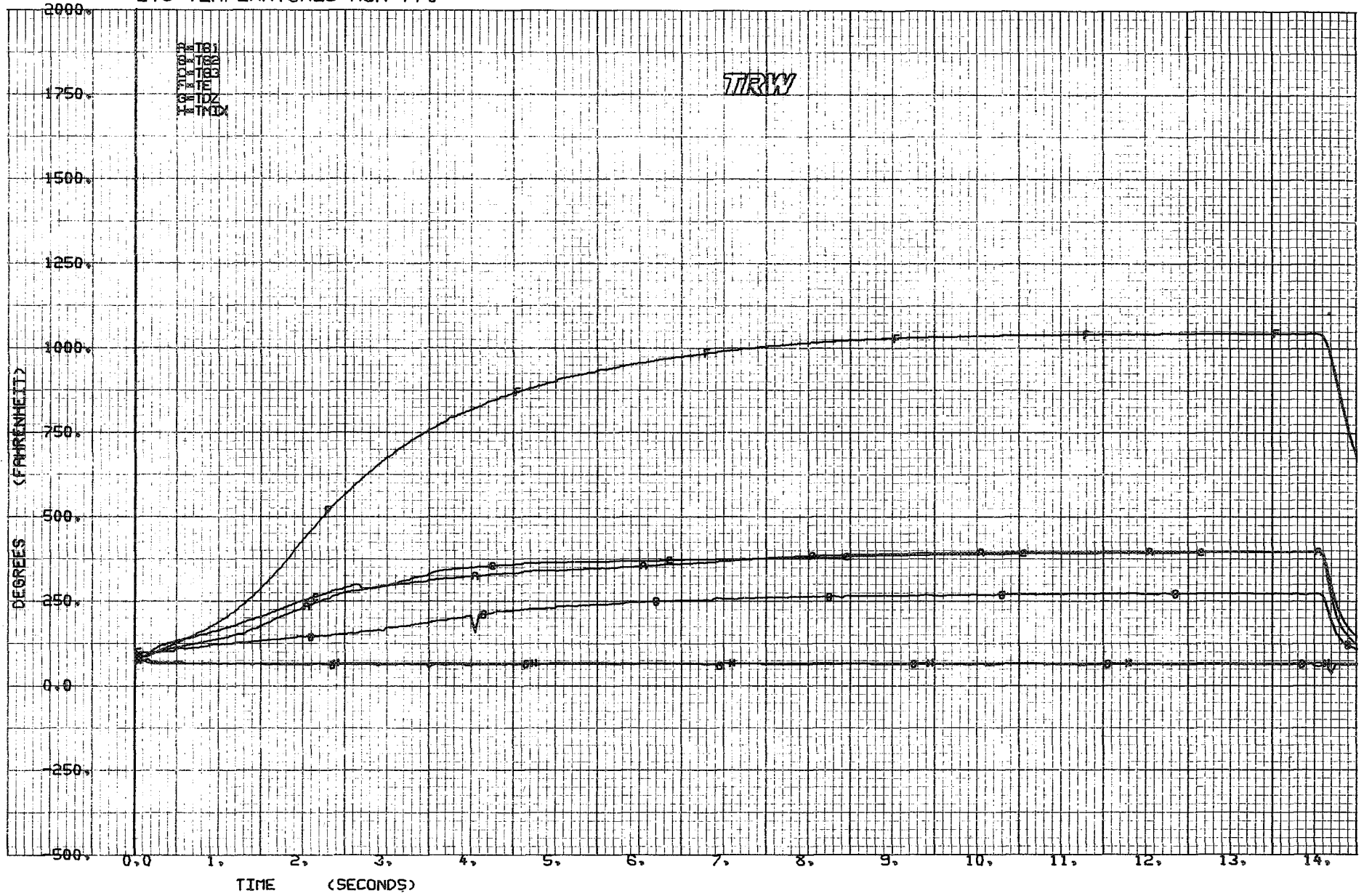

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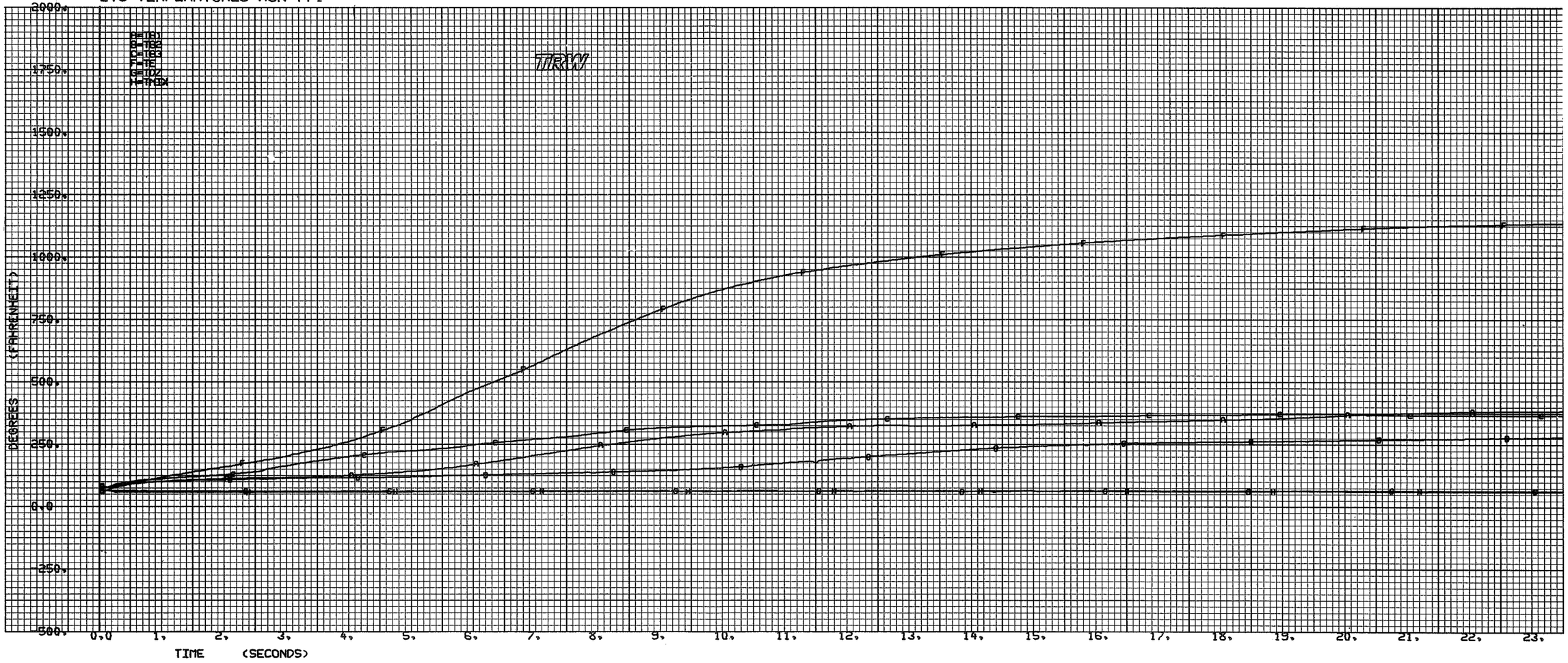
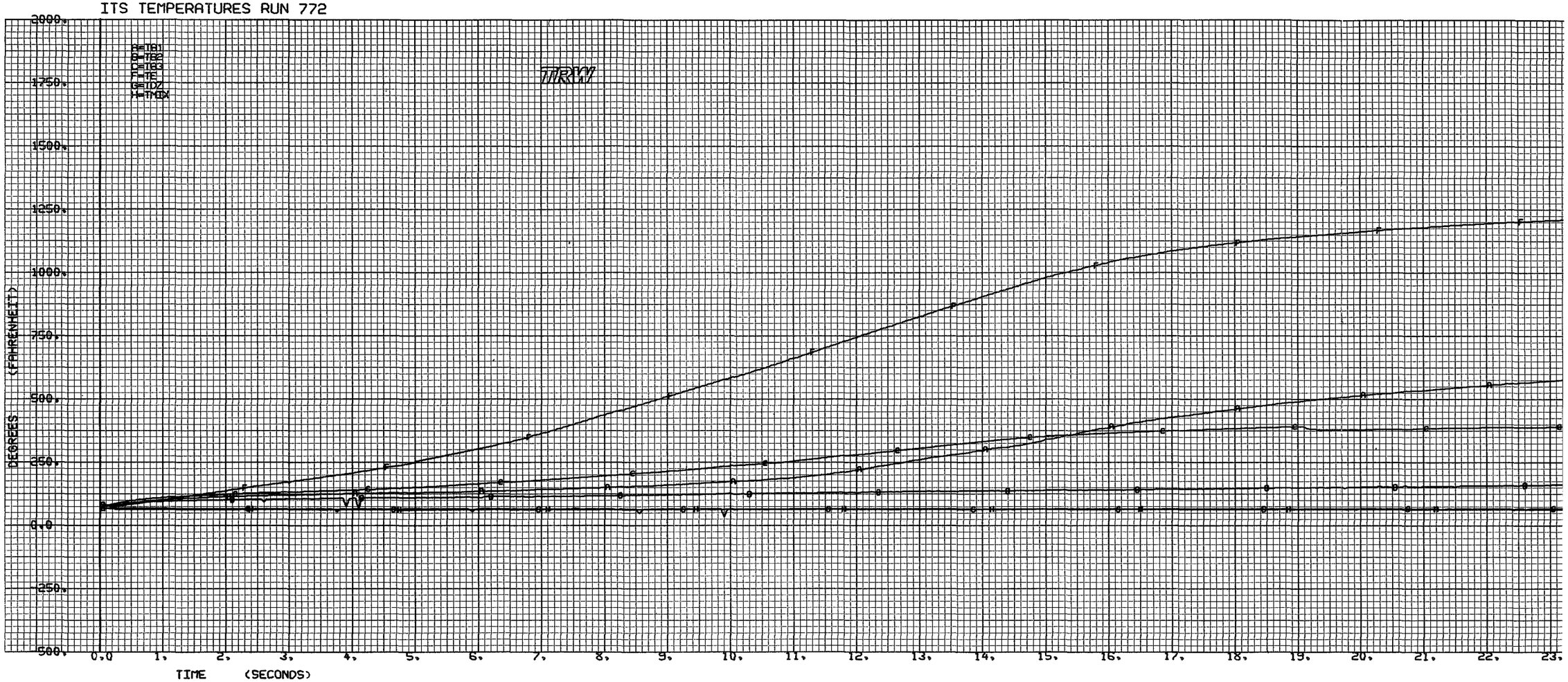
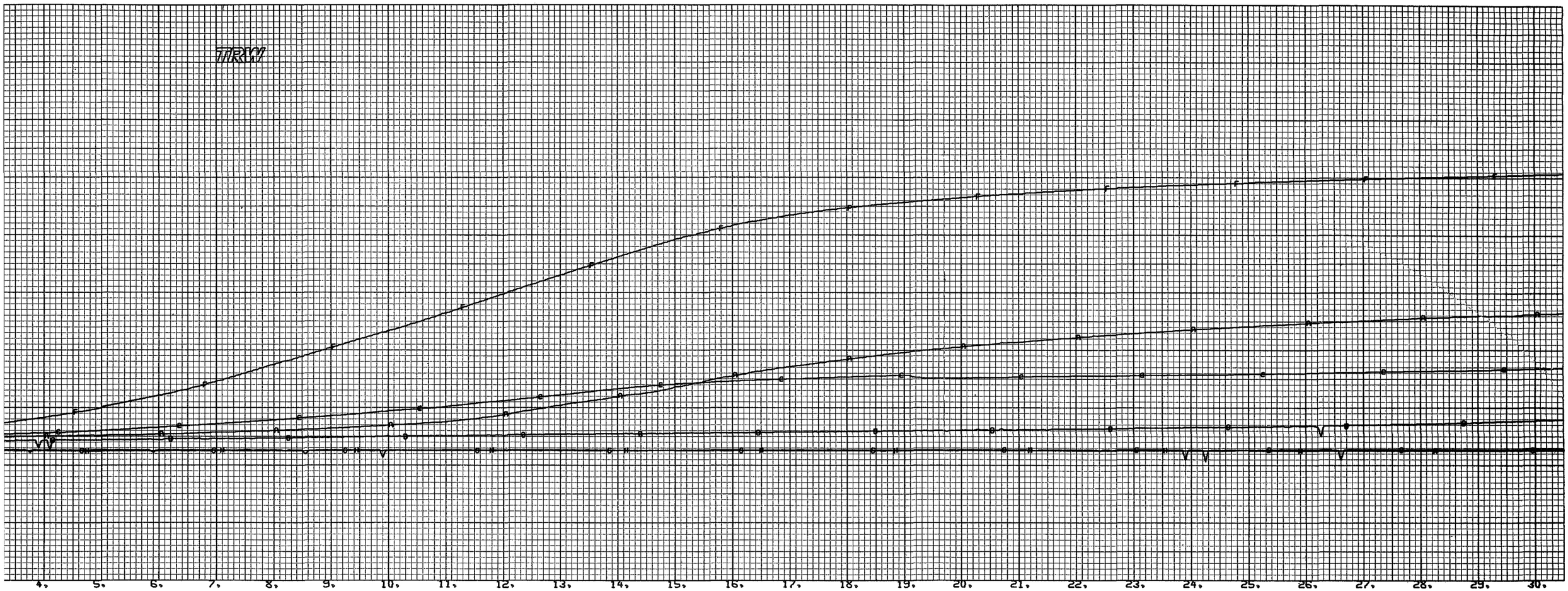
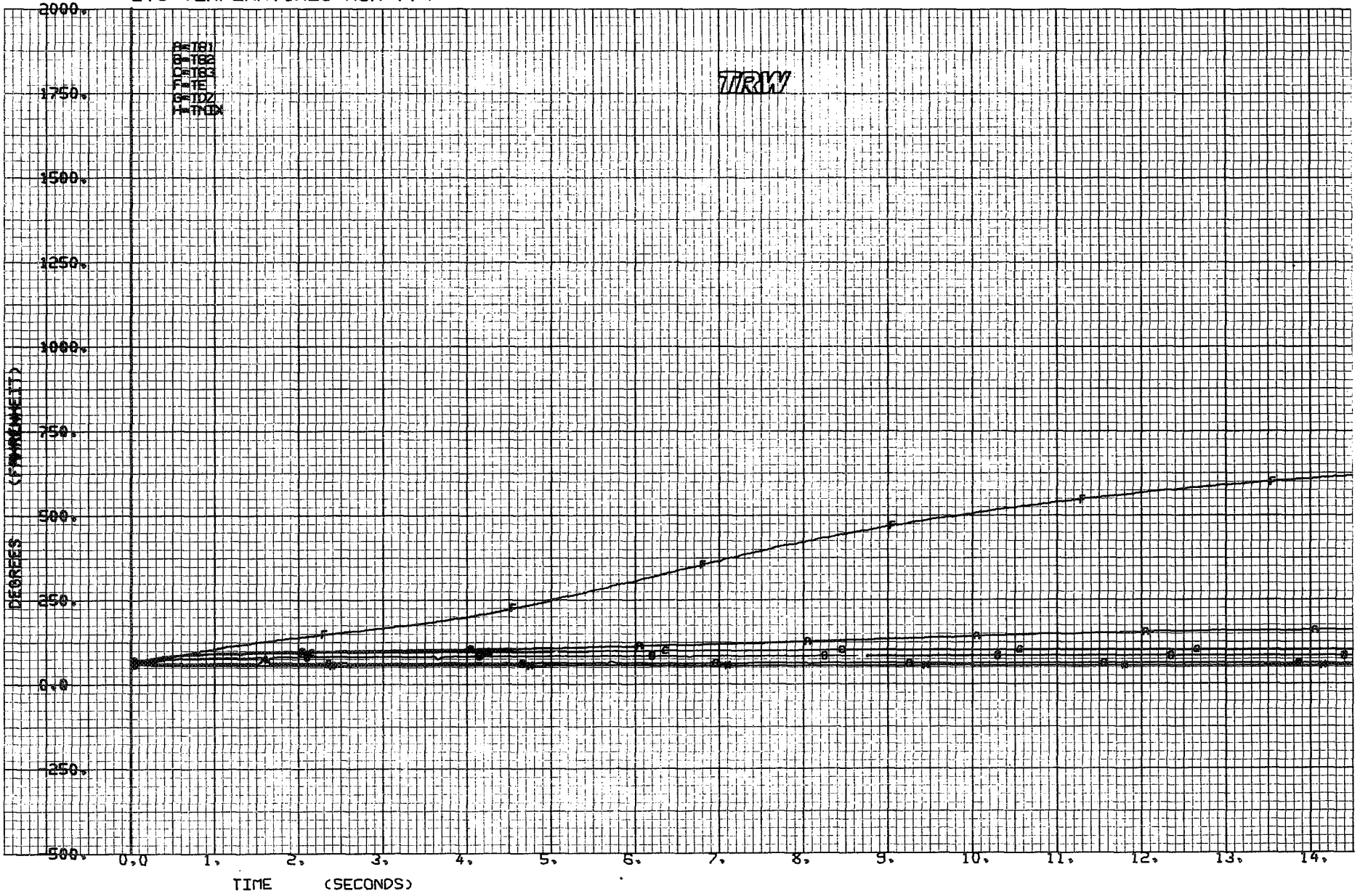


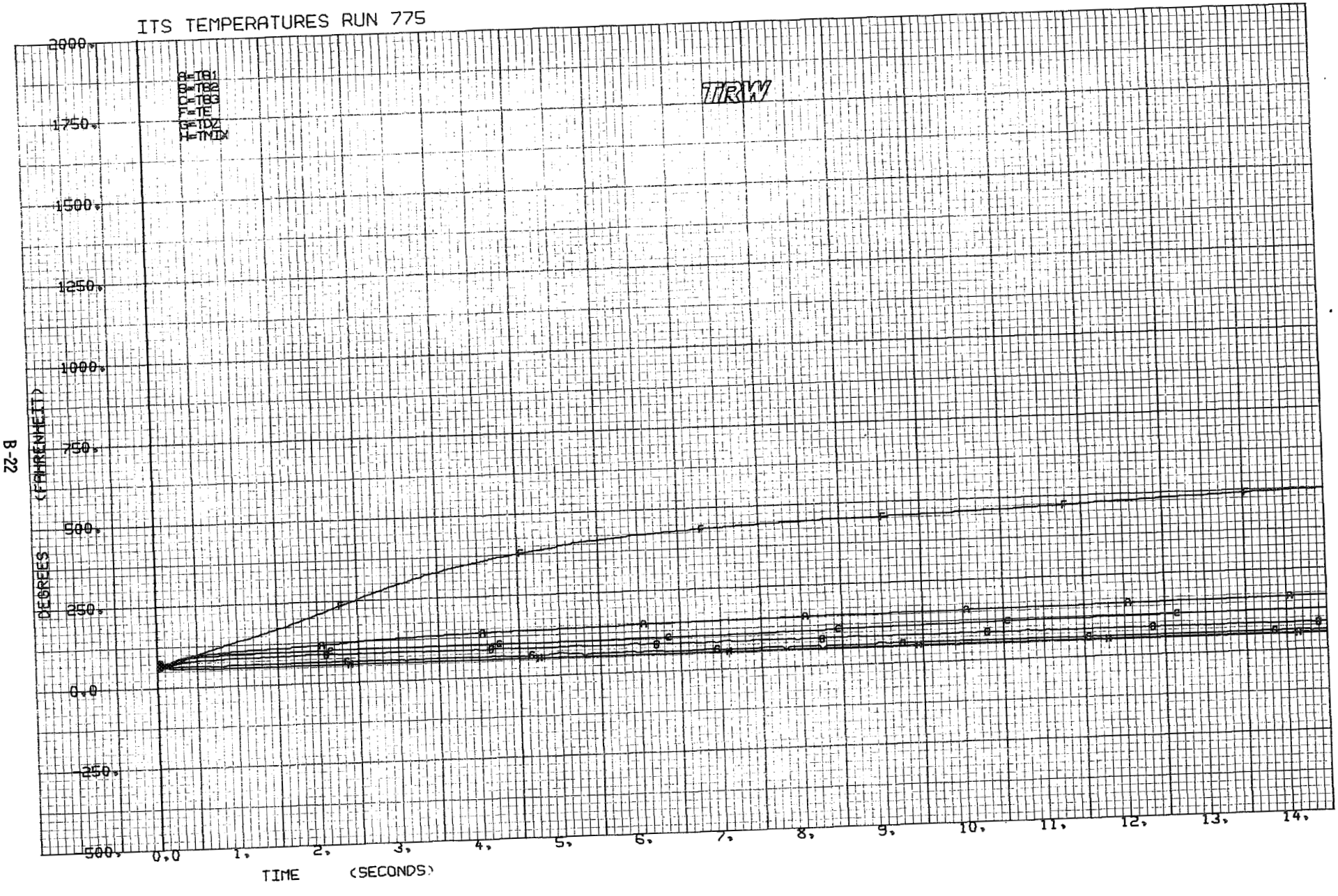
ITS TEMPERATURES RUN 772



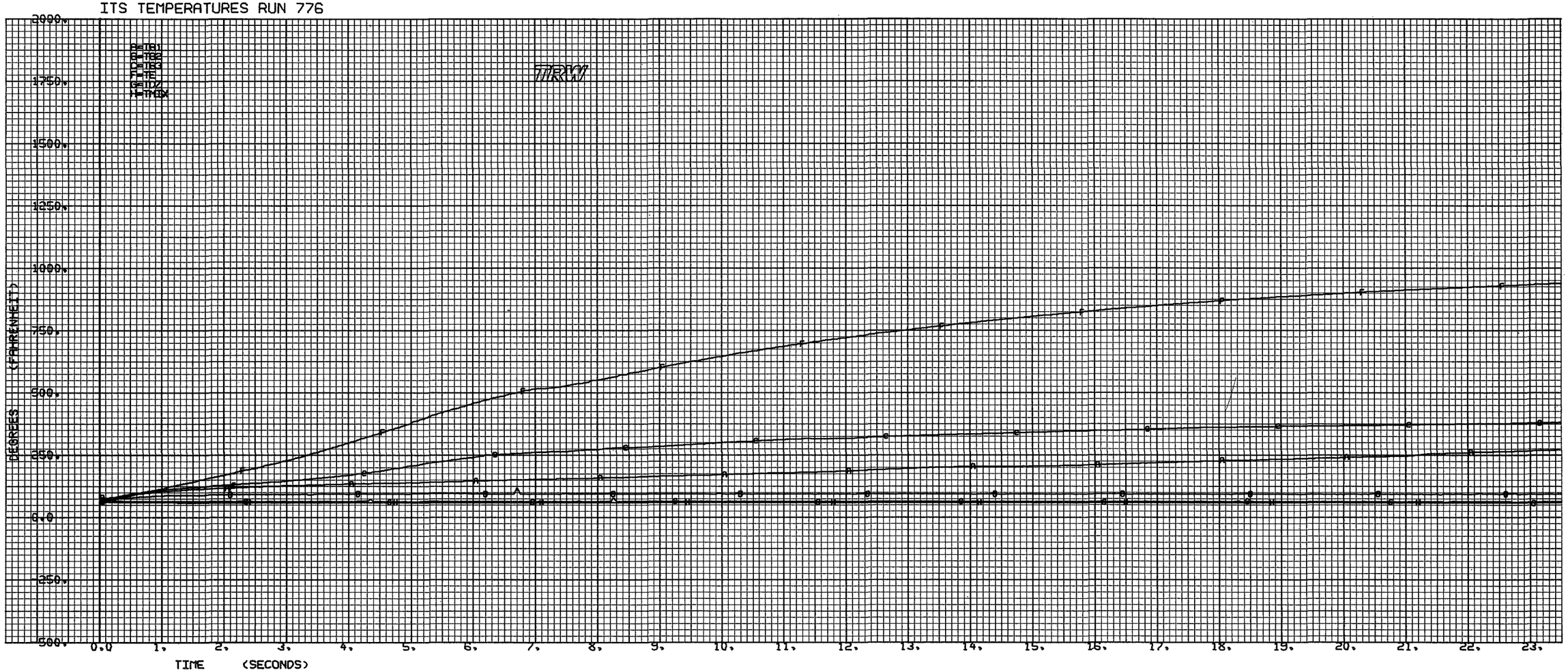

TIRCV
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
15.
16.
17.
18.
19.
20.
21.
22.
23.
24.
25.
26.
27.
28.
29.
30.

ITS TEMPERATURES RUN 774

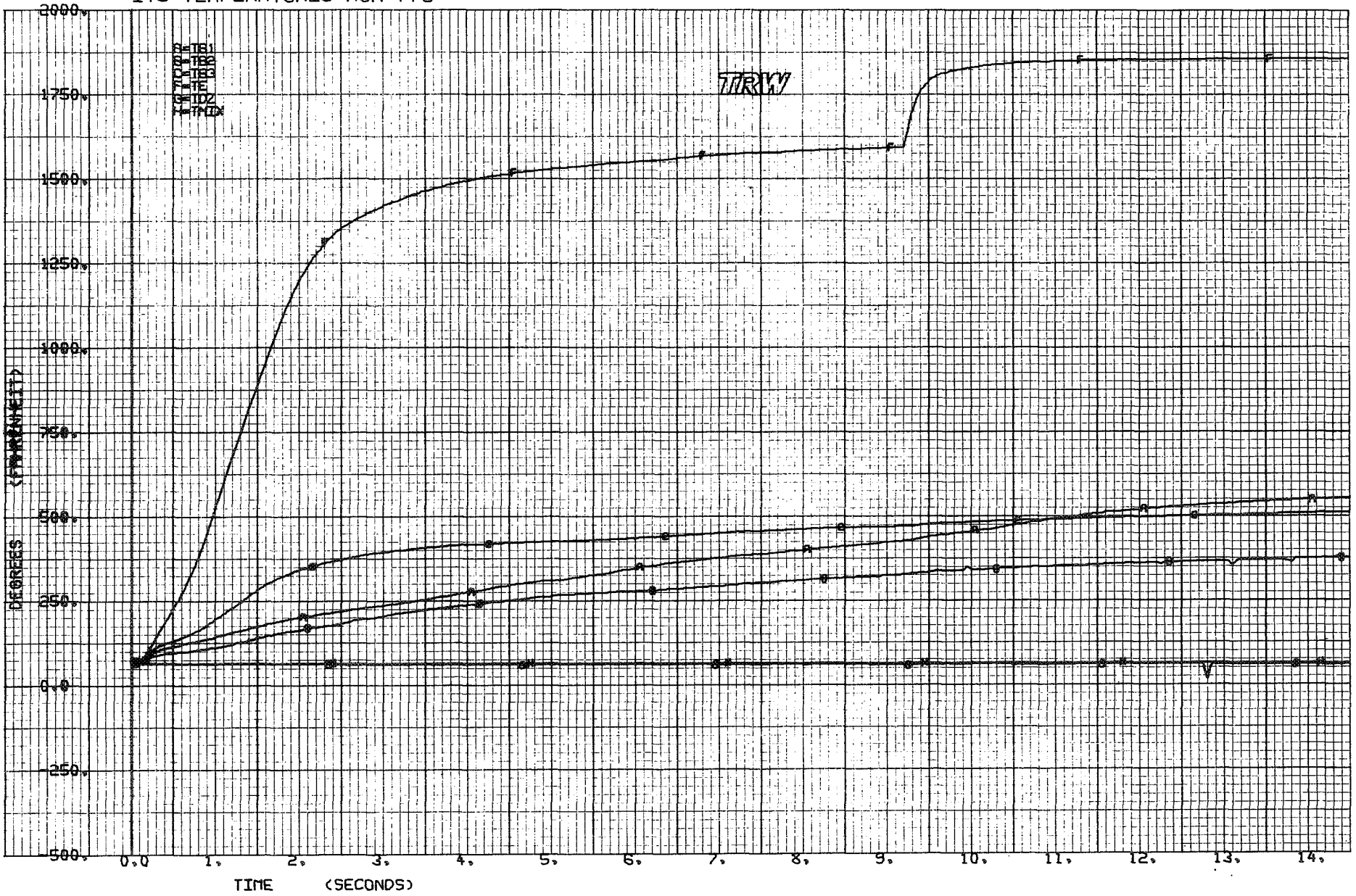



ITS TEMPERATURES RUN 775
A=TB1
B=TB2
C=TBG
F=TE
G=TDZ
H=TMIX
TRW
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
TIME (SECONDS)

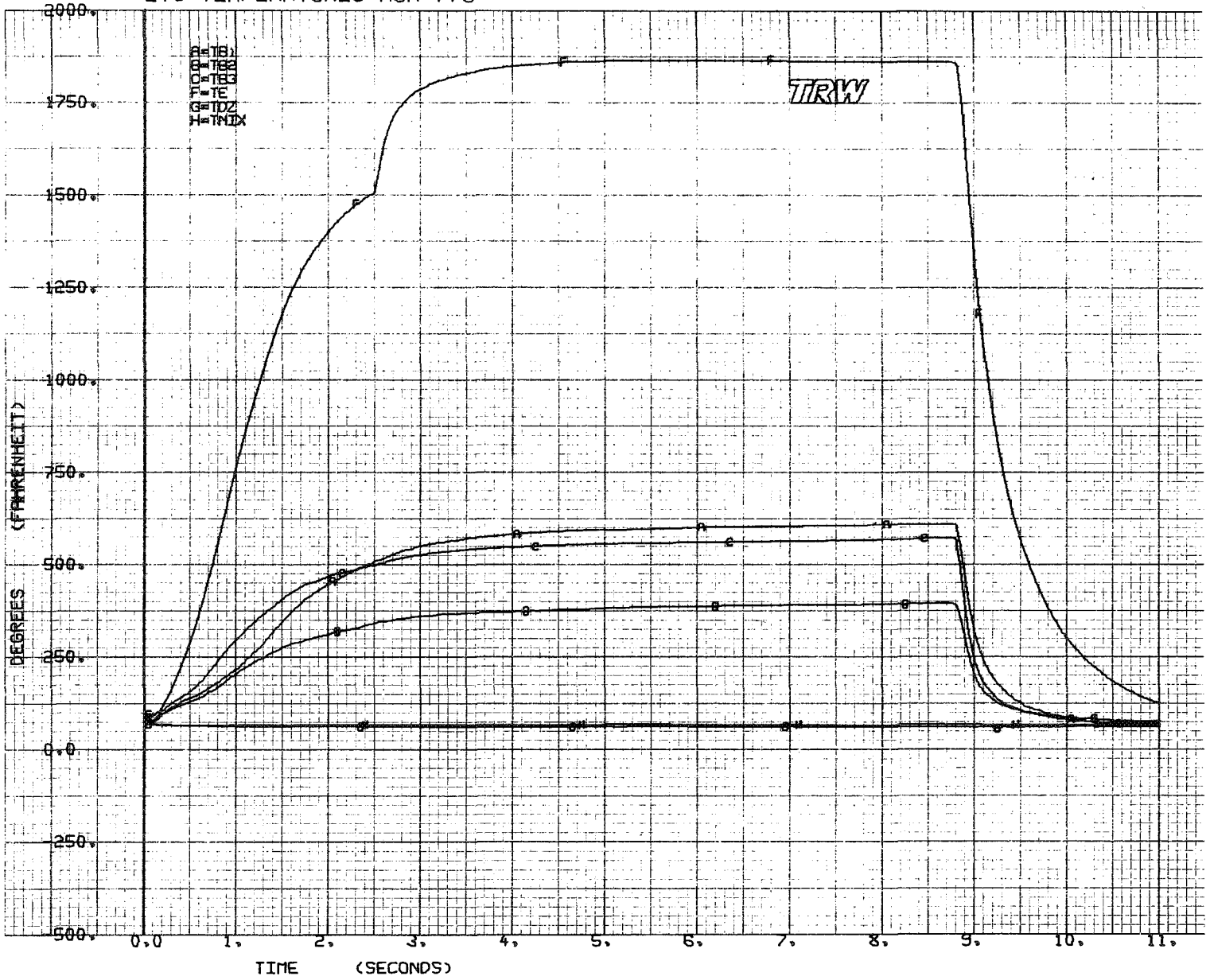
ITS TEMPERATURES RUN 776

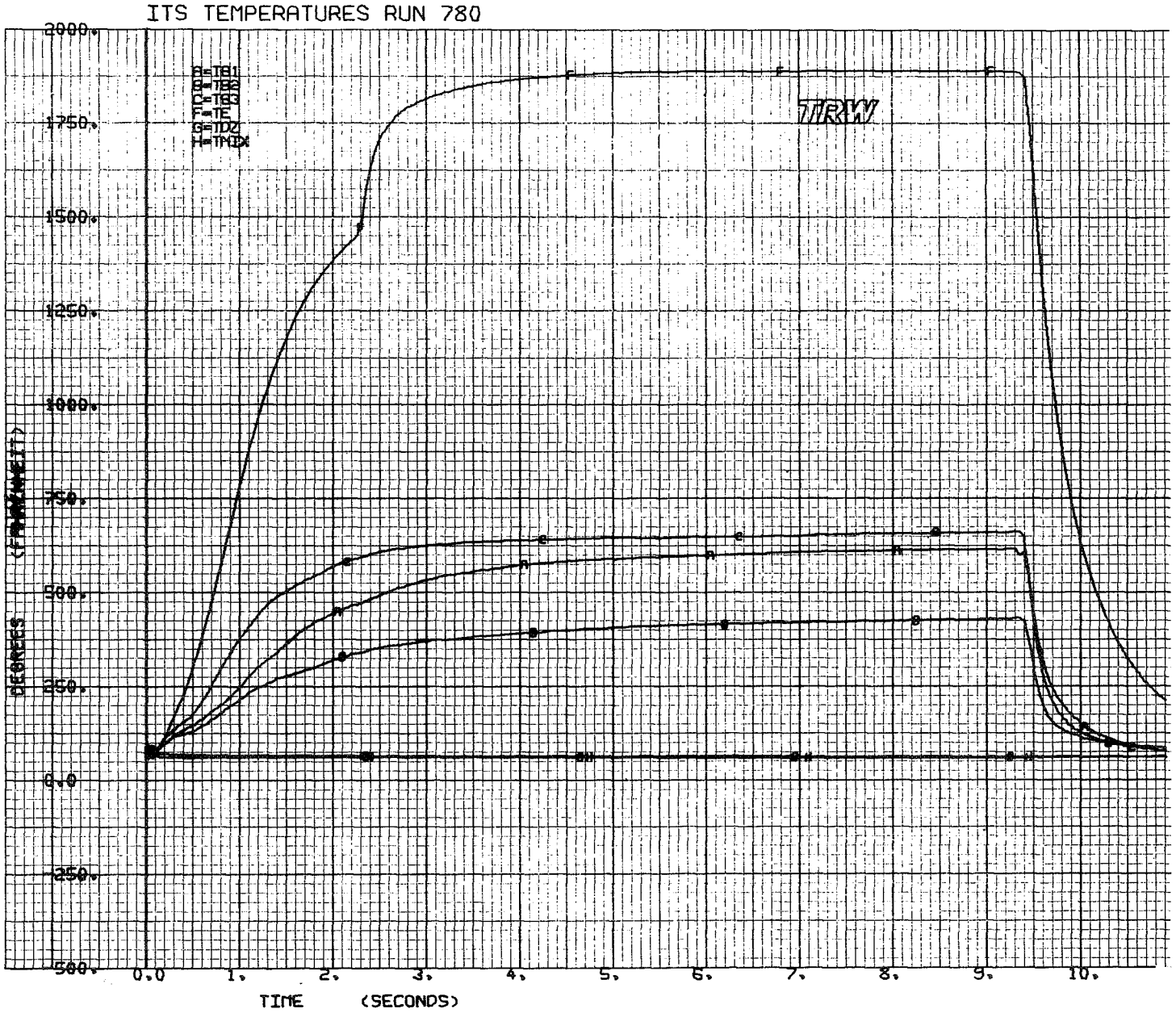


ITS TEMPERATURES RUN 778

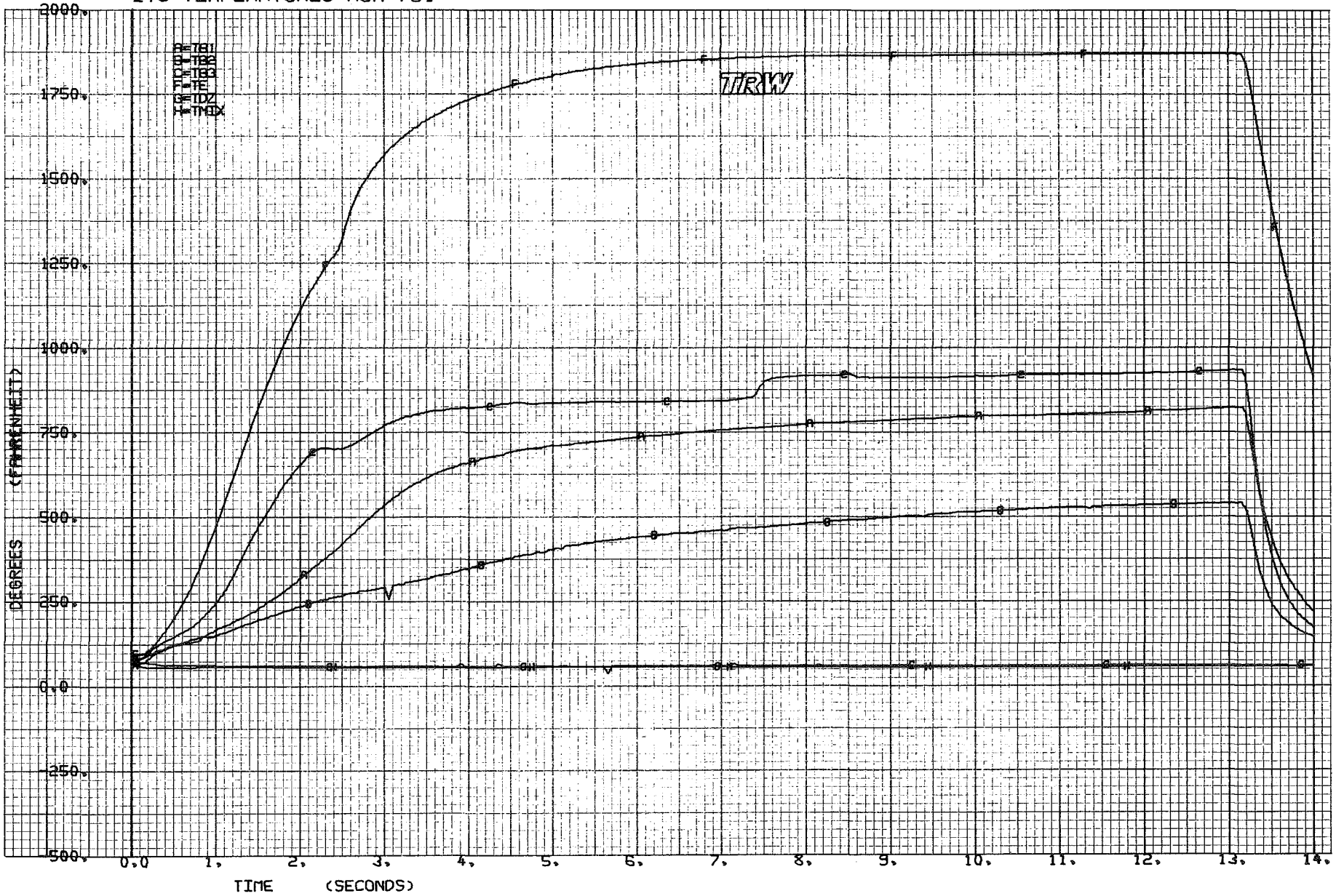


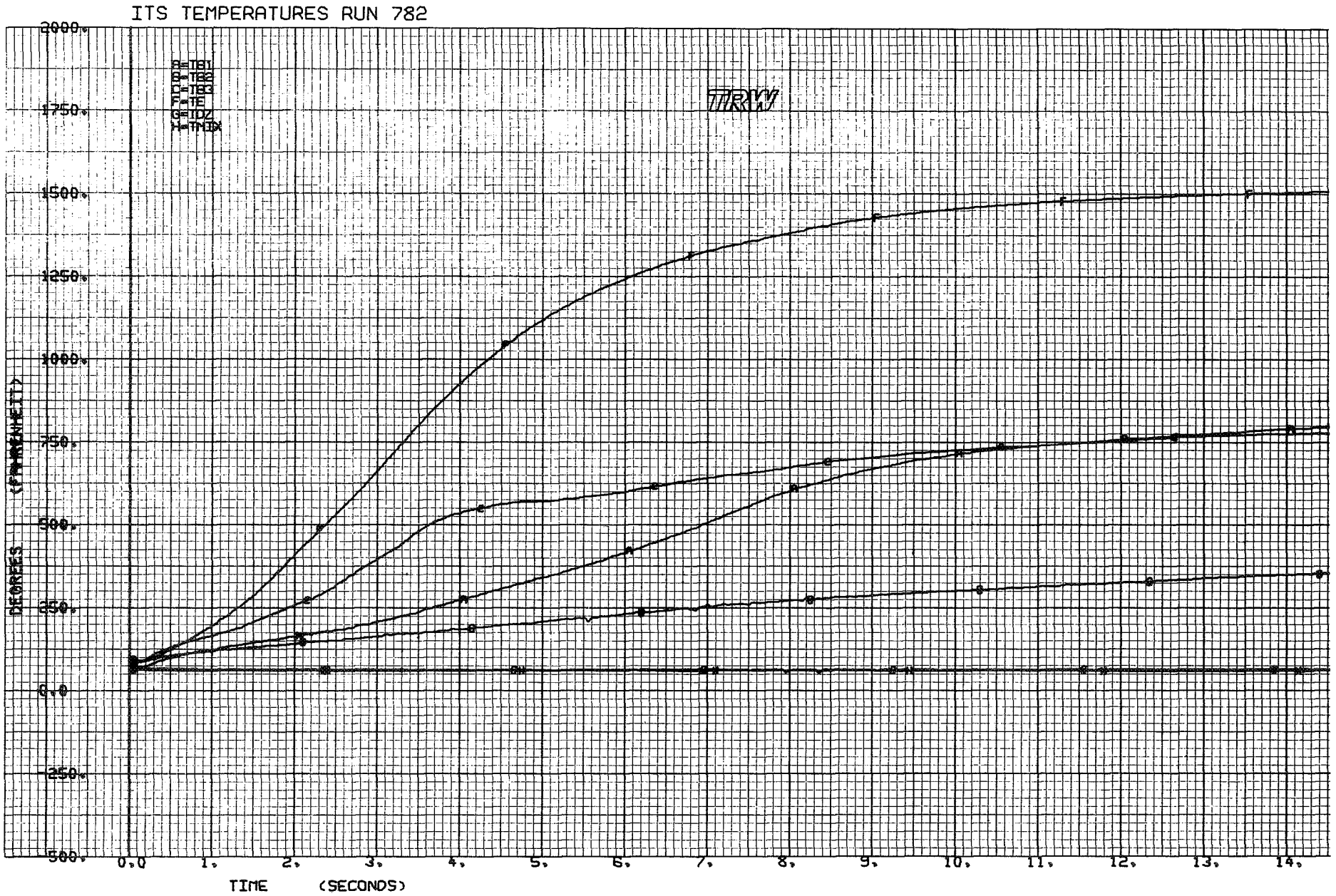
ITS TEMPERATURES RUN 779

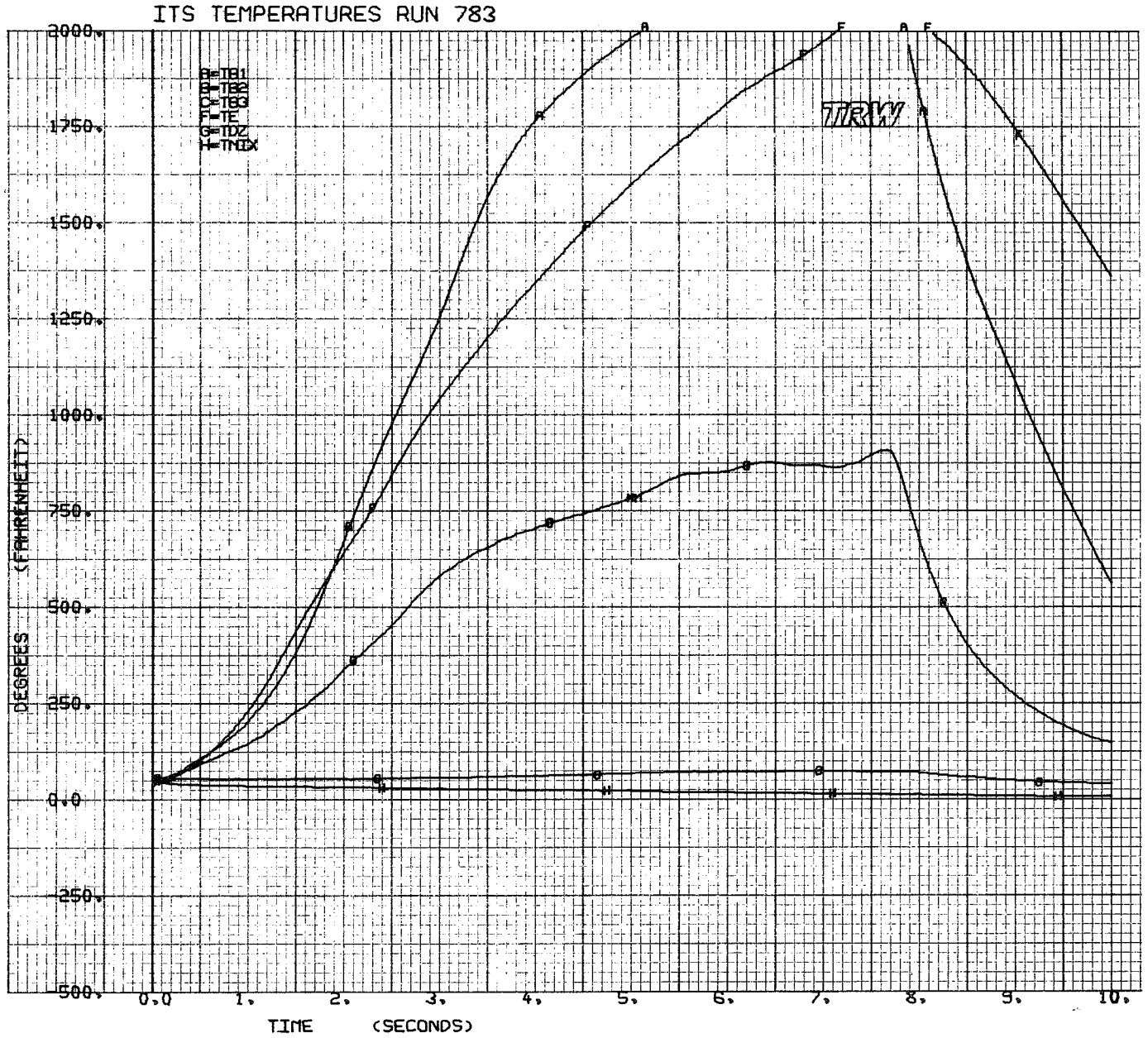



ITS TEMPERATURES RUN 780
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
TIME (SECONDS)

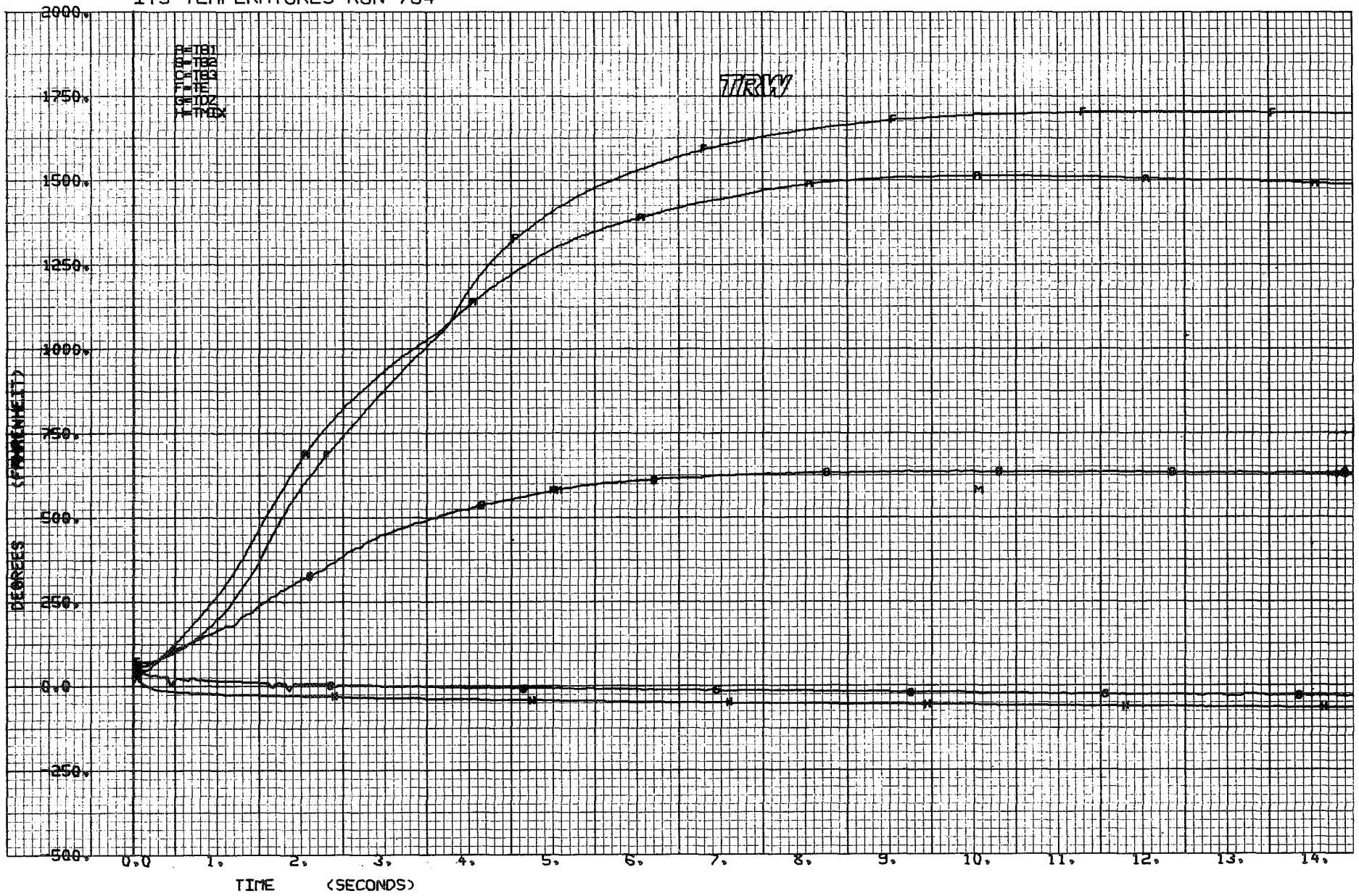
ITS TEMPERATURES RUN 781

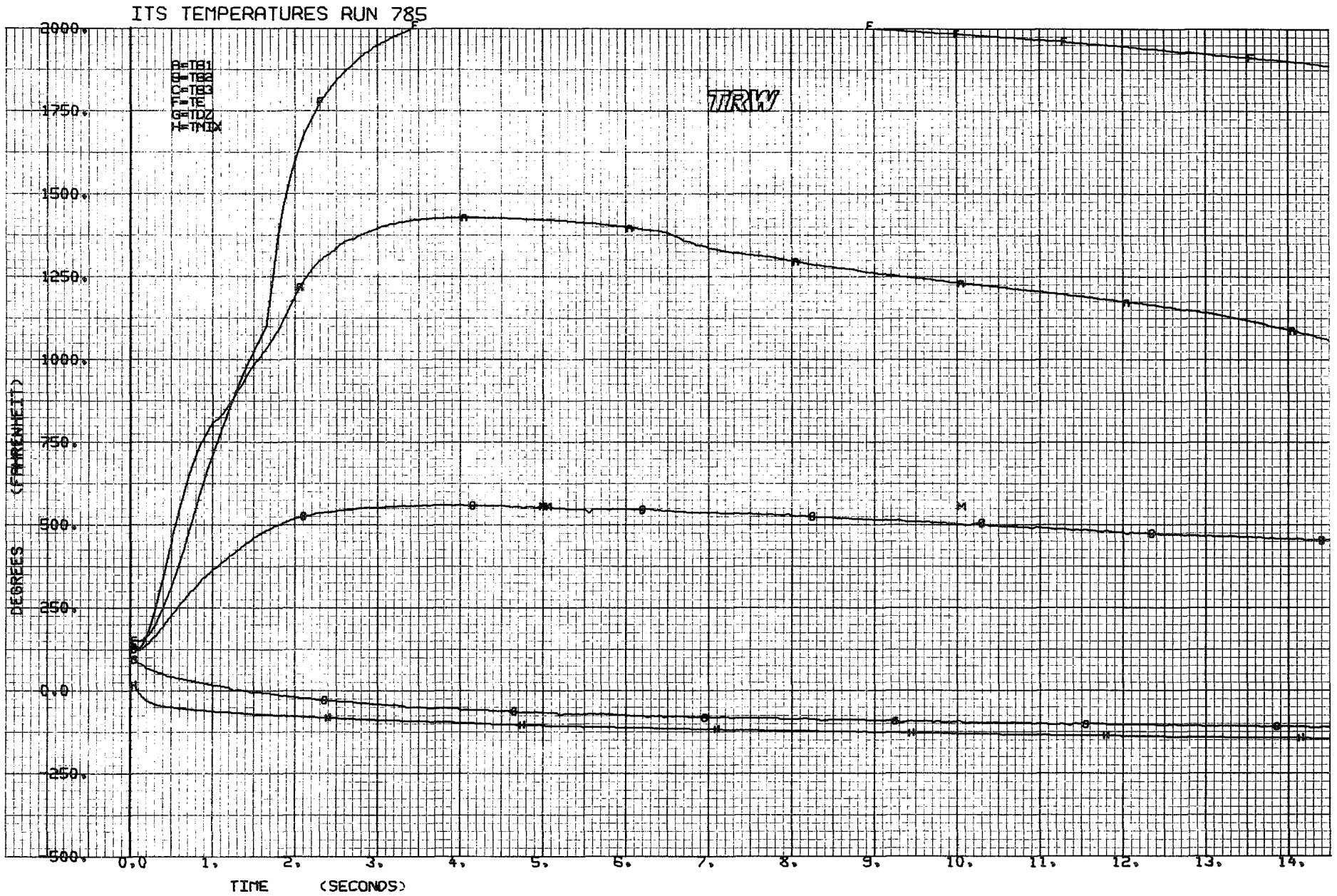



ITS TEMPERATURES RUN 782
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
TIME (SECONDS)

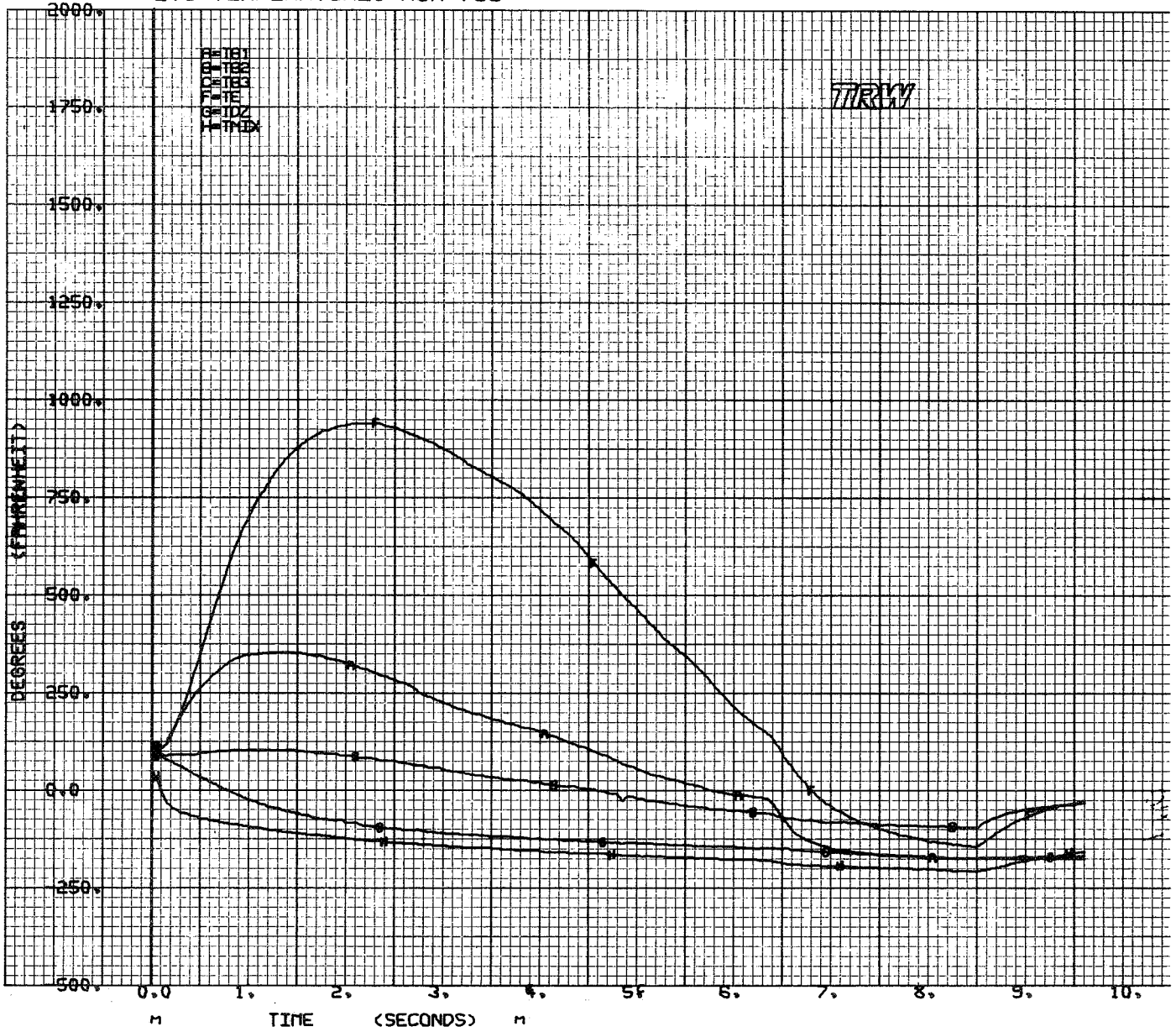

ITS TEMPERATURES RUN 783
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TNIX
TRW
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
TIME (SECONDS)

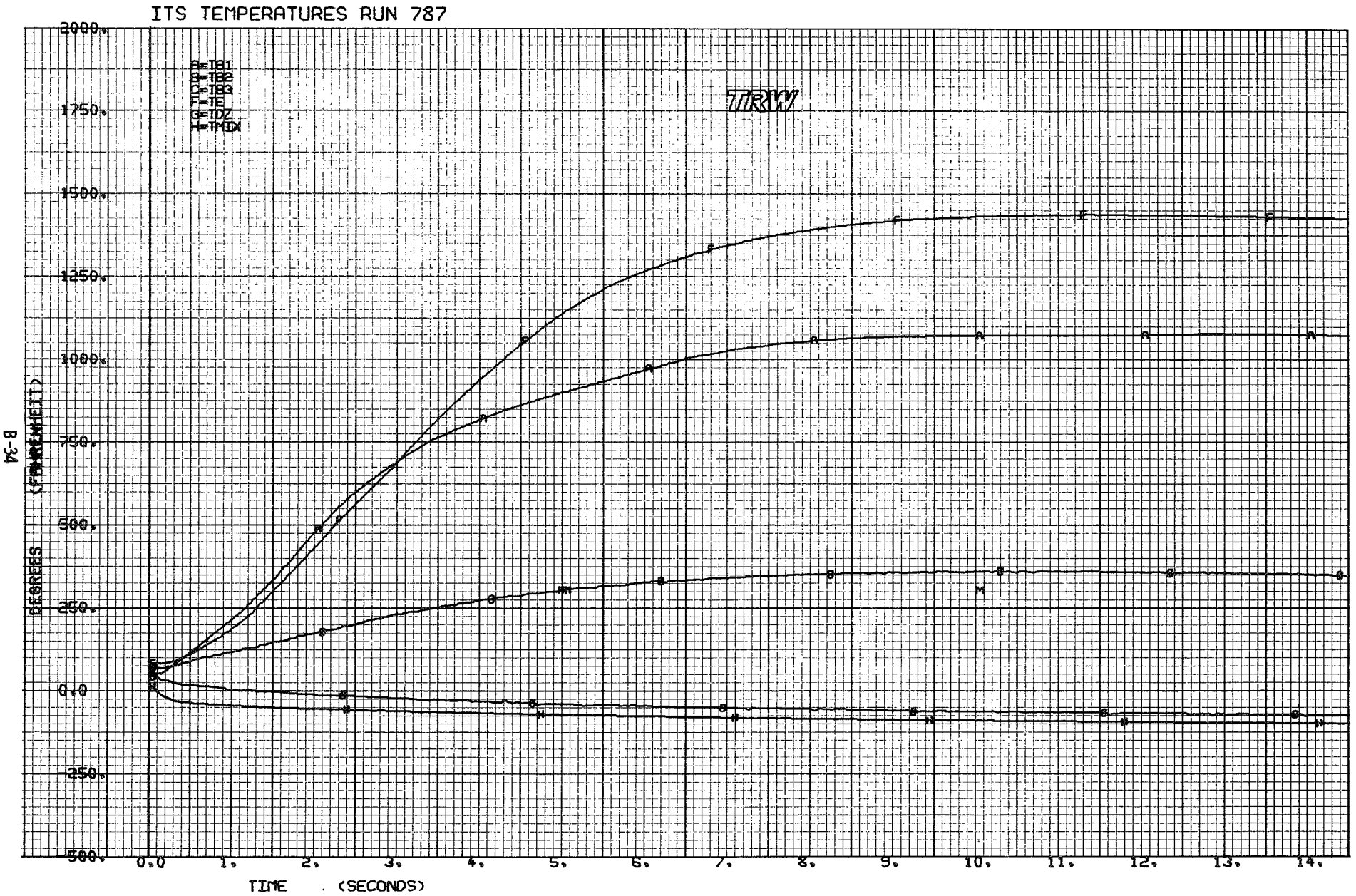
ITS TEMPERATURES RUN 784



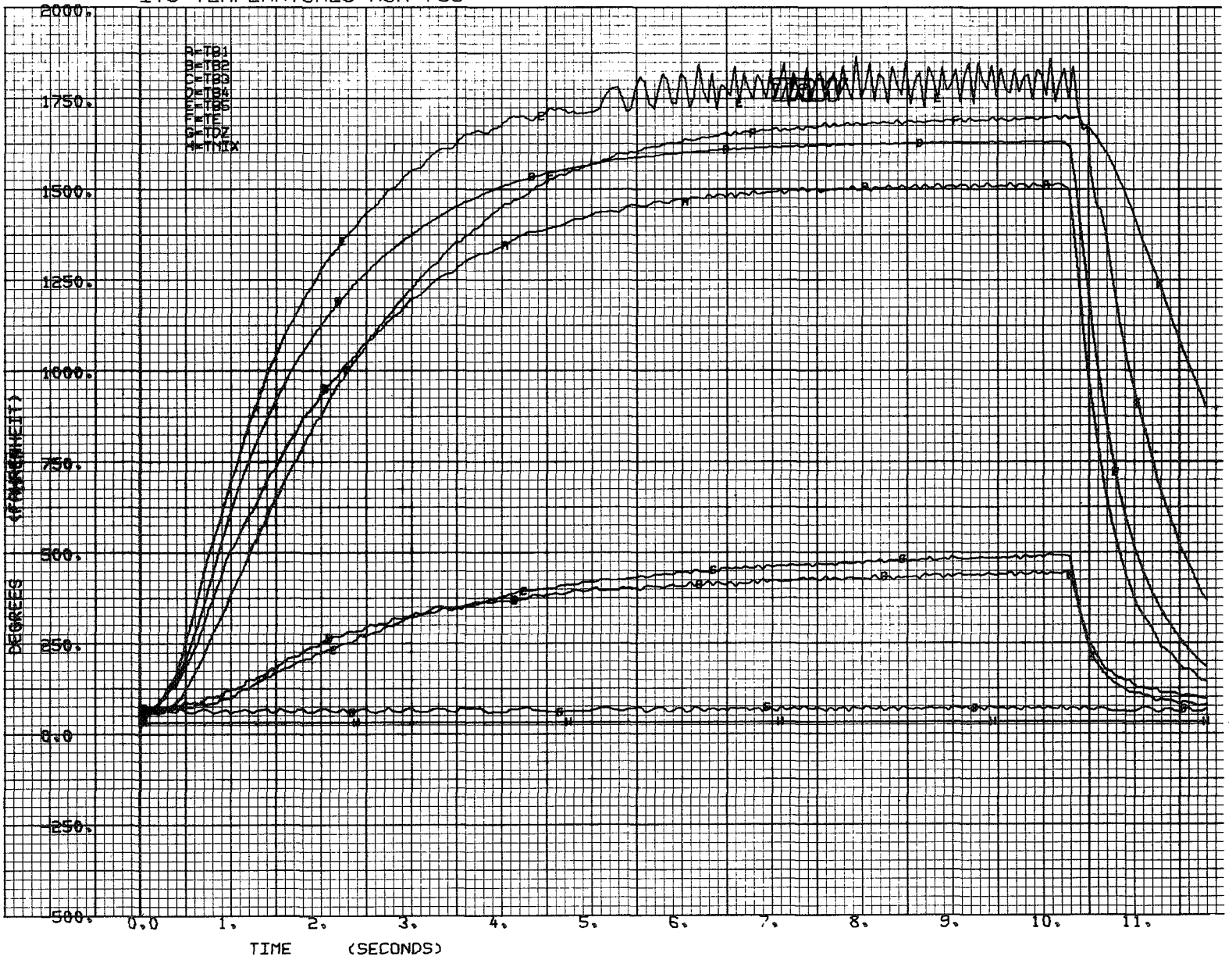

ITS TEMPERATURES RUN 785
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
TIME (SECONDS)

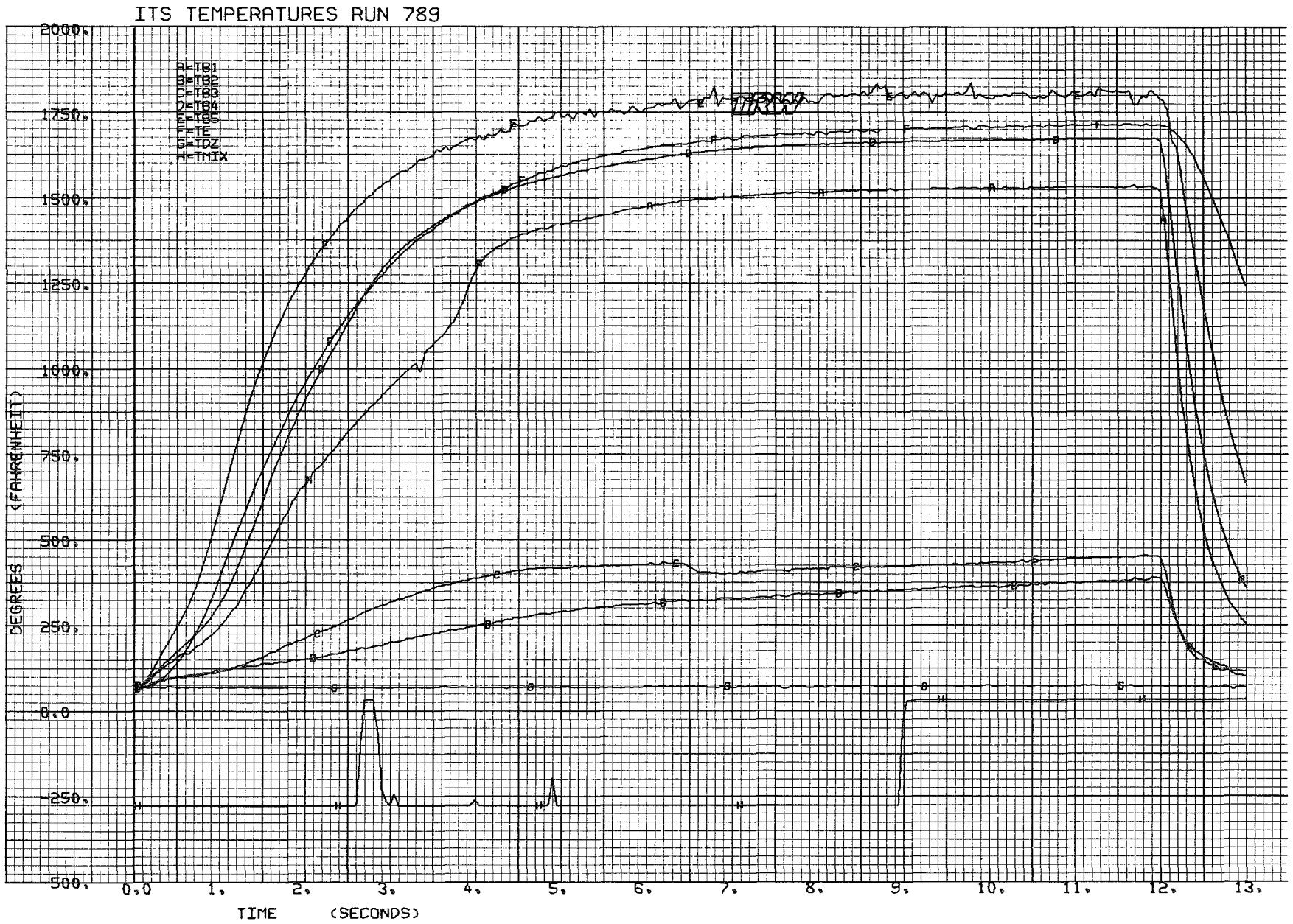
ITS TEMPERATURES RUN 786



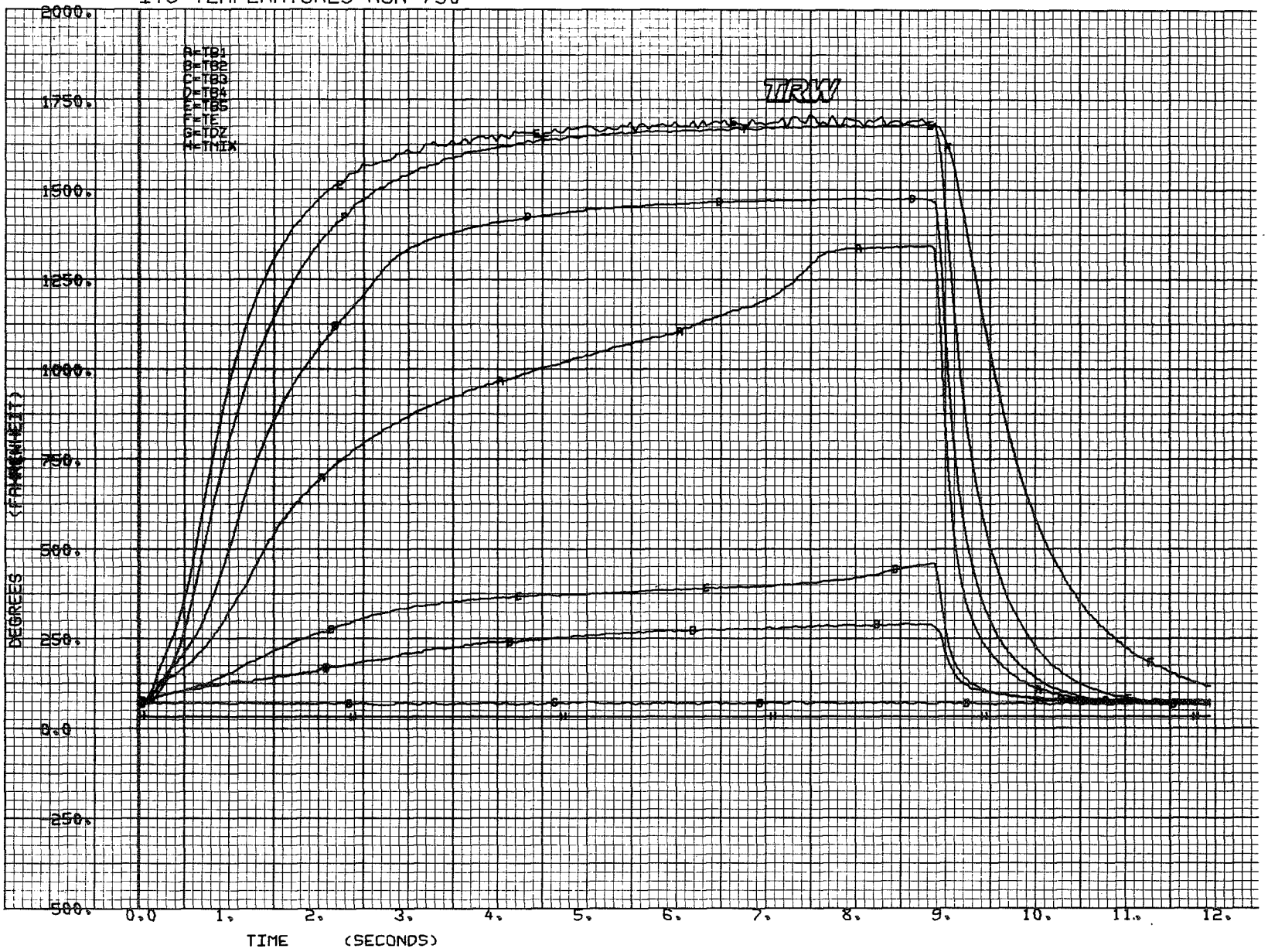

ITS TEMPERATURES RUN 787
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
TIME (SECONDS)

ITS TEMPERATURES RUN 788

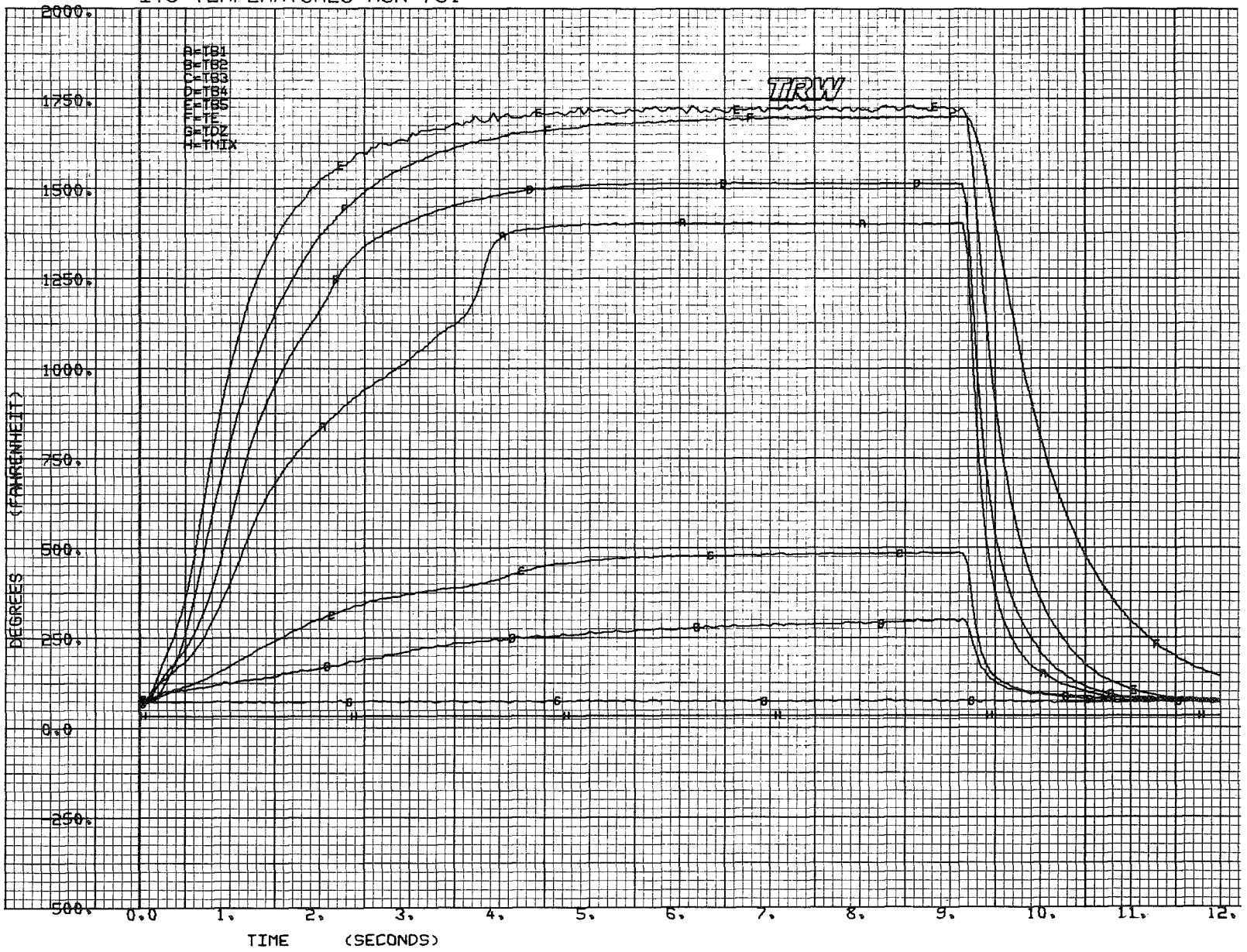



ITS TEMPERATURES RUN 789
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TDZ
H=TMIX
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
TIME (SECONDS)

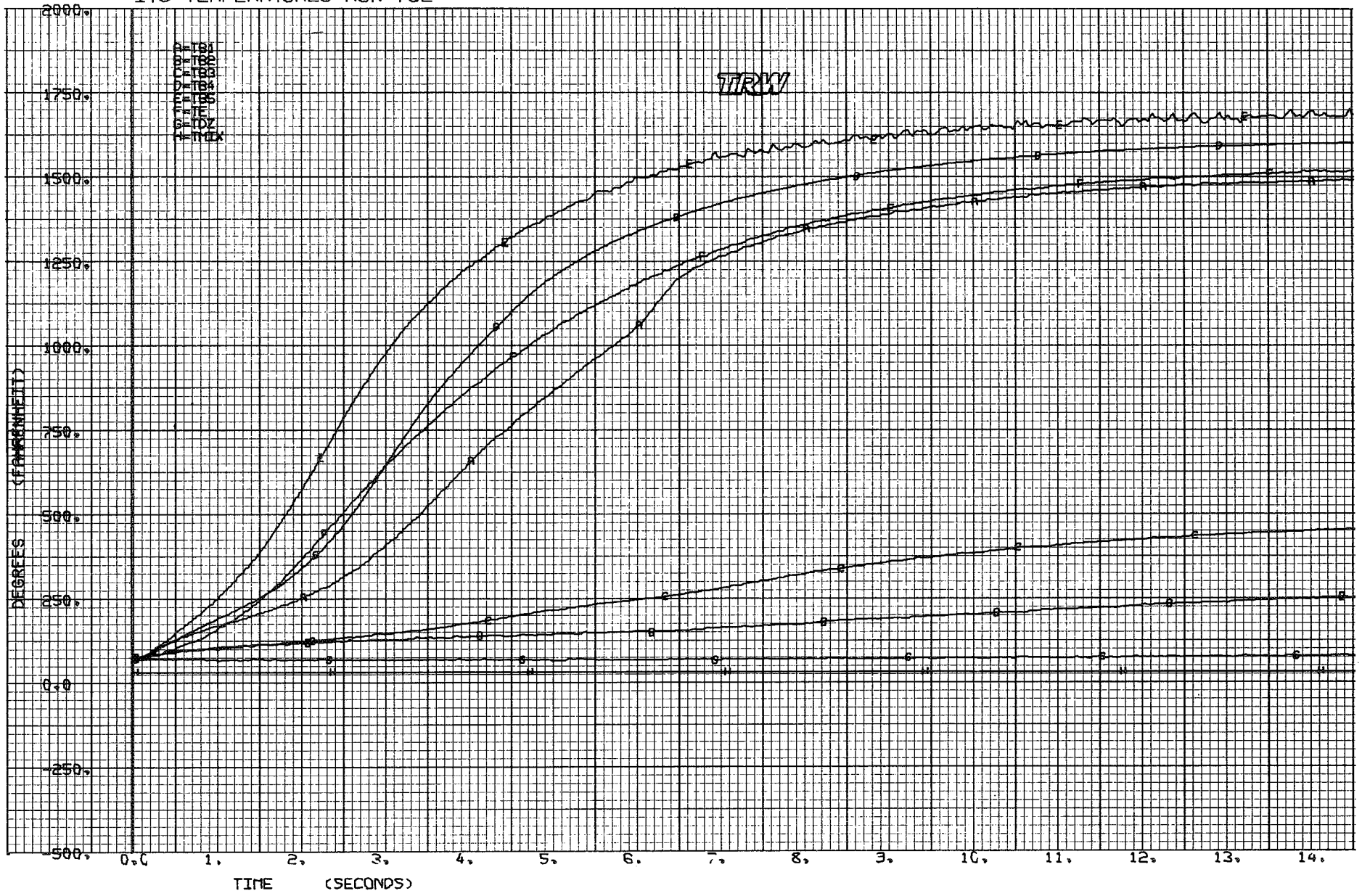
ITS TEMPERATURES RUN 790



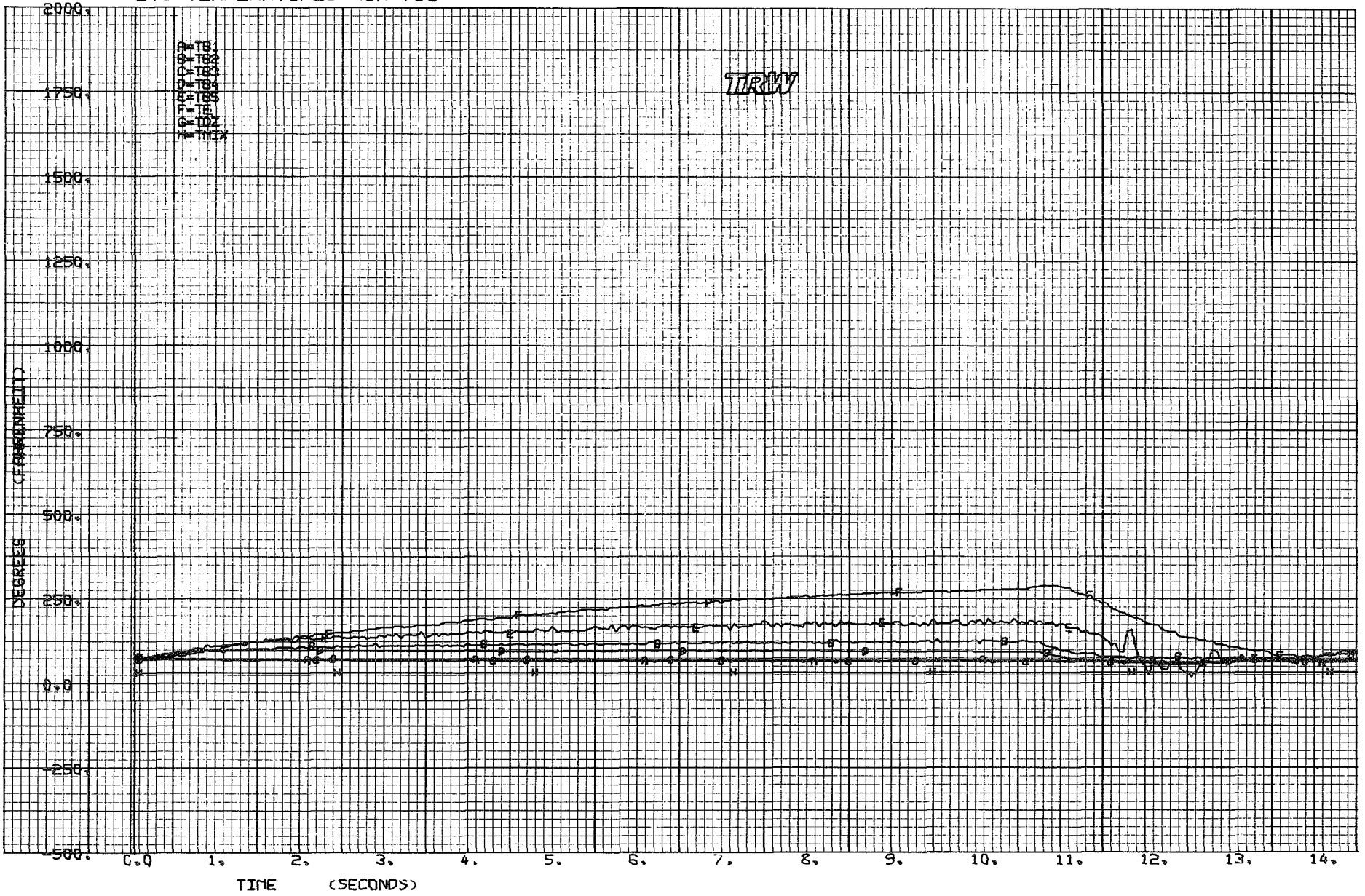
ITS TEMPERATURES RUN 791



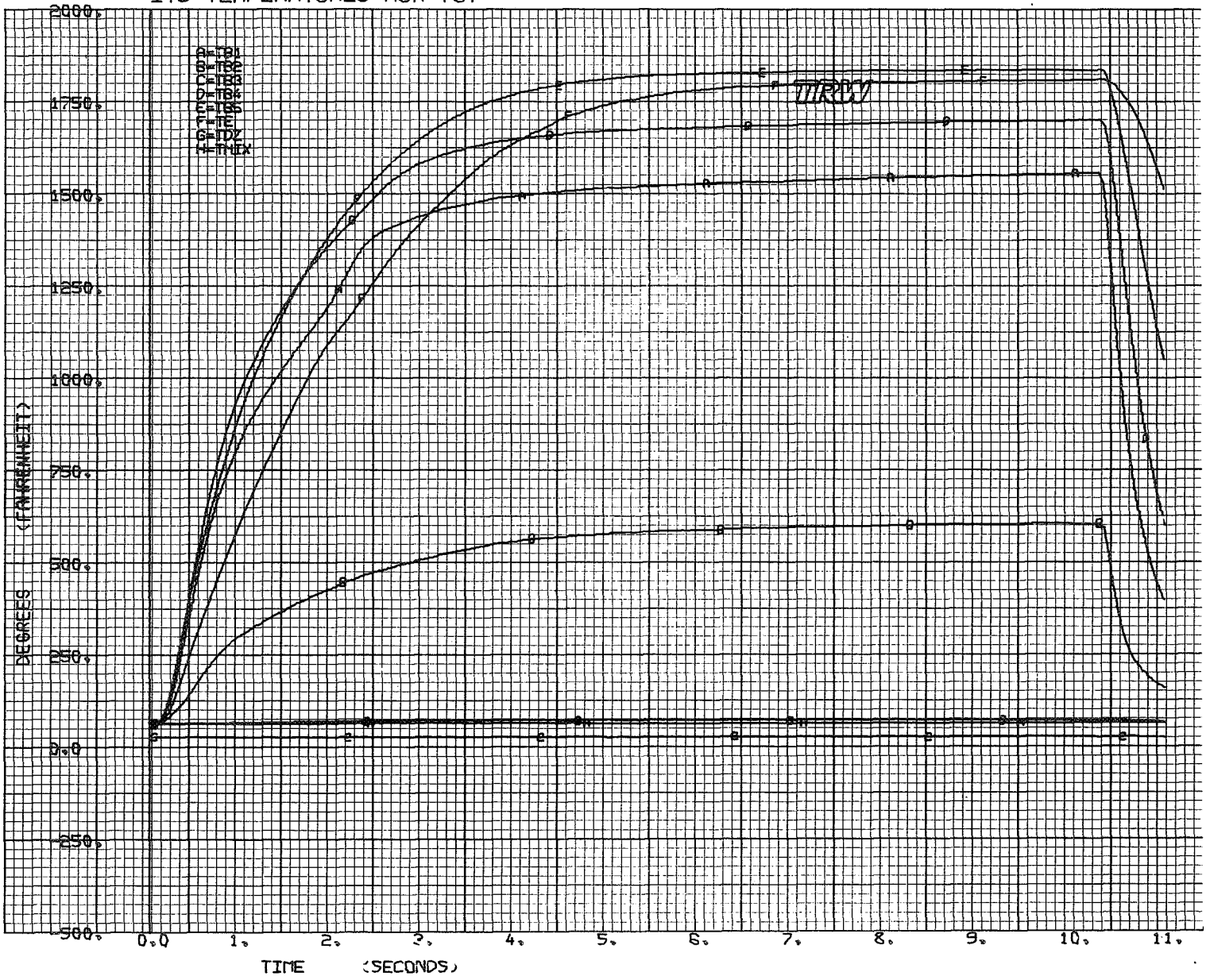
ITS TEMPERATURES RUN 792

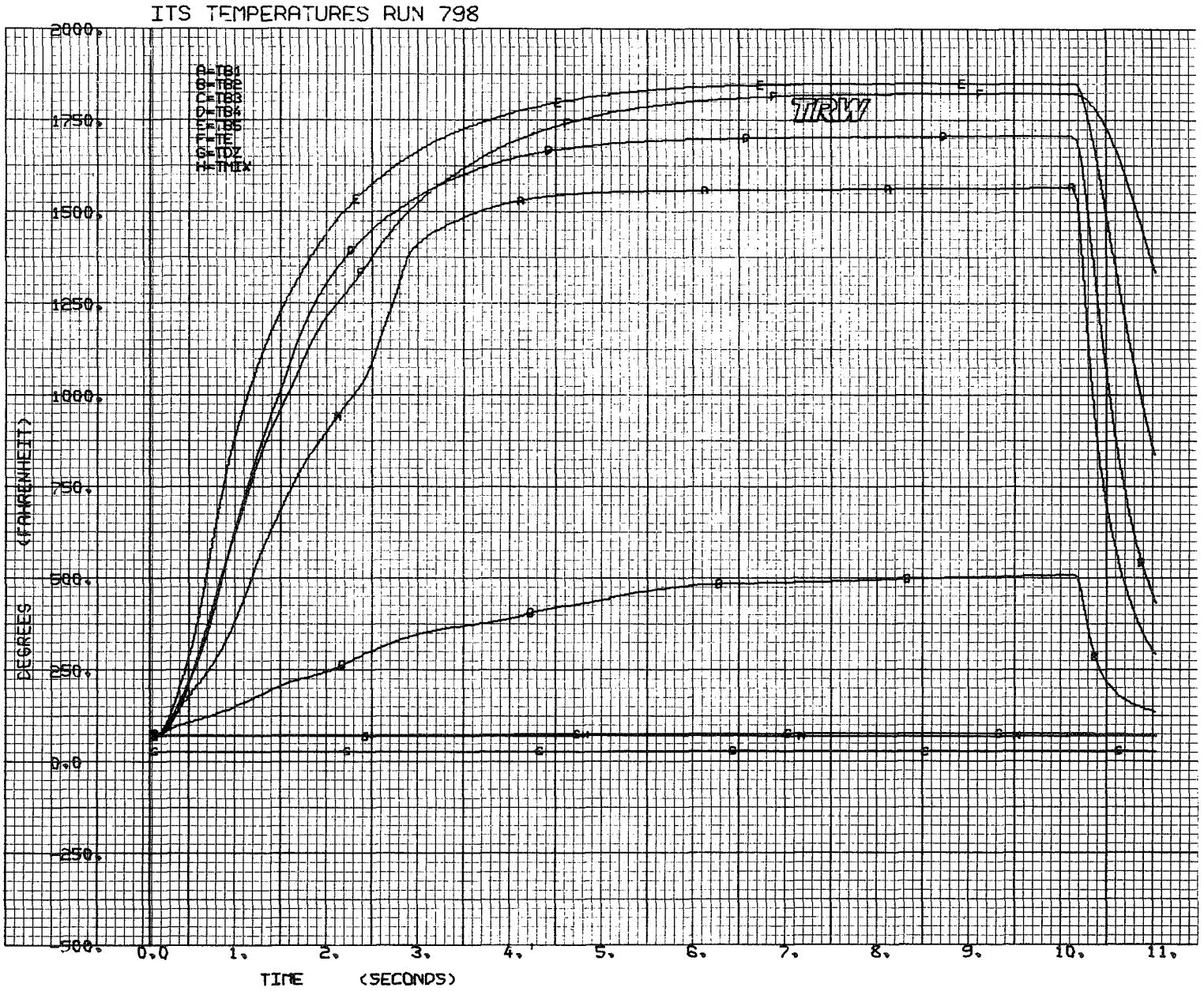


ITS TEMPERATURES RUN 796

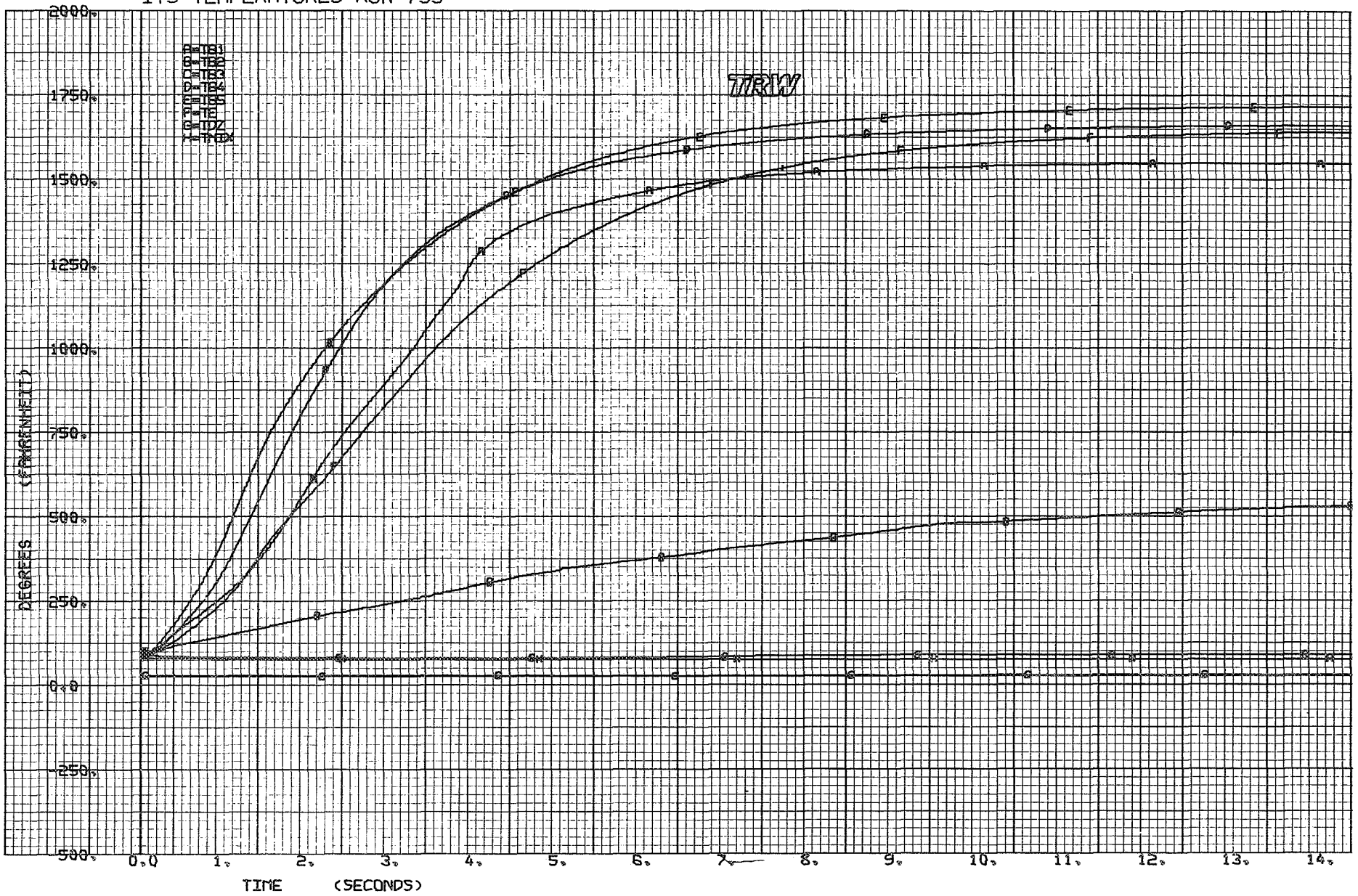


ITS TEMPERATURES RUN 797

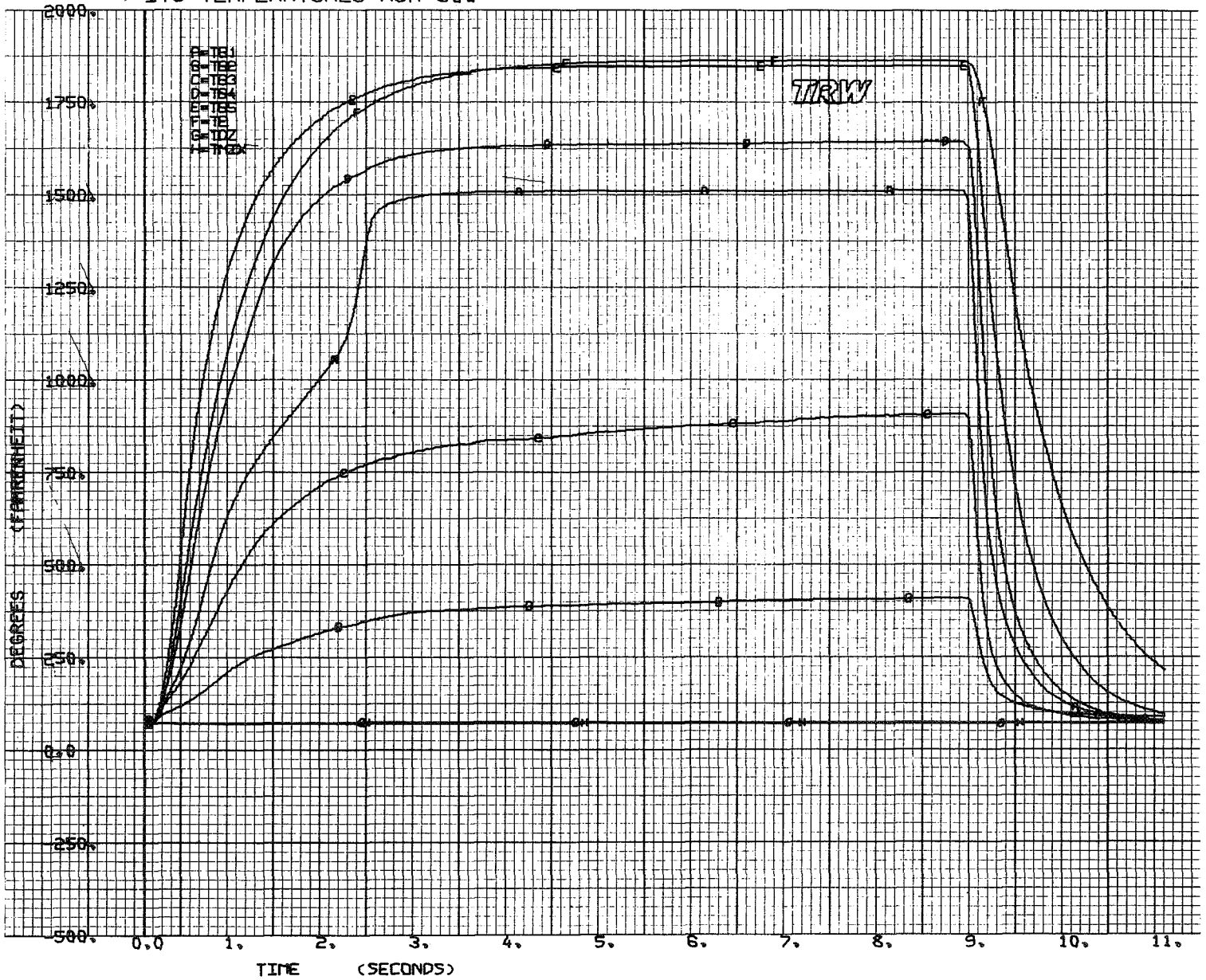



ITS TEMPERATURES RUN 798
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TDZ
H=THEX
TRW
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
TIME (SECONDS)

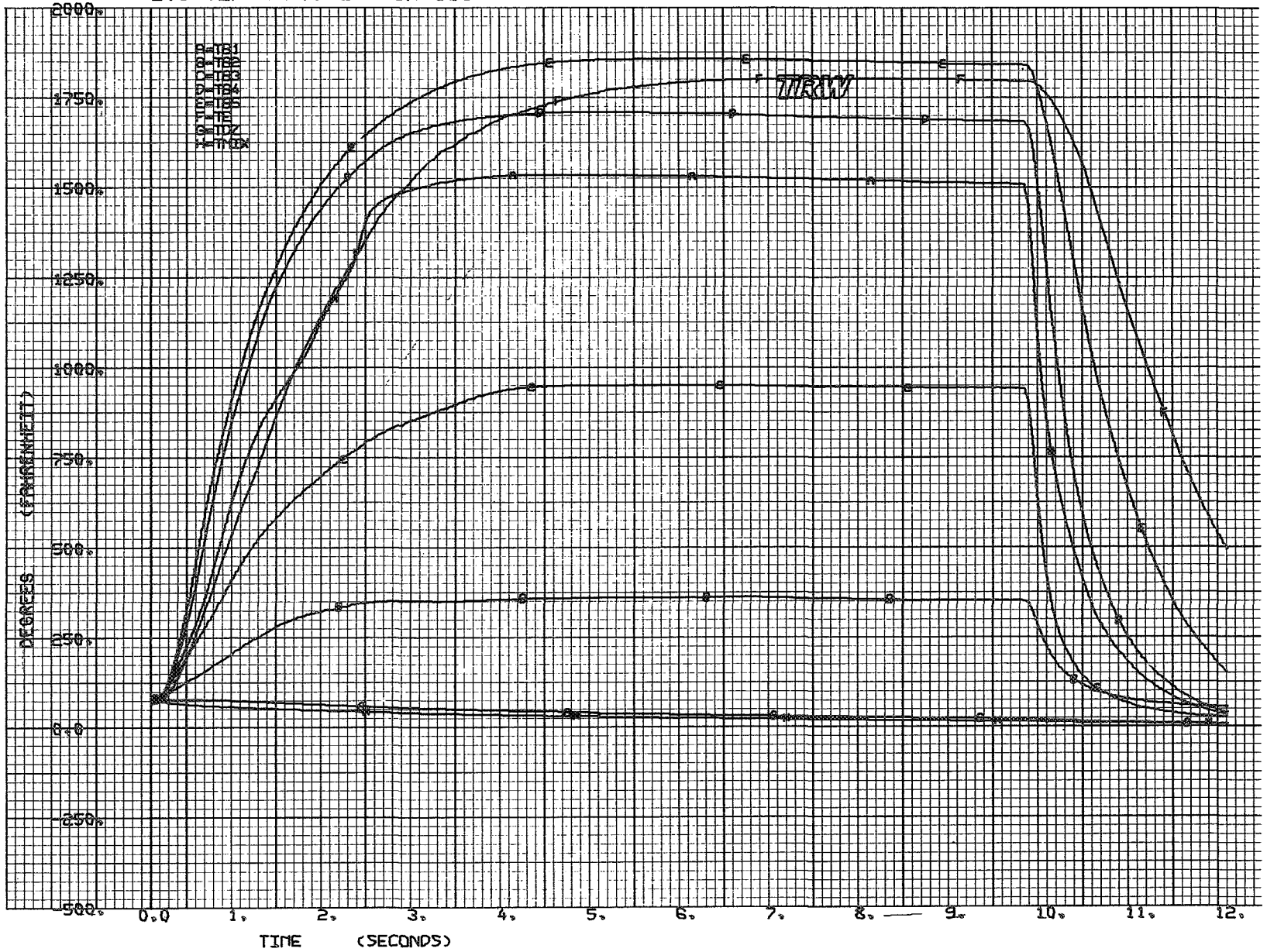
ITS TEMPERATURES RUN 799

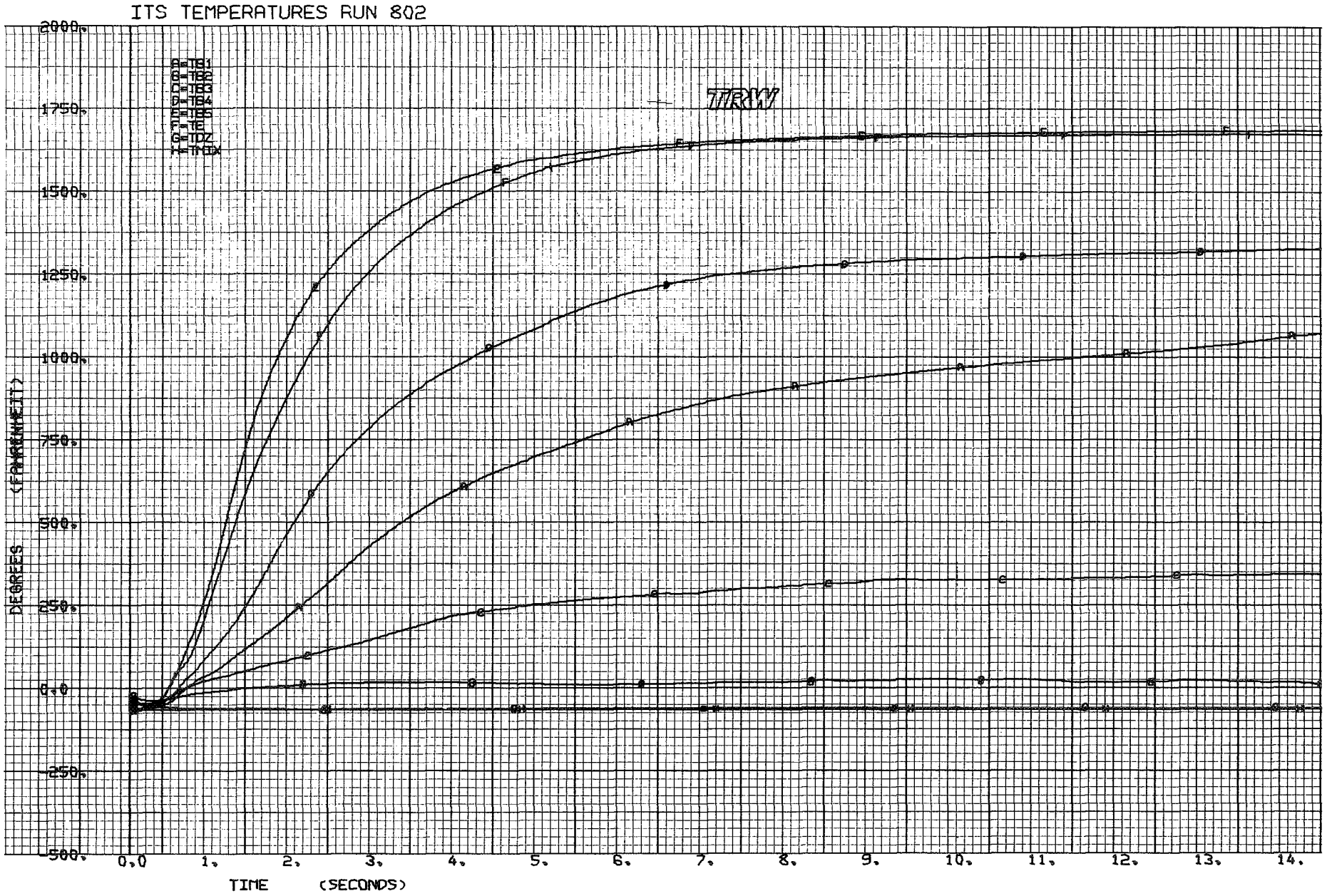


ITS TEMPERATURES RUN 800

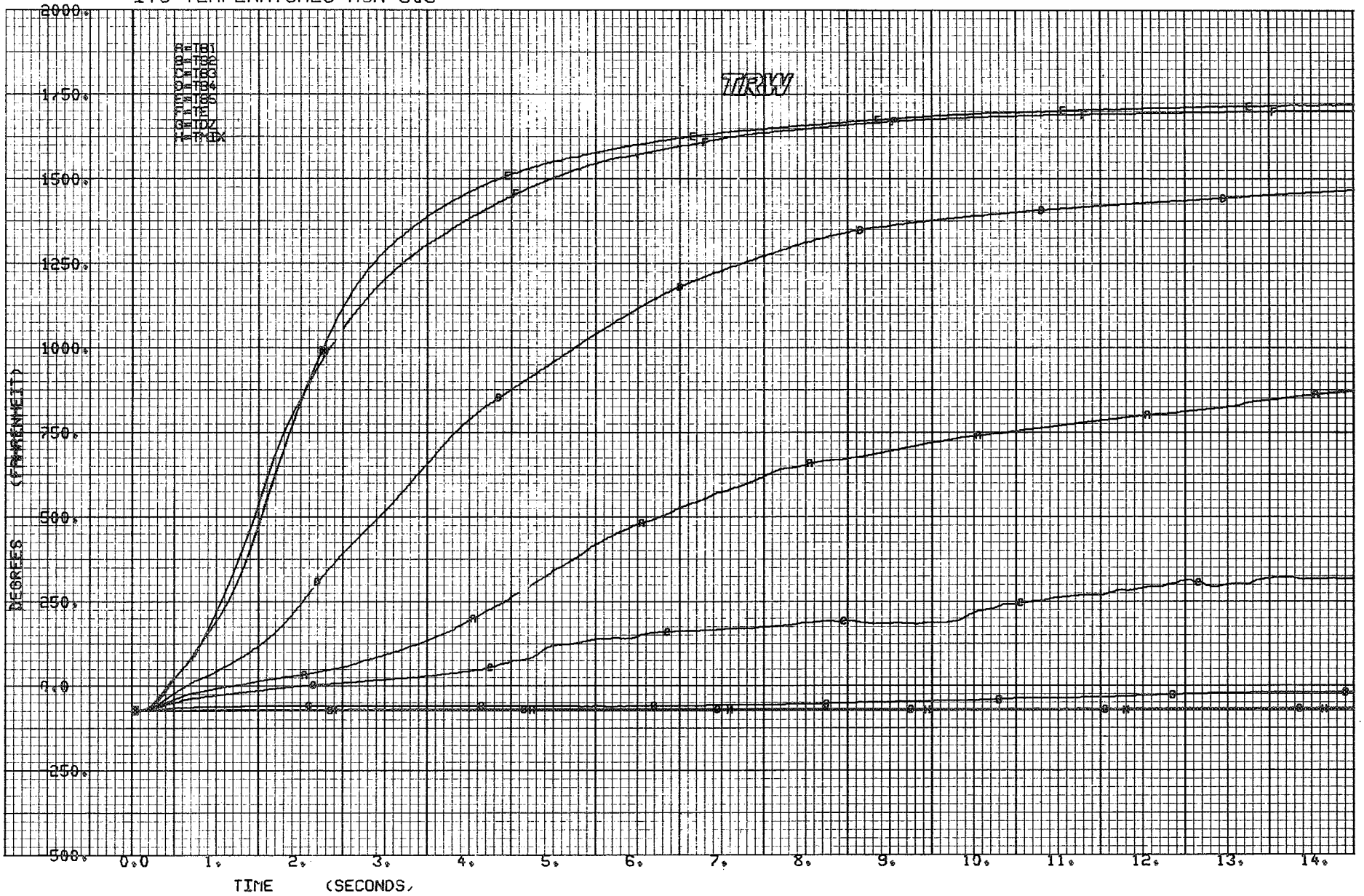


ITS TEMPERATURES RUN 801

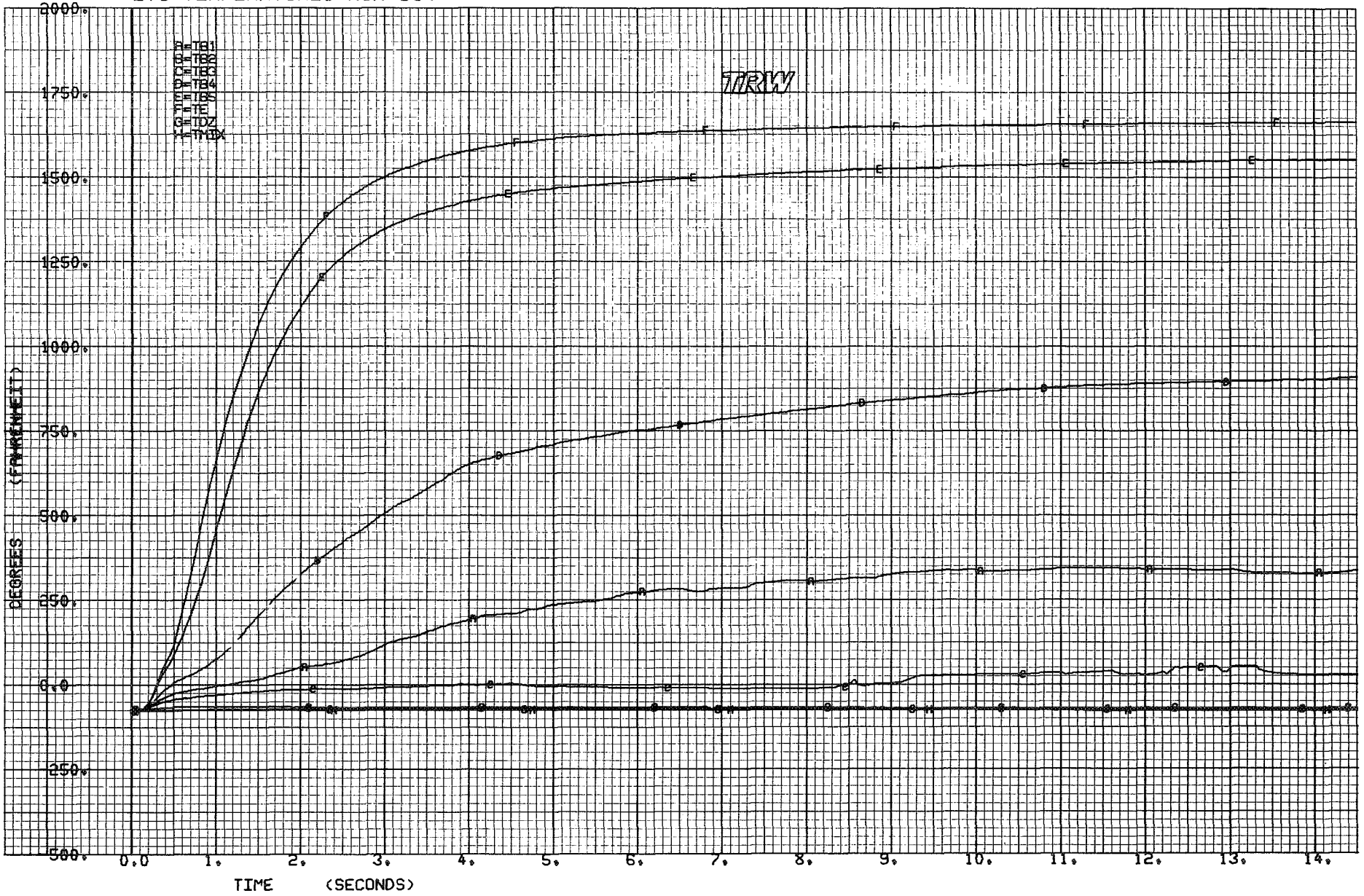



ITS TEMPERATURES RUN 802
B=TB1
C=TB2
D=TB3
E=TB4
F=TB5
G=TDZ
H=TNDX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
TIME (SECONDS)

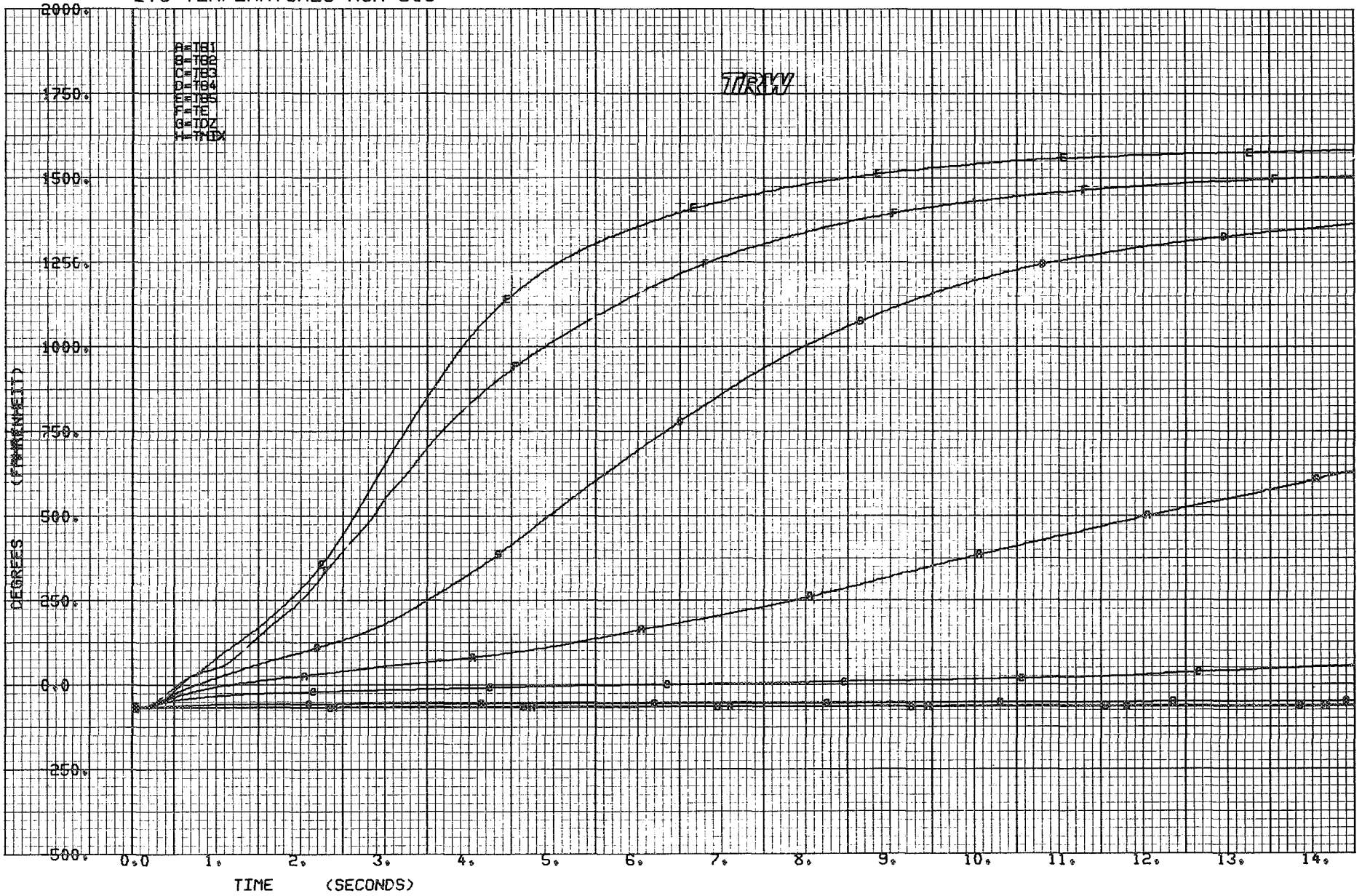
ITS TEMPERATURES RJN 803



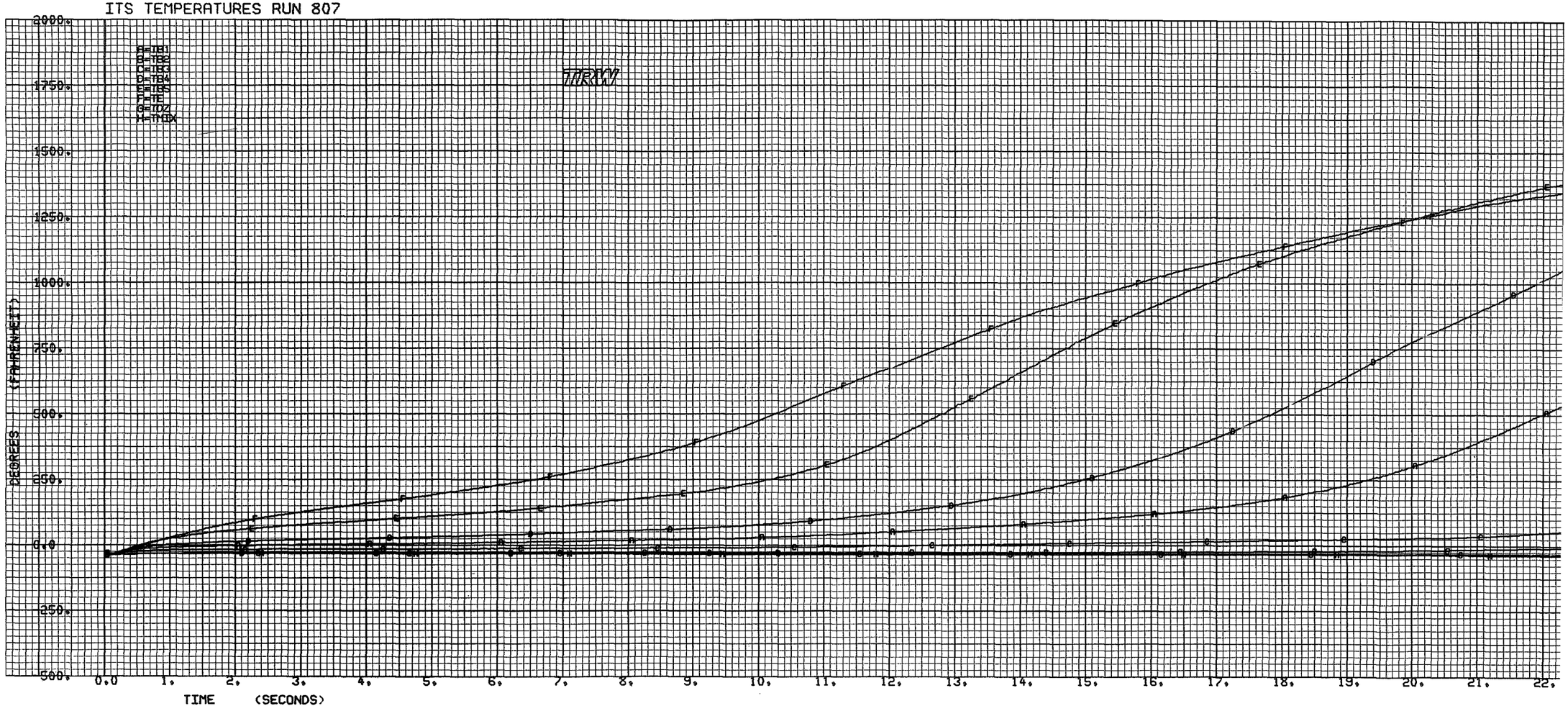
ITS TEMPERATURES RUN 804

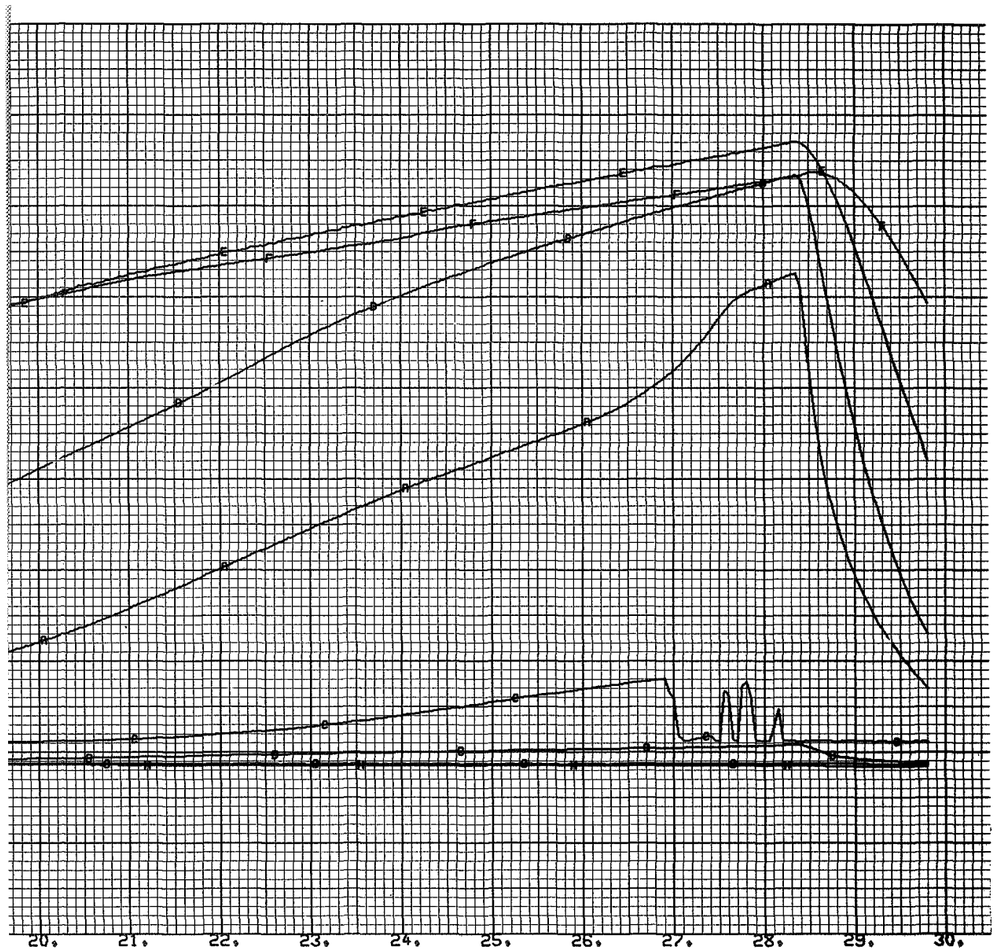


ITS TEMPERATURES RUN 805

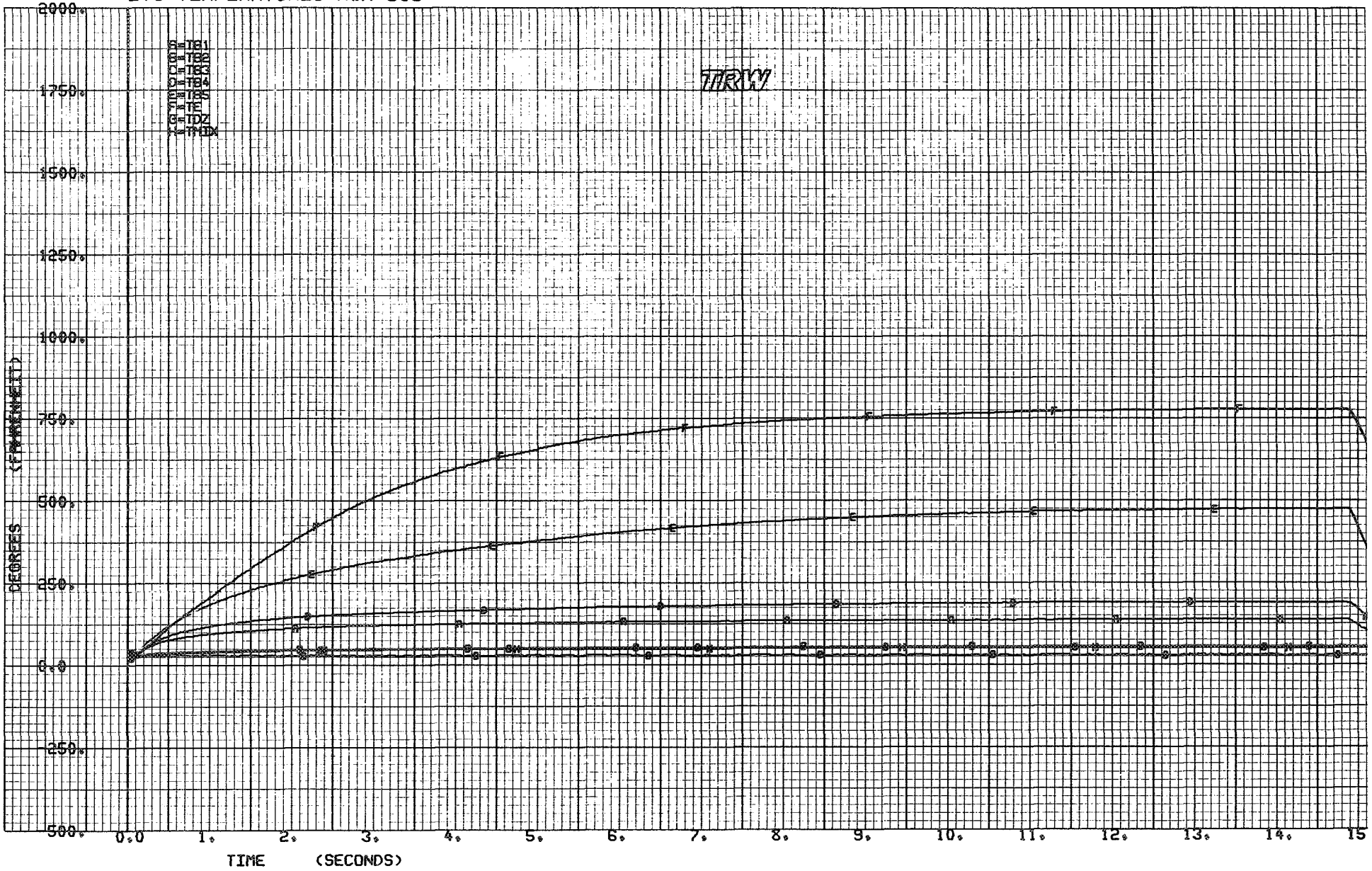


ITS TEMPERATURES RUN 807

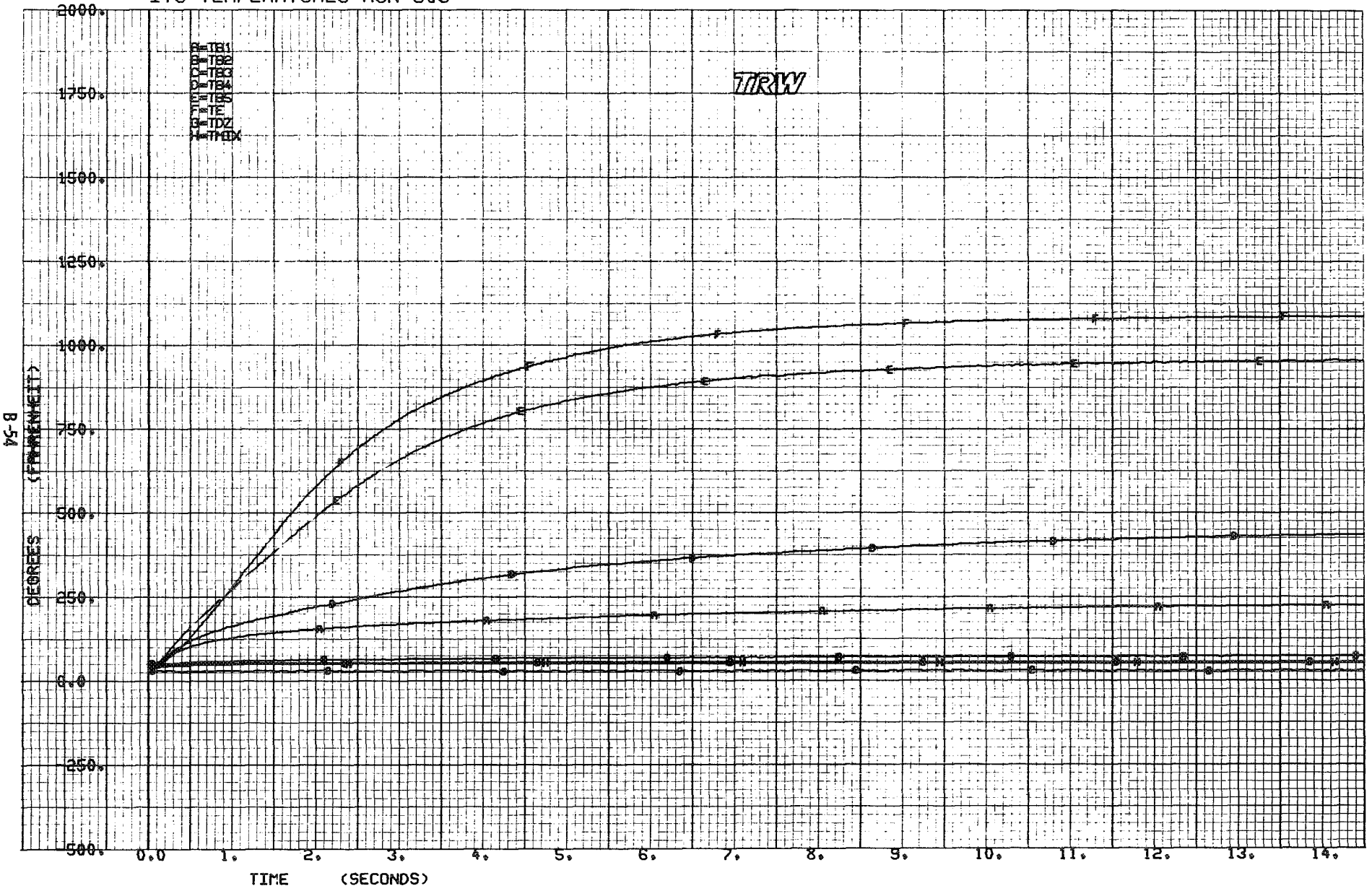



20.
21.
22.
23.
24.
25.
26.
27.
28.
29.
30.

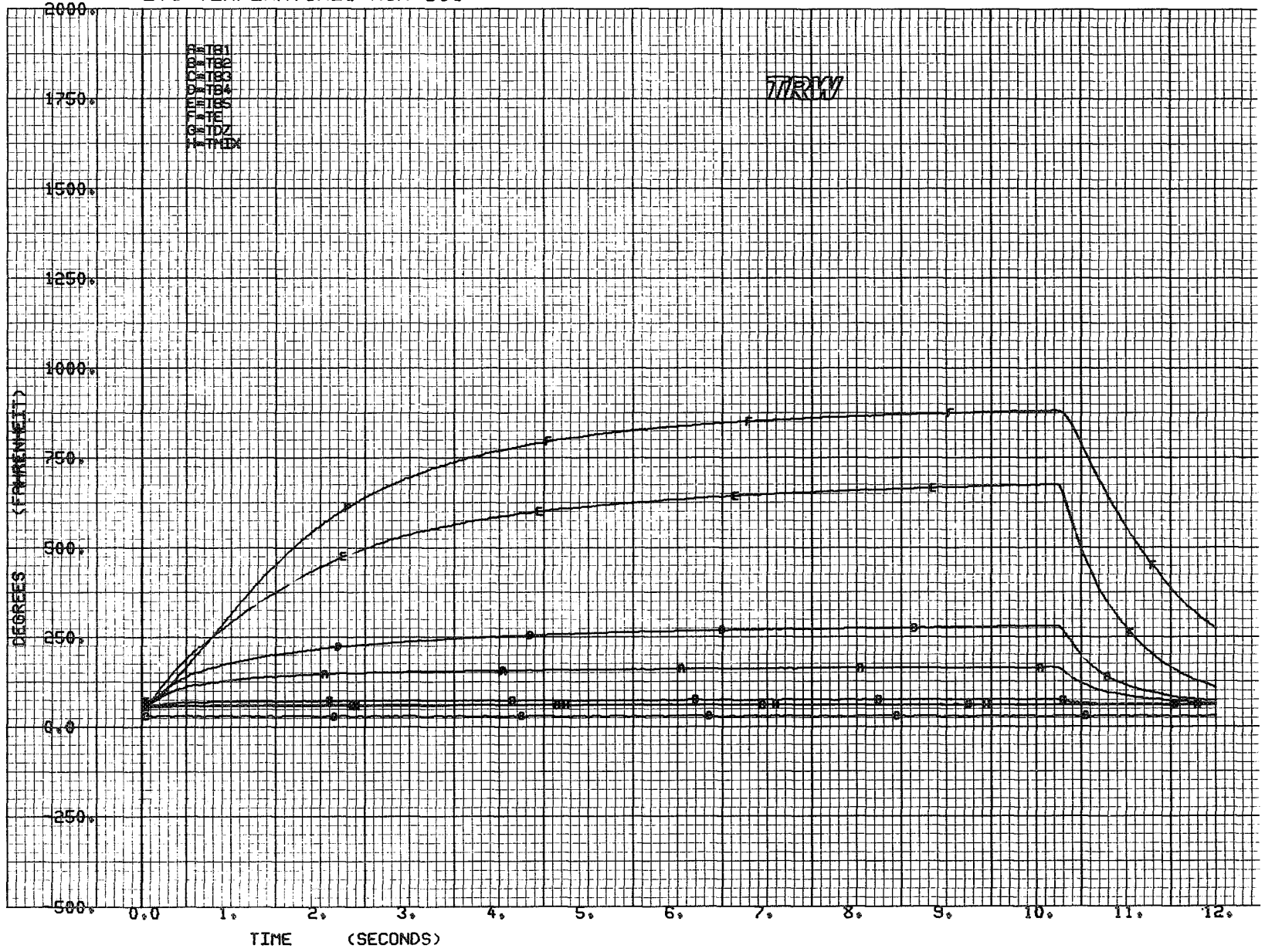
ITS TEMPERATURES RUN 808

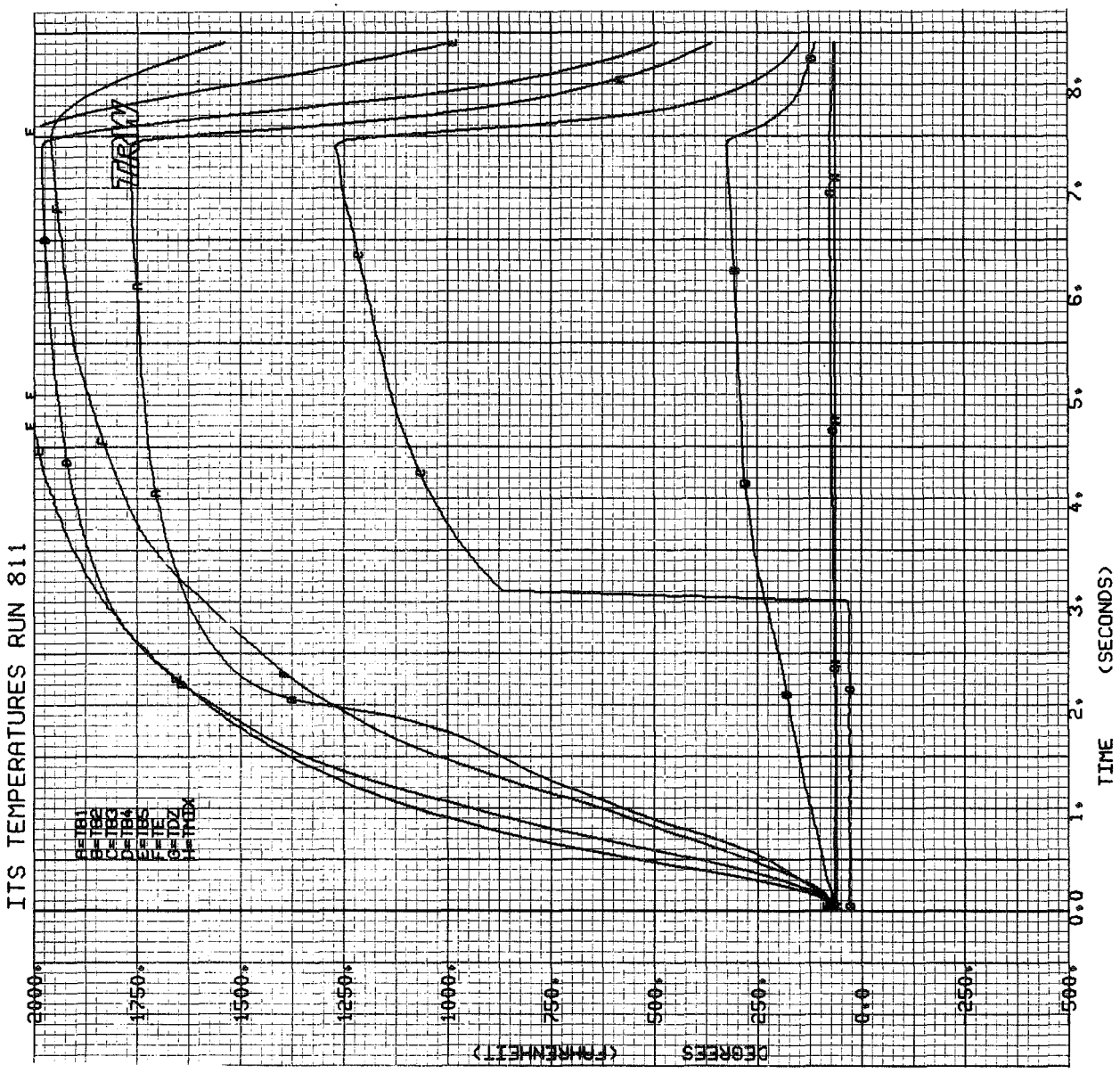


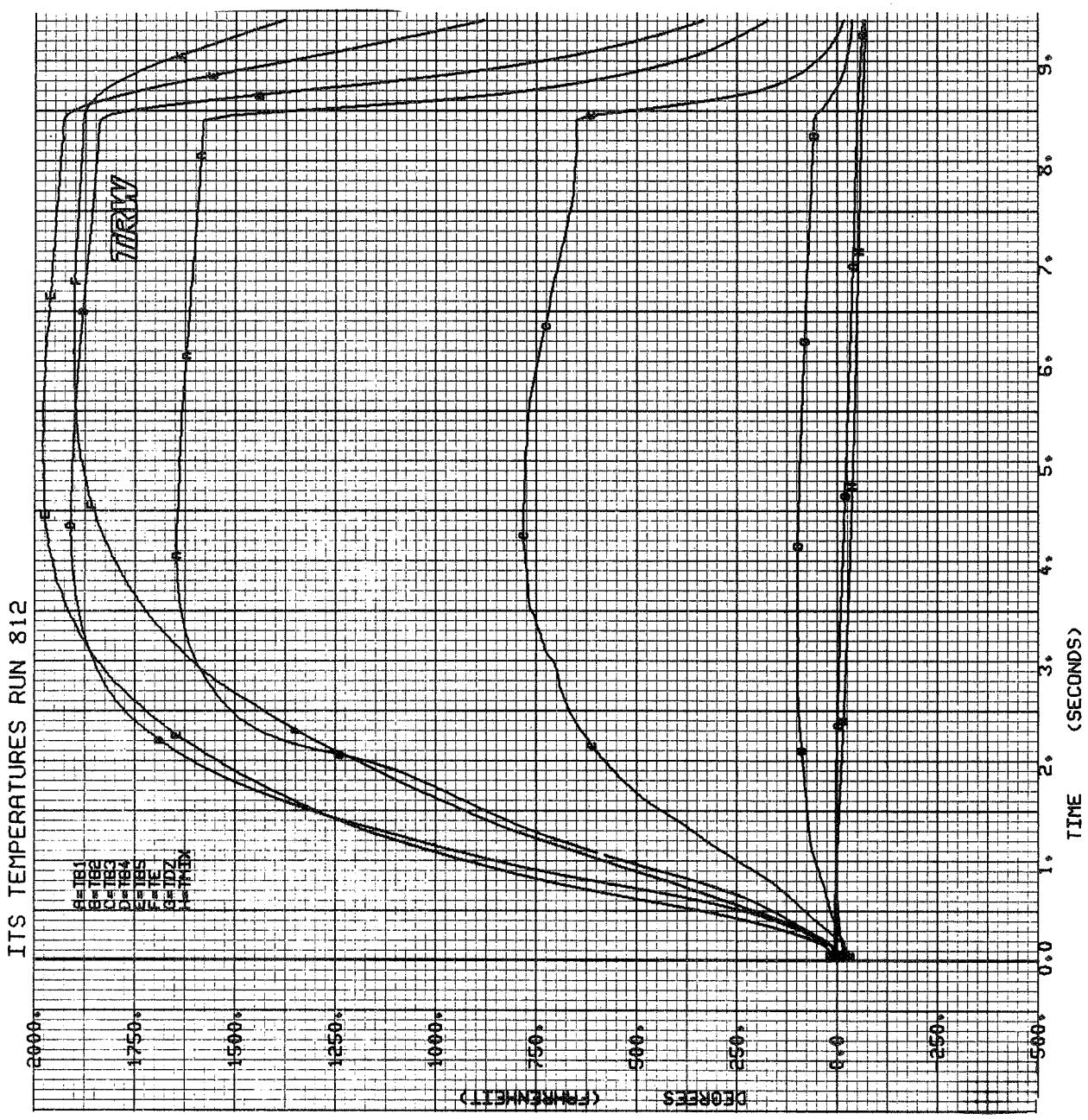
ITS TEMPERATURES RUN 809



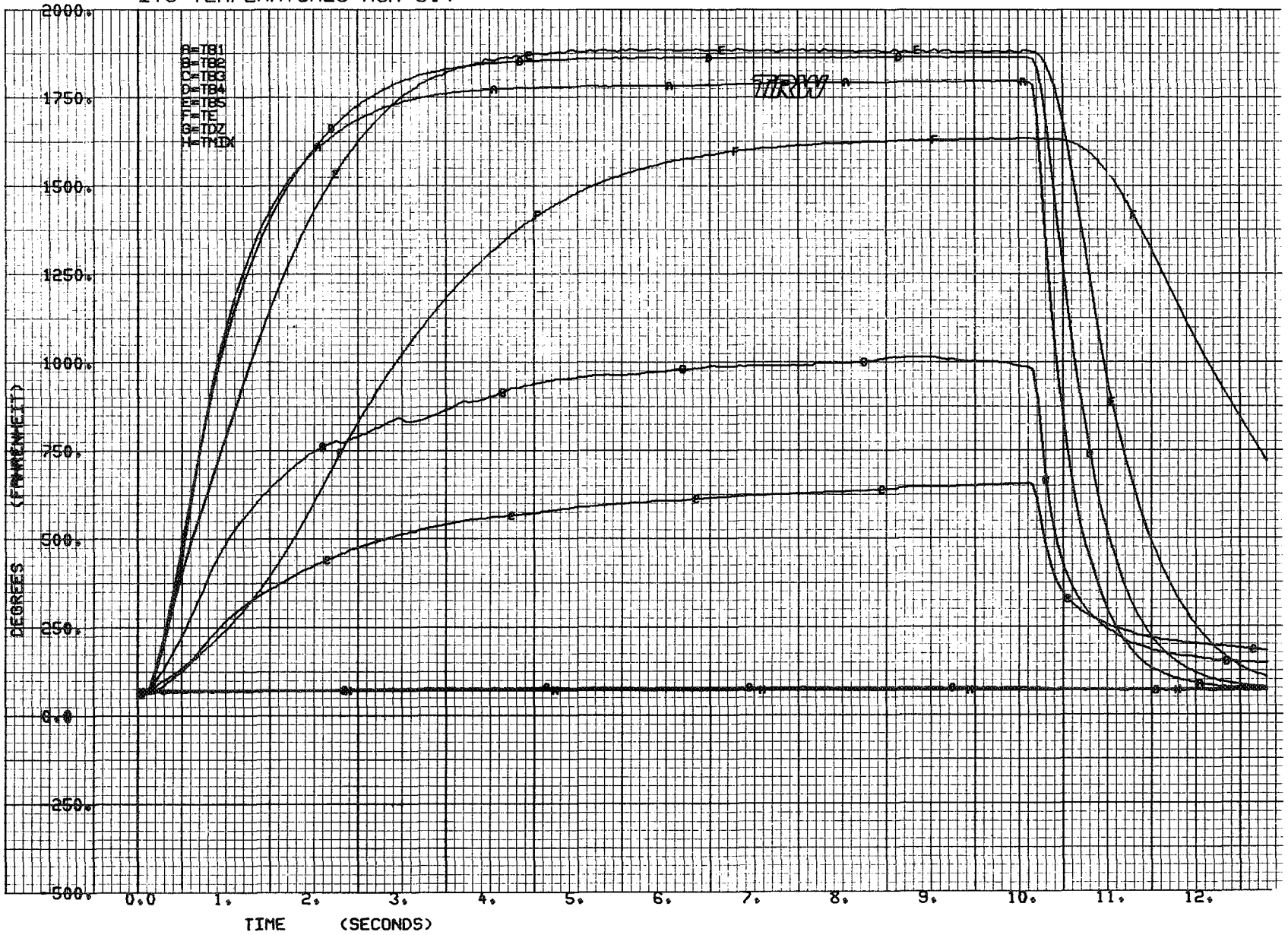
ITS TEMPERATURES RUN 810



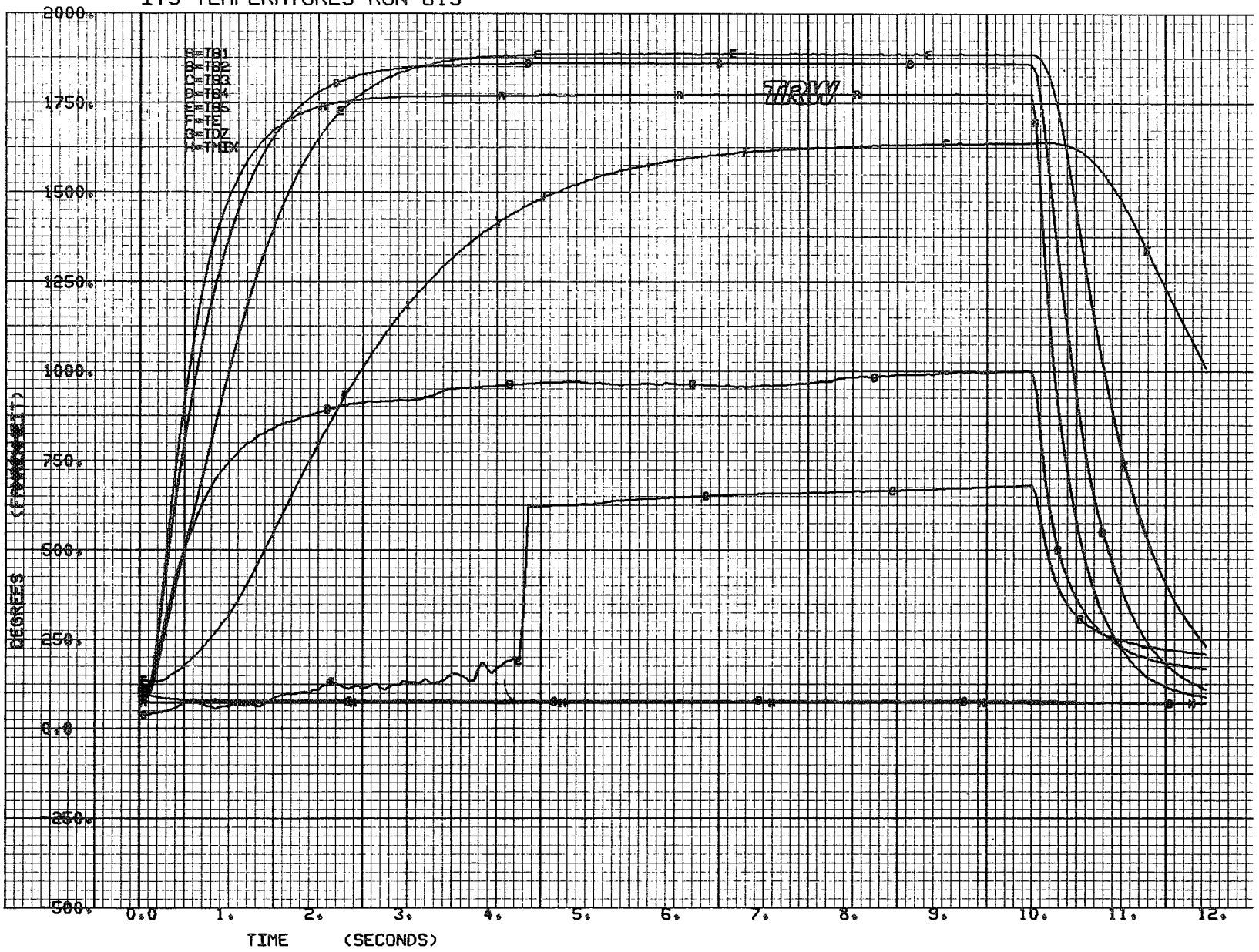

ITS TEMPERATURES RUN 811
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TD2
H=TBX
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
250.
500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
TIME (SECONDS)


ITS TEMPERATURES RUN 812
A = TB1
B = TB2
C = TB3
D = TB4
E = TB5
F = TE
G = TD2
H = TMEX
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
TIME (SECONDS)

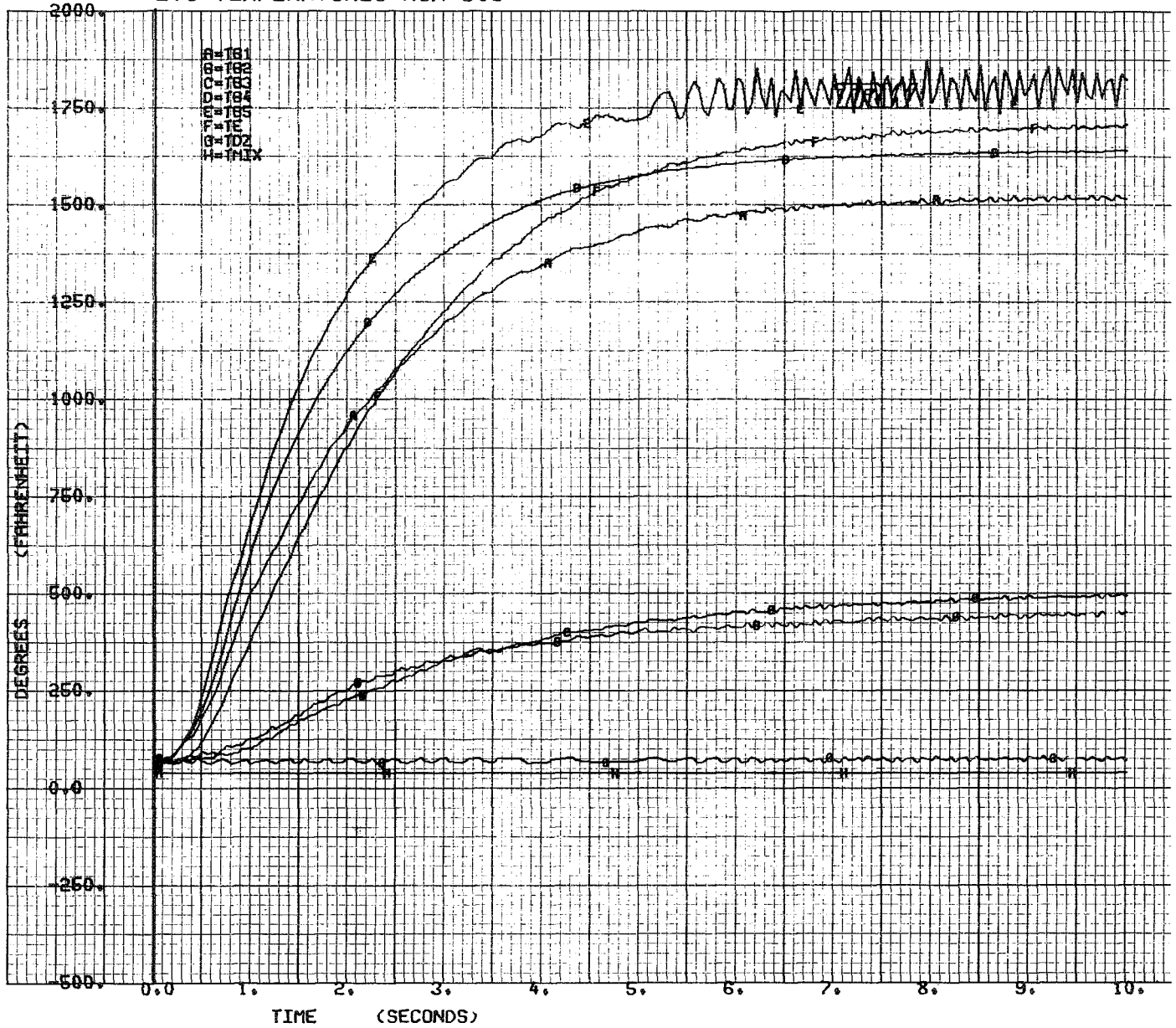
ITS TEMPERATURES RUN 814



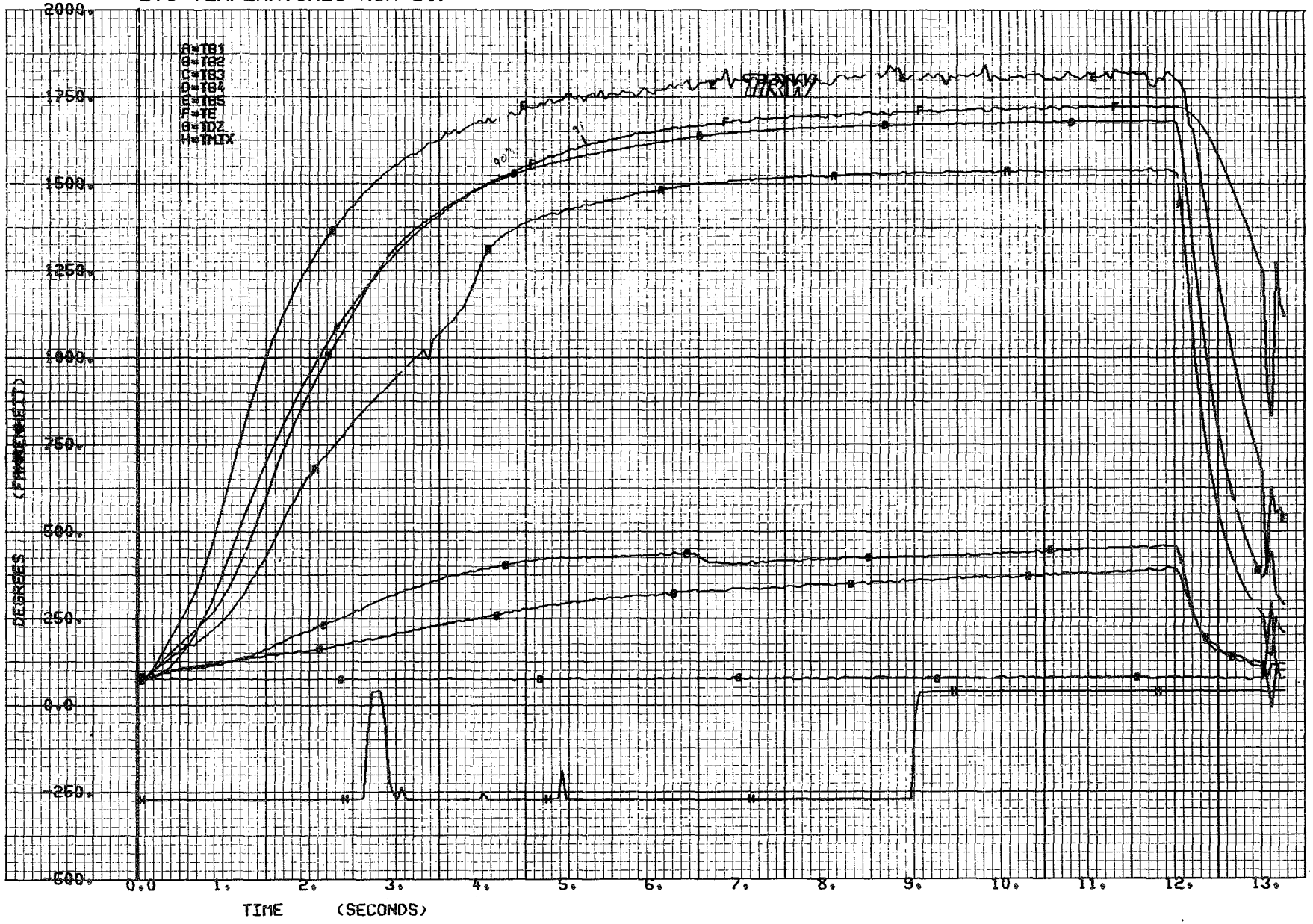
ITS TEMPERATURES RUN 815



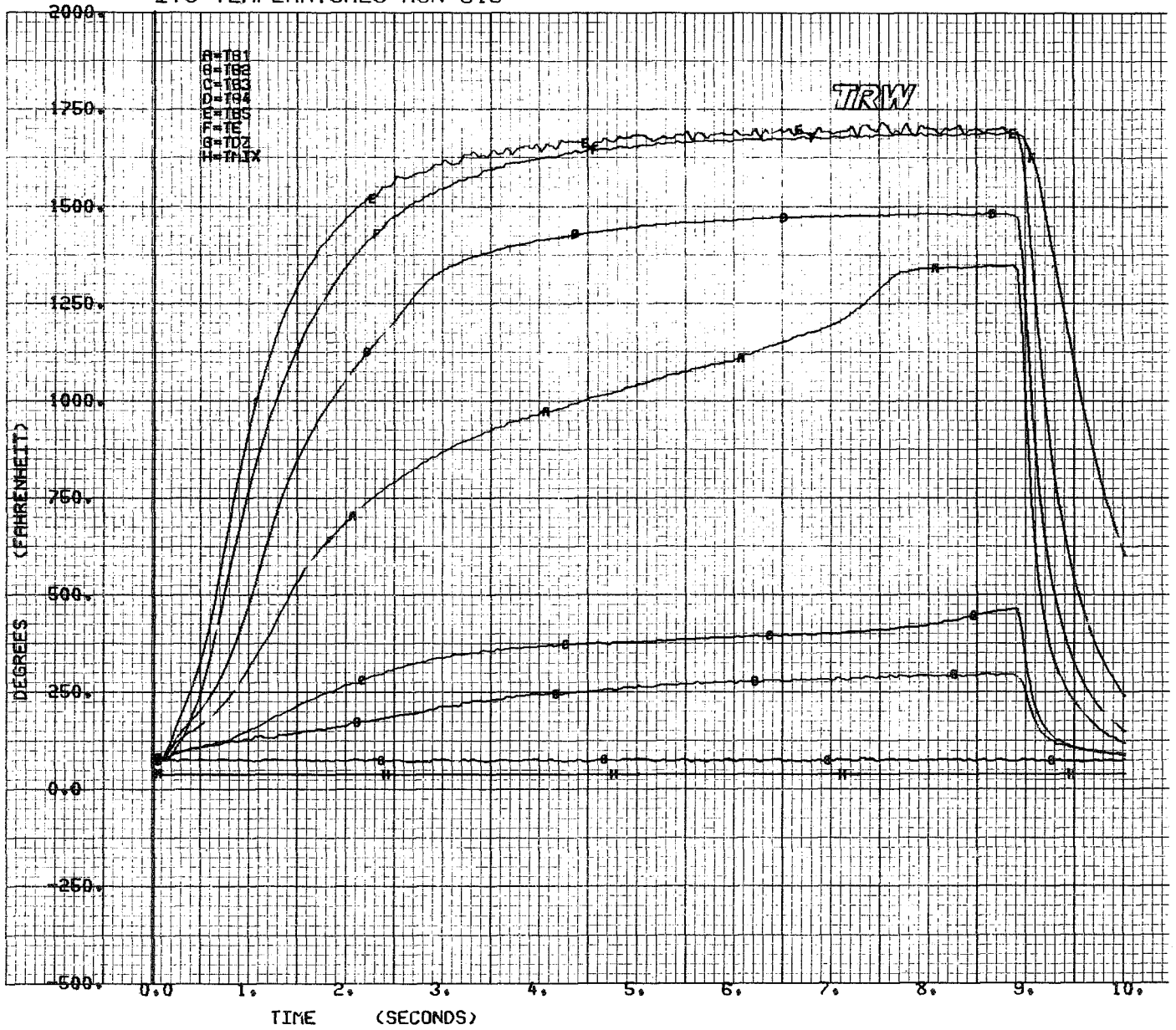
ITS TEMPERATURES RUN 816



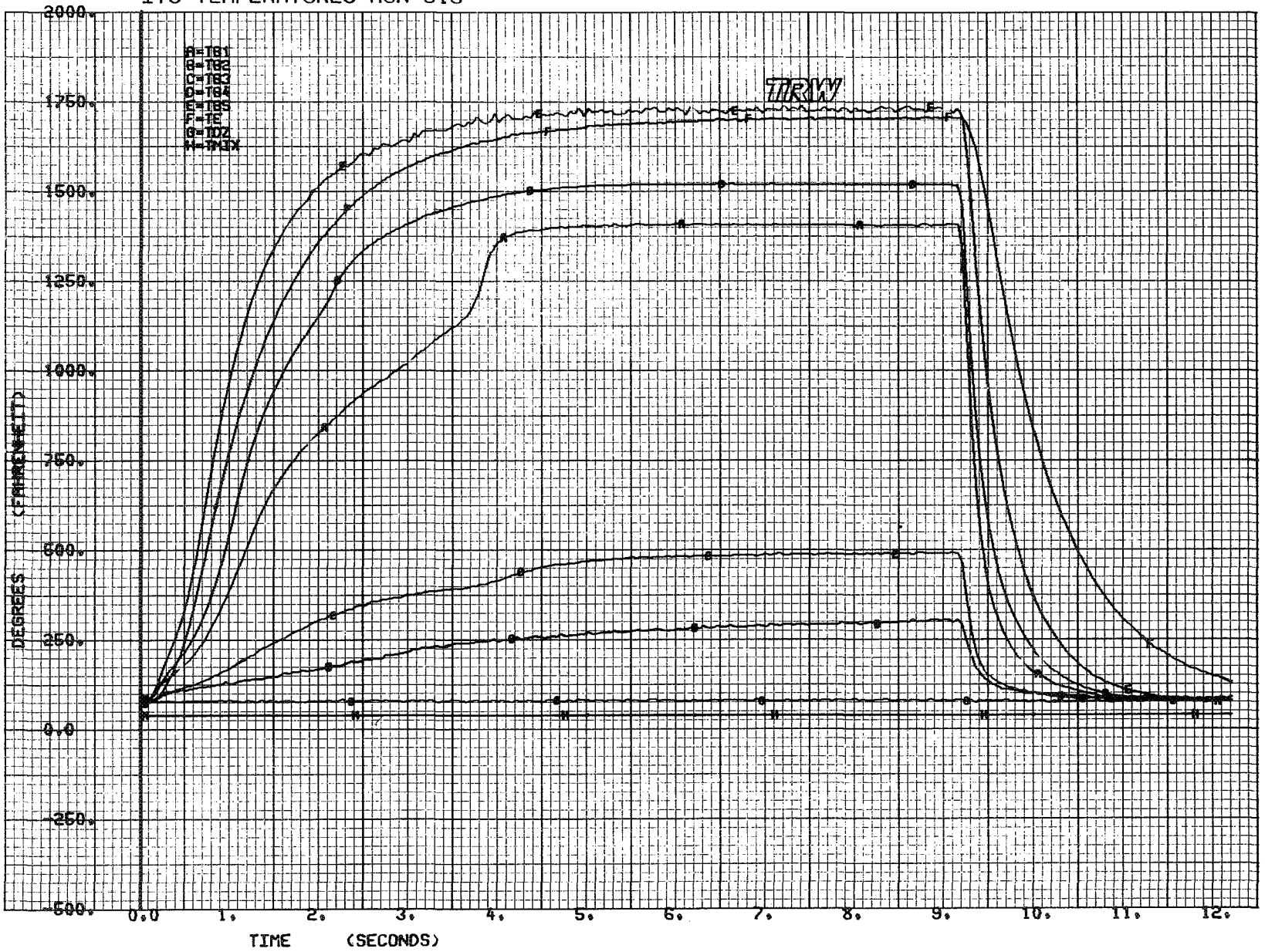
ITS TEMPERATURES RUN 817



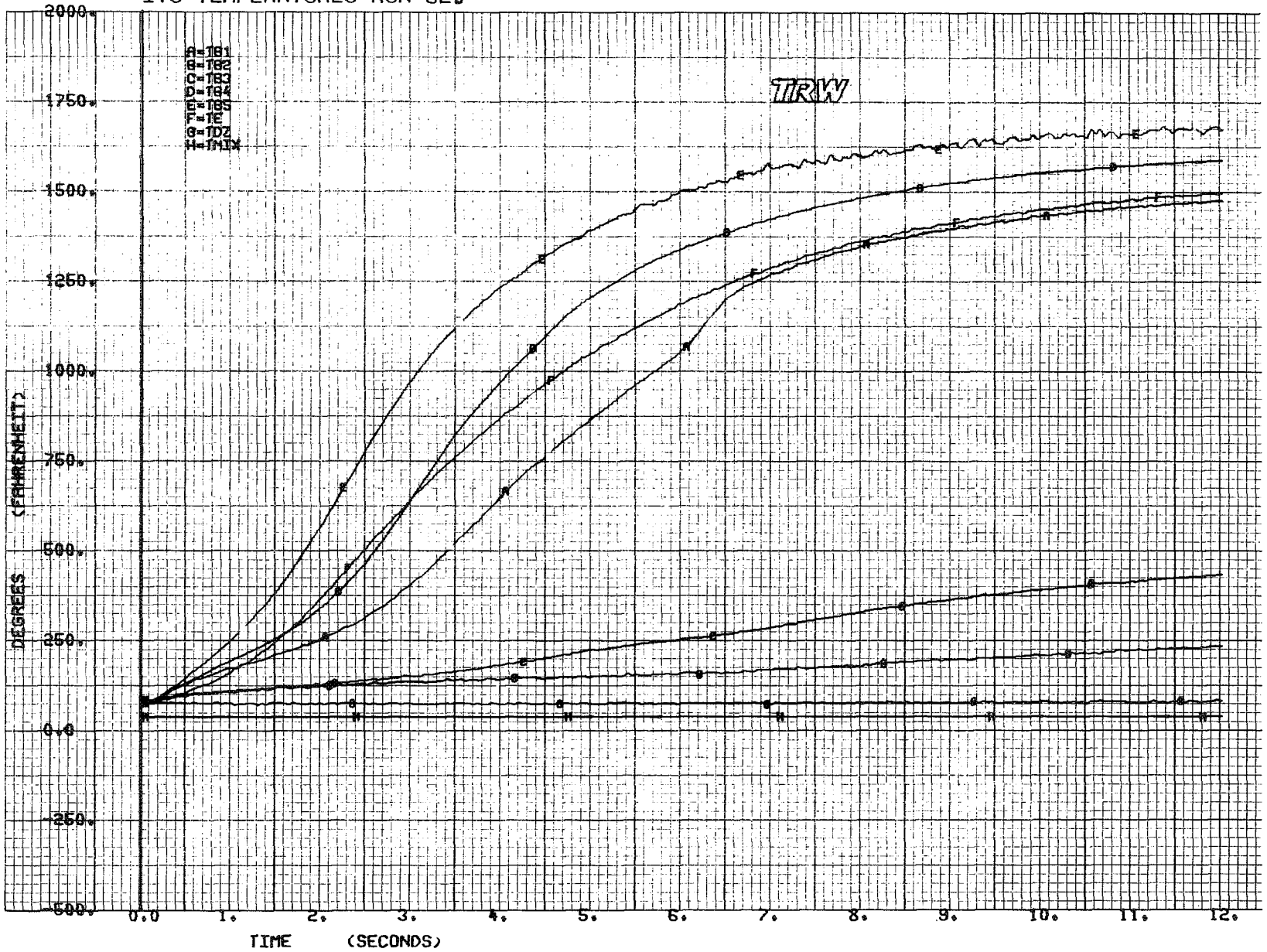
ITS TEMPERATURES RUN 818



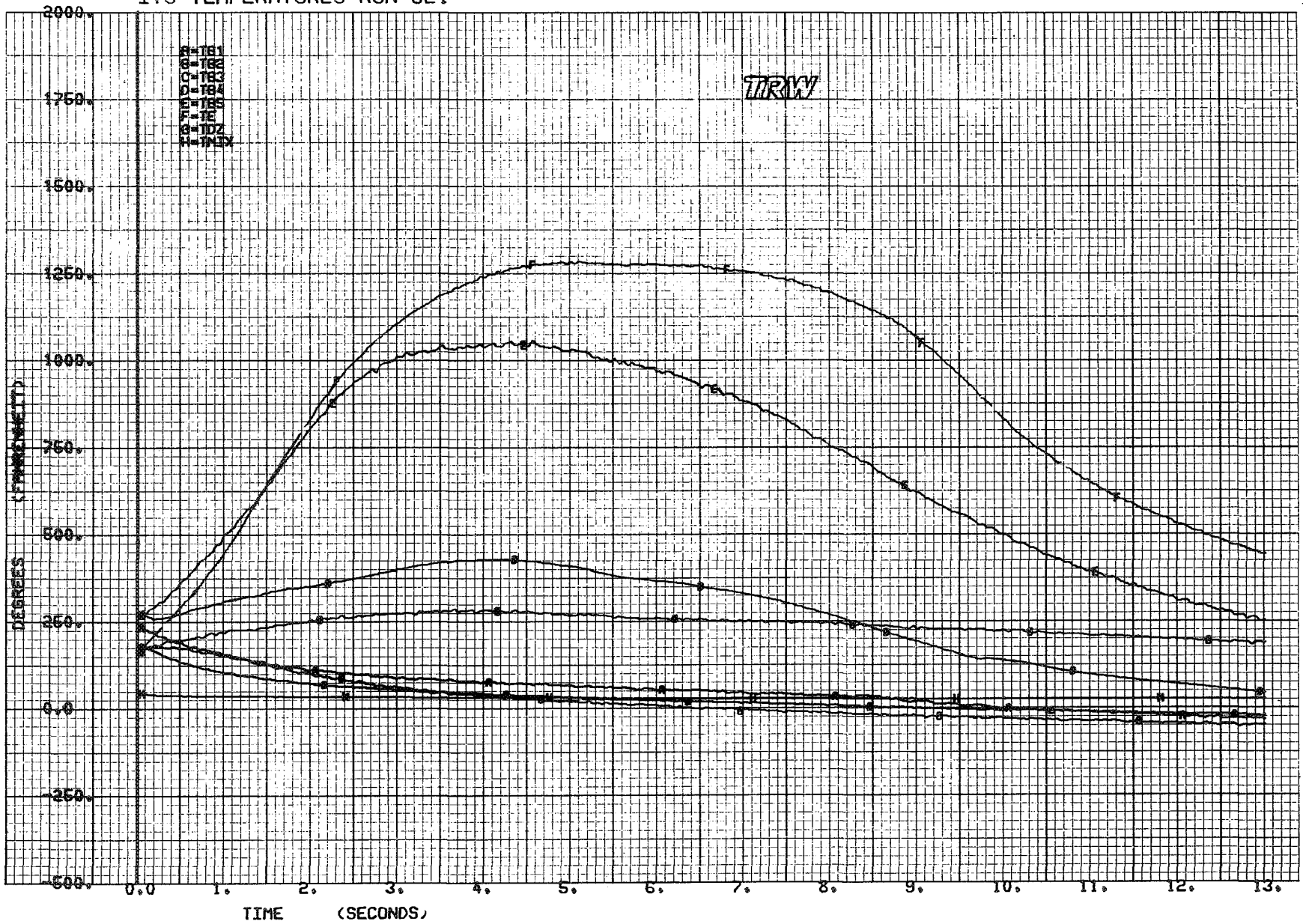
ITS TEMPERATURES RUN 819

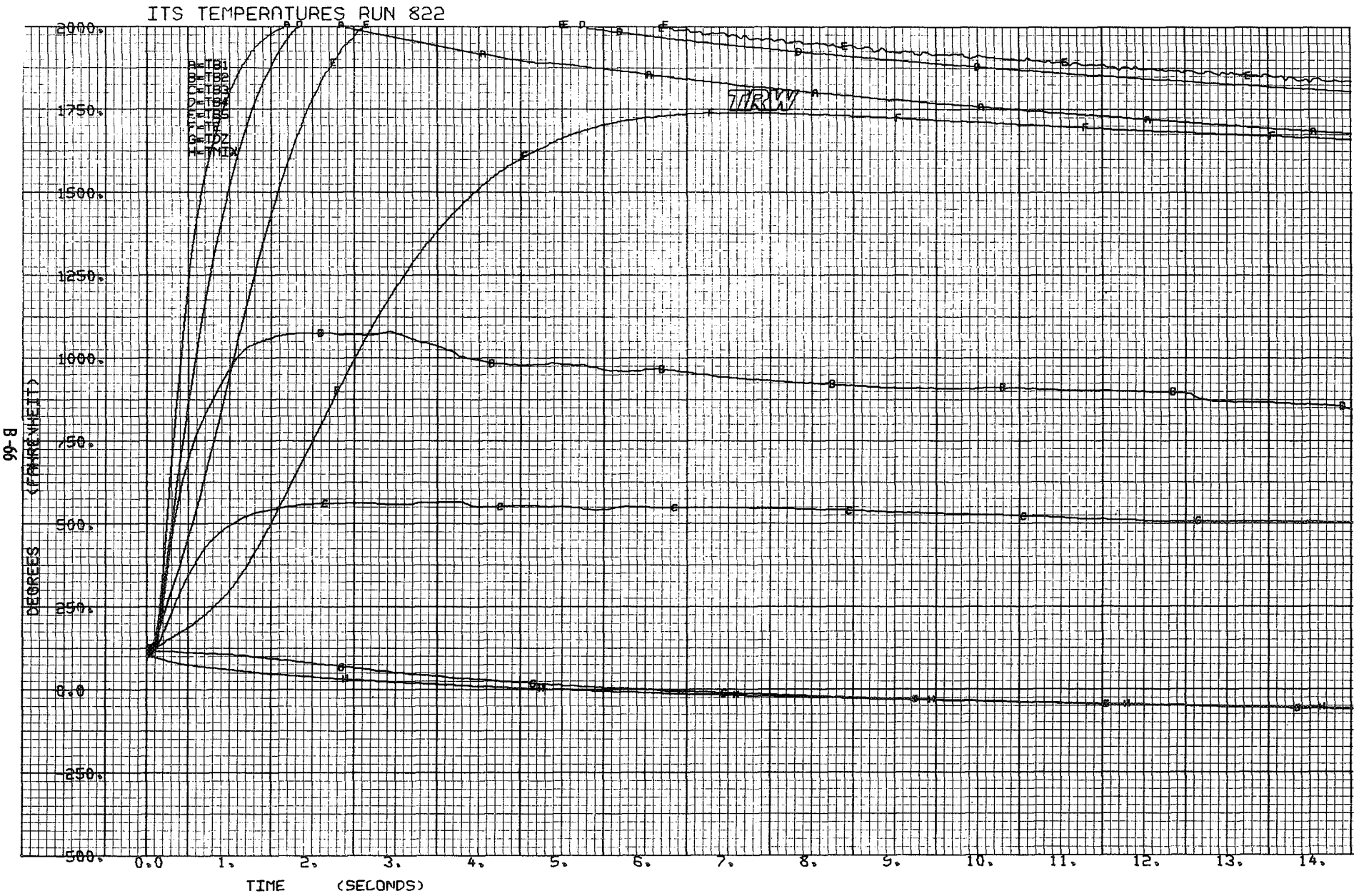


ITS TEMPERATURES RUN 820

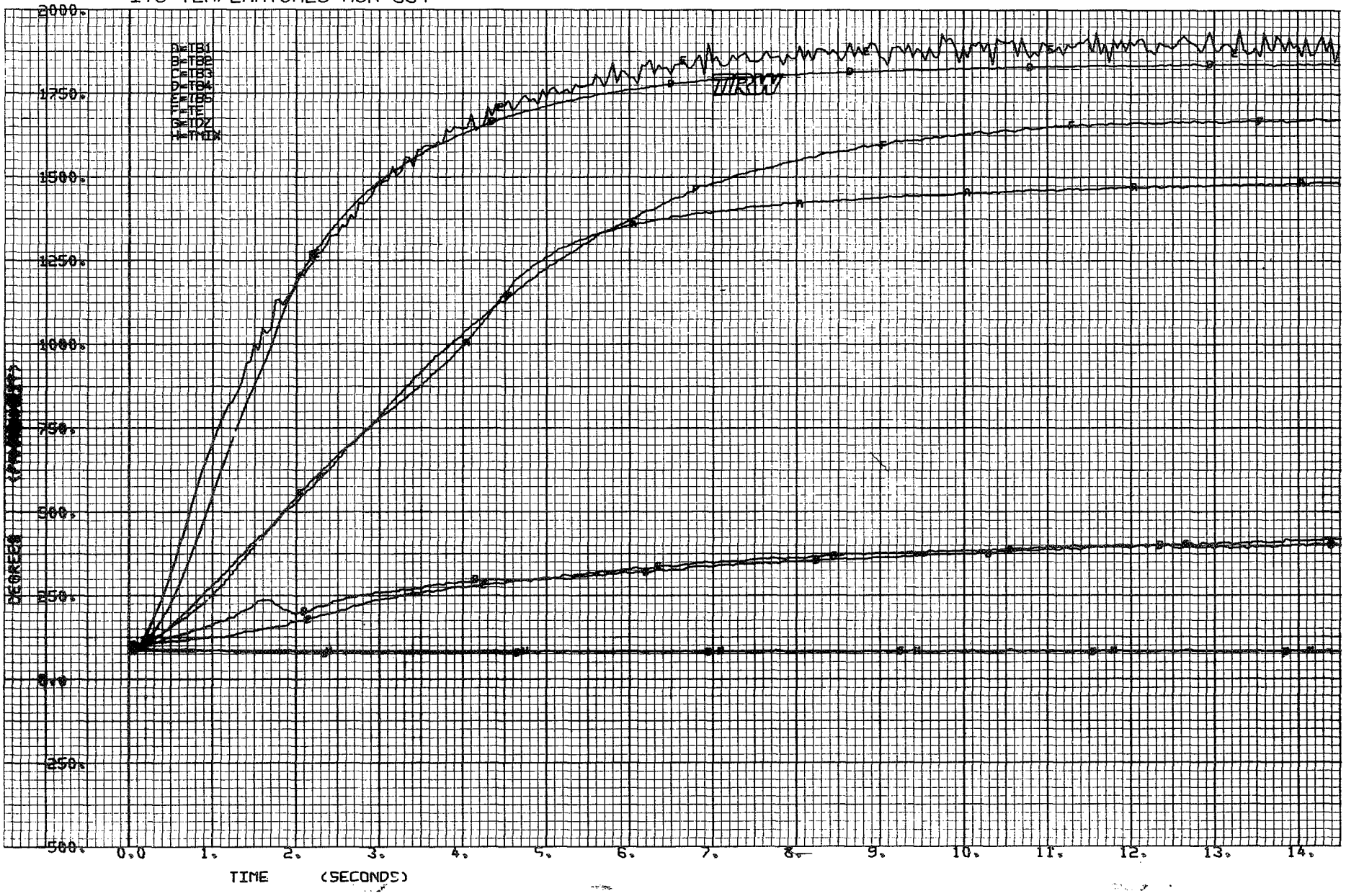


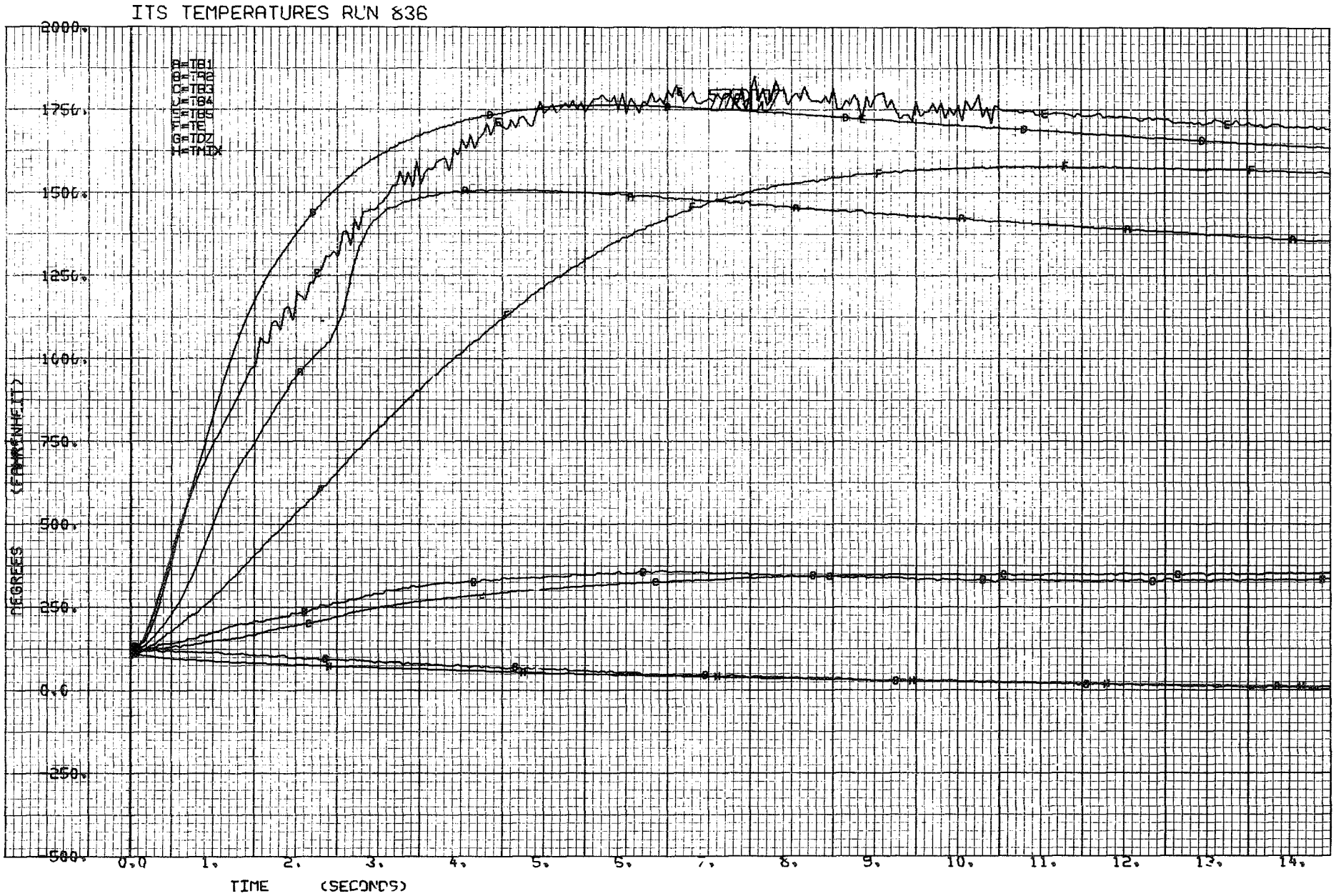
ITS TEMPERATURES RUN 821



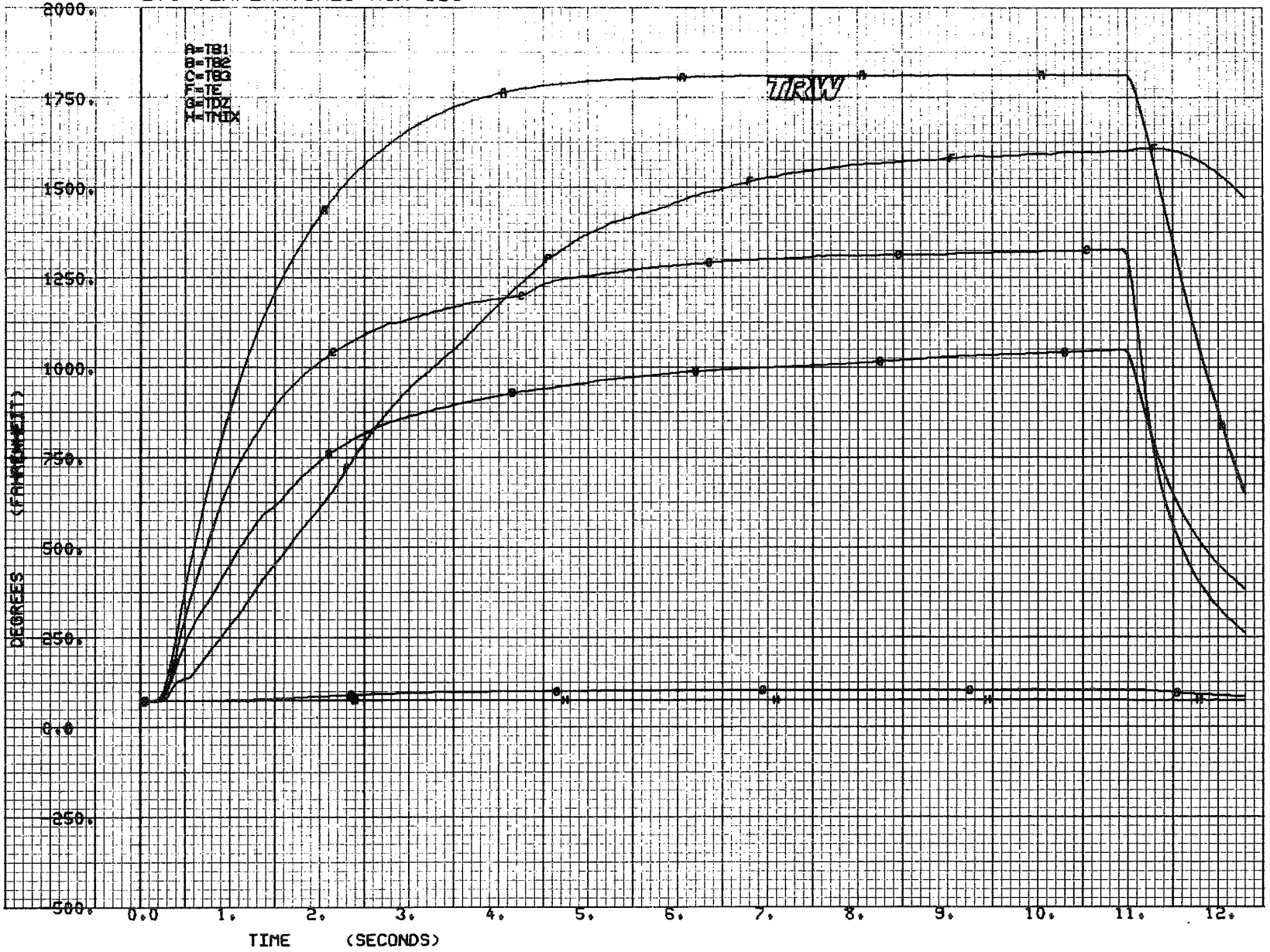

ITS TEMPERATURES RUN 822
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TF
G=TDZ
H=TMIN
TRW
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
TIME (SECONDS)

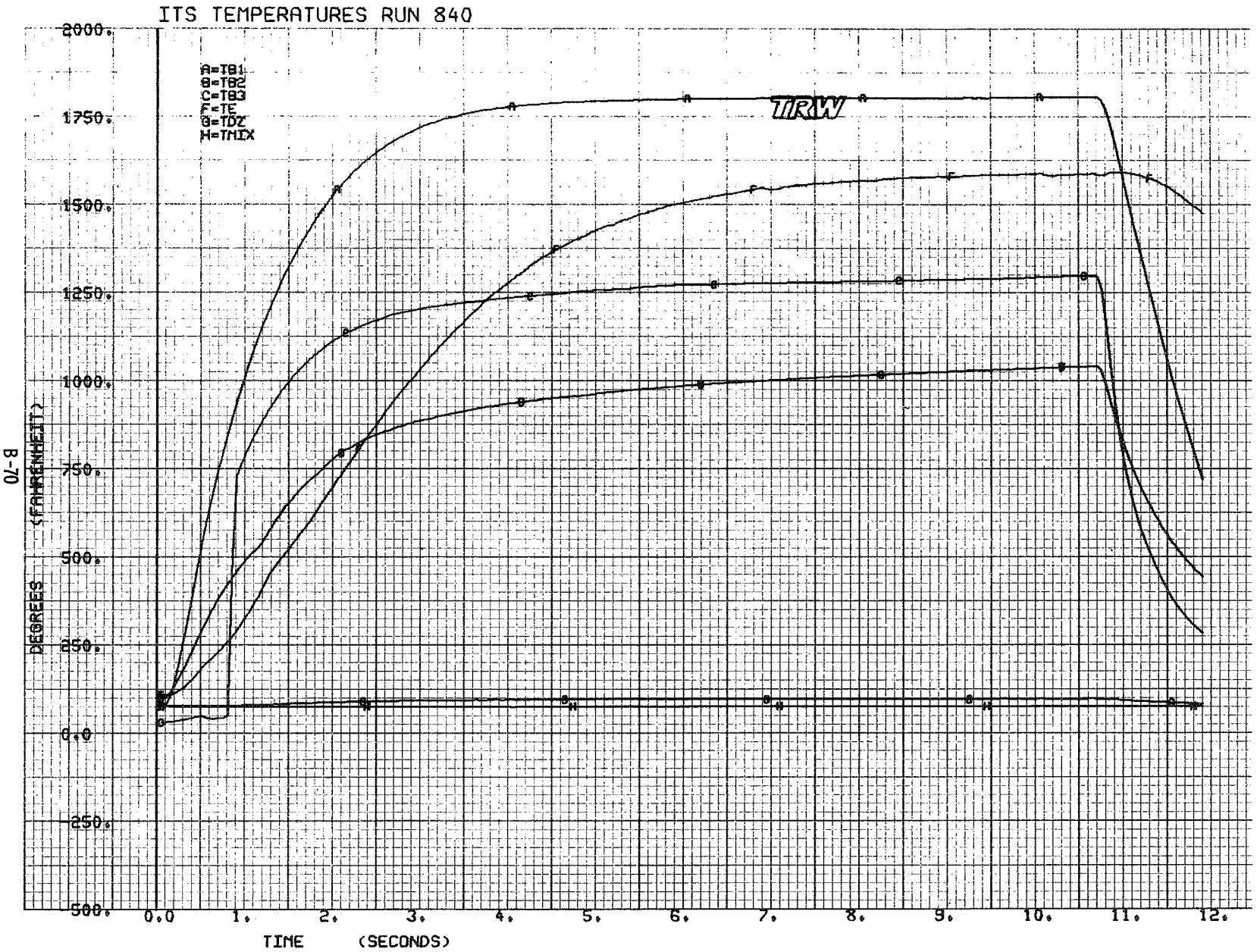
ITS TEMPERATURES RUN 834

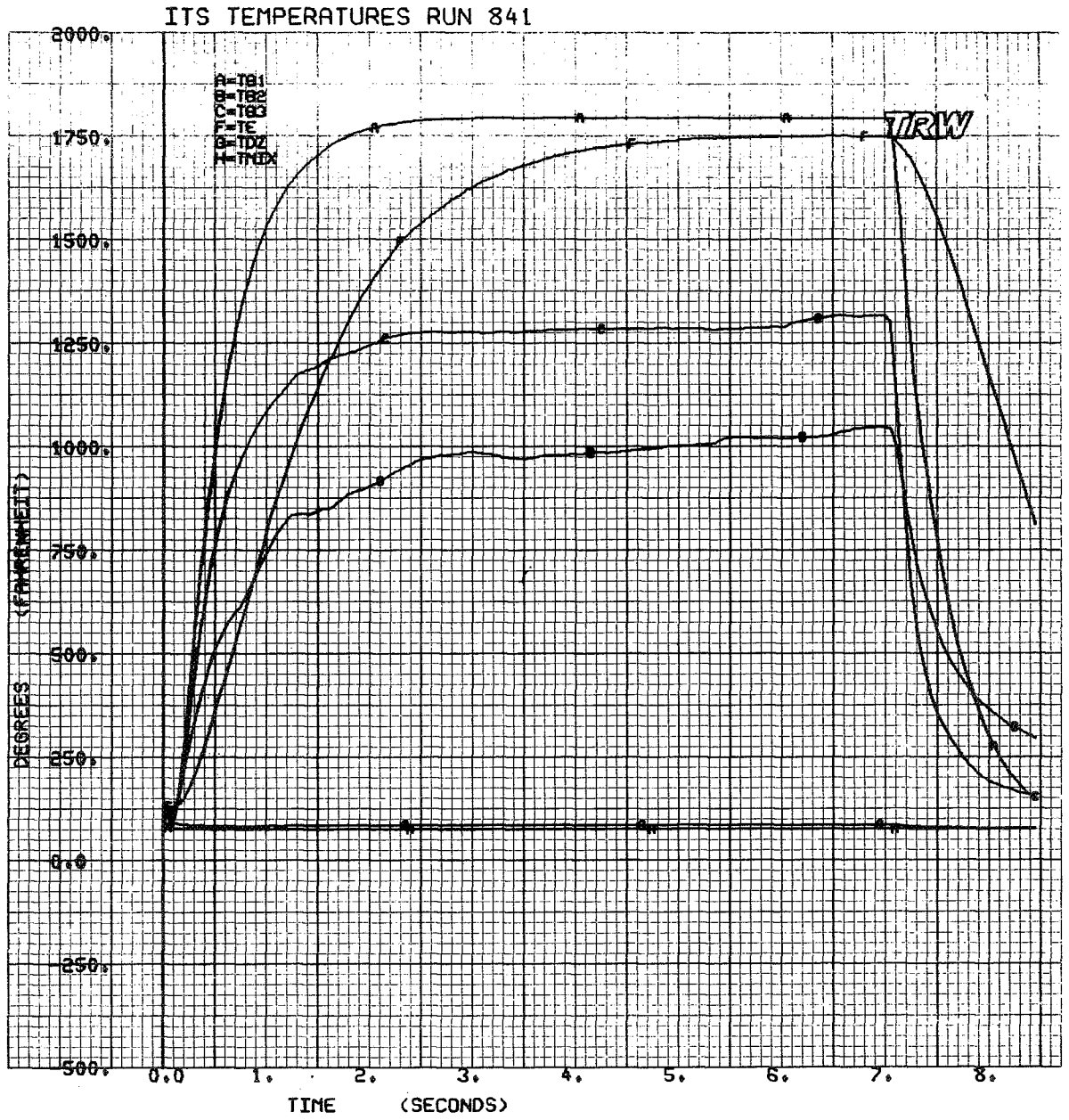



ITS TEMPERATURES RUN 836
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TDZ
H=TMIX
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
TIME (SECONDS)

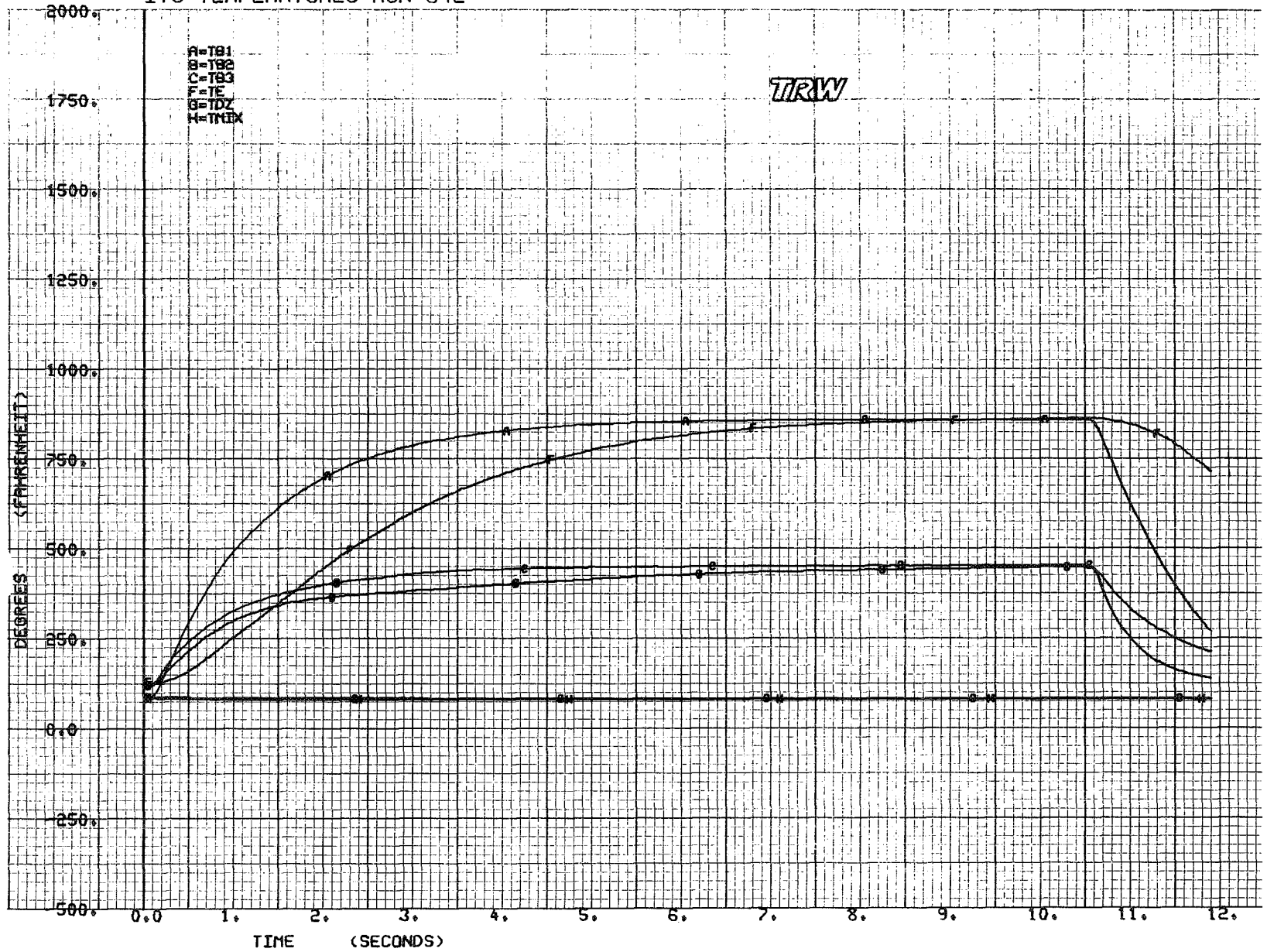
ITS TEMPERATURES RUN 839

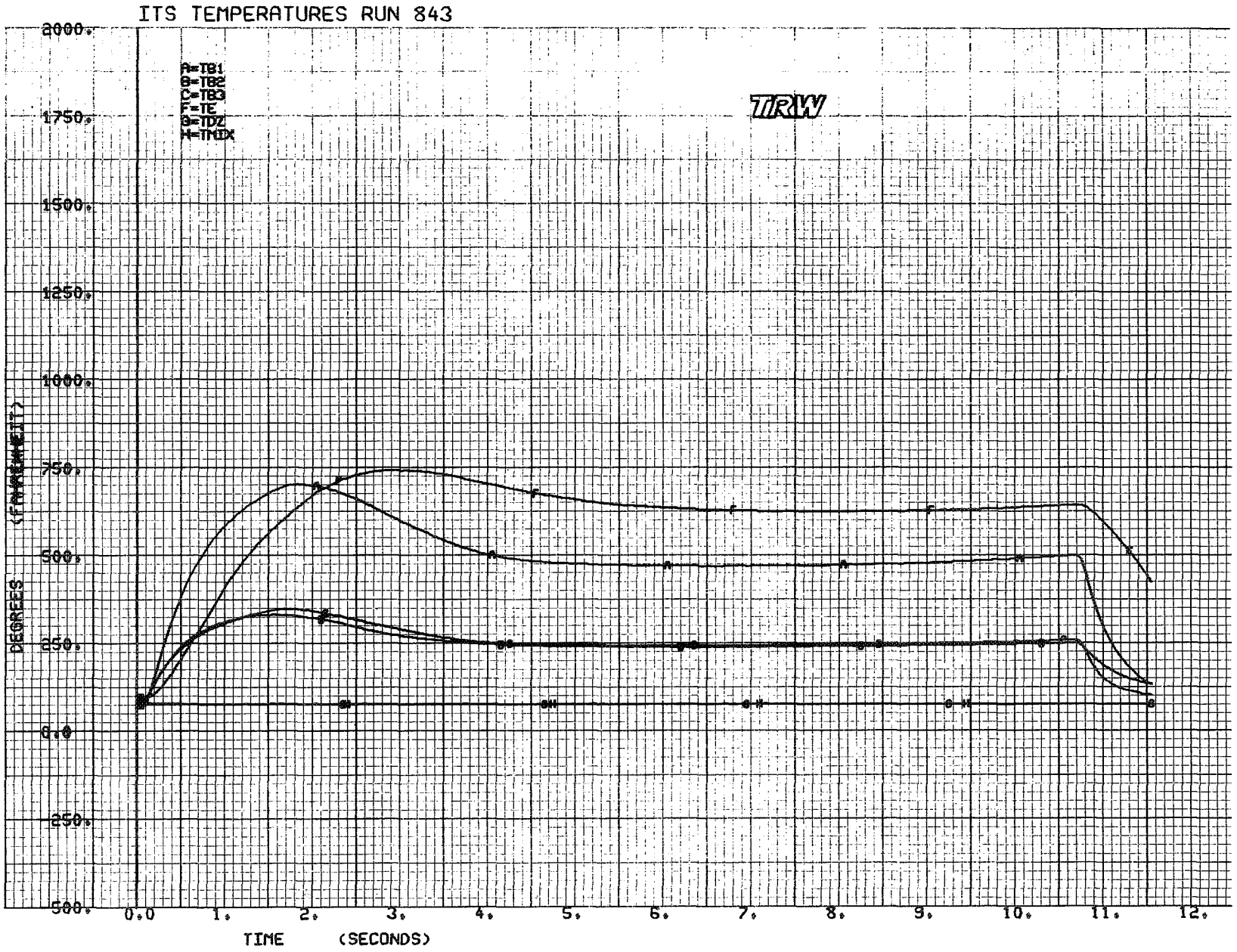


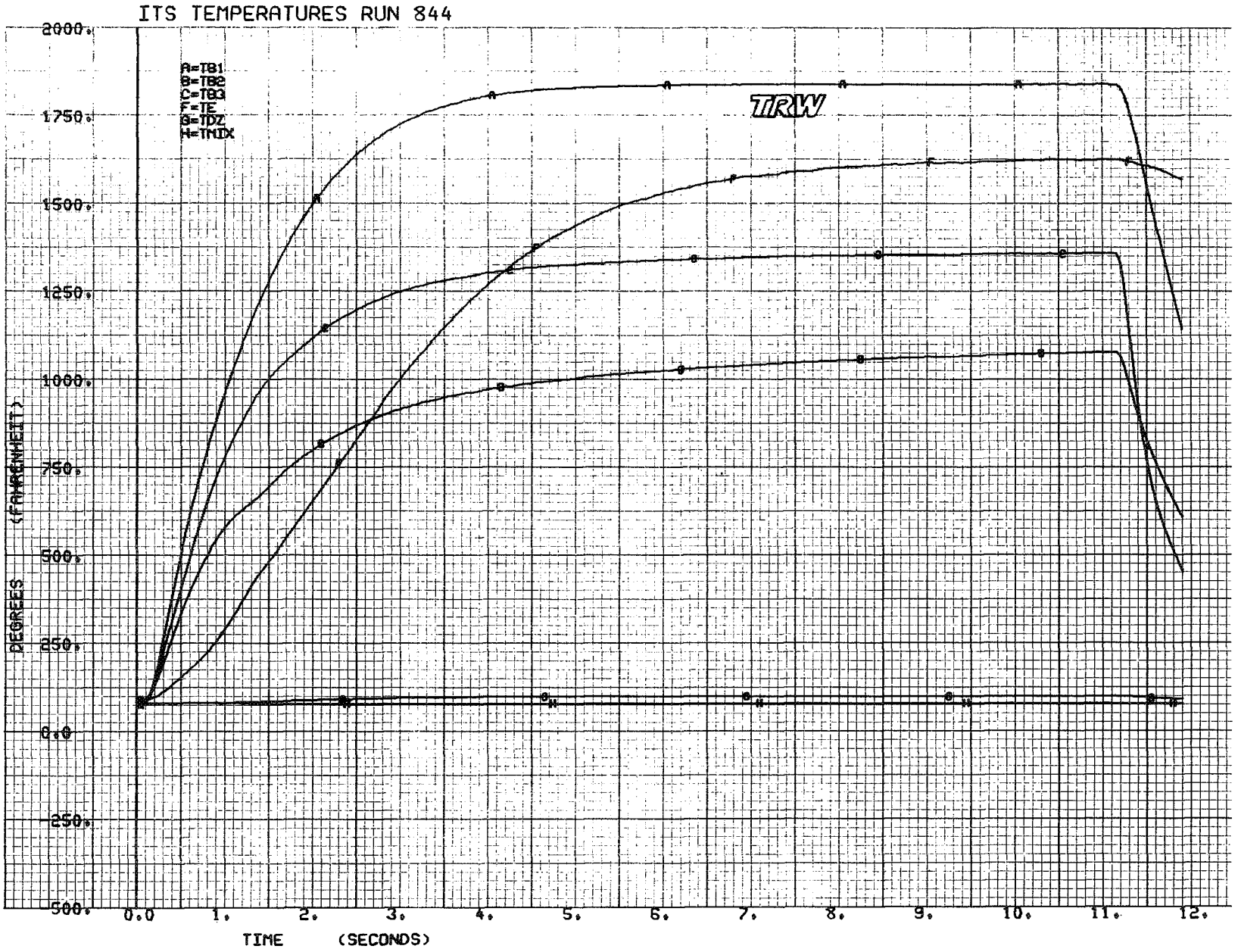

ITS TEMPERATURES RUN 840
A=T01
B=T02
C=T03
F=TE
G=TDZ
H=TMIX
TRW
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
TIME (SECONDS)


ITS TEMPERATURES RUN 841
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=THDX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
TIME (SECONDS)

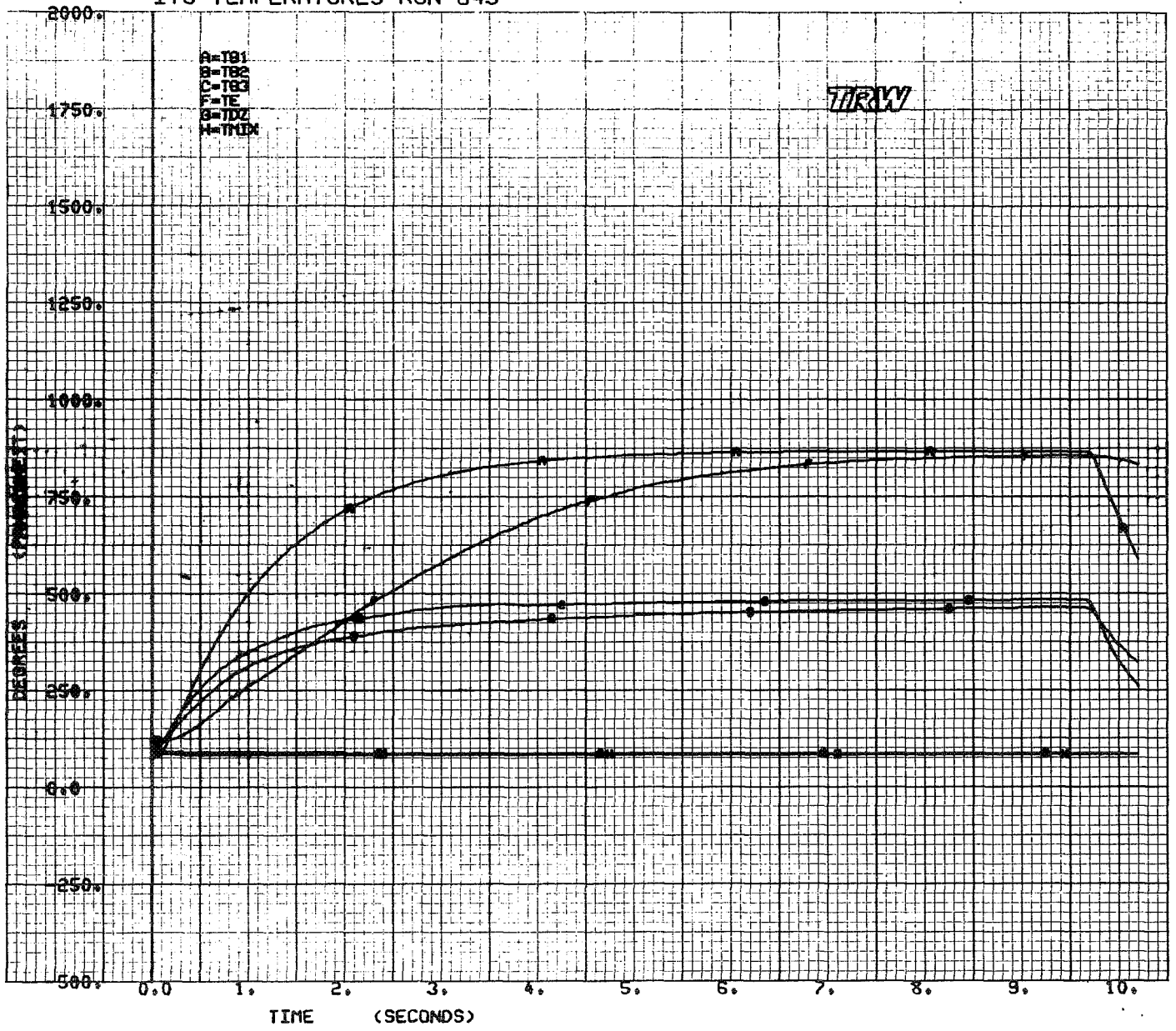
ITS TEMPERATURES RUN 842

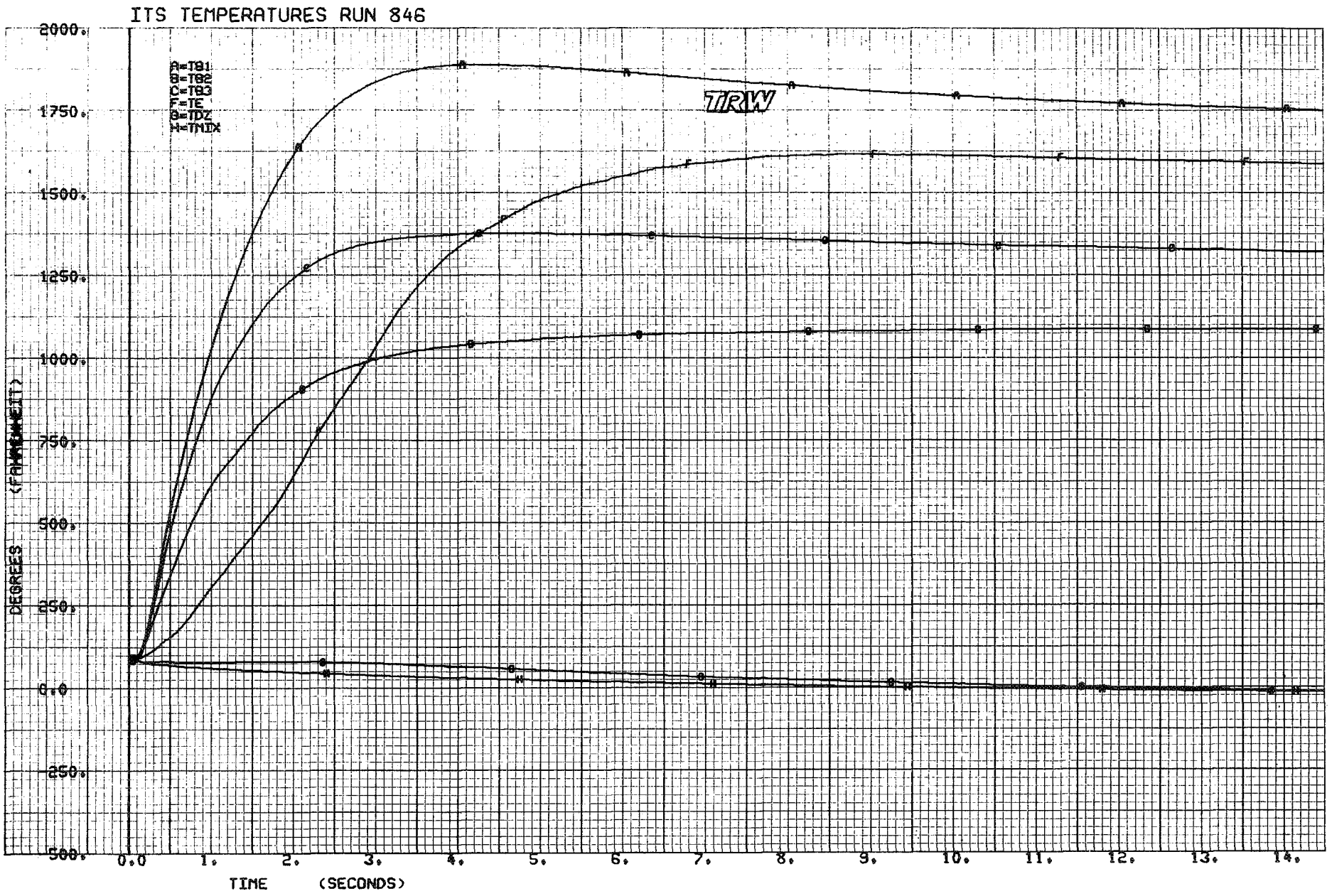


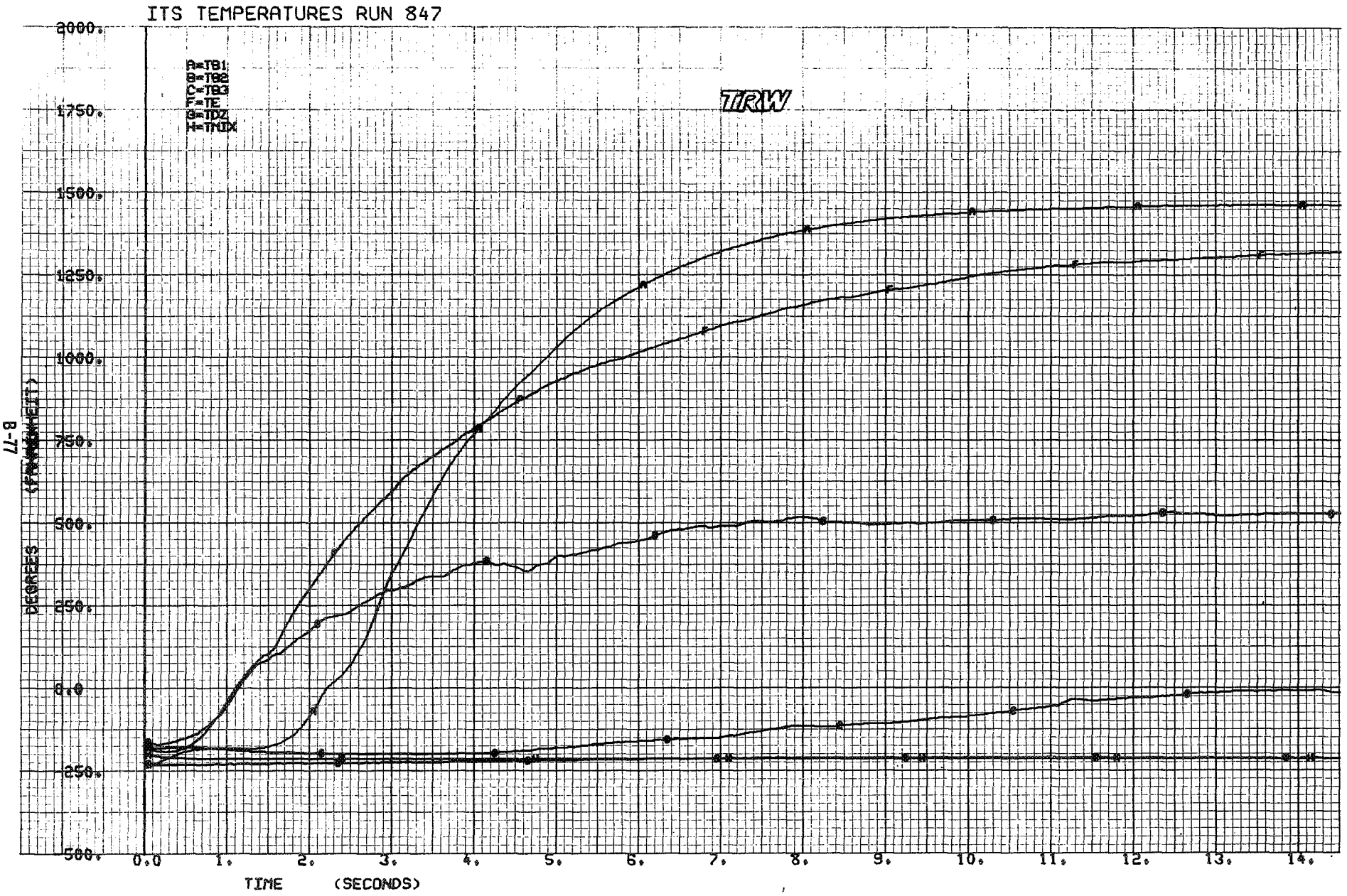

ITS TEMPERATURES RUN 843
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
TIME (SECONDS)

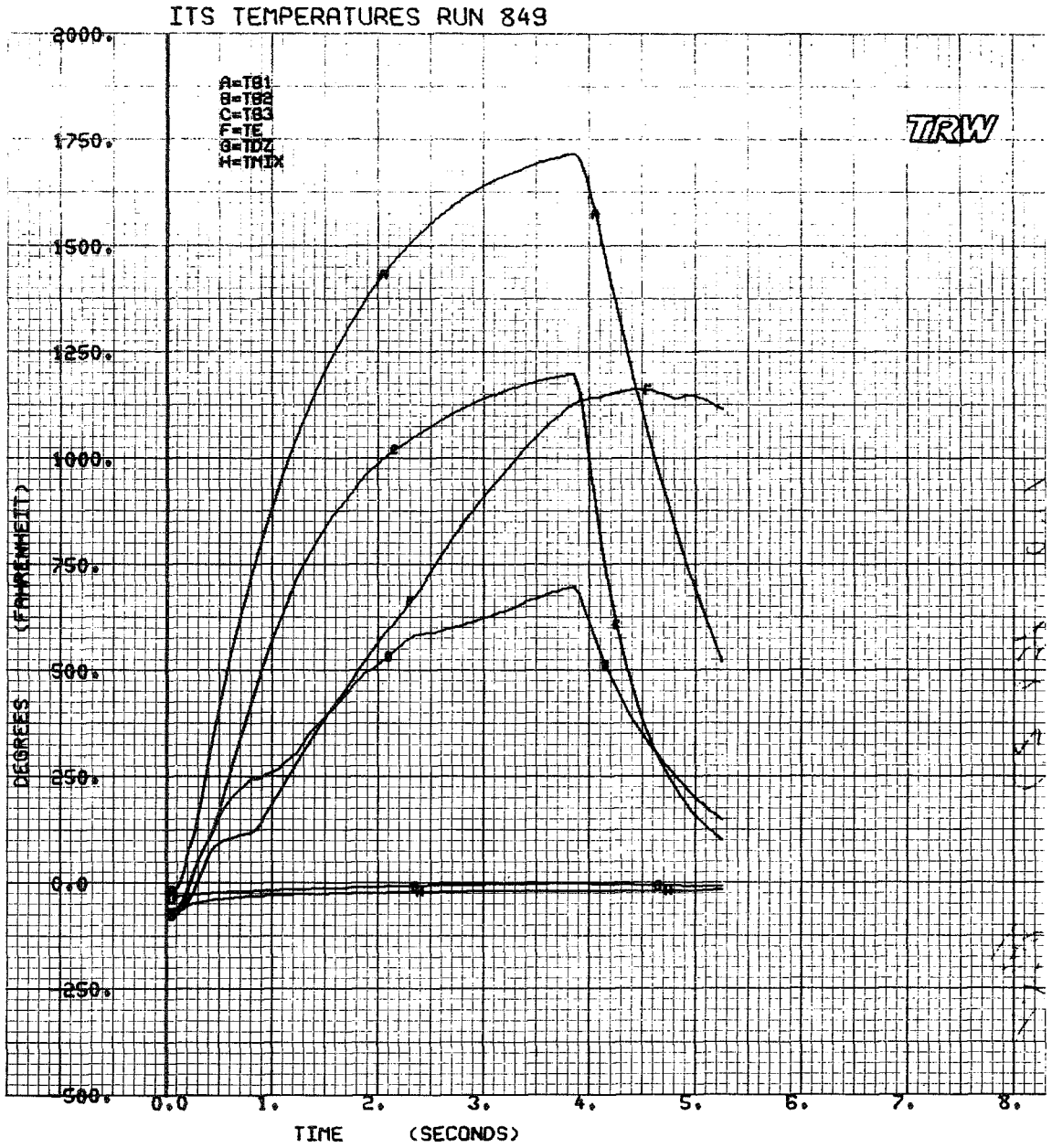

ITS TEMPERATURES RUN 844
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
TIME (SECONDS)

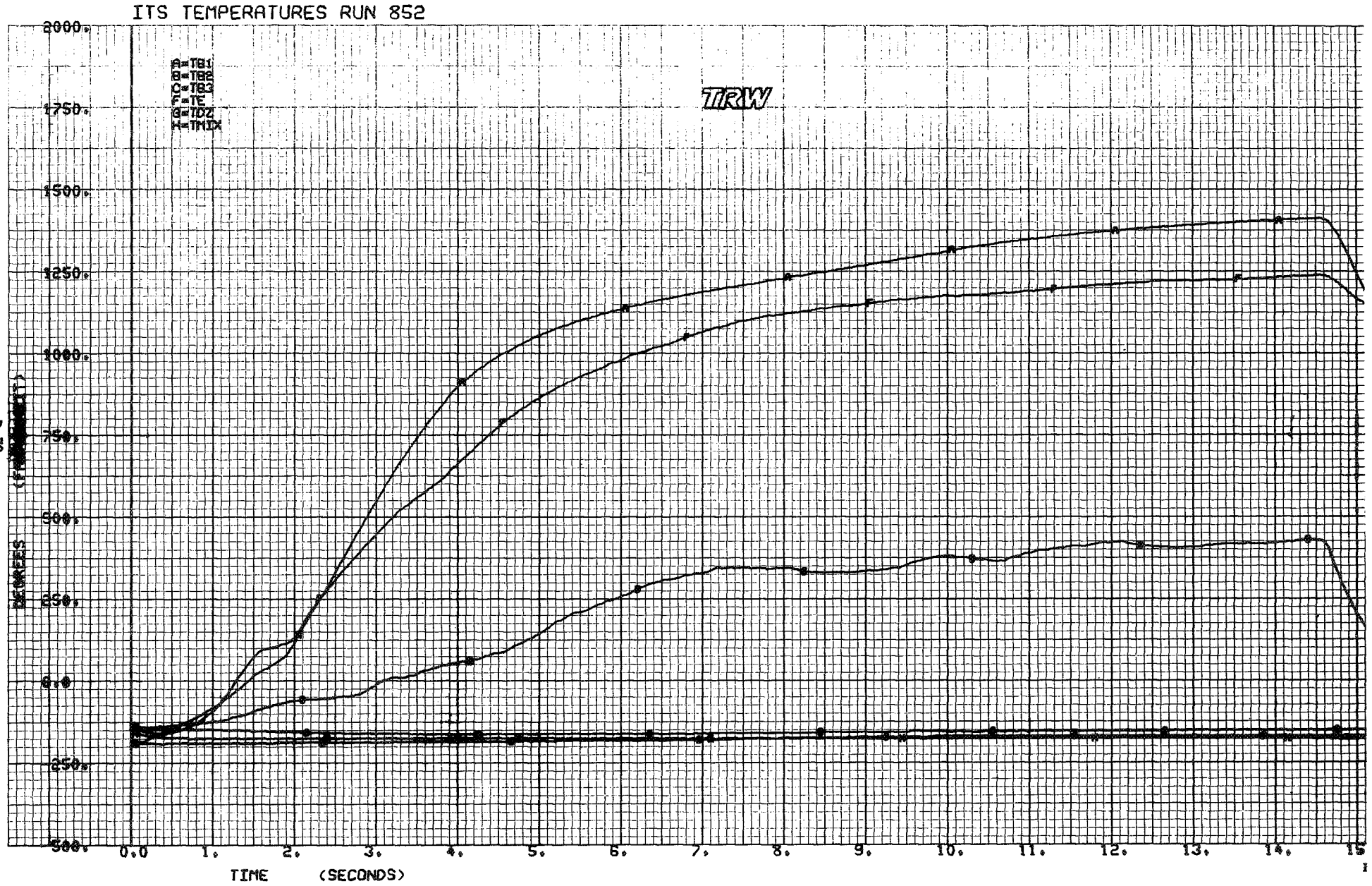
ITS TEMPERATURES RUN 845



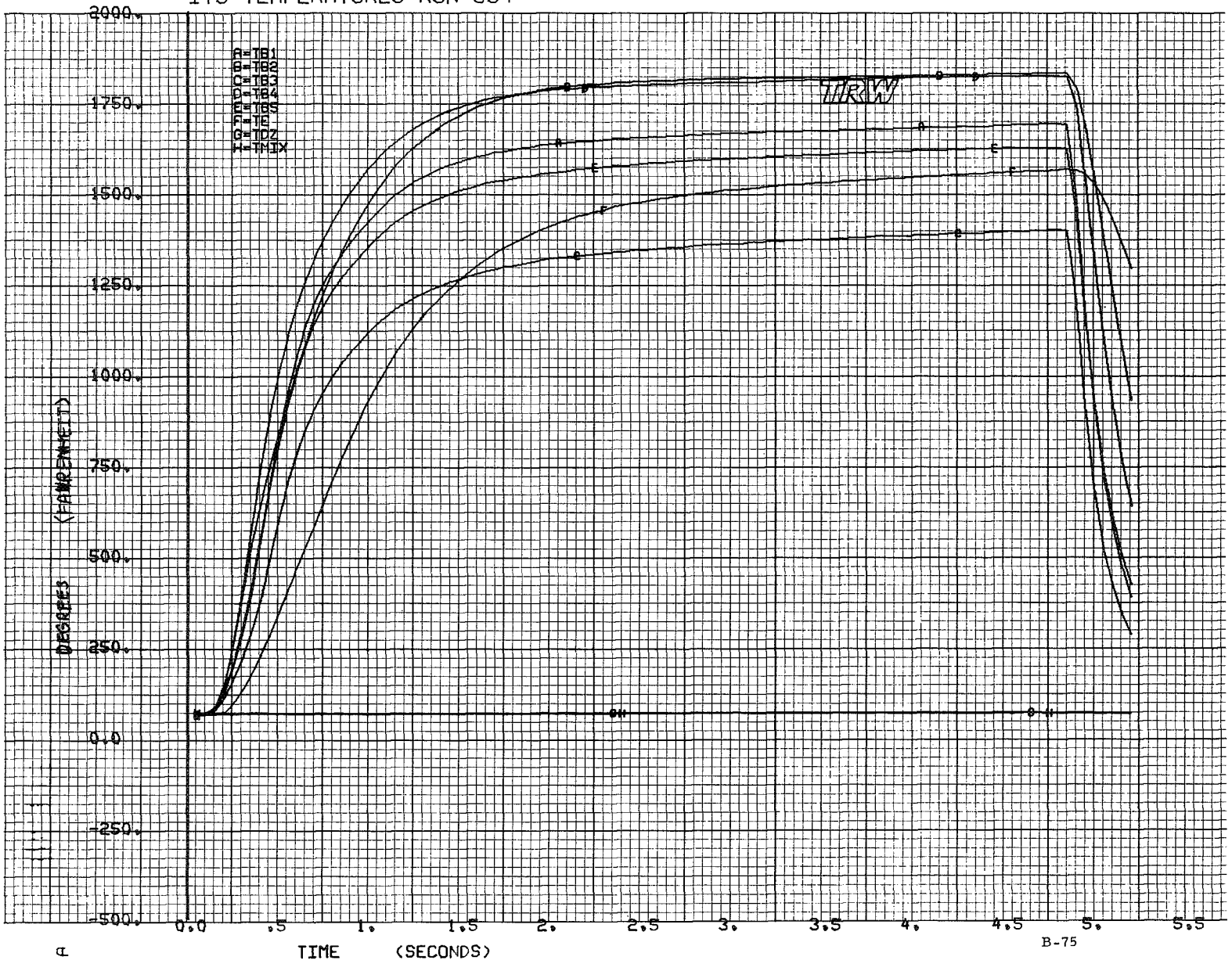

ITS TEMPERATURES RUN 846
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
TIME (SECONDS)


ITS TEMPERATURES RUN 847
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
TIME (SECONDS)


ITS TEMPERATURES RUN 849
A=T81
B=T82
C=T83
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
1.
2.
3.
4.
5.
6.
7.
8.
TIME (SECONDS)

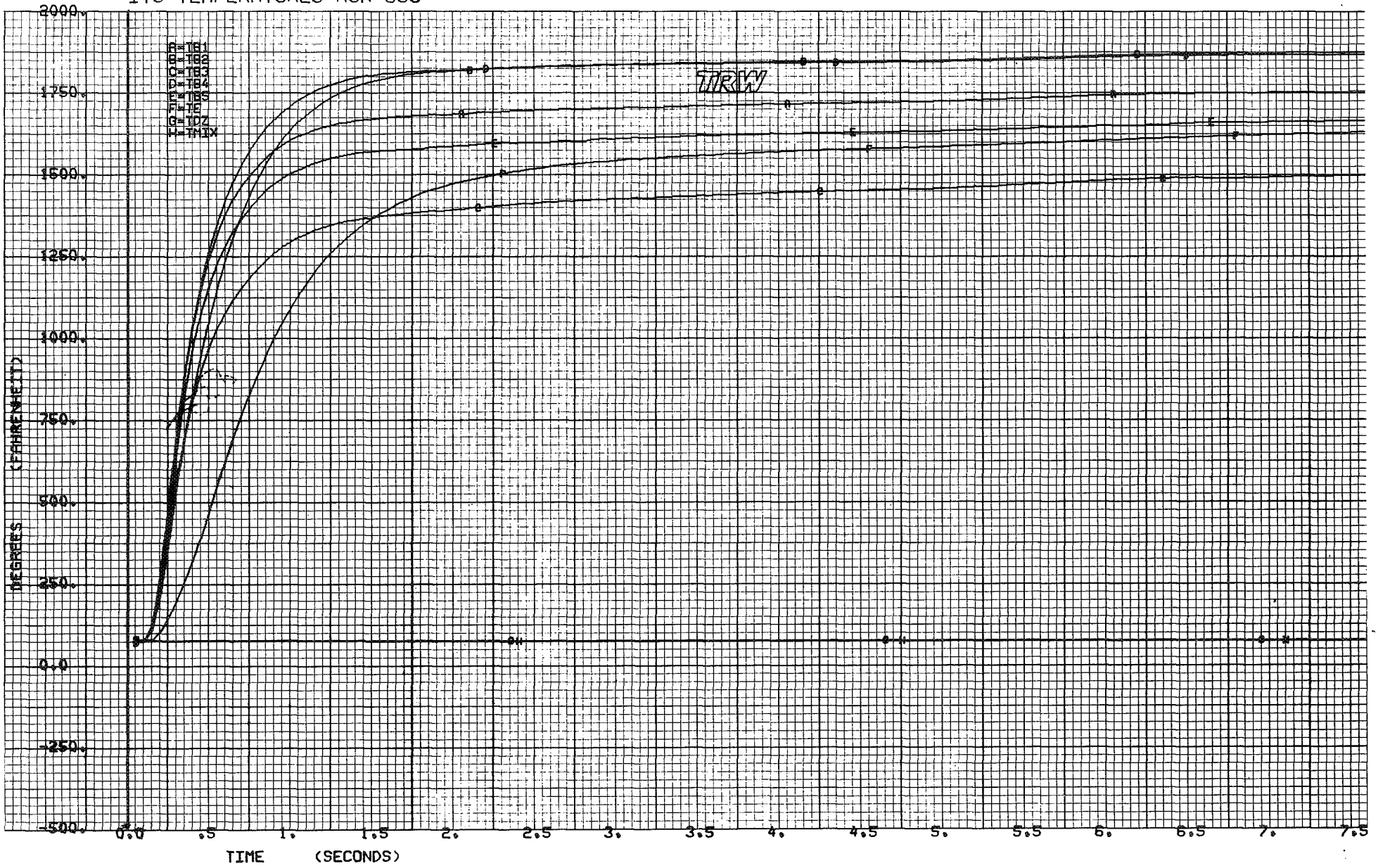

ITS TEMPERATURES RUN 852
A=TB1
B=TB2
C=TB3
F=TE
G=TD2
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
1.
2.
3.
4.
5.
6.
7.
8.
9.
10.
11.
12.
13.
14.
15
TIME (SECONDS)

ITS TEMPERATURES RUN 854

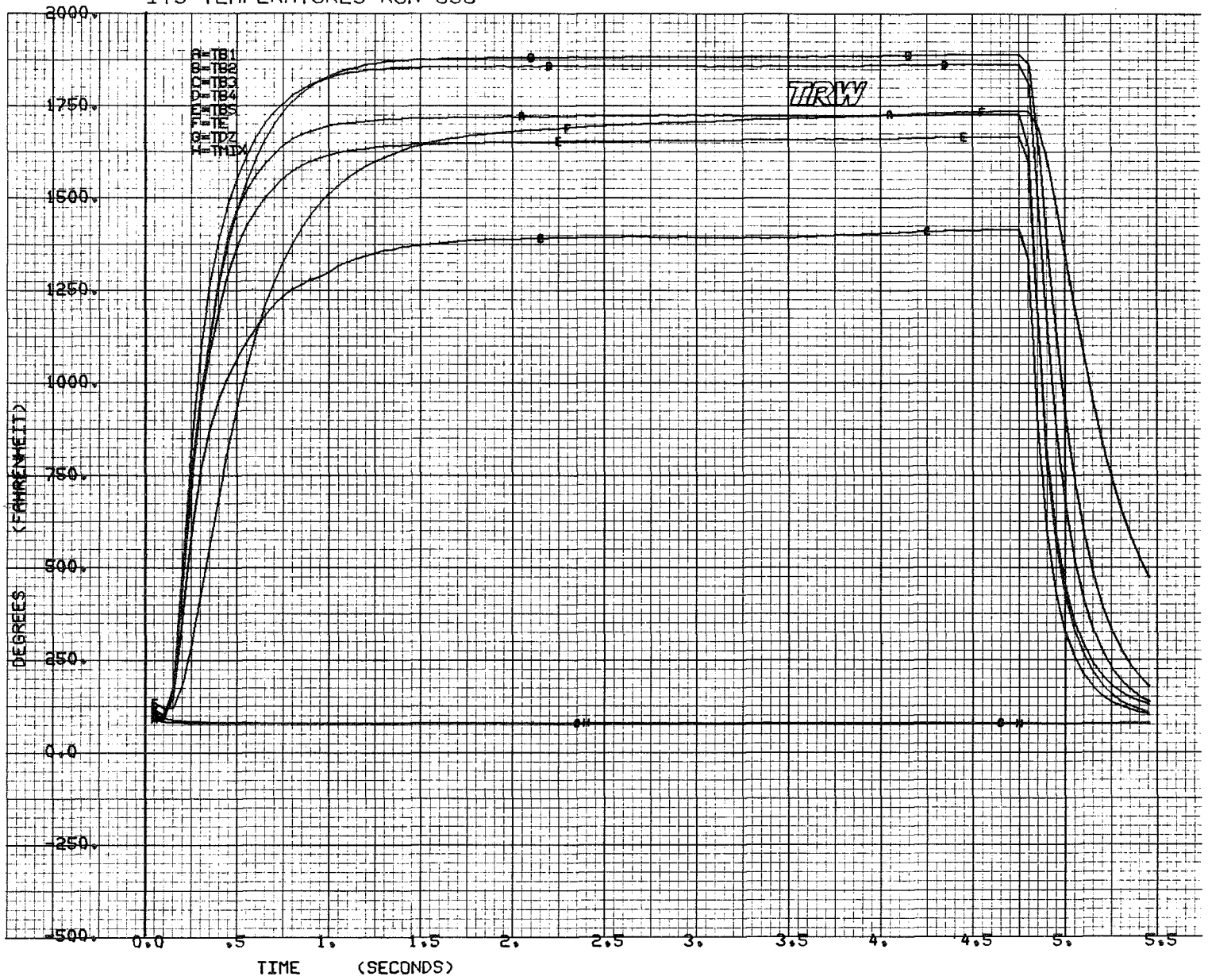


B-75

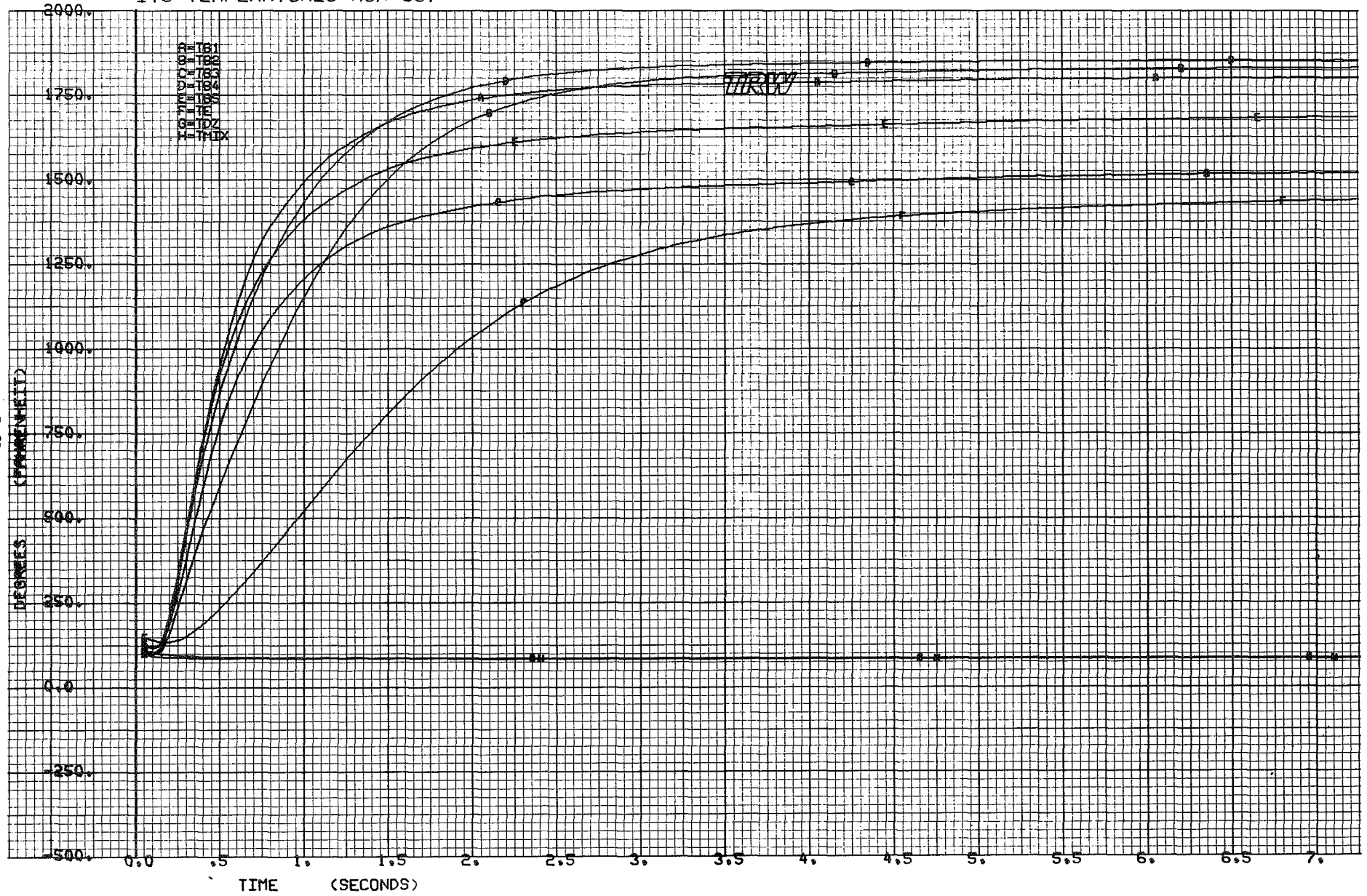
ITS TEMPERATURES RUN 855



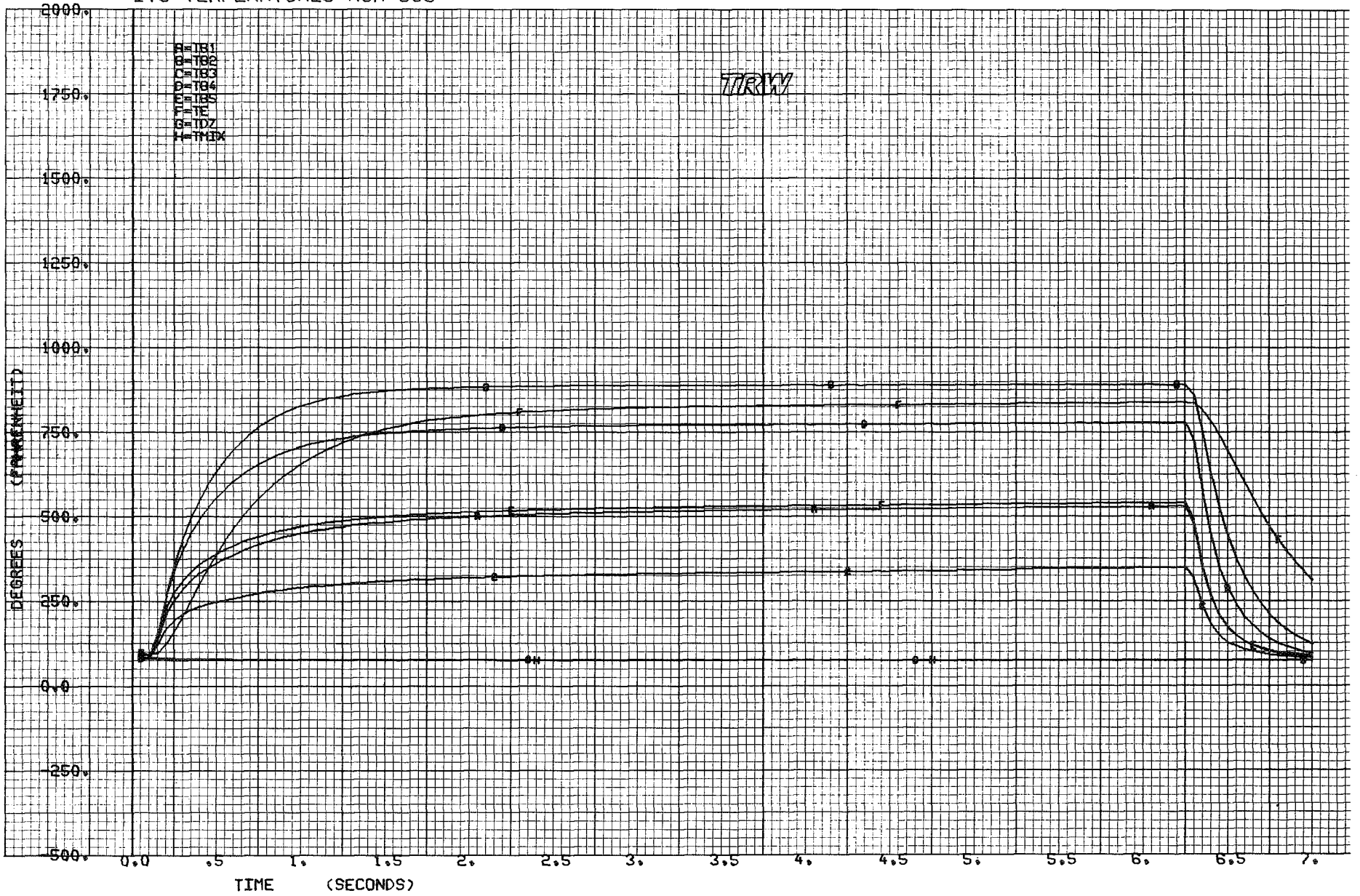
ITS TEMPERATURES RUN 856



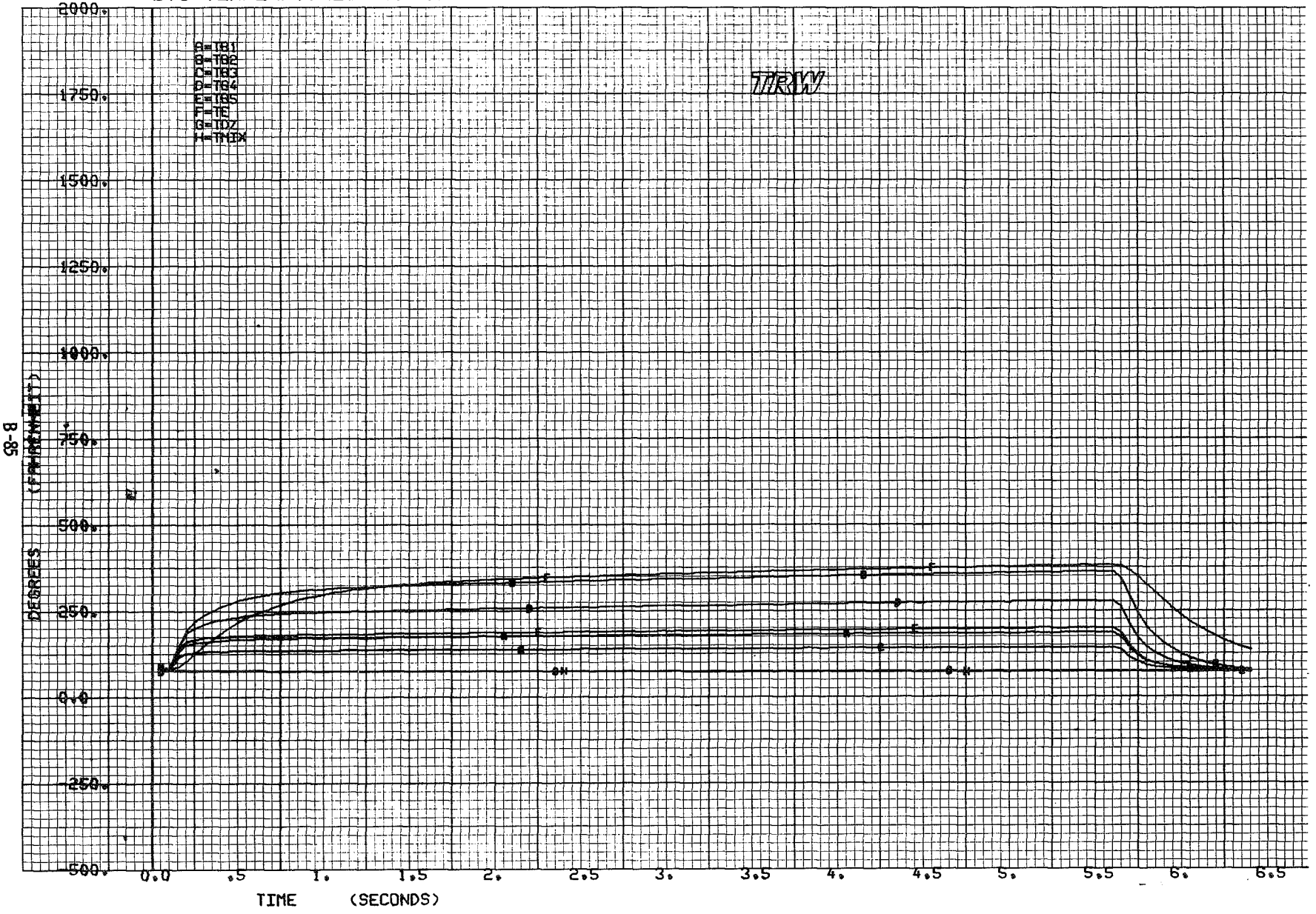
ITS TEMPERATURES RUN 857



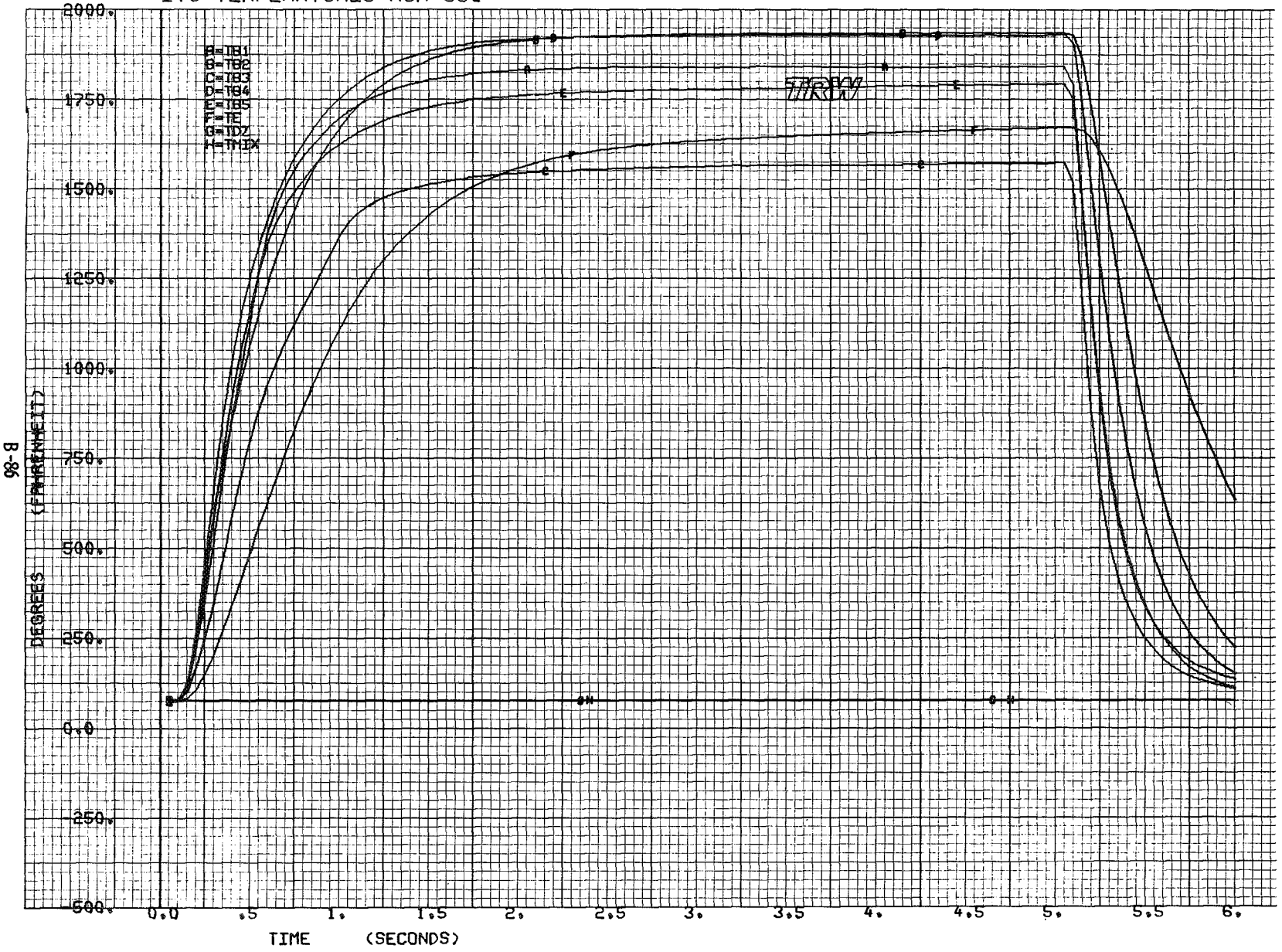
ITS TEMPERATURES RUN 858



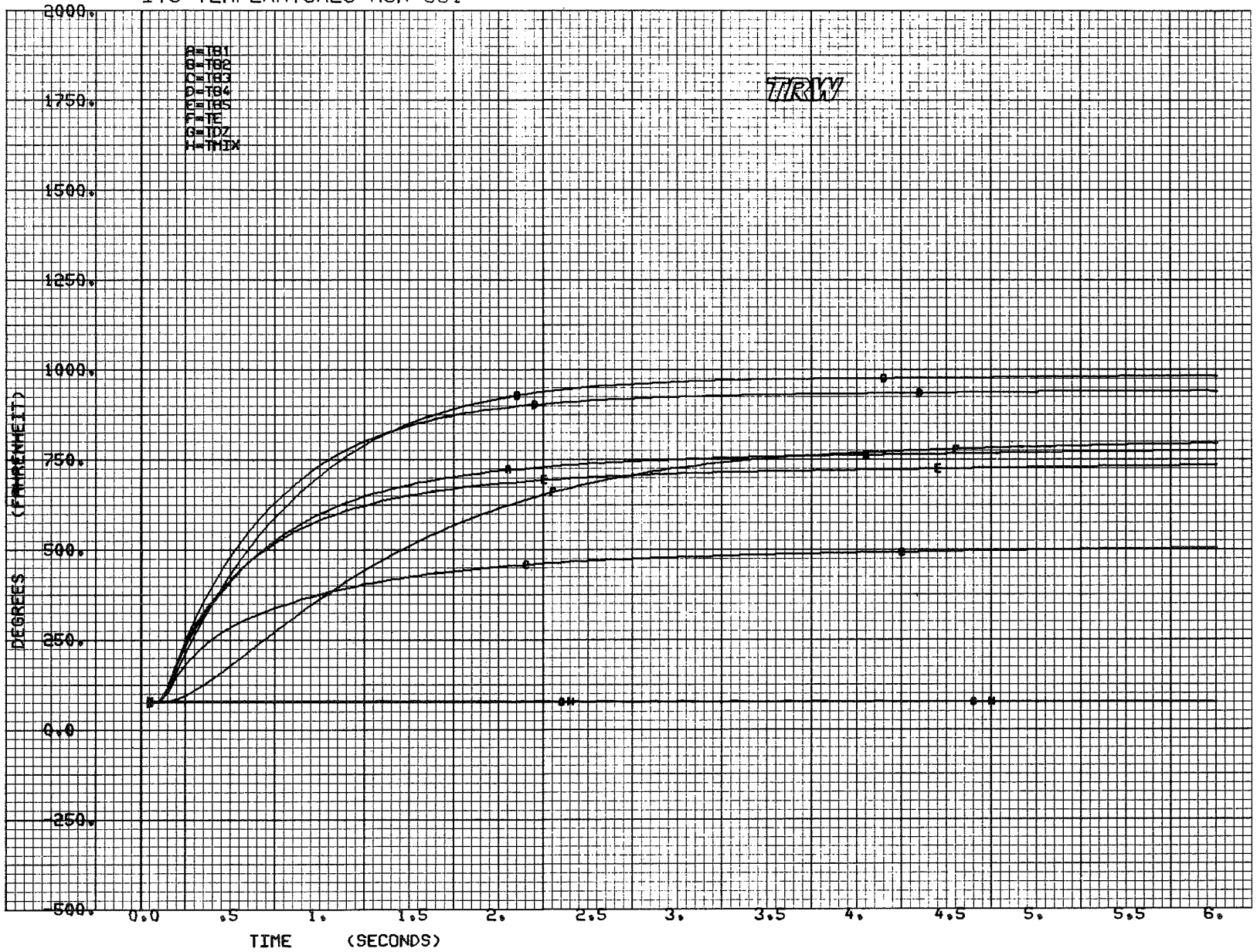
ITS TEMPERATURES RUN 859



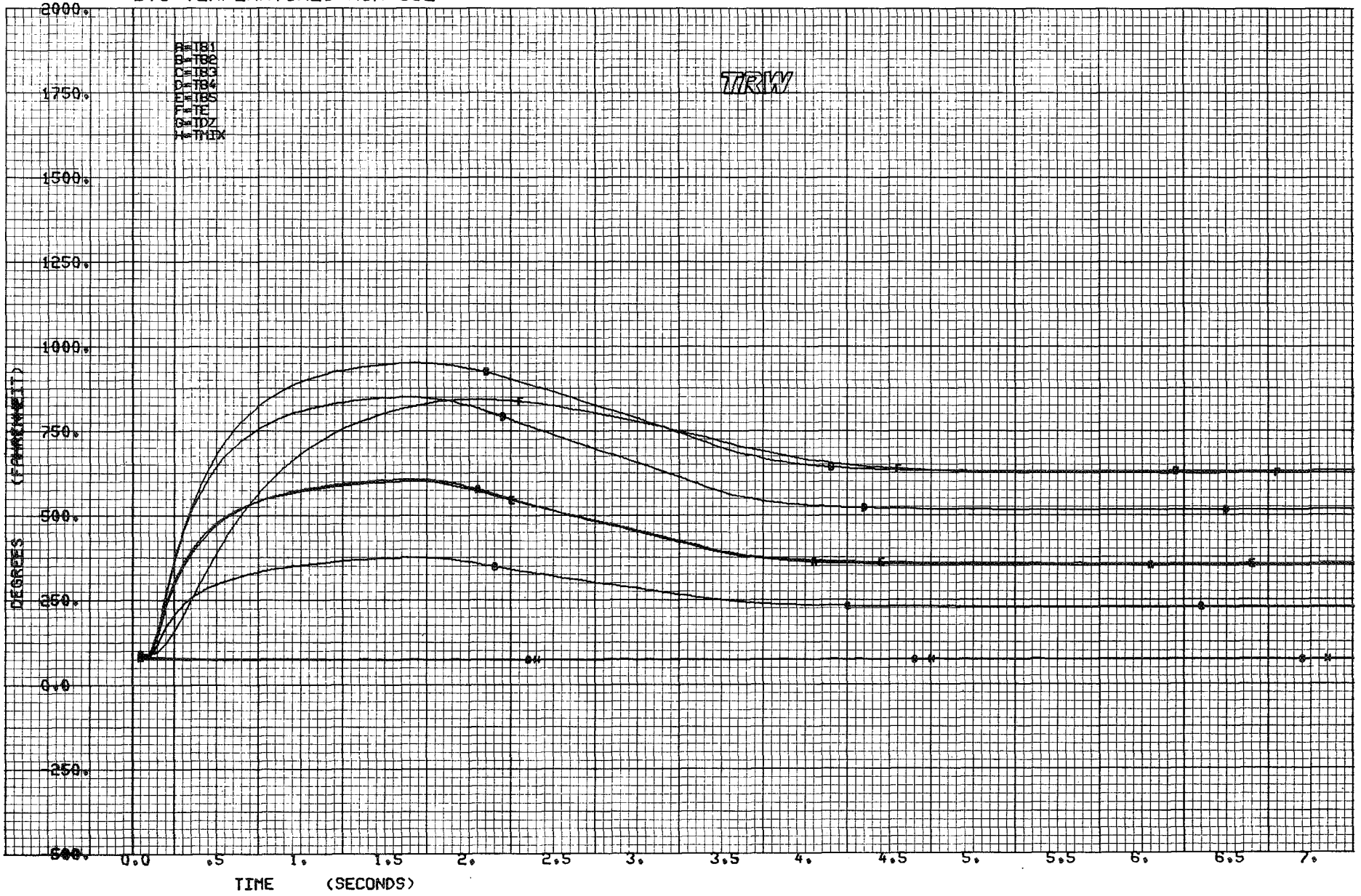
ITS TEMPERATURES RUN 860



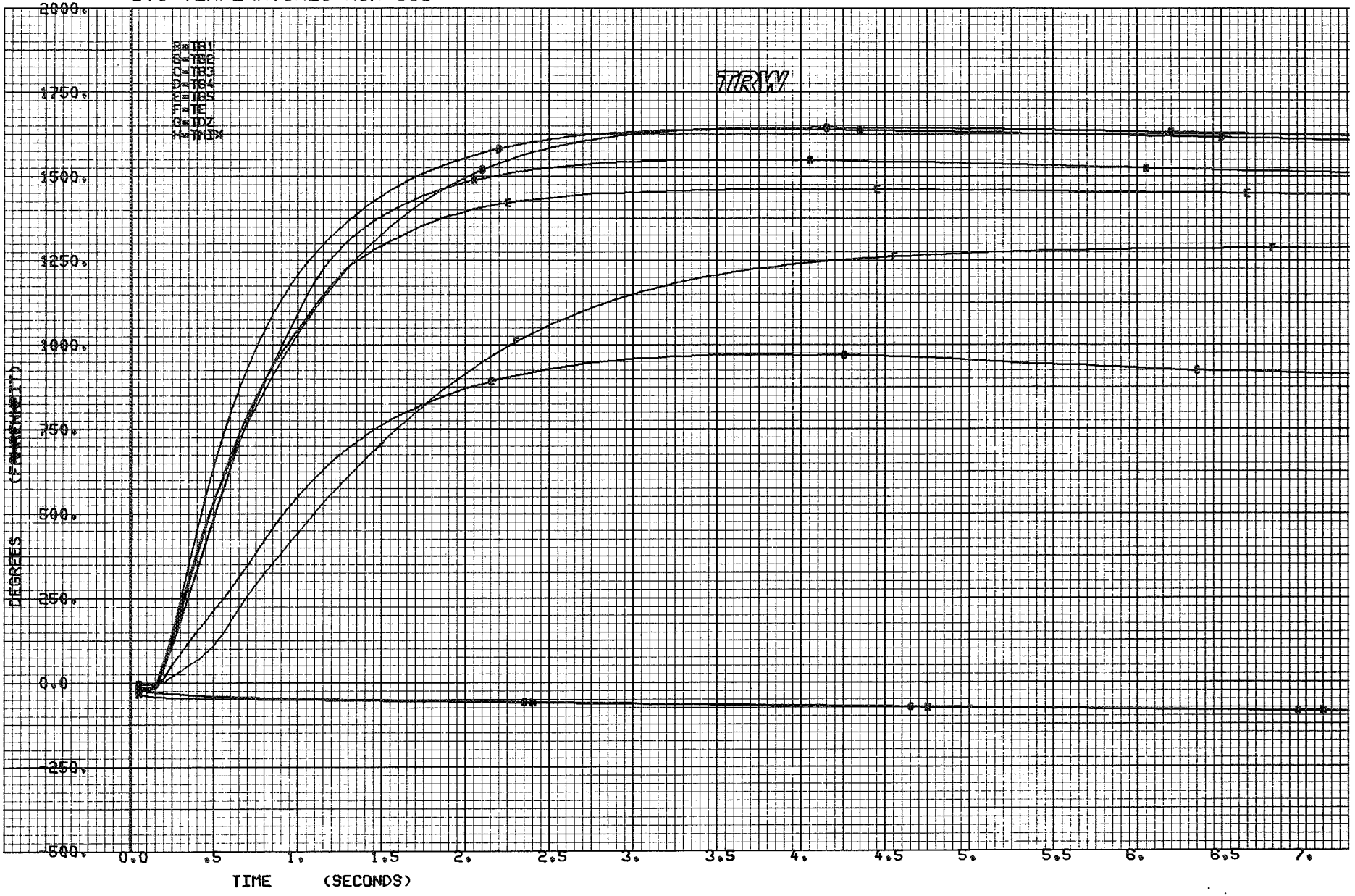
ITS TEMPERATURES RUN 861



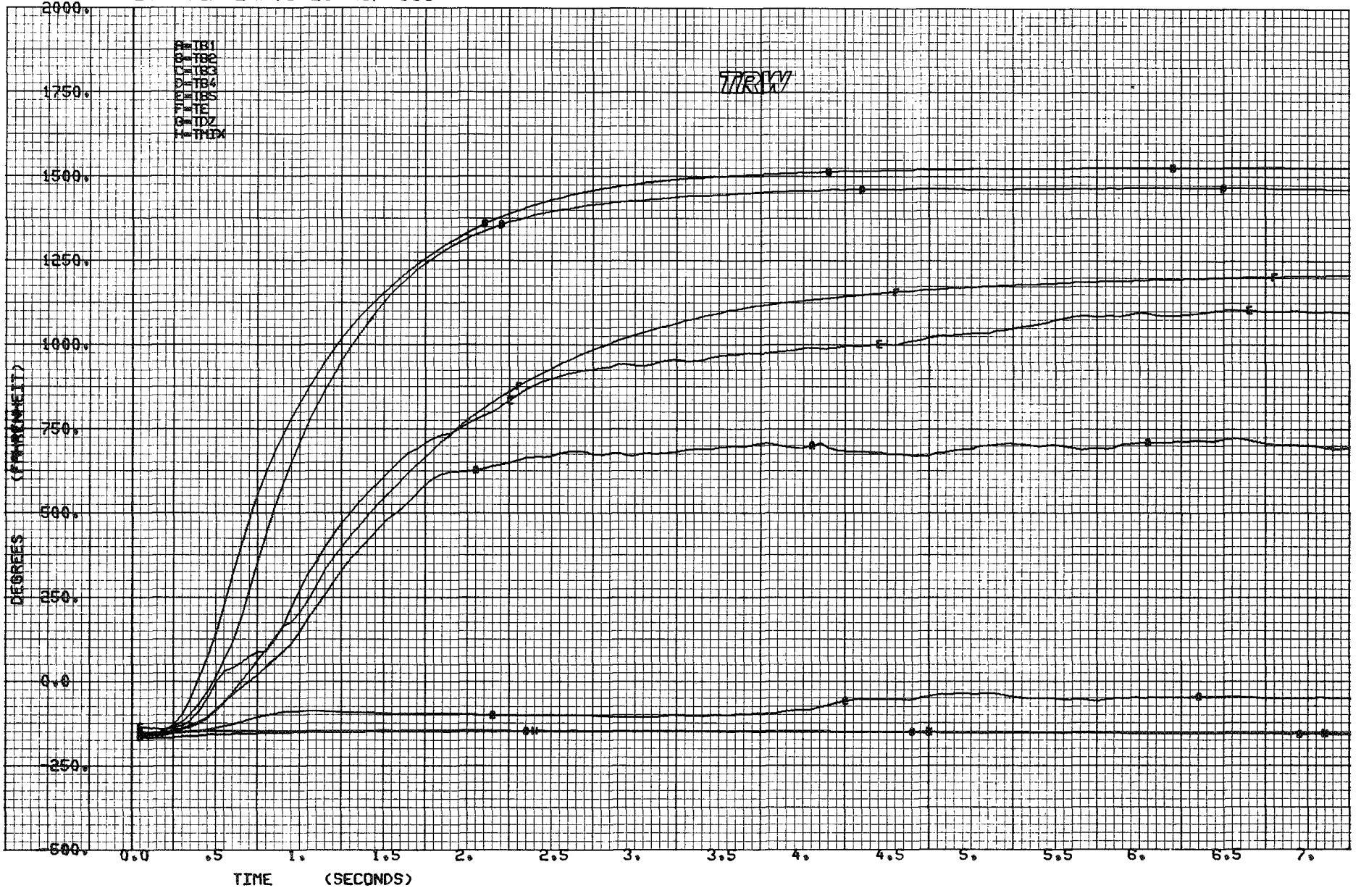
ITS TEMPERATURES RUN 862



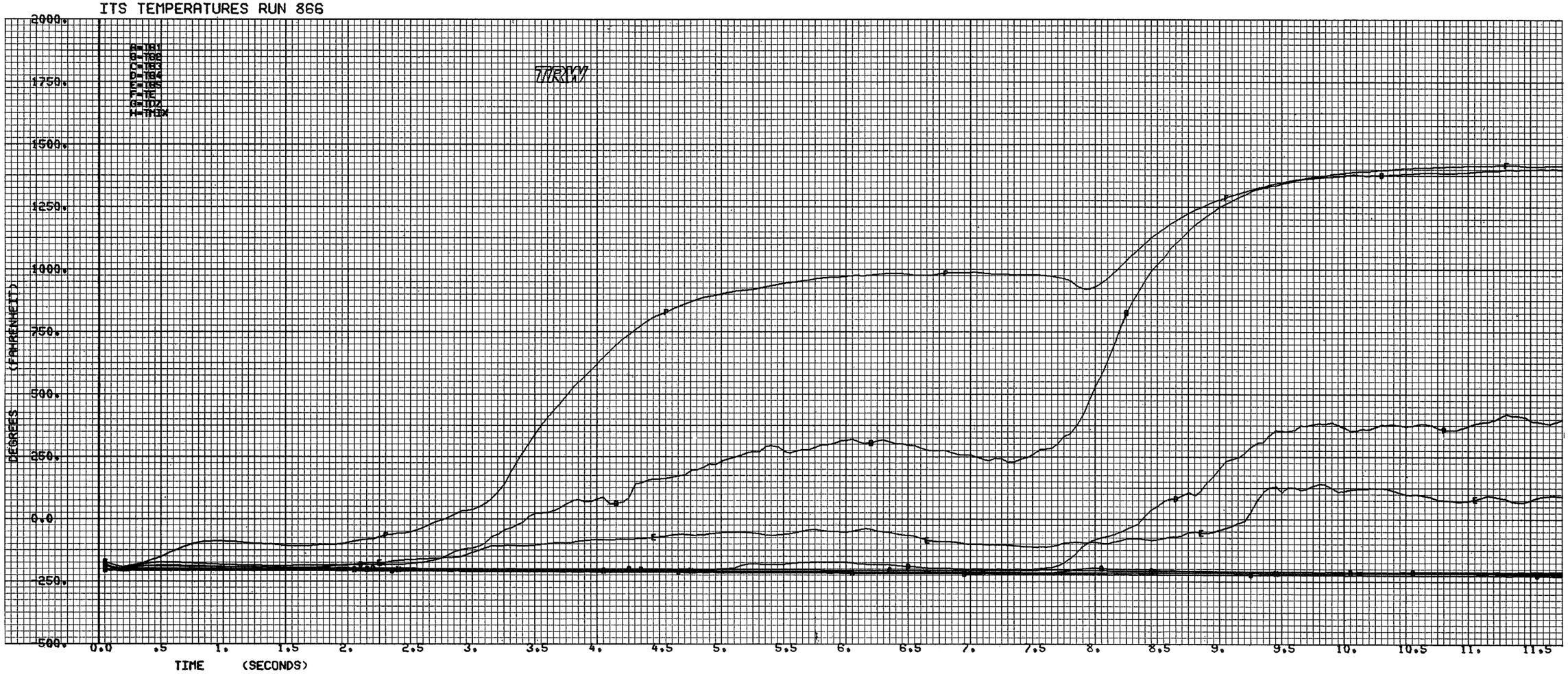
ITS TEMPERATURES RUN 863



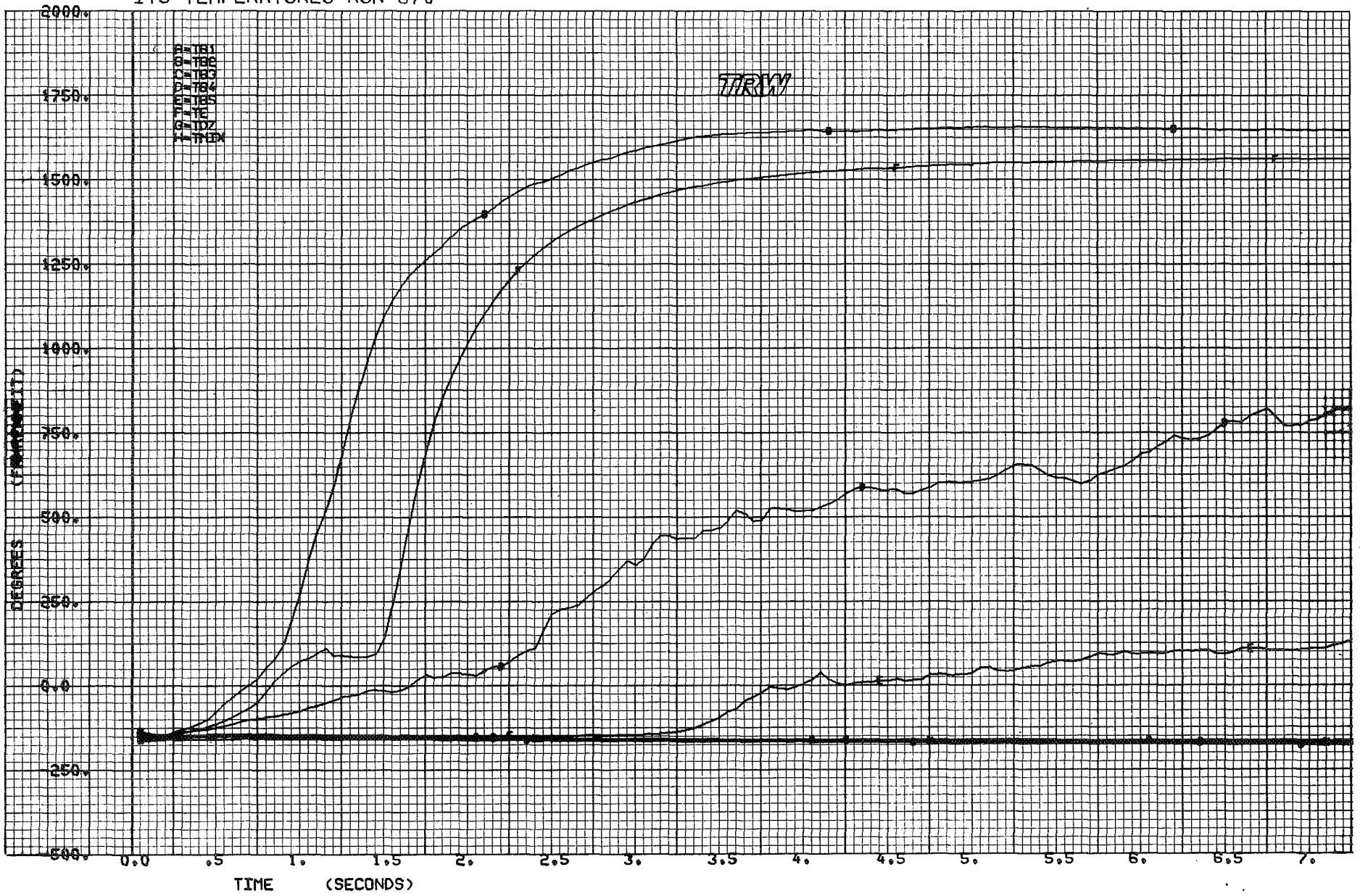
ITS TEMPERATURES RUN 865

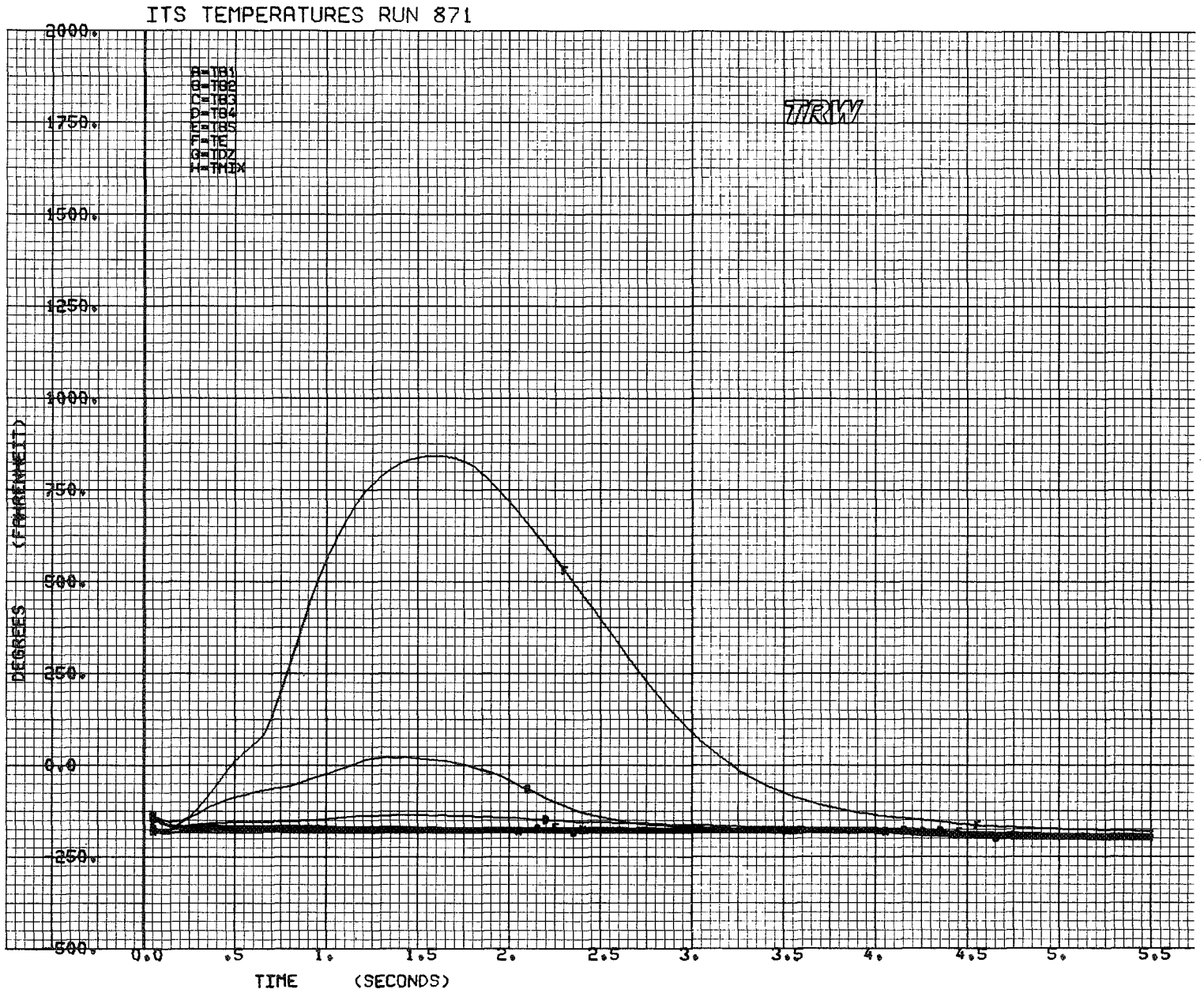


ITS TEMPERATURES RUN 866

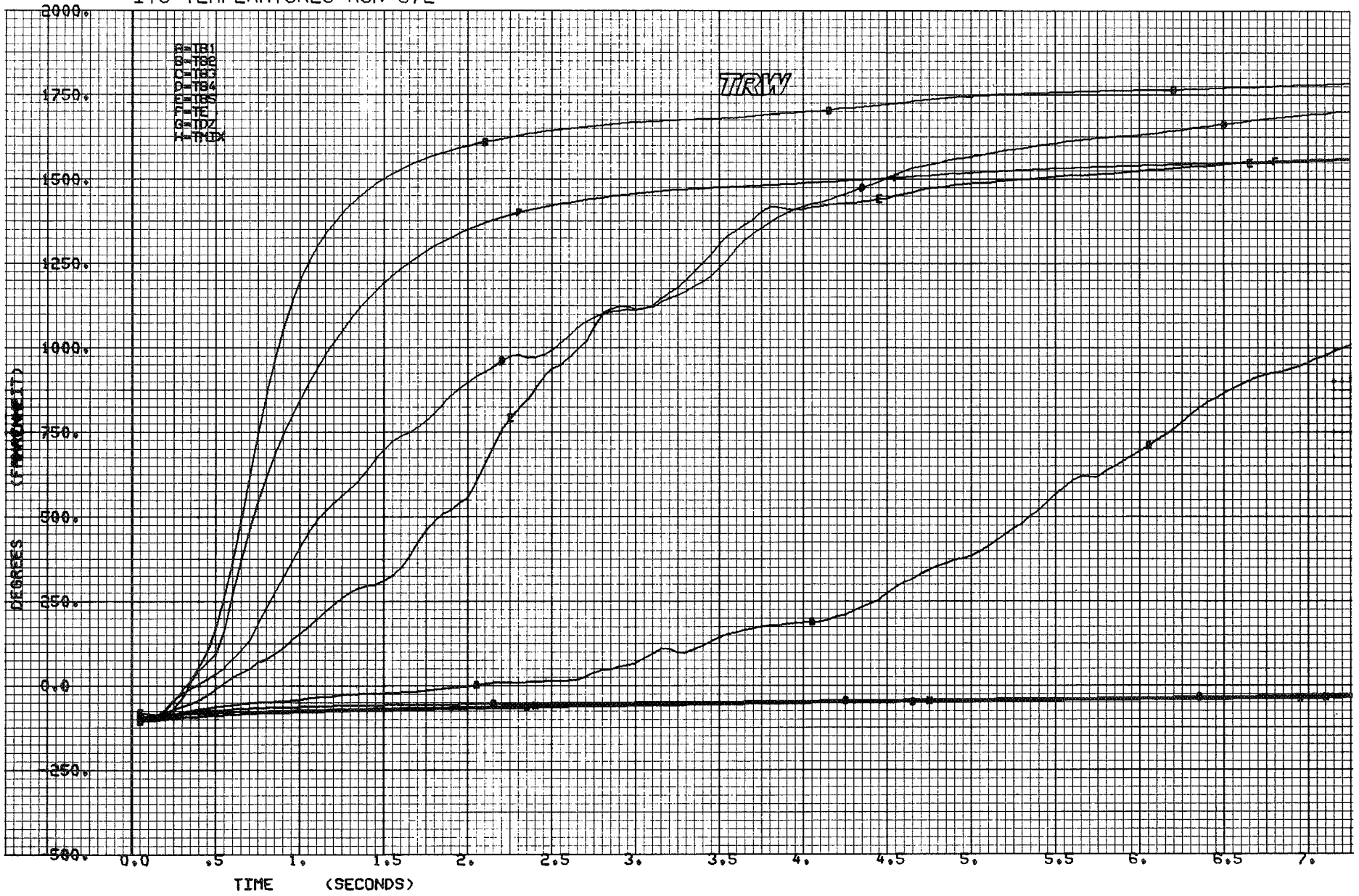


ITS TEMPERATURES RUN 870

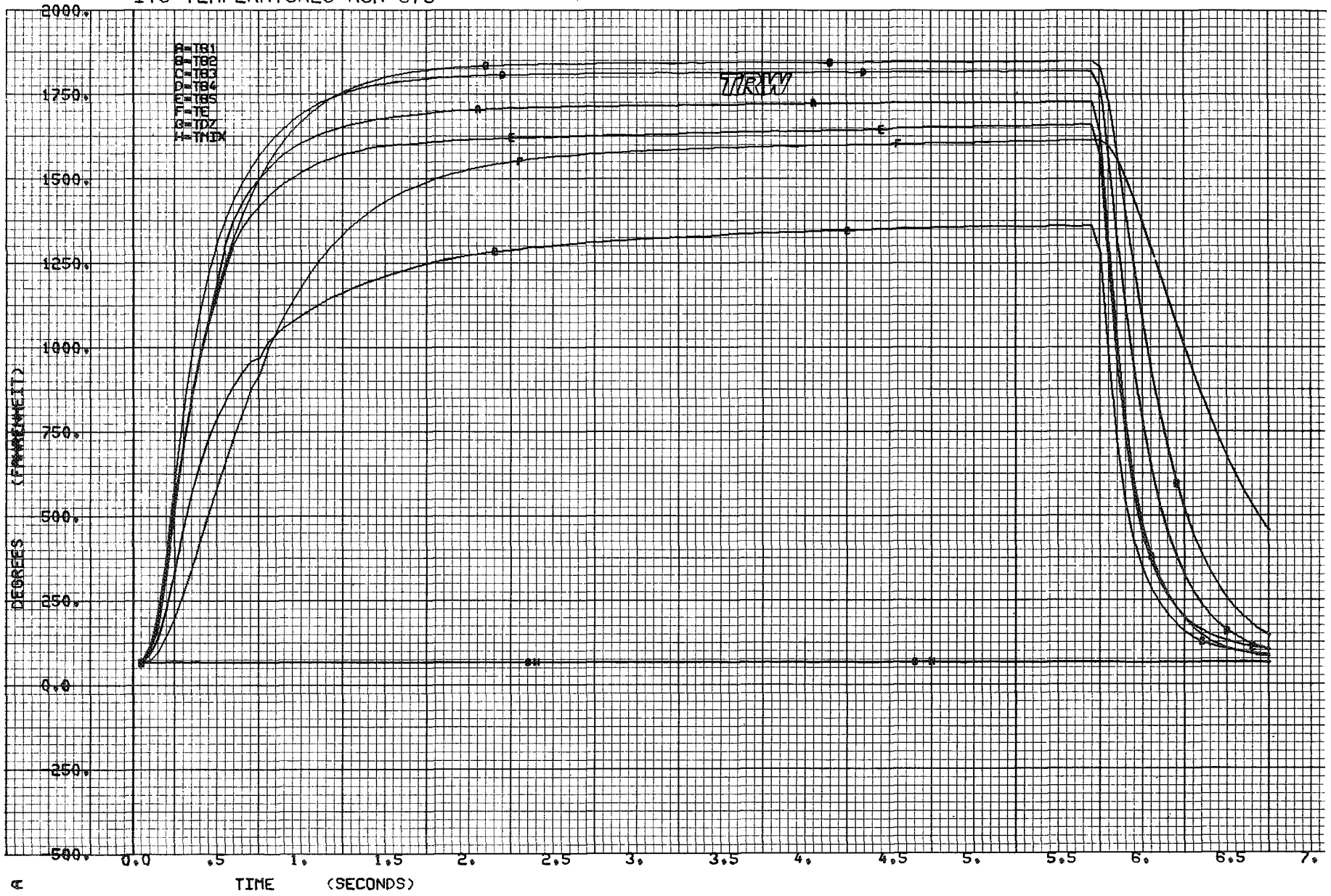


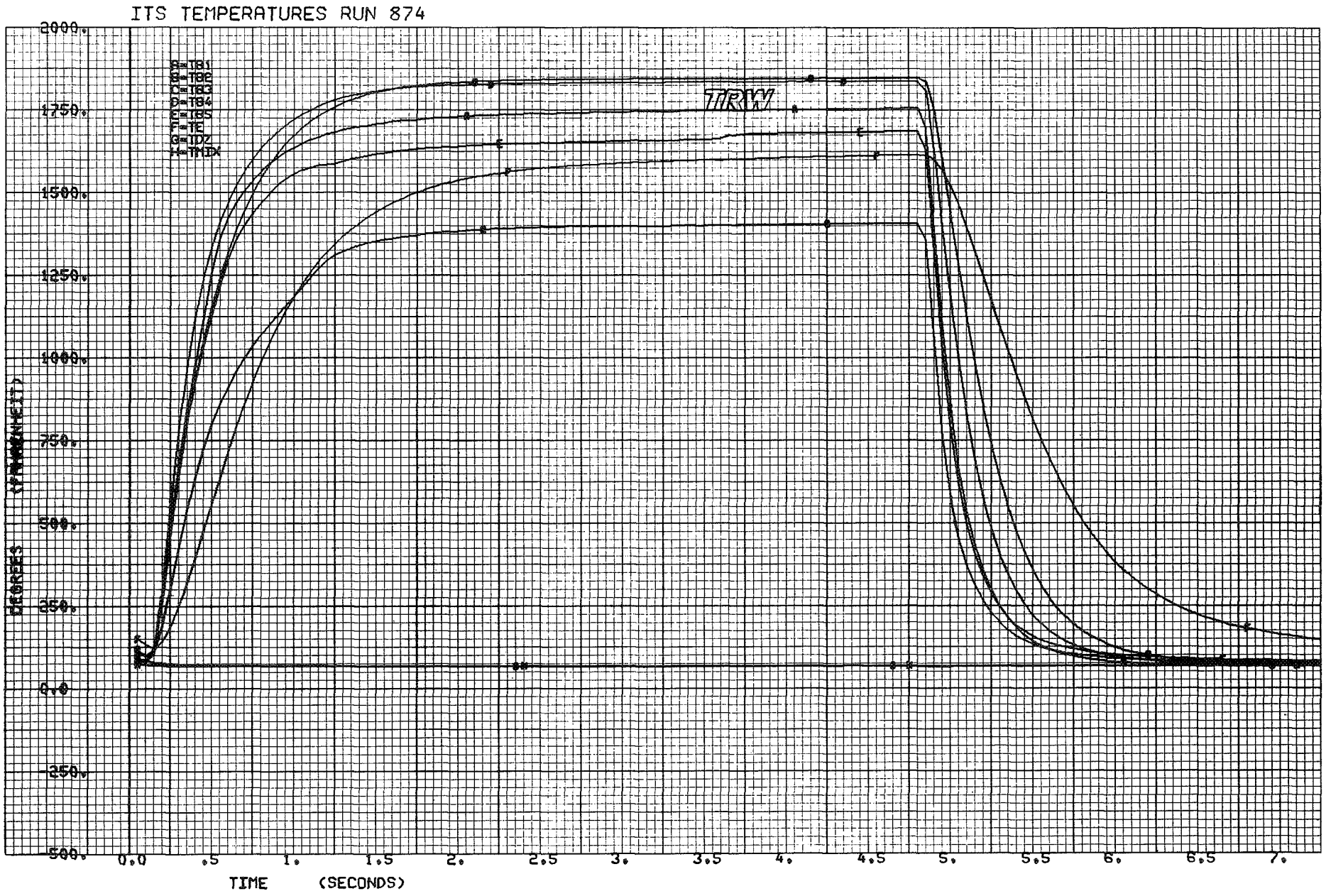

ITS TEMPERATURES RUN 871
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TF
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
TIME (SECONDS)

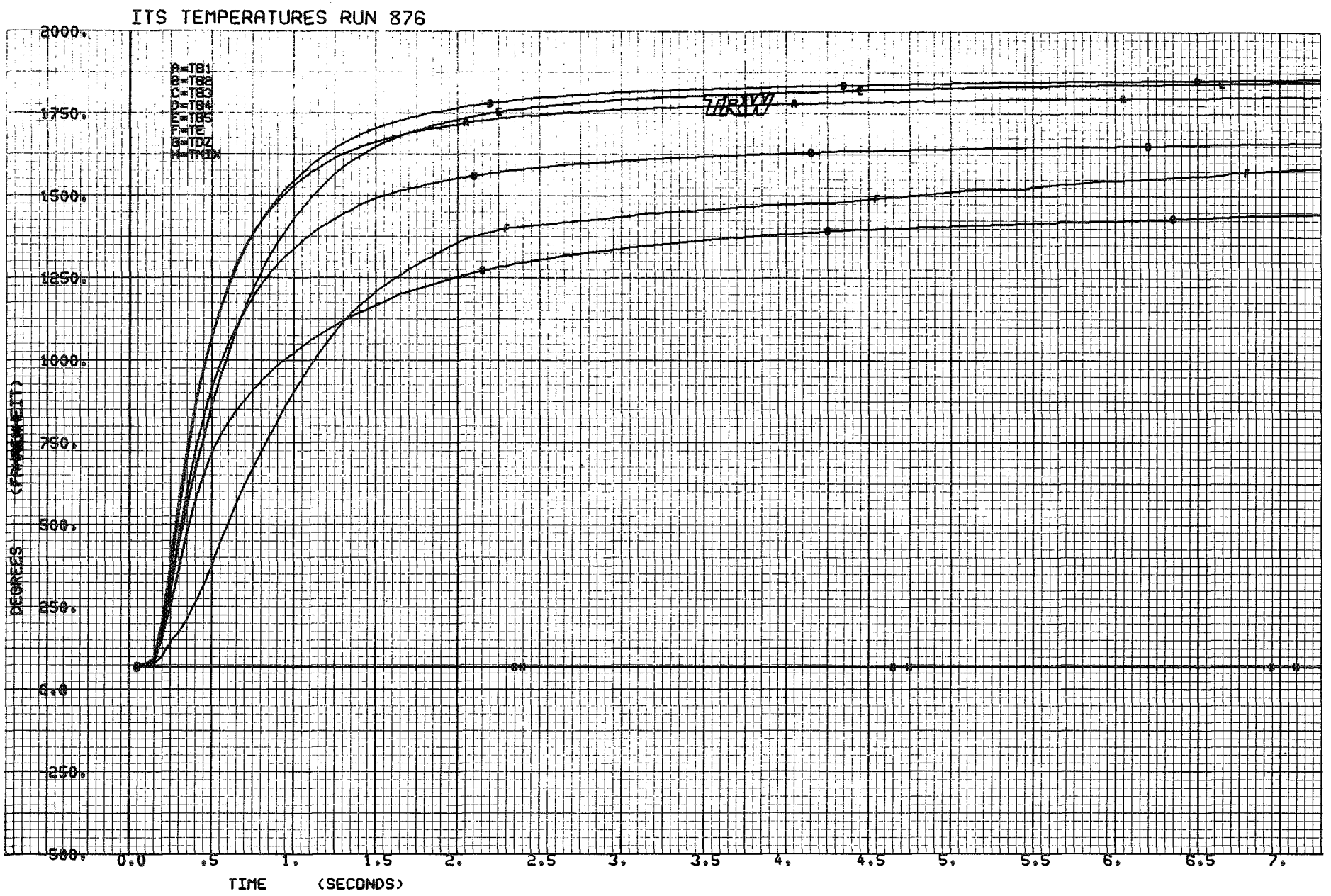
ITS TEMPERATURES RUN 872



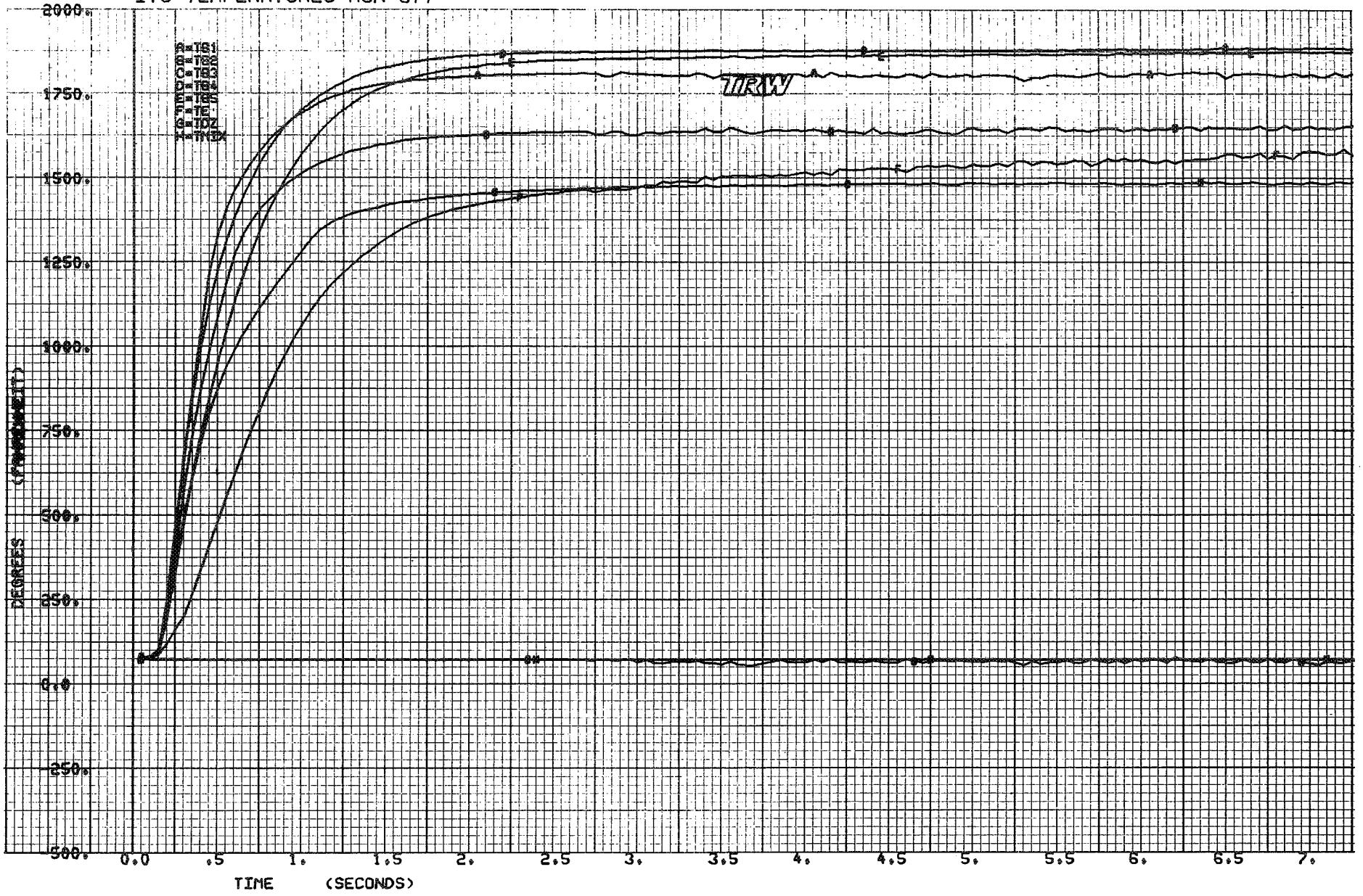
ITS TEMPERATURES RUN 873



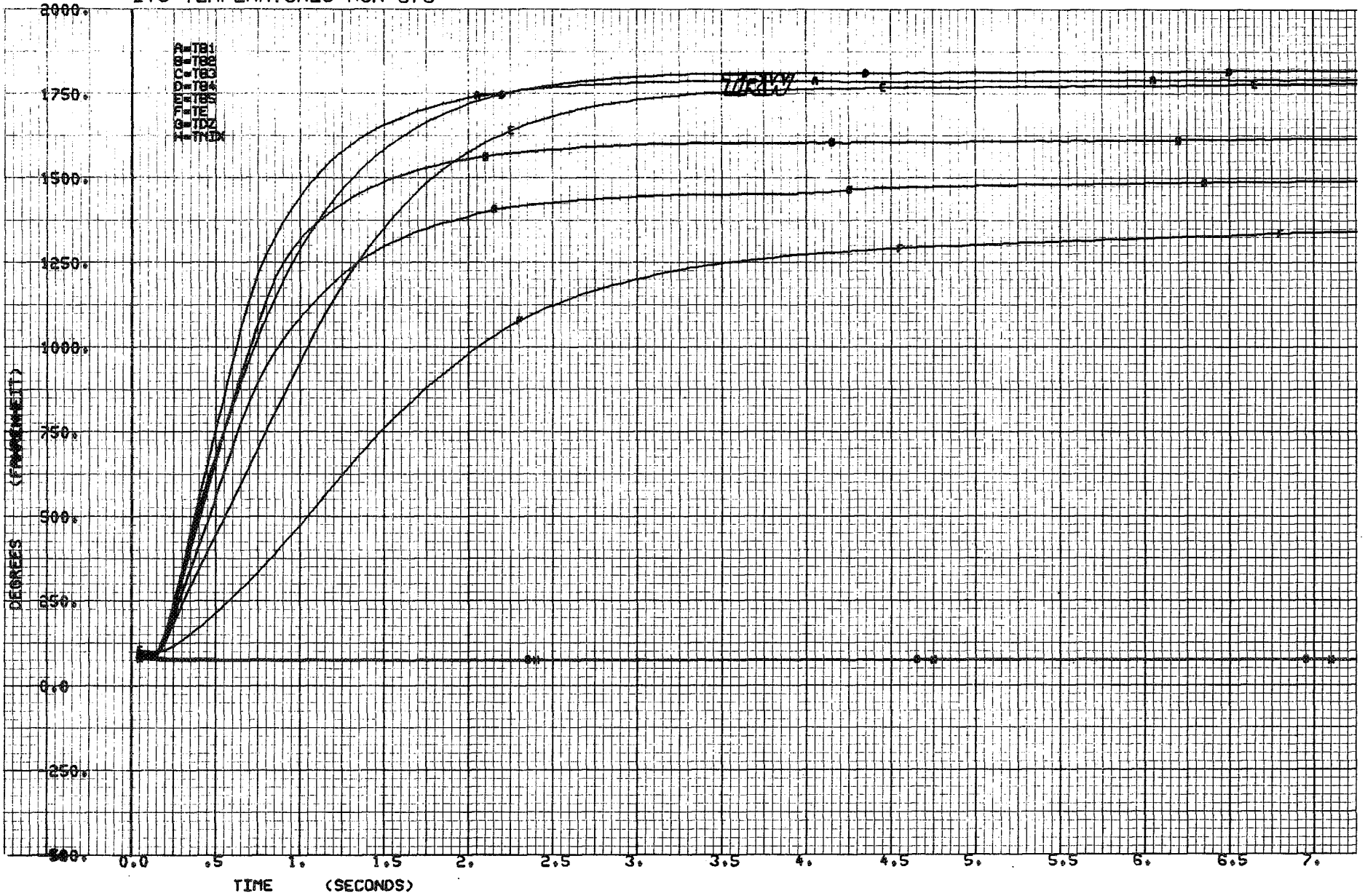

ITS TEMPERATURES RUN 874
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TDZ
H=TNIX
TIRKW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
7.
TIME (SECONDS)

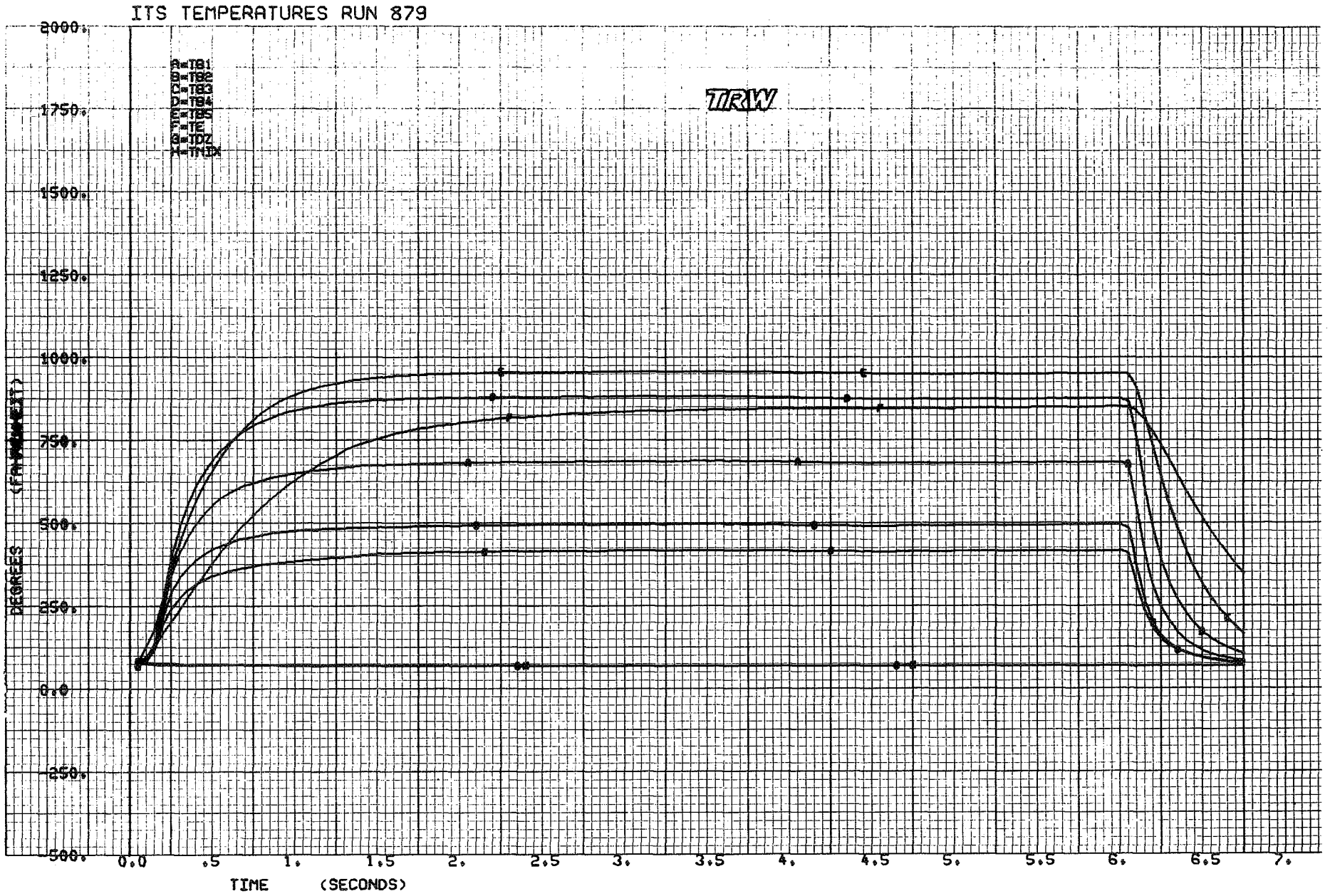

ITS TEMPERATURES RUN 876
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TDZ
H=TMIX
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
7.
TIME (SECONDS)

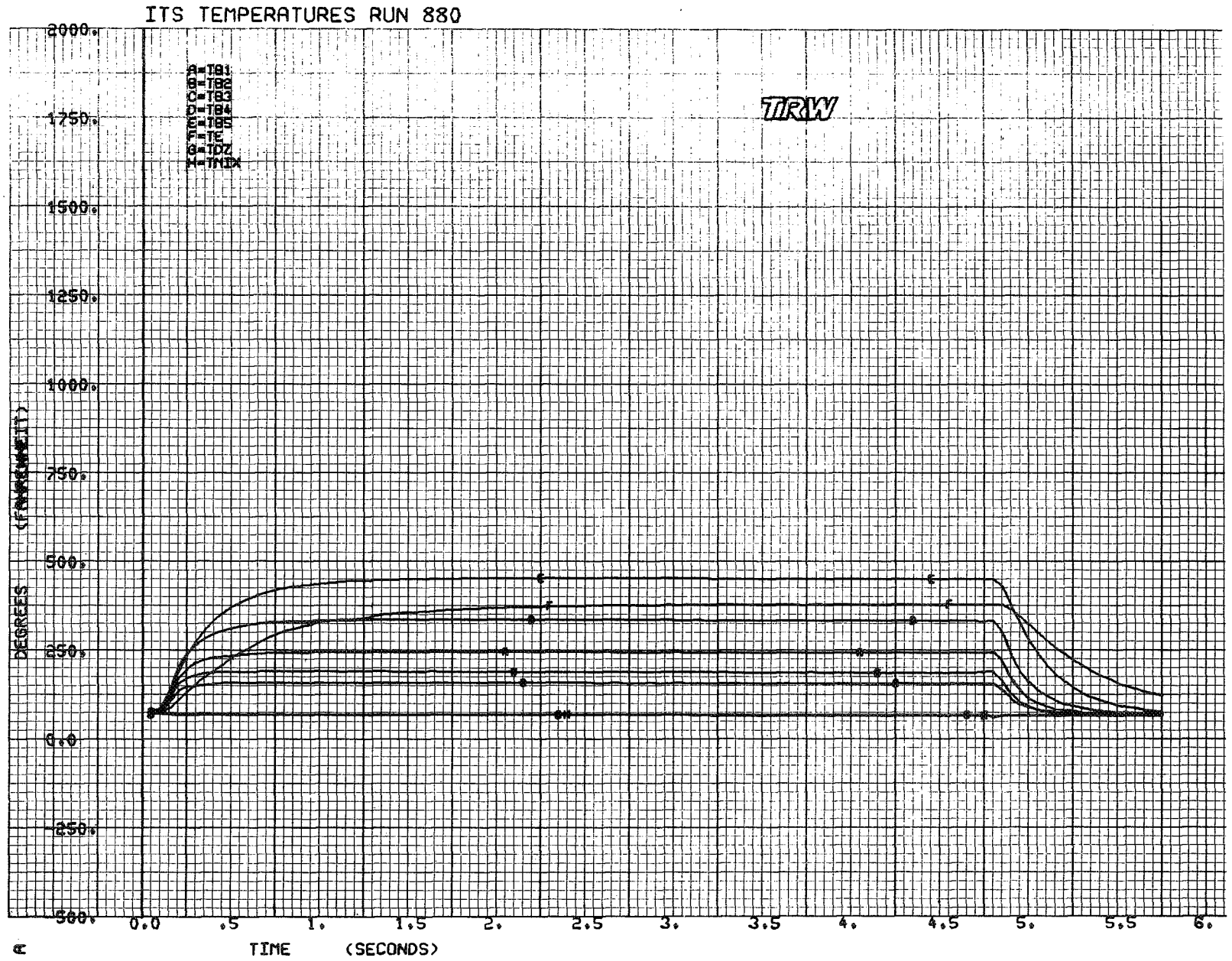
ITS TEMPERATURES RUN 877



ITS TEMPERATURES RUN 878

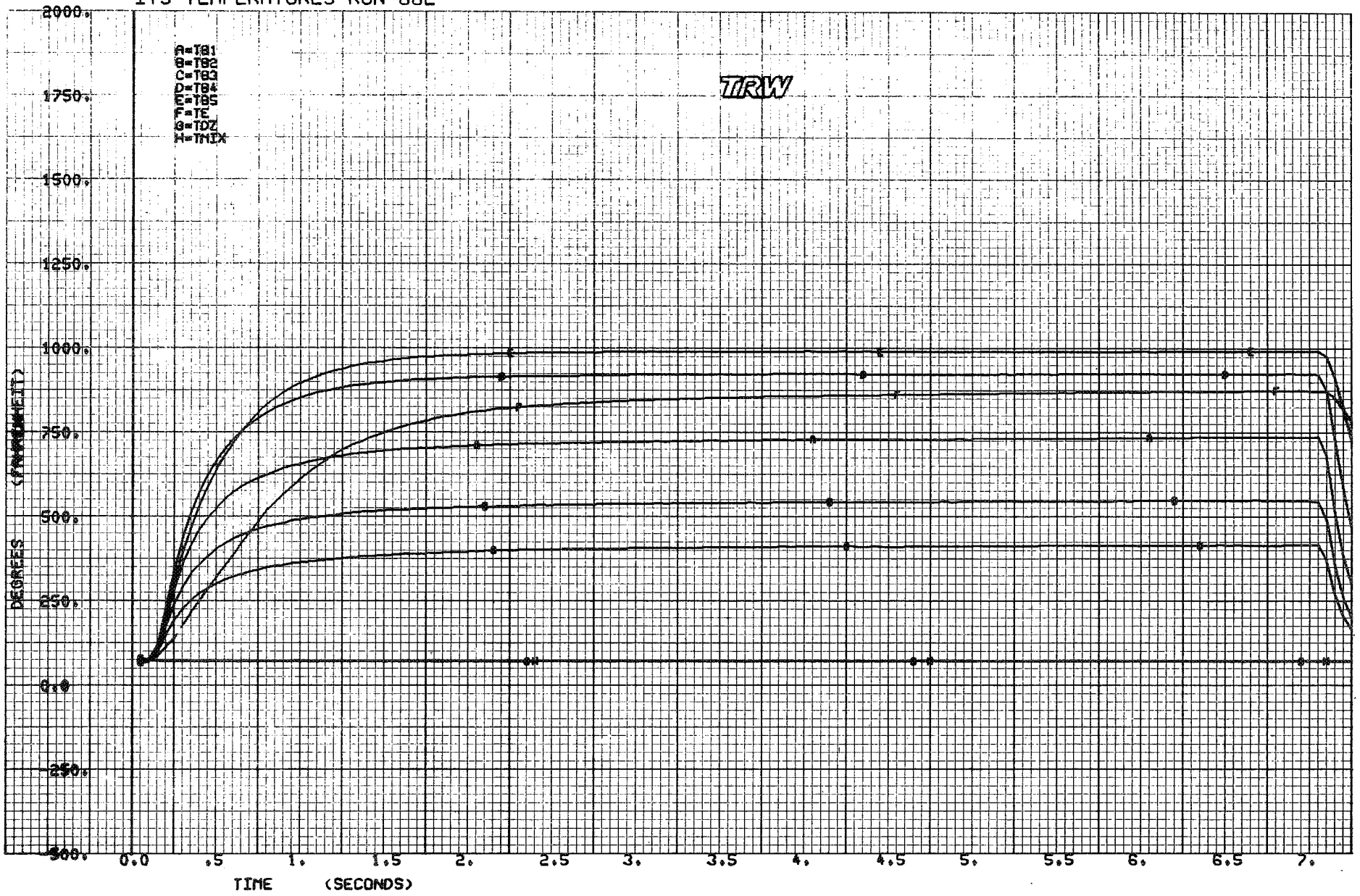



ITS TEMPERATURES RUN 879
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TDZ
H=TNIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
7.
TIME (SECONDS)

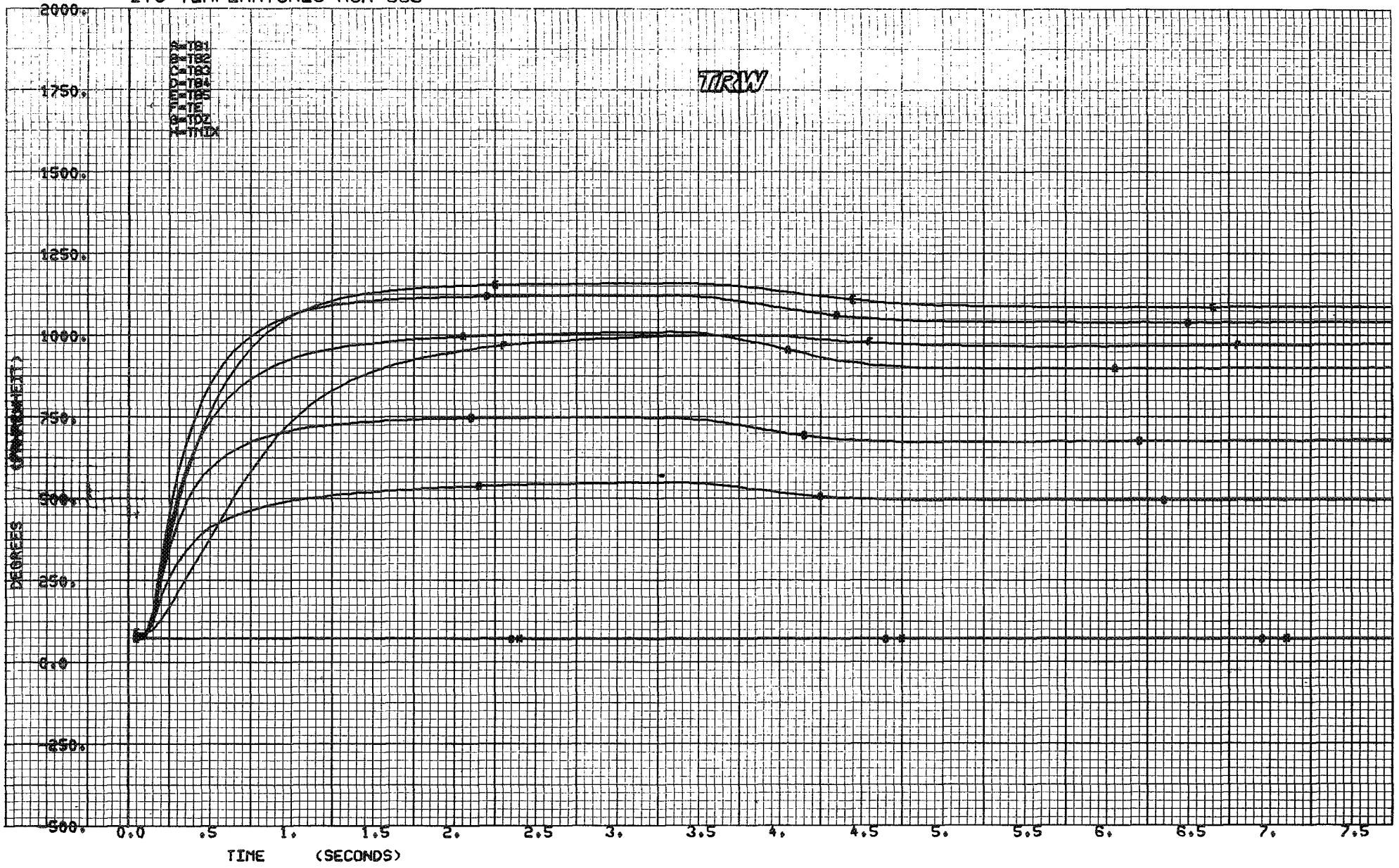

ITS TEMPERATURES RUN 880
A=T01
B=T02
C=T03
D=T04
E=T05
F=TC
G=T07
H=TMIX
TRW
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
TIME (SECONDS)

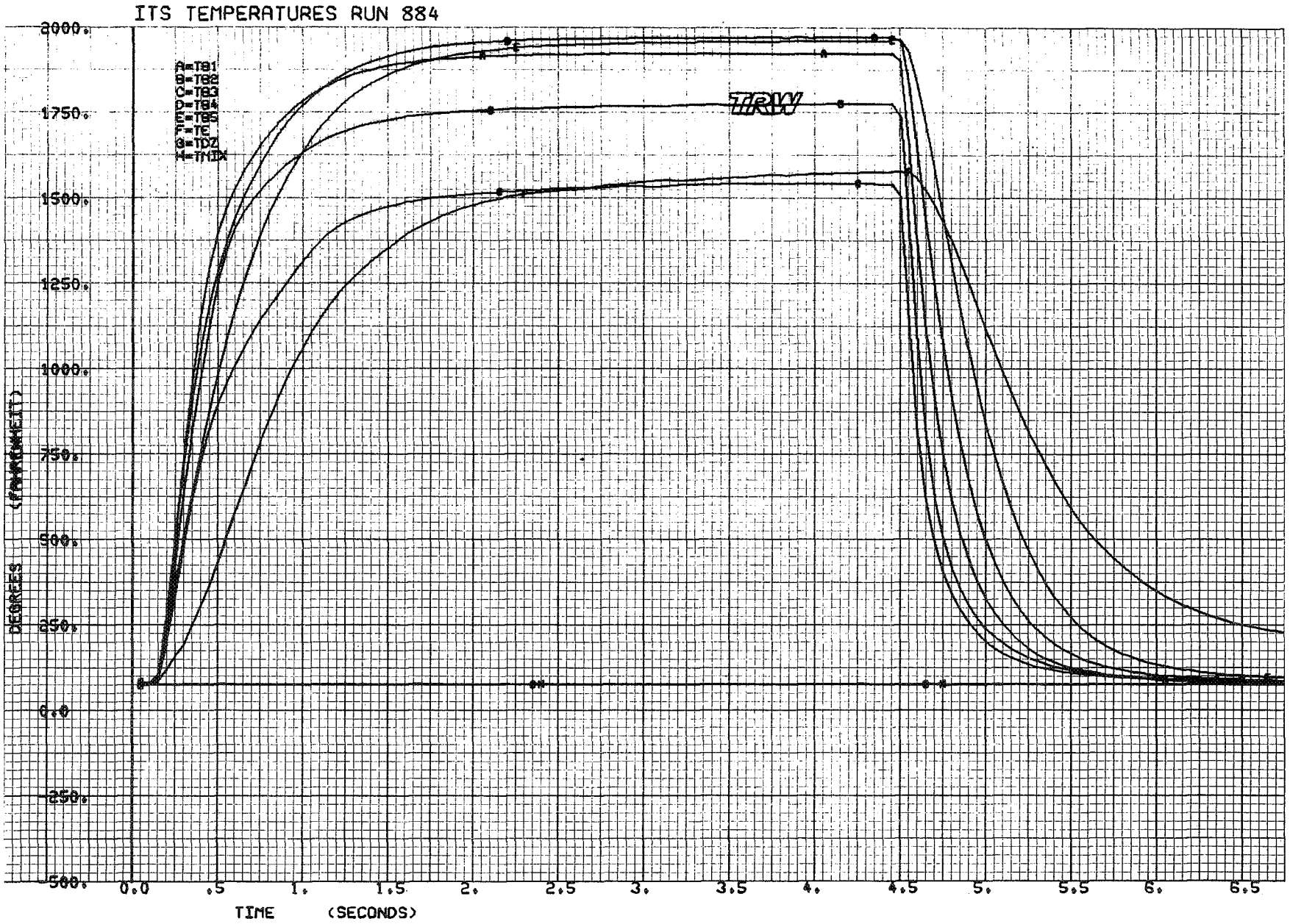
ITS TEMPERATURES RUN 881

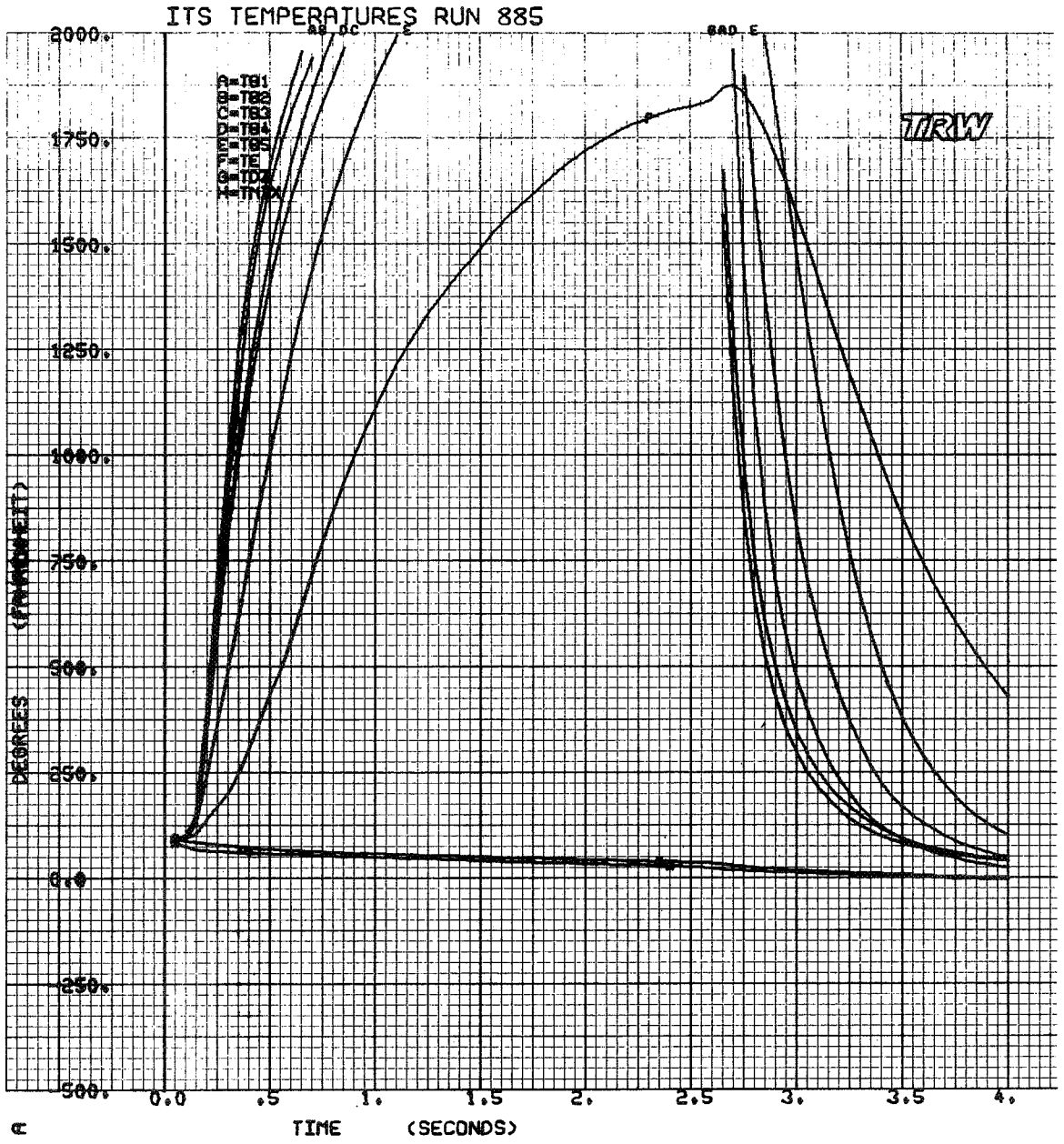
ITS TEMPERATURES RUN 882

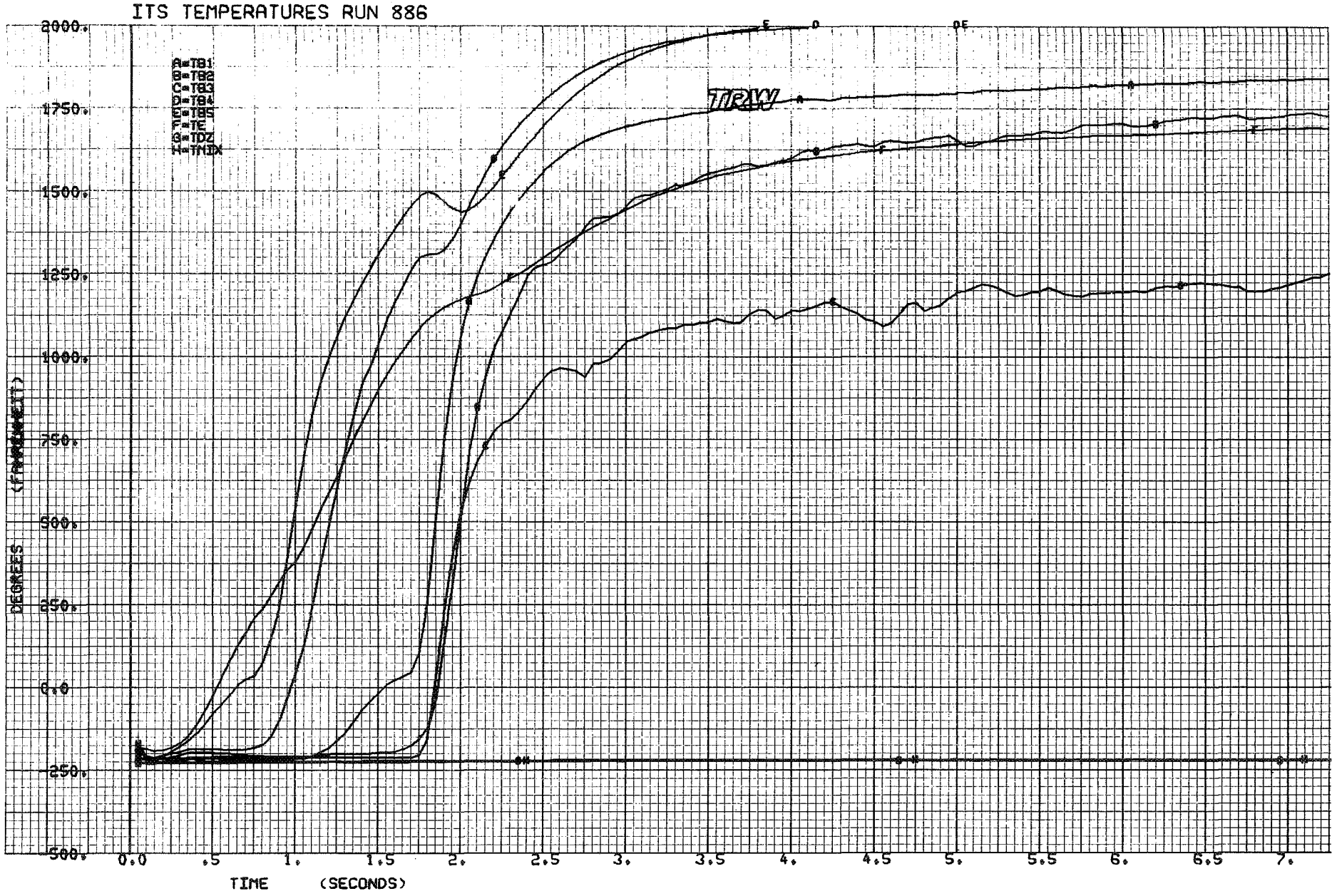


ITS TEMPERATURES RUN 883

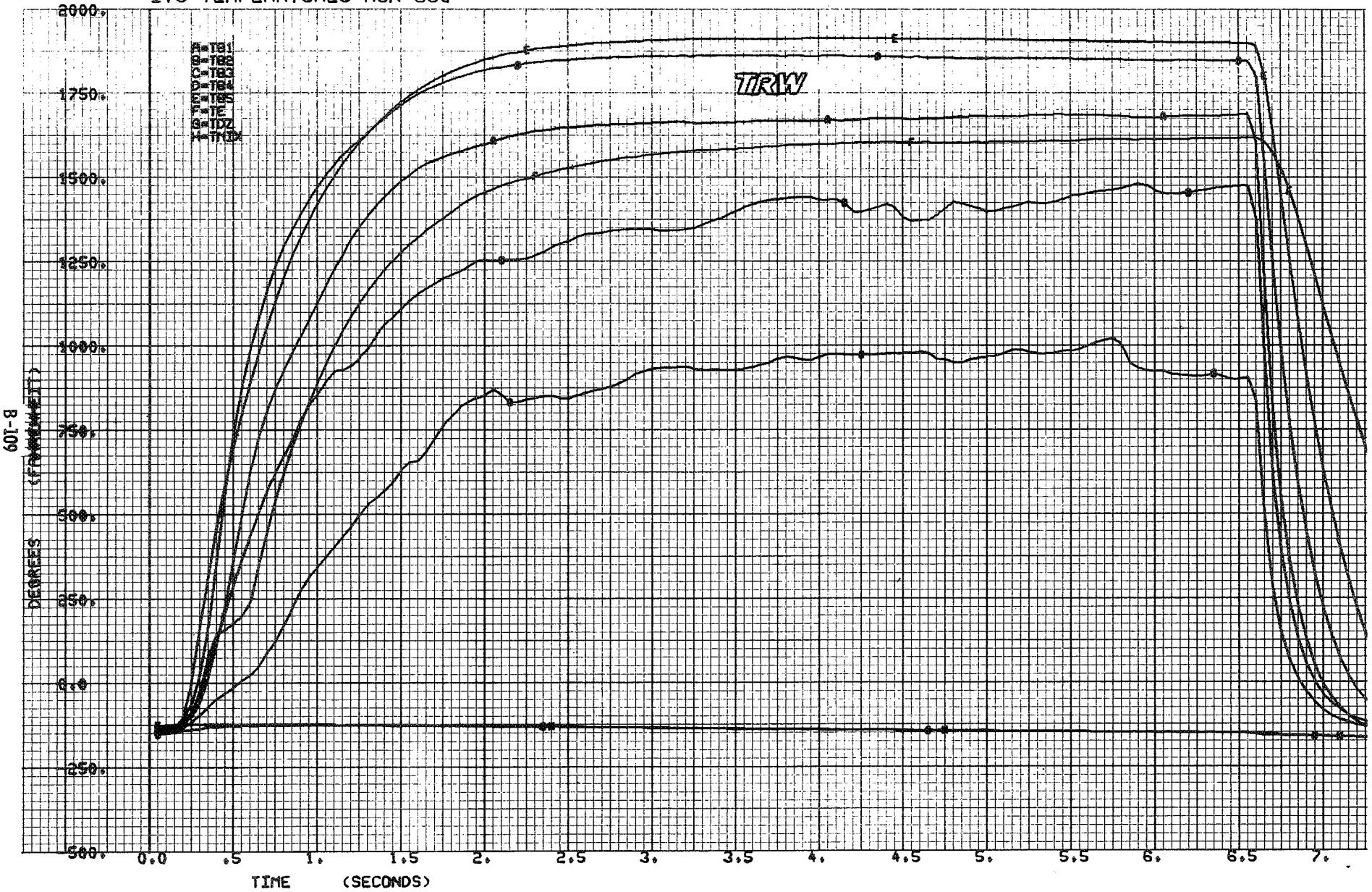



ITS TEMPERATURES RUN 884
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TC
G=TDZ
H=TMIX
TRW
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
TIME (SECONDS)

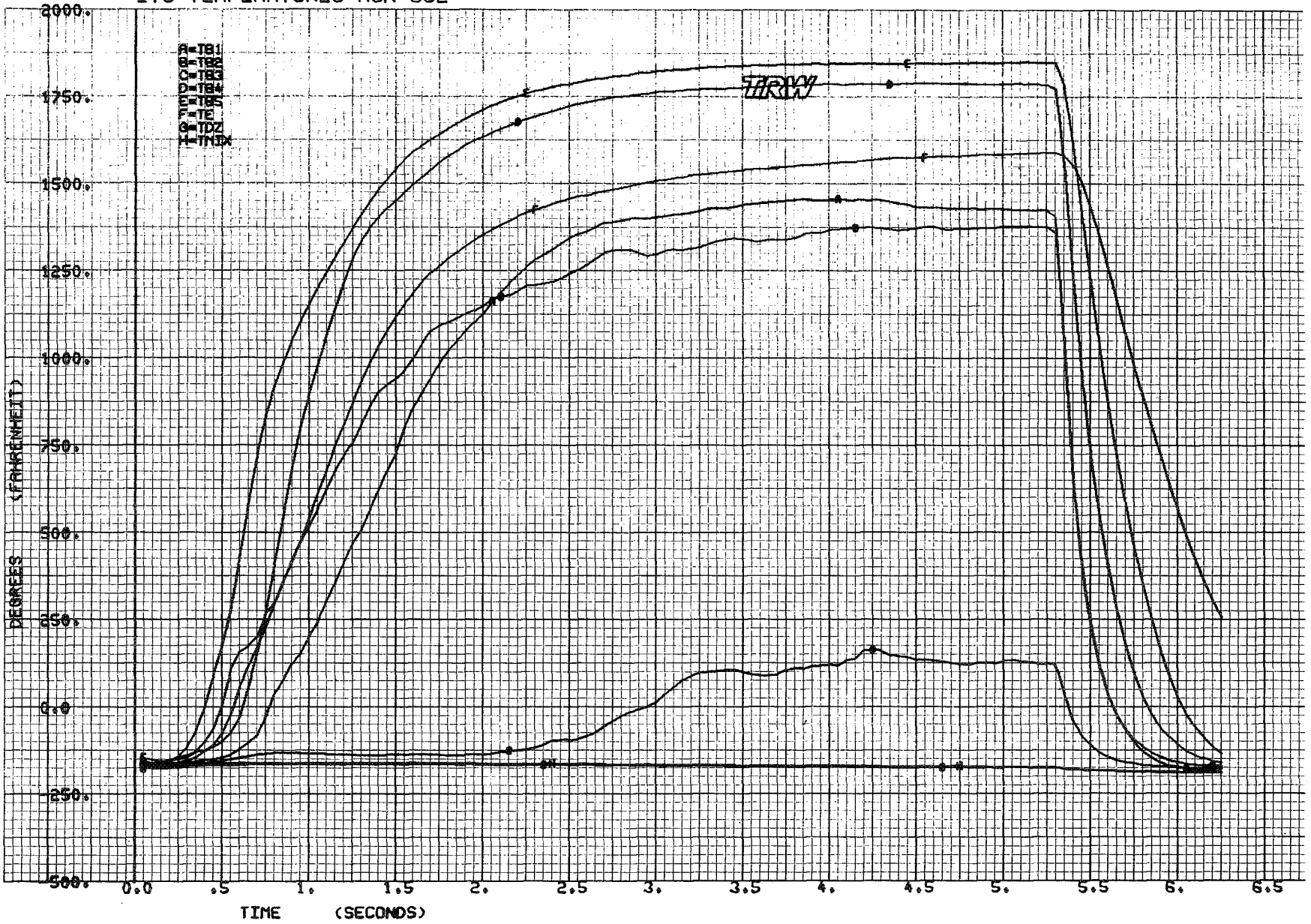

ITS TEMPERATURES RUN 885
A=T01
B=T02
C=T03
D=T04
E=T05
F=TE
G=TDX
H=TYCN
AB DC
E
BAD E
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
TIME (SECONDS)

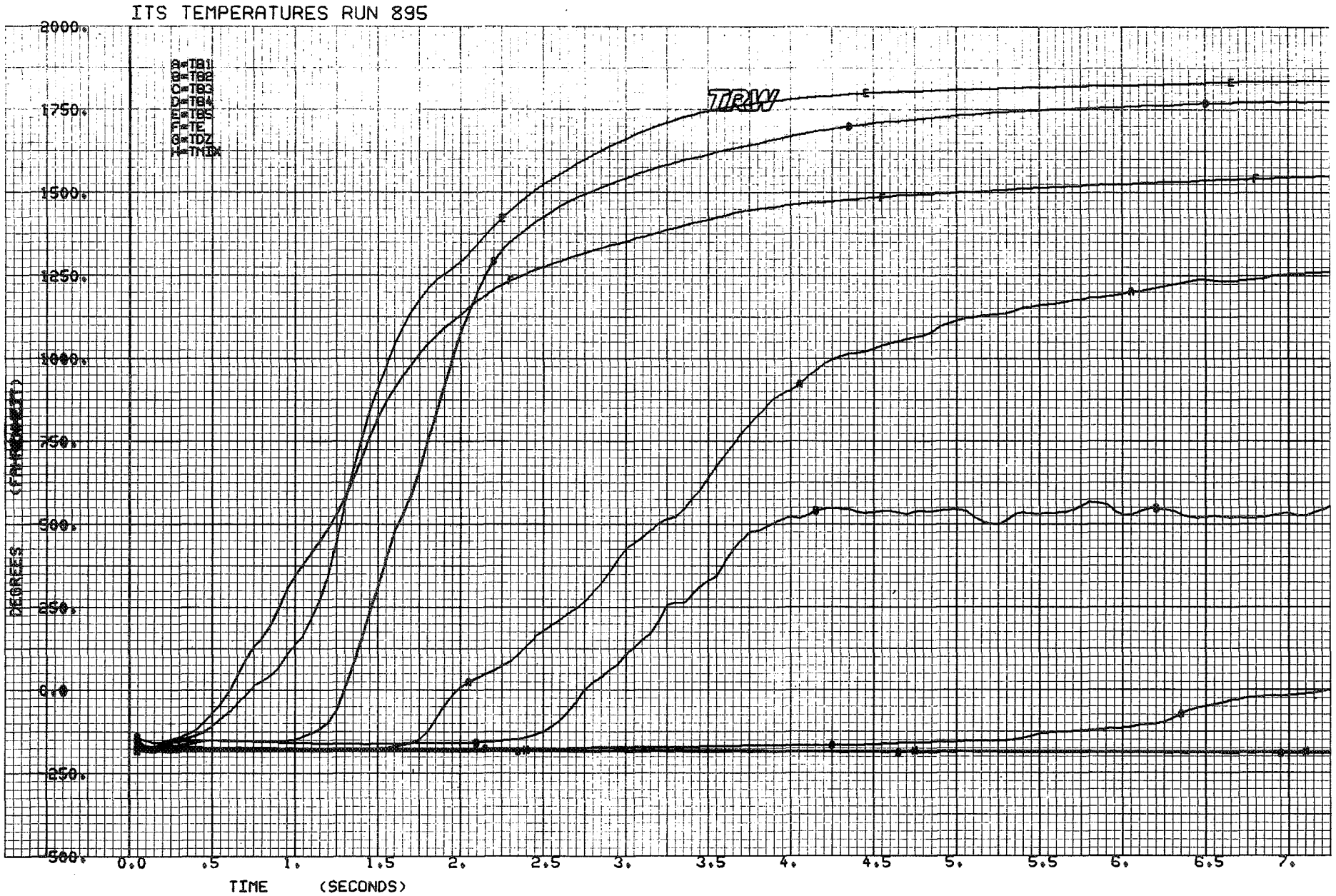

ITS TEMPERATURES RUN 886
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TDZ
H=TNIX
TRAY
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
7.
TIME (SECONDS)

ITS TEMPERATURES RUN 890

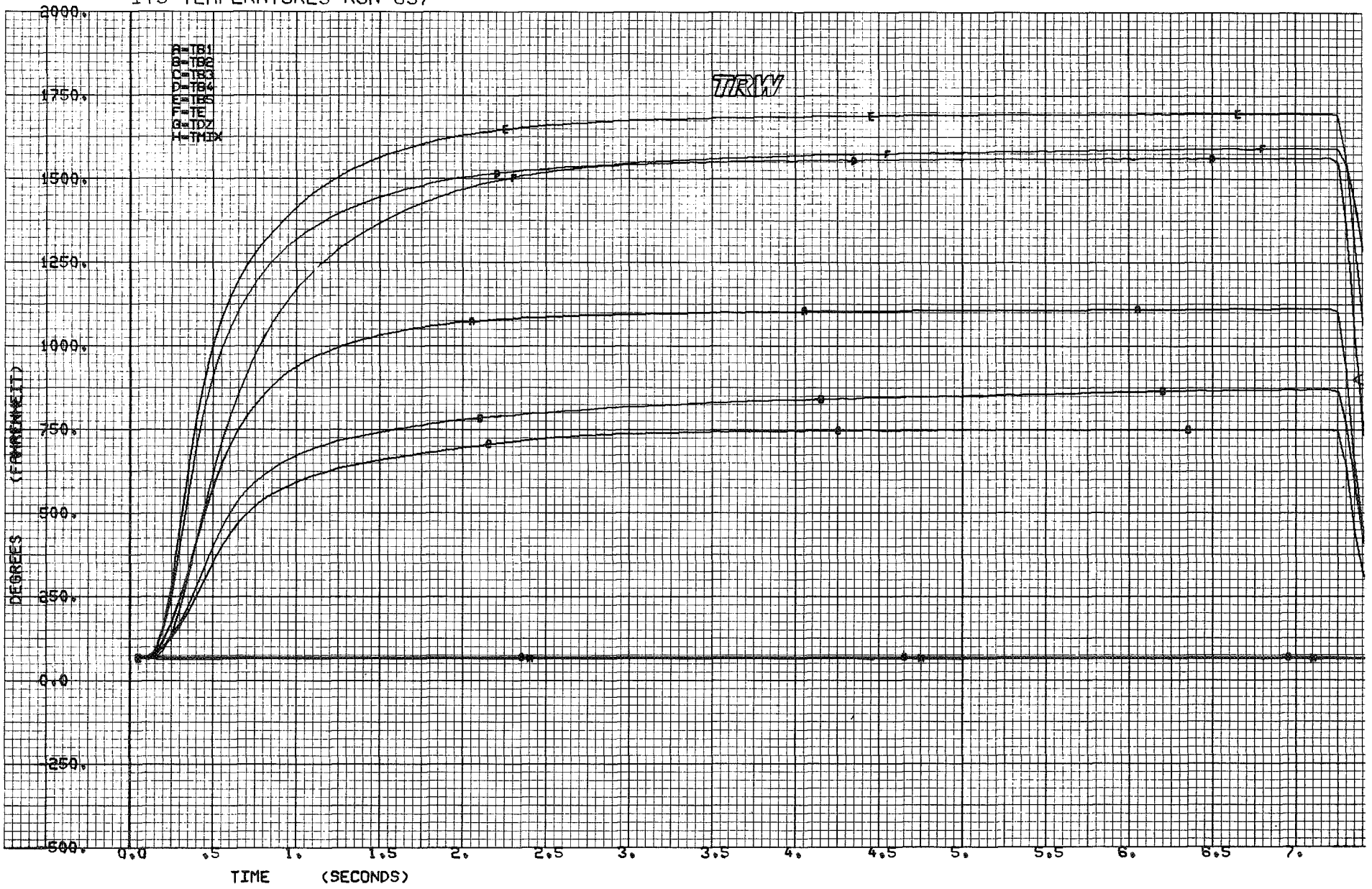


ITS TEMPERATURES RUN 892

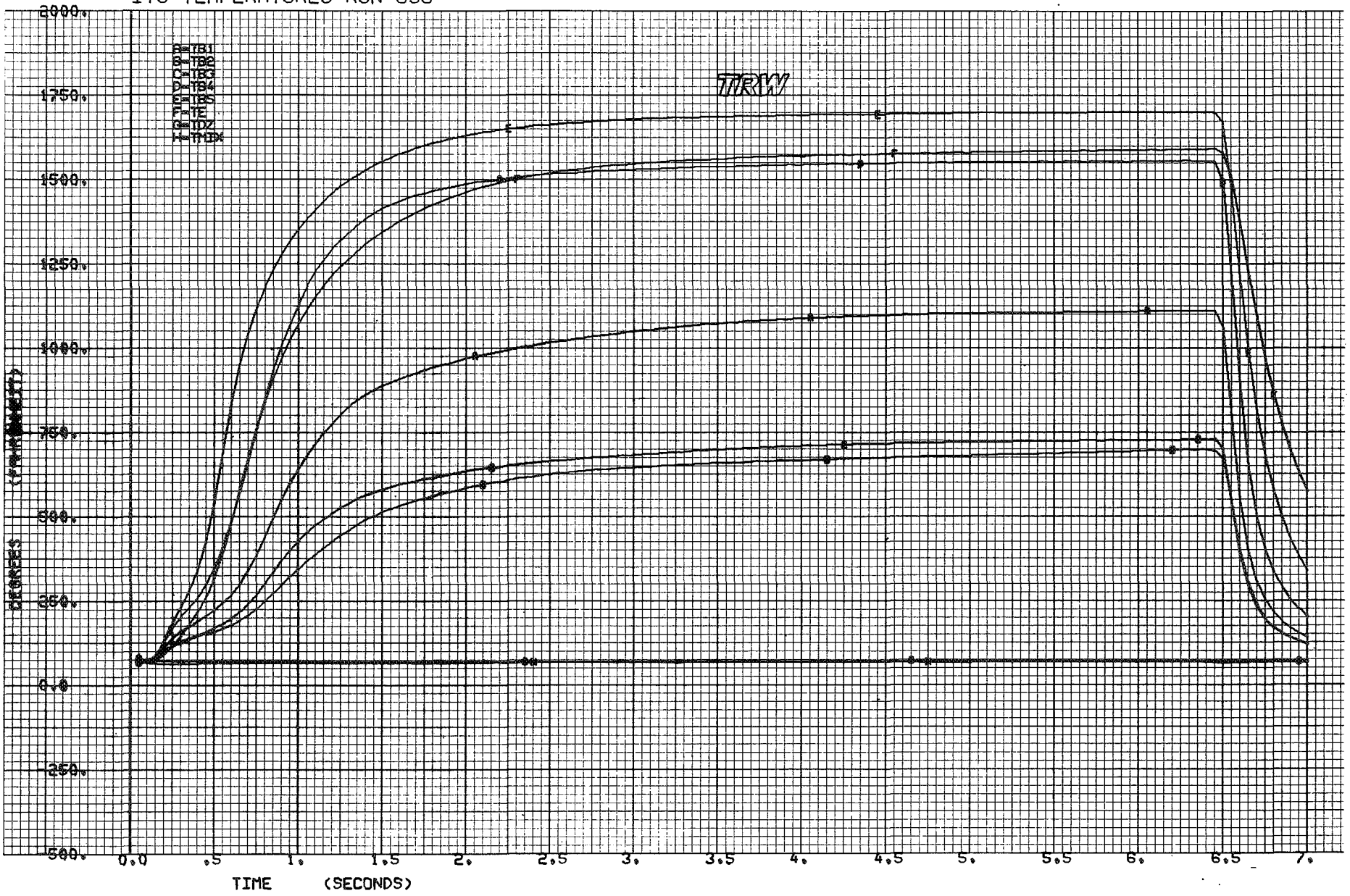



ITS TEMPERATURES RUN 895
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TDZ
H=TNDX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
7.
TIME (SECONDS)
DEGREES (FAHRENHEIT)

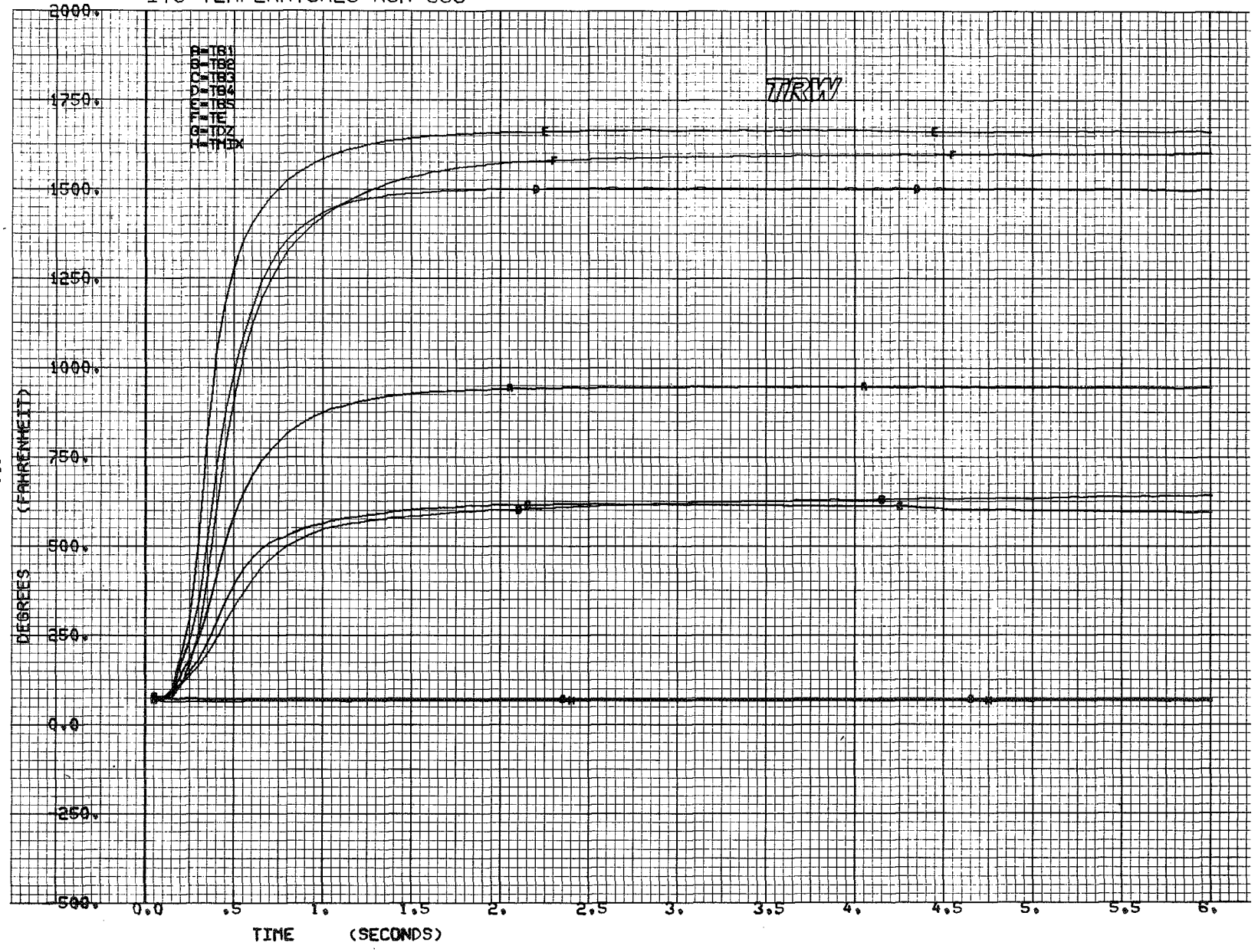
ITS TEMPERATURES RUN 897



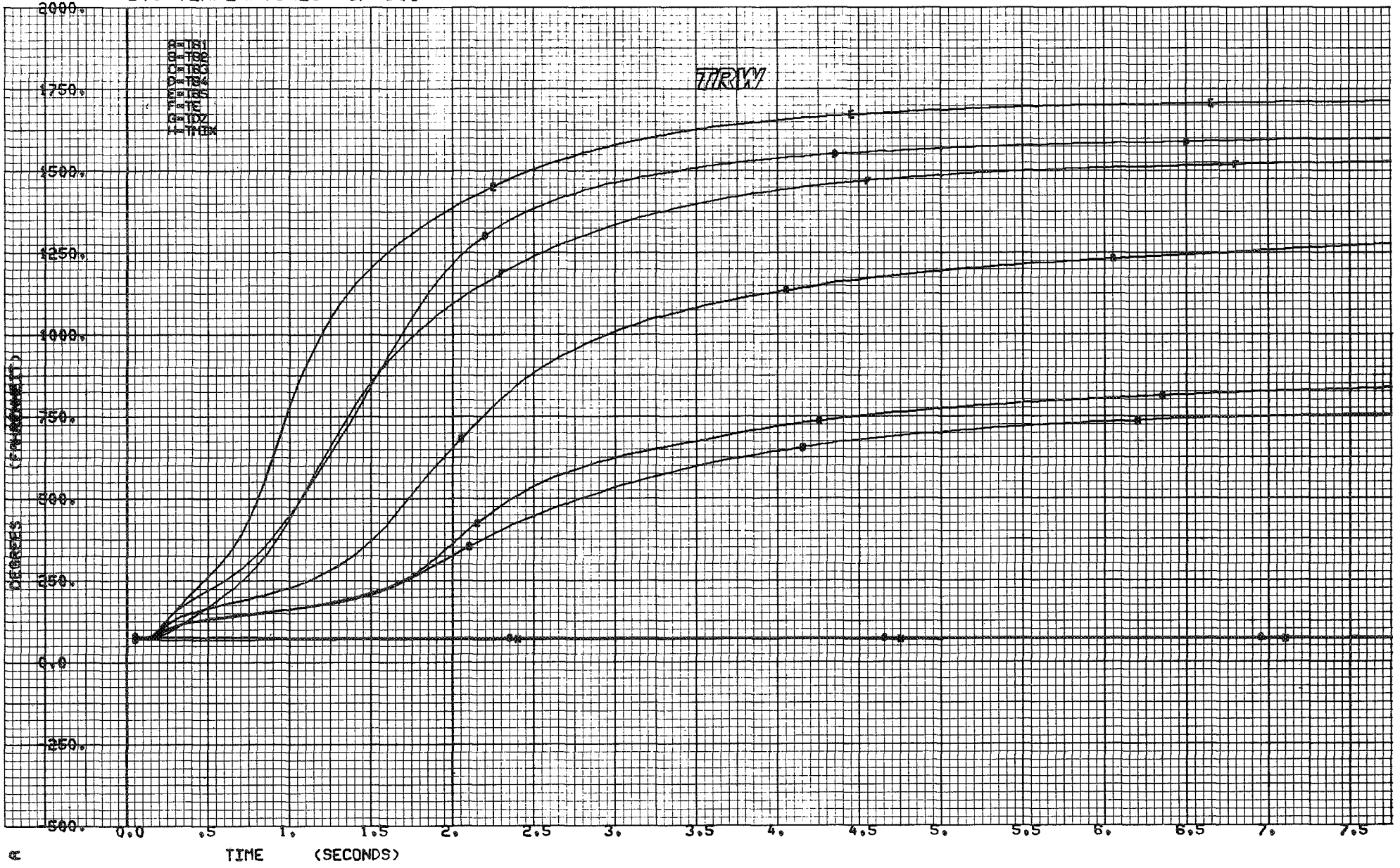
ITS TEMPERATURES RUN 898



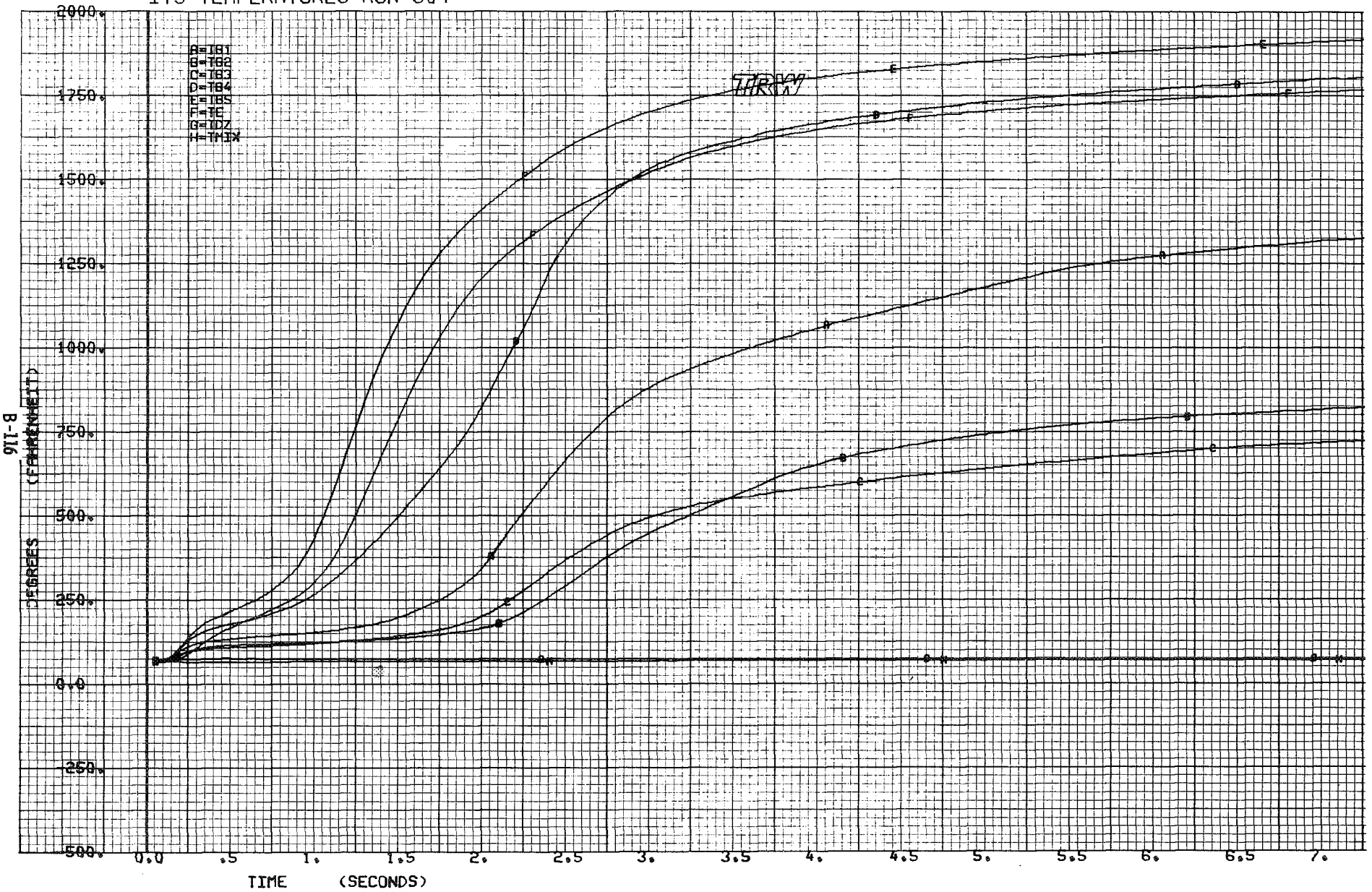
ITS TEMPERATURES RUN 899



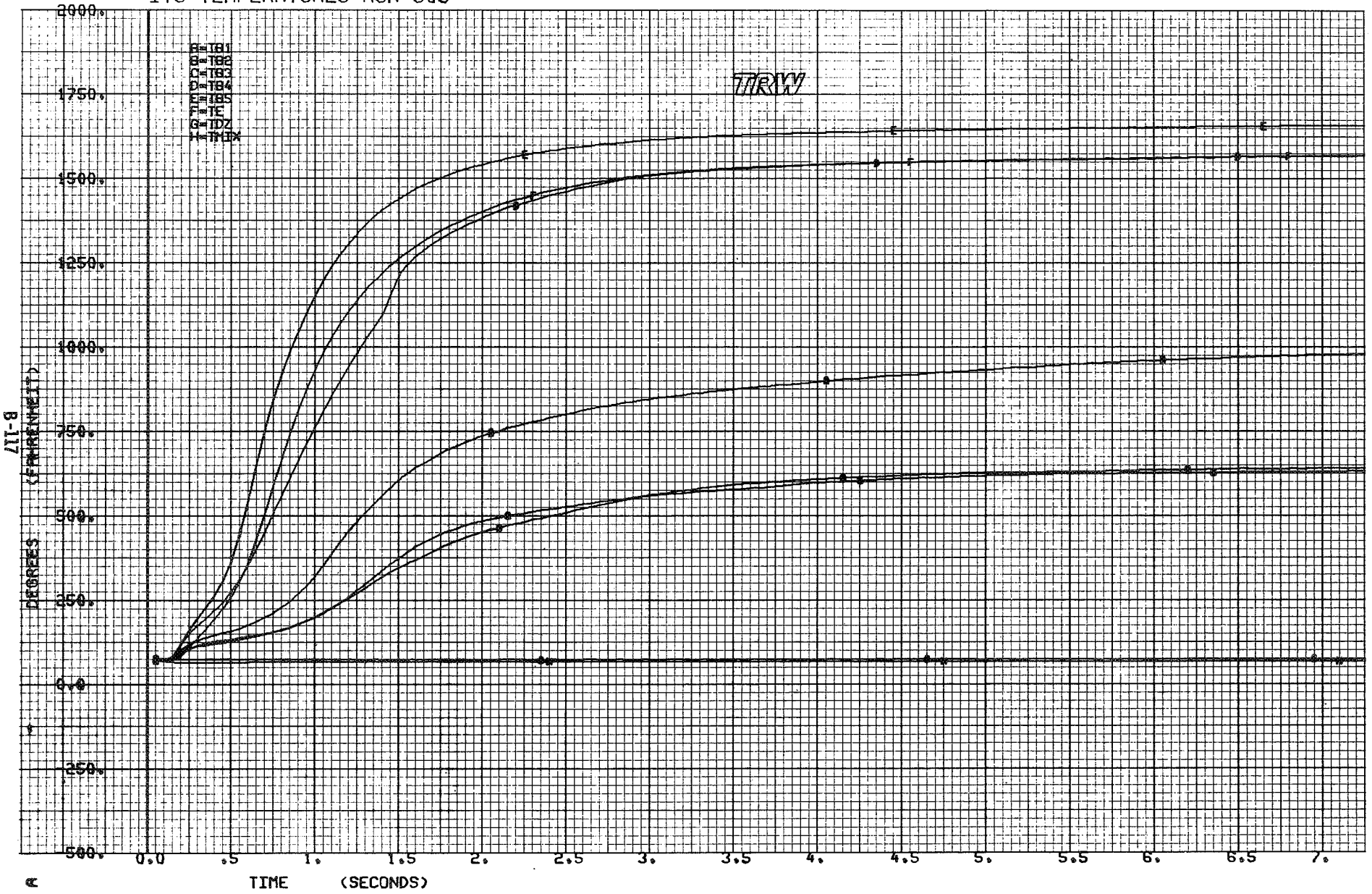
ITS TEMPERATURES RUN 900



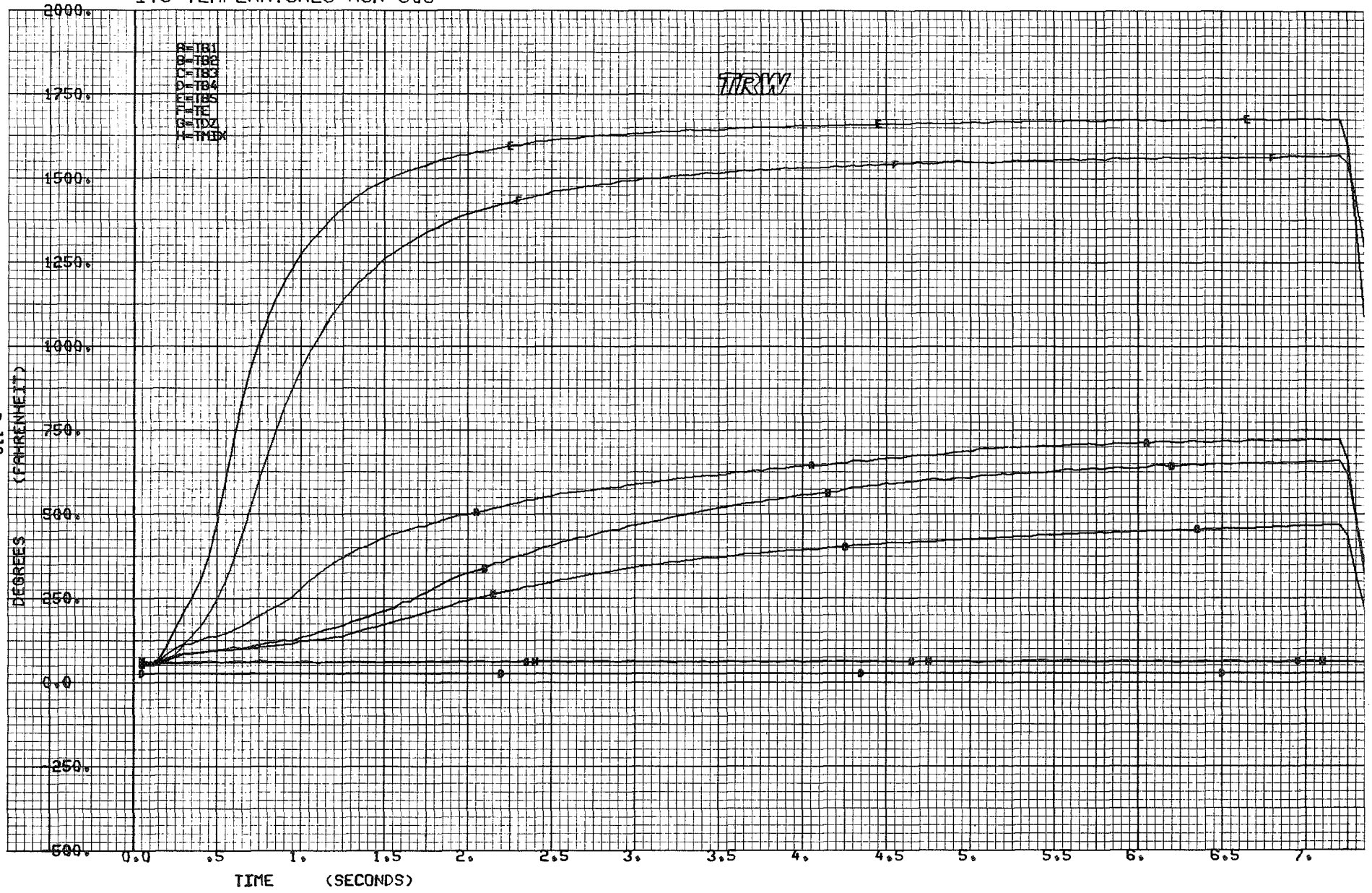
ITS TEMPERATURES RUN 904



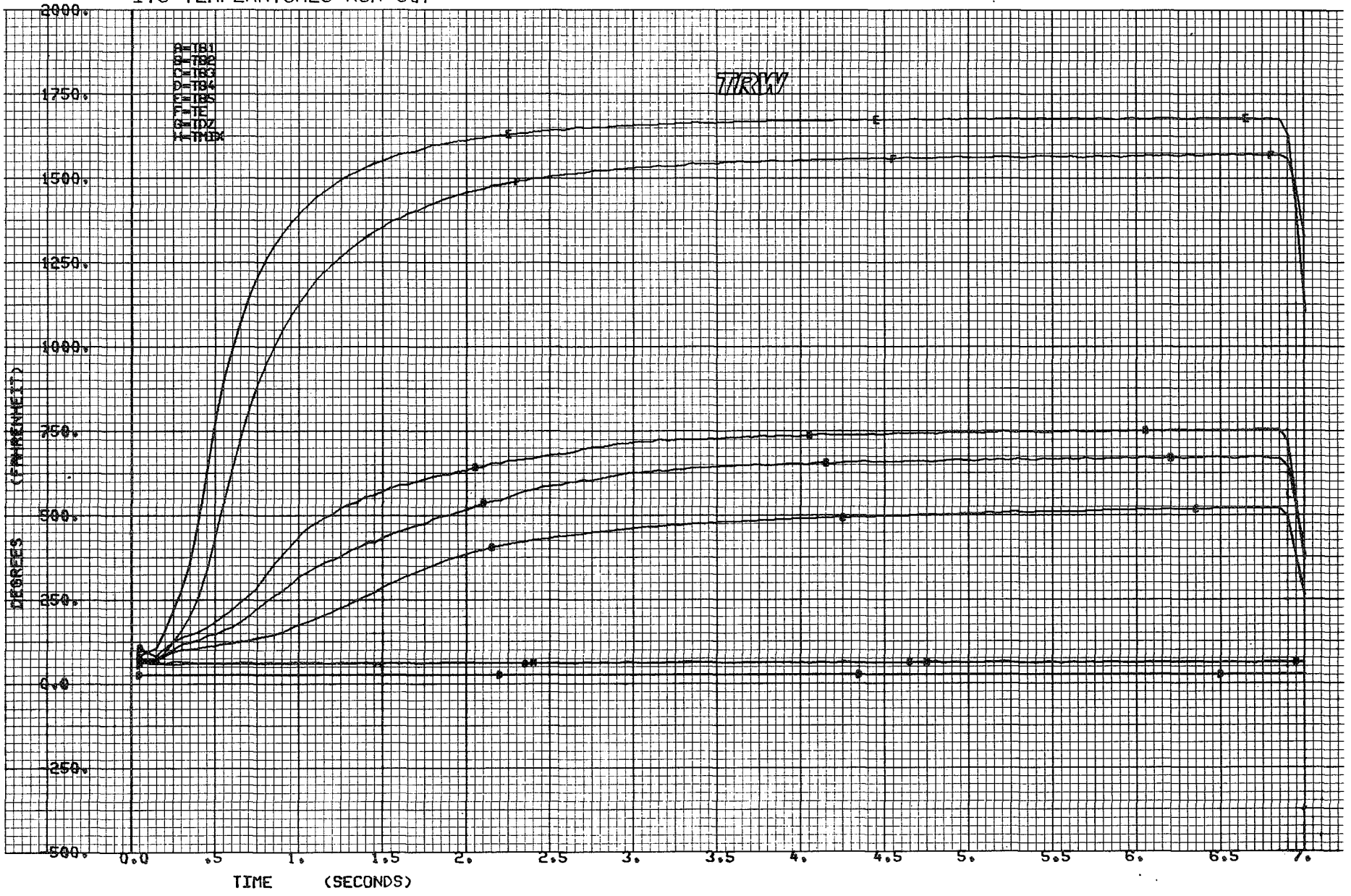
ITS TEMPERATURES RUN 905



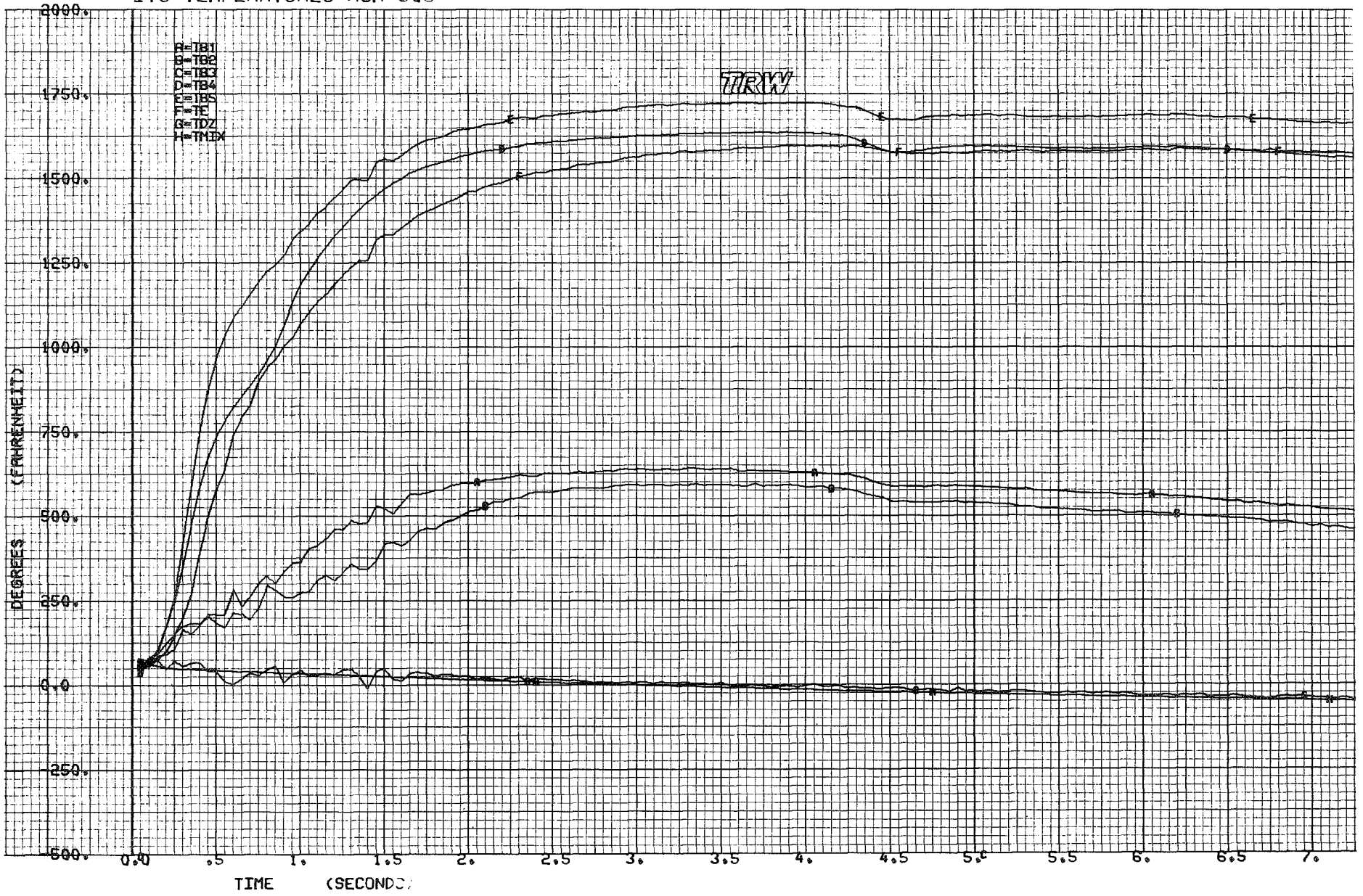
ITS TEMPERATURES RUN 906



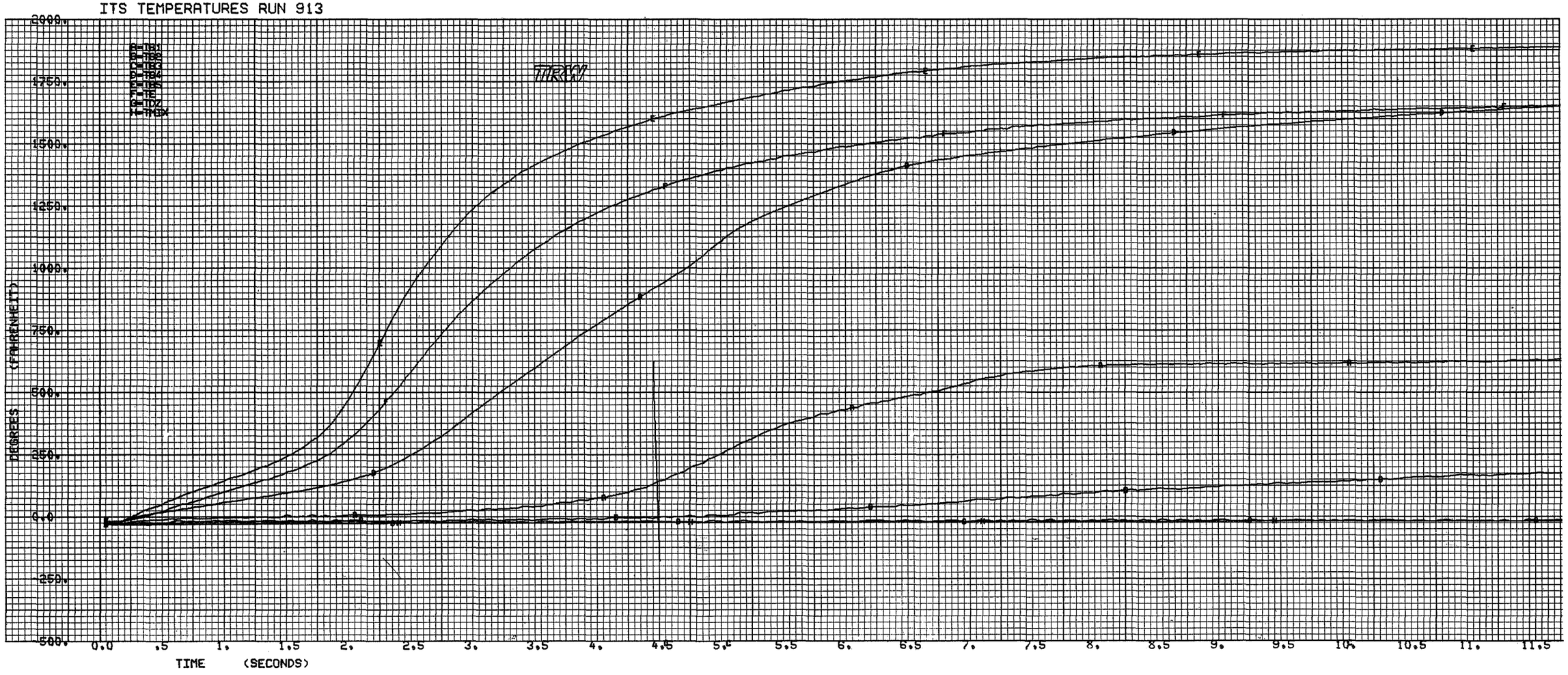
ITS TEMPERATURES RUN 907



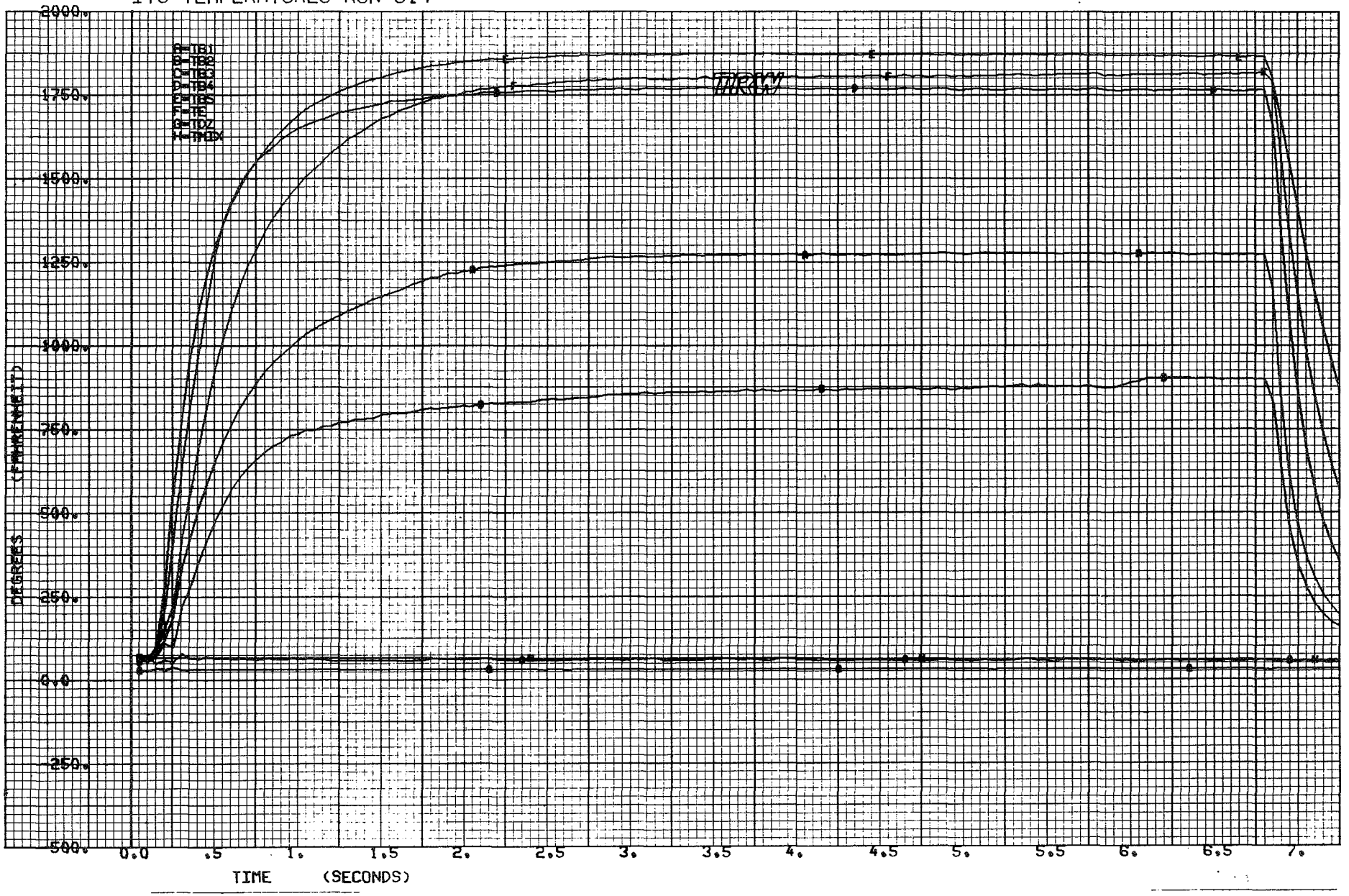
ITS TEMPERATURES RUN 908



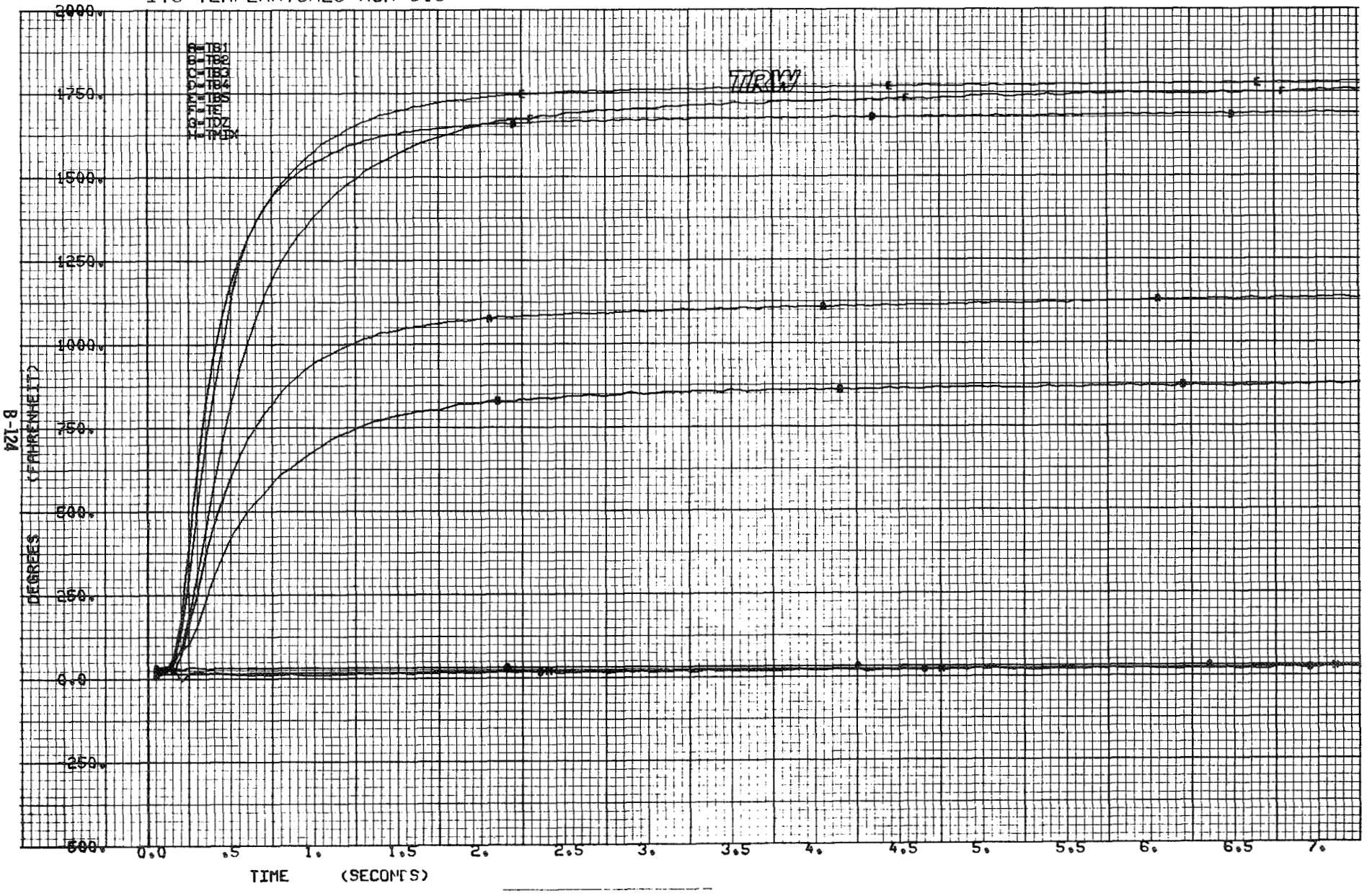
ITS TEMPERATURES RUN 913



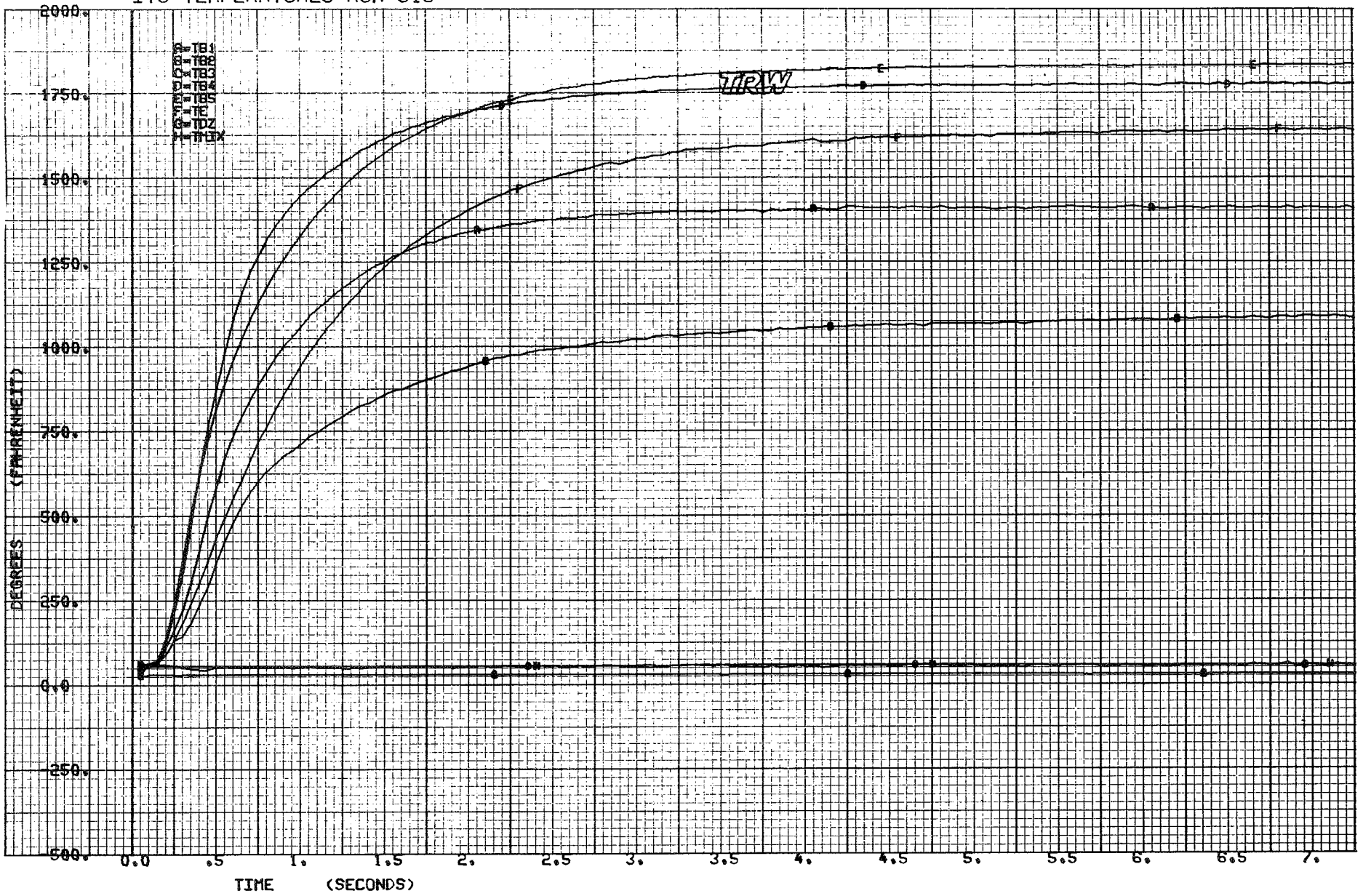
ITS TEMPERATURES RUN 914



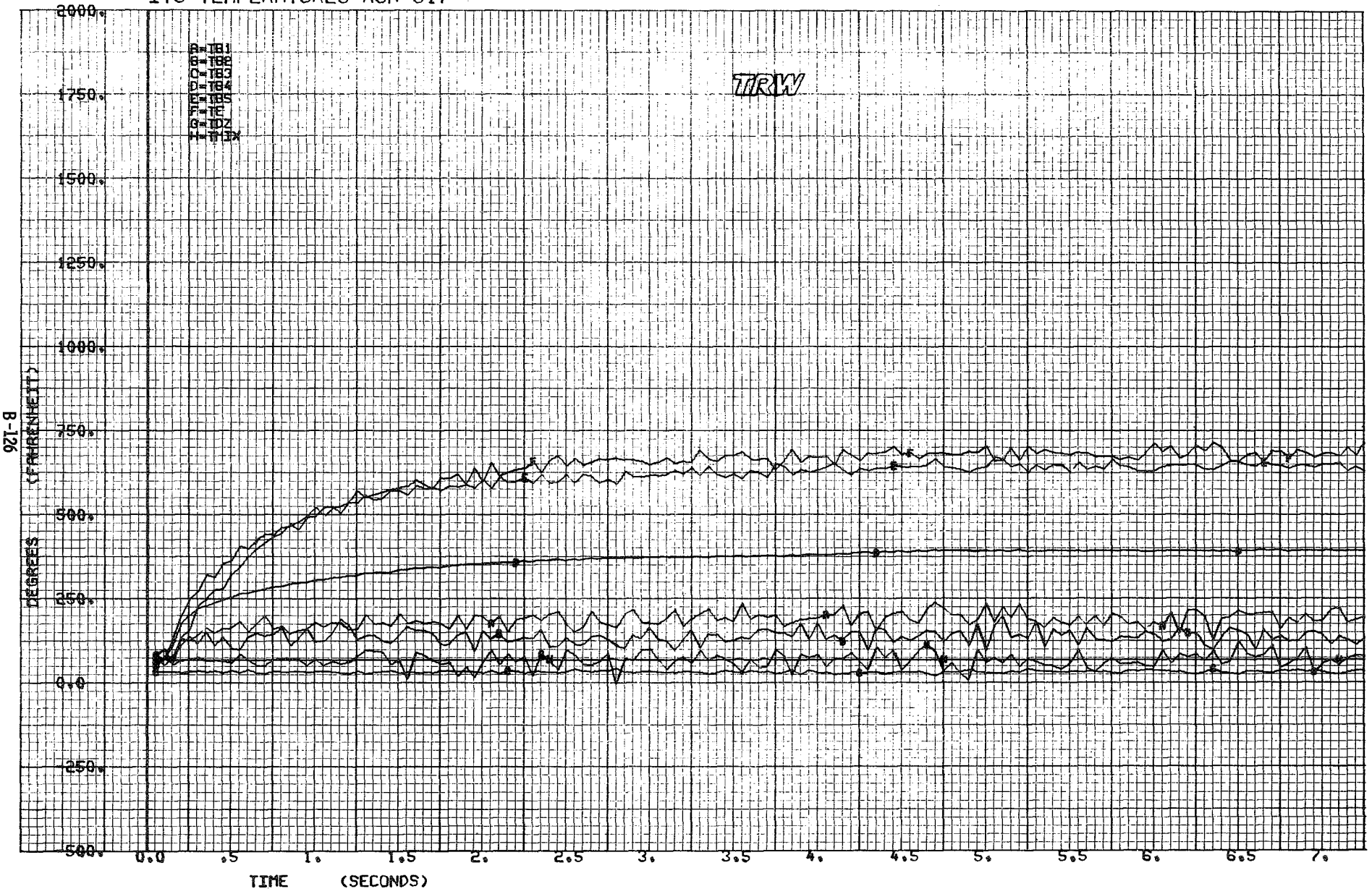
ITS TEMPERATURES RUN 915



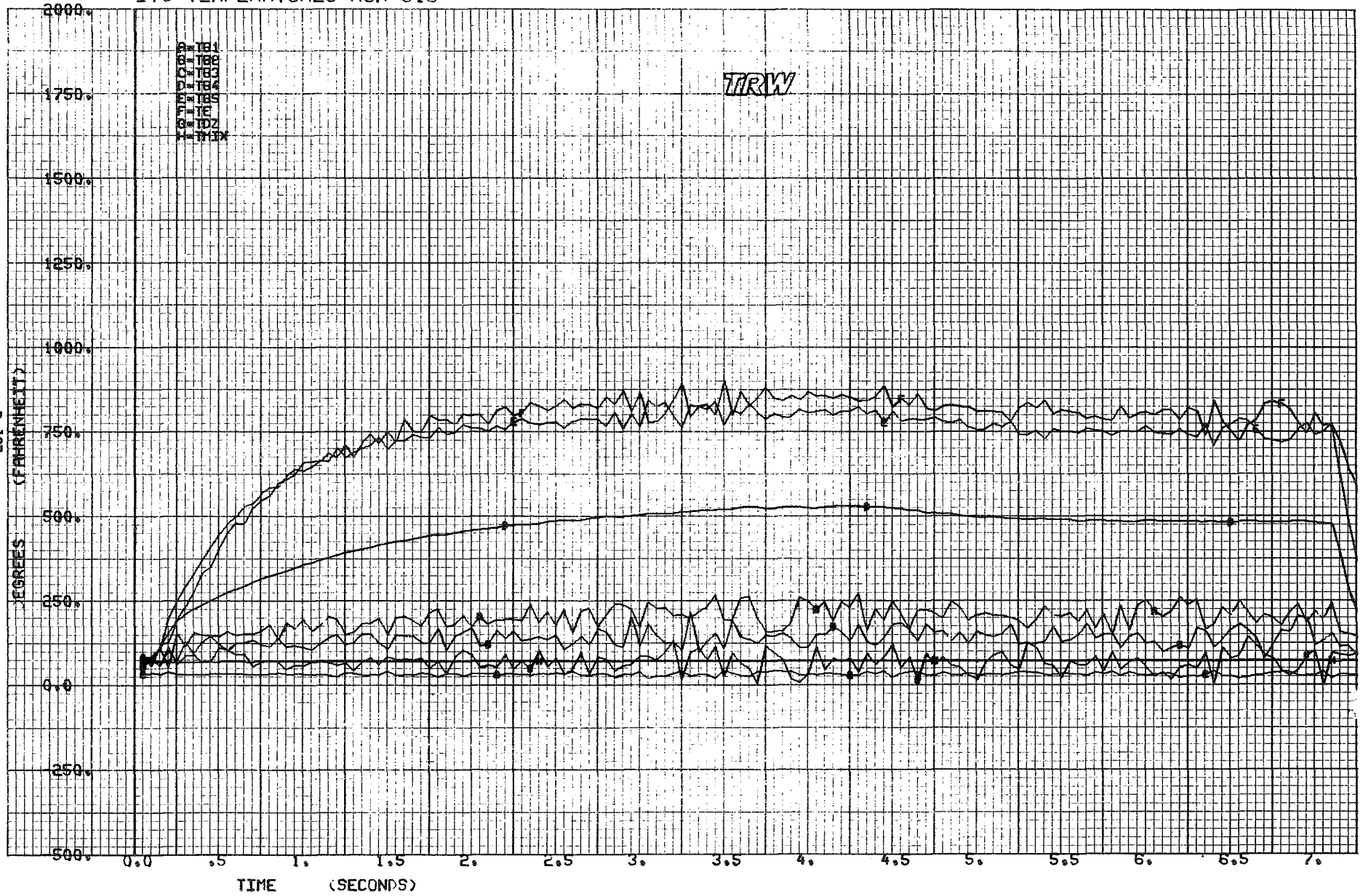
ITS TEMPERATURES RUN 916



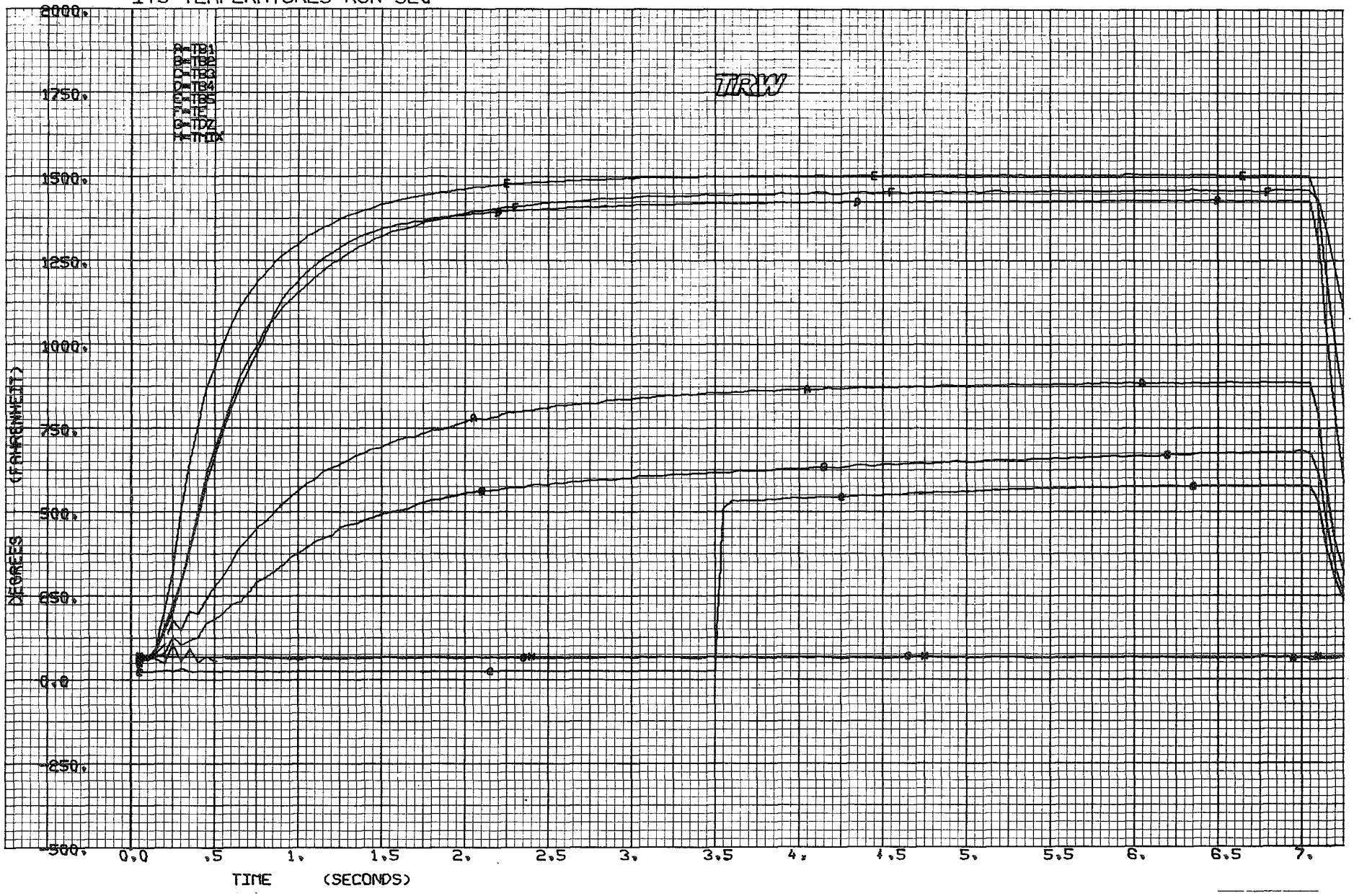
ITS TEMPERATURES RUN 917



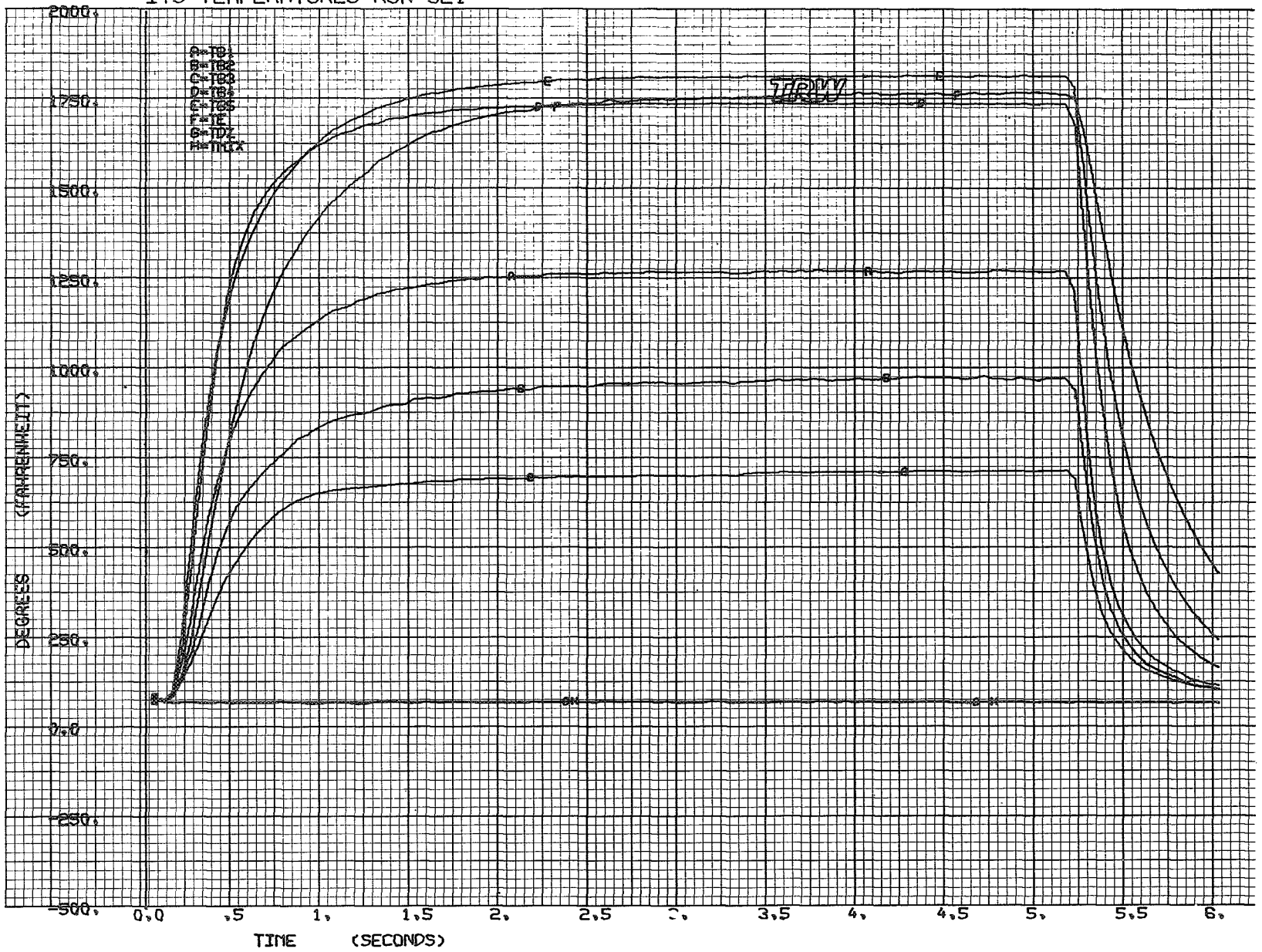
ITS TEMPERATURES RUN 918



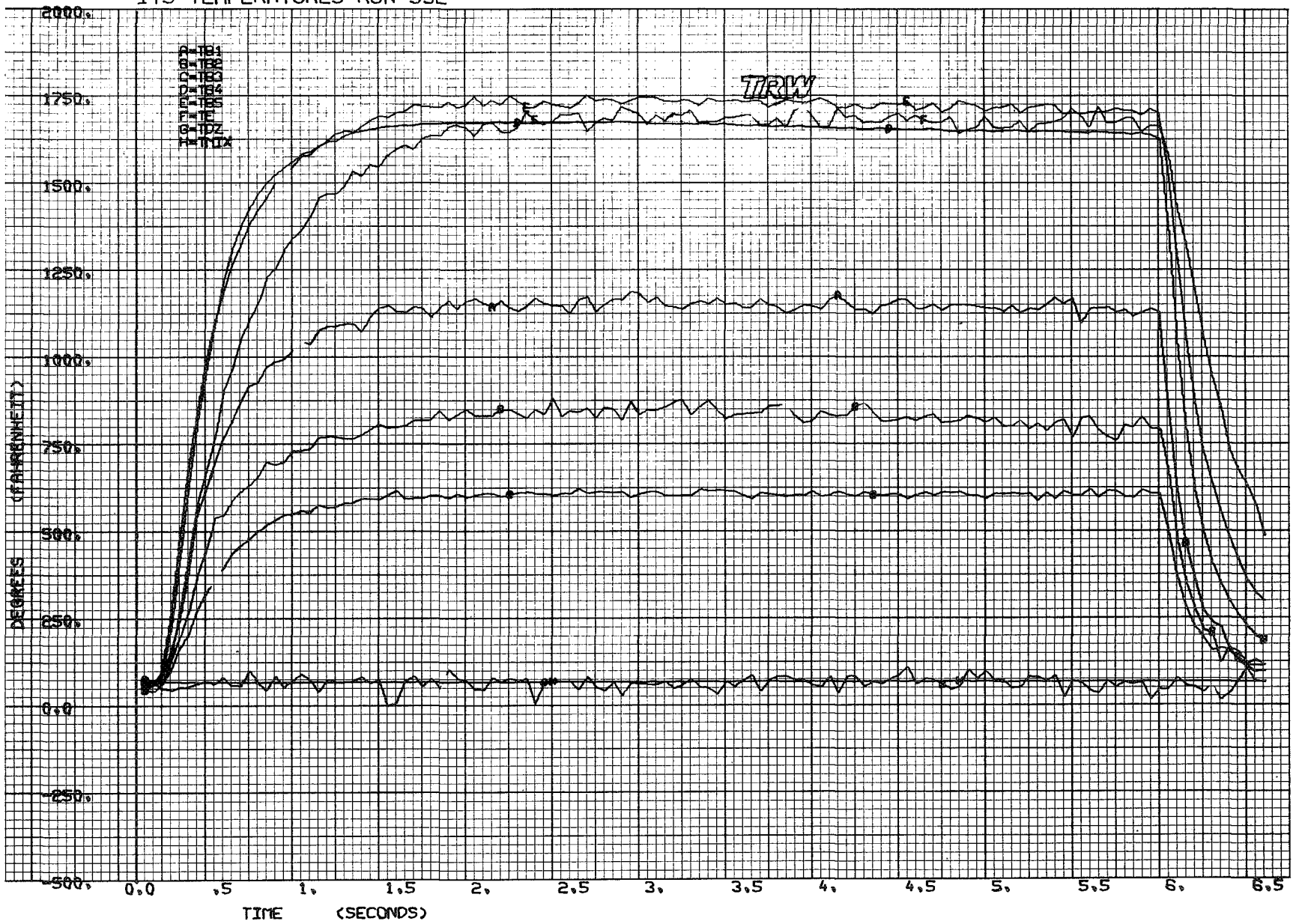
ITS TEMPERATURES RUN 92Q



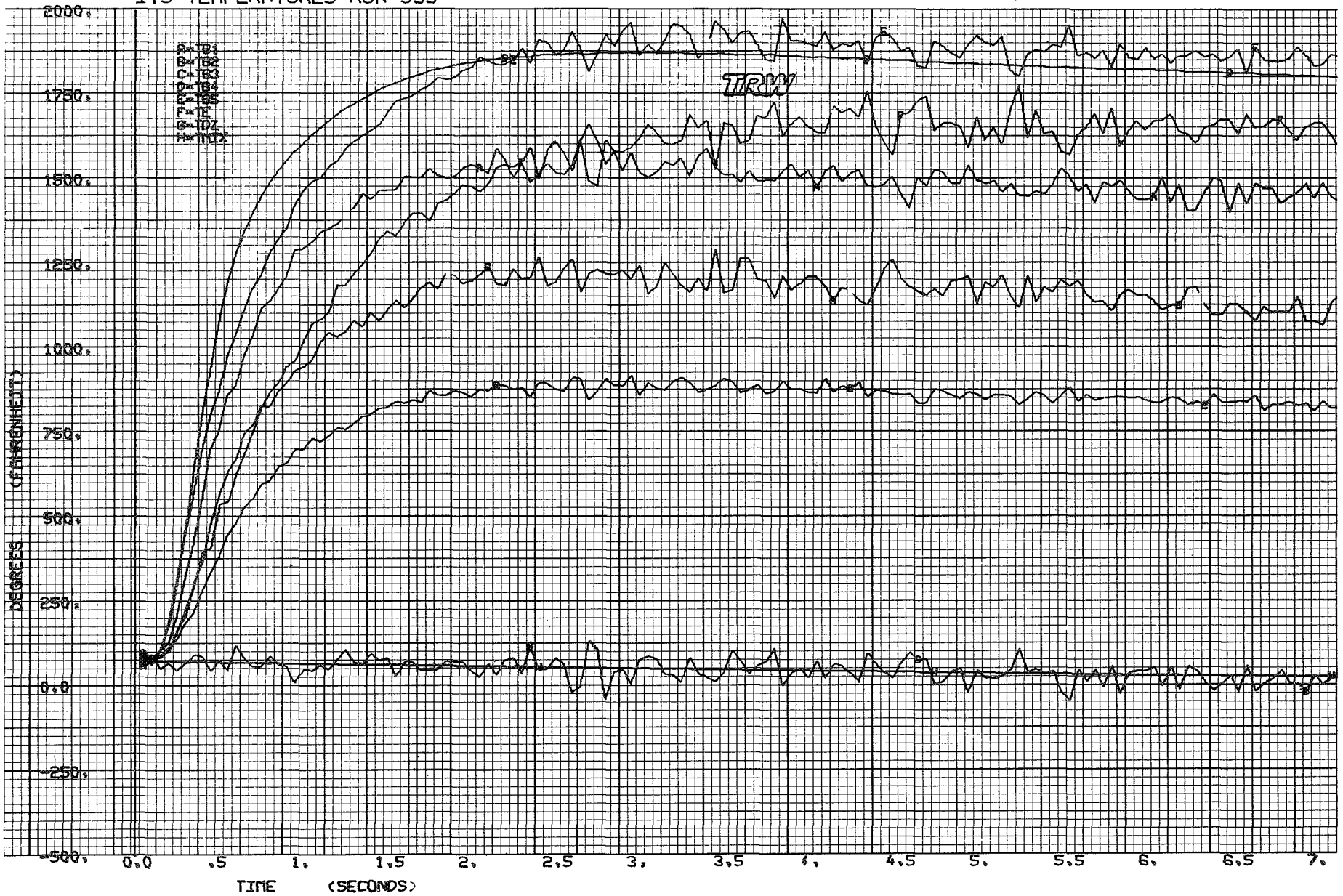
ITS TEMPERATURES RUN 921

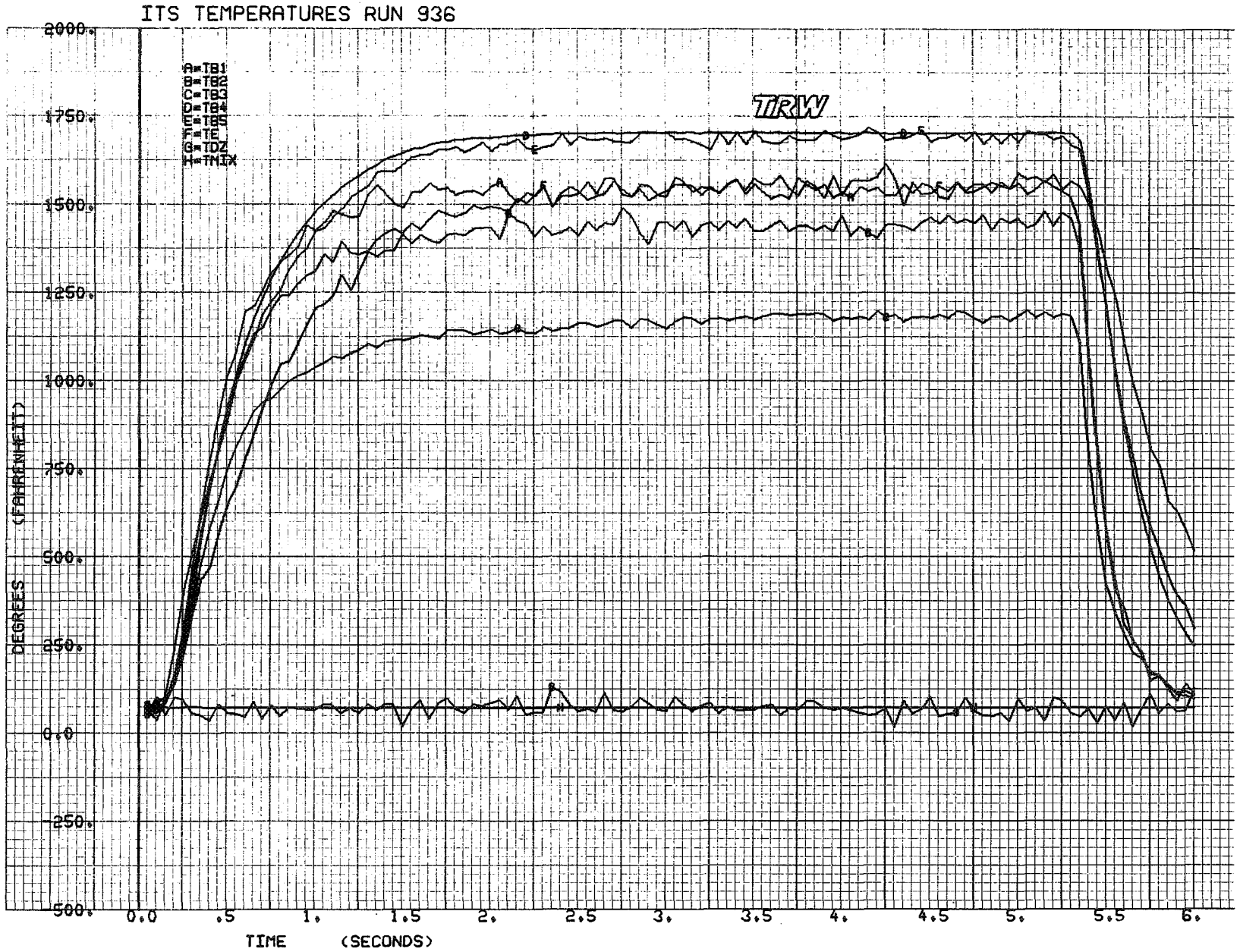


ITS TEMPERATURES RUN 932

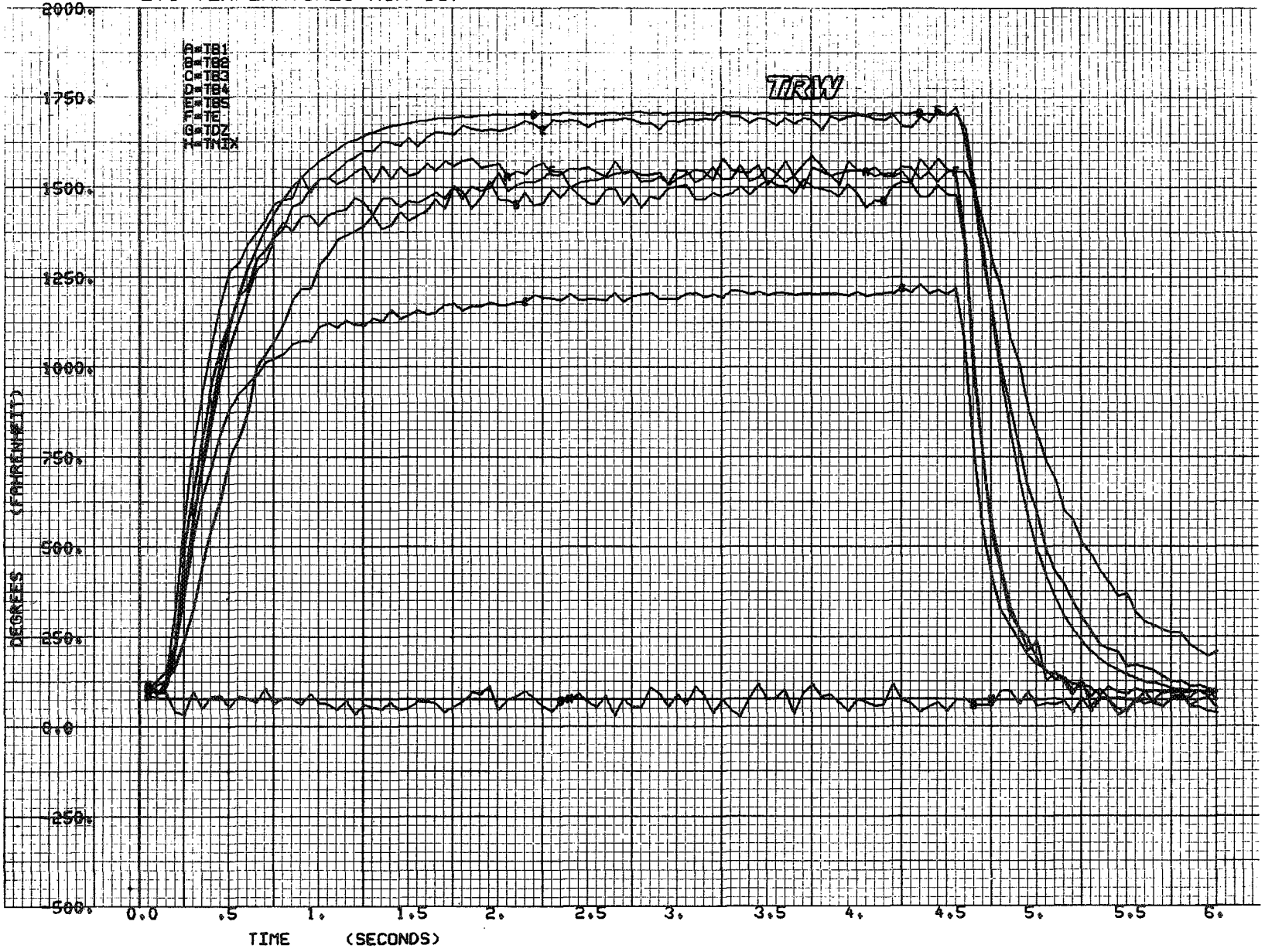


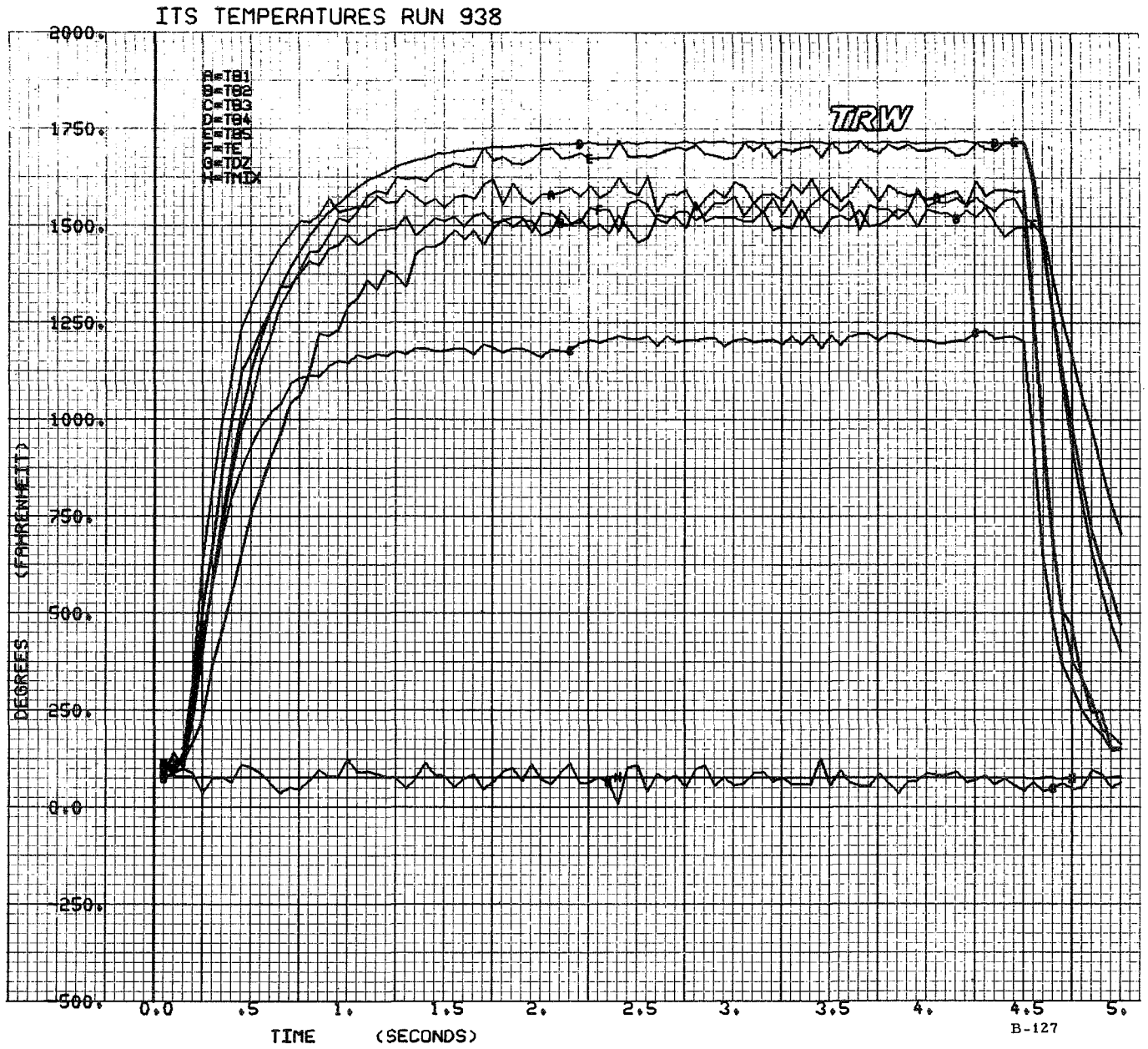
ITS TEMPERATURES RUN 933

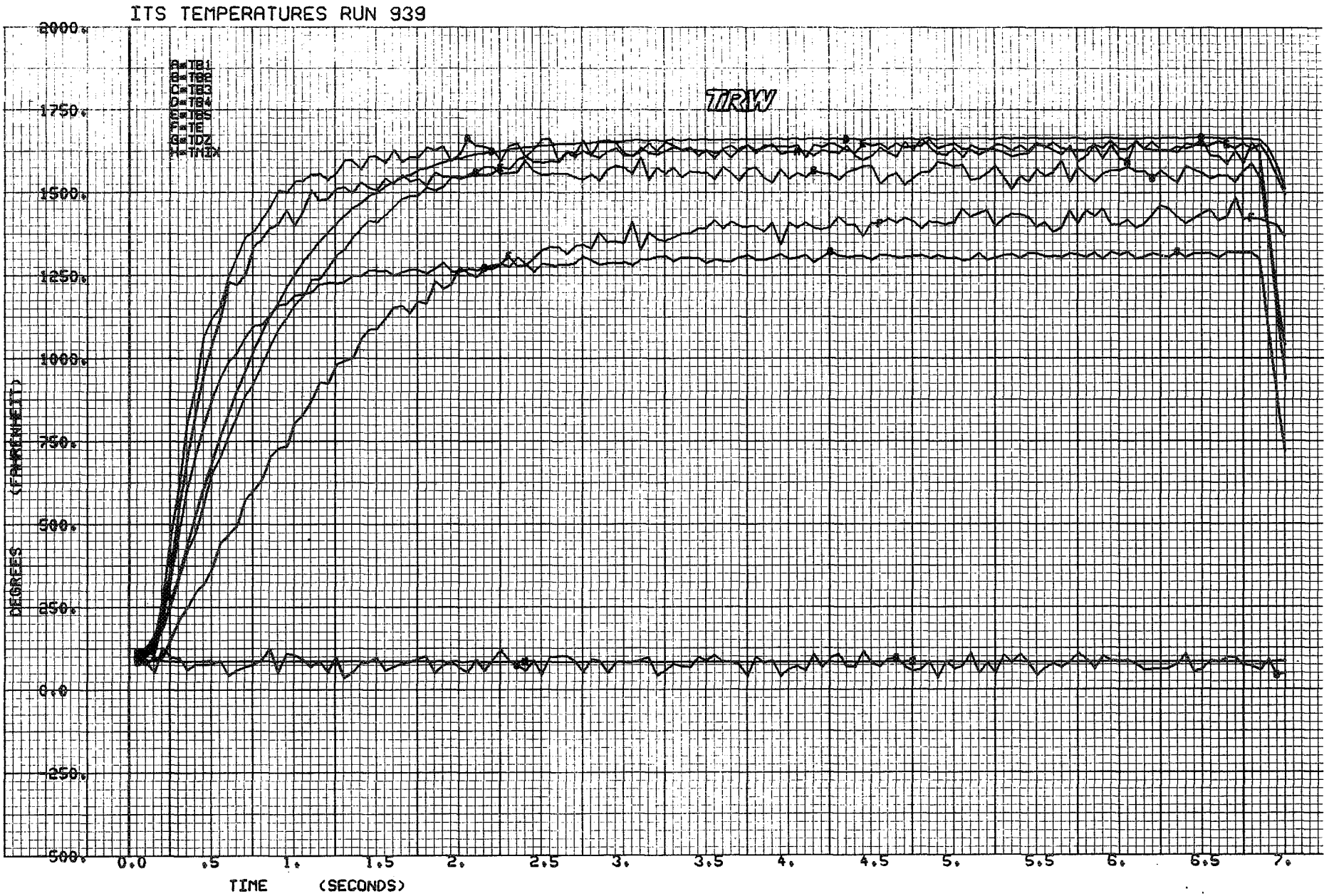



ITS TEMPERATURES RUN 936
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
TIME (SECONDS)

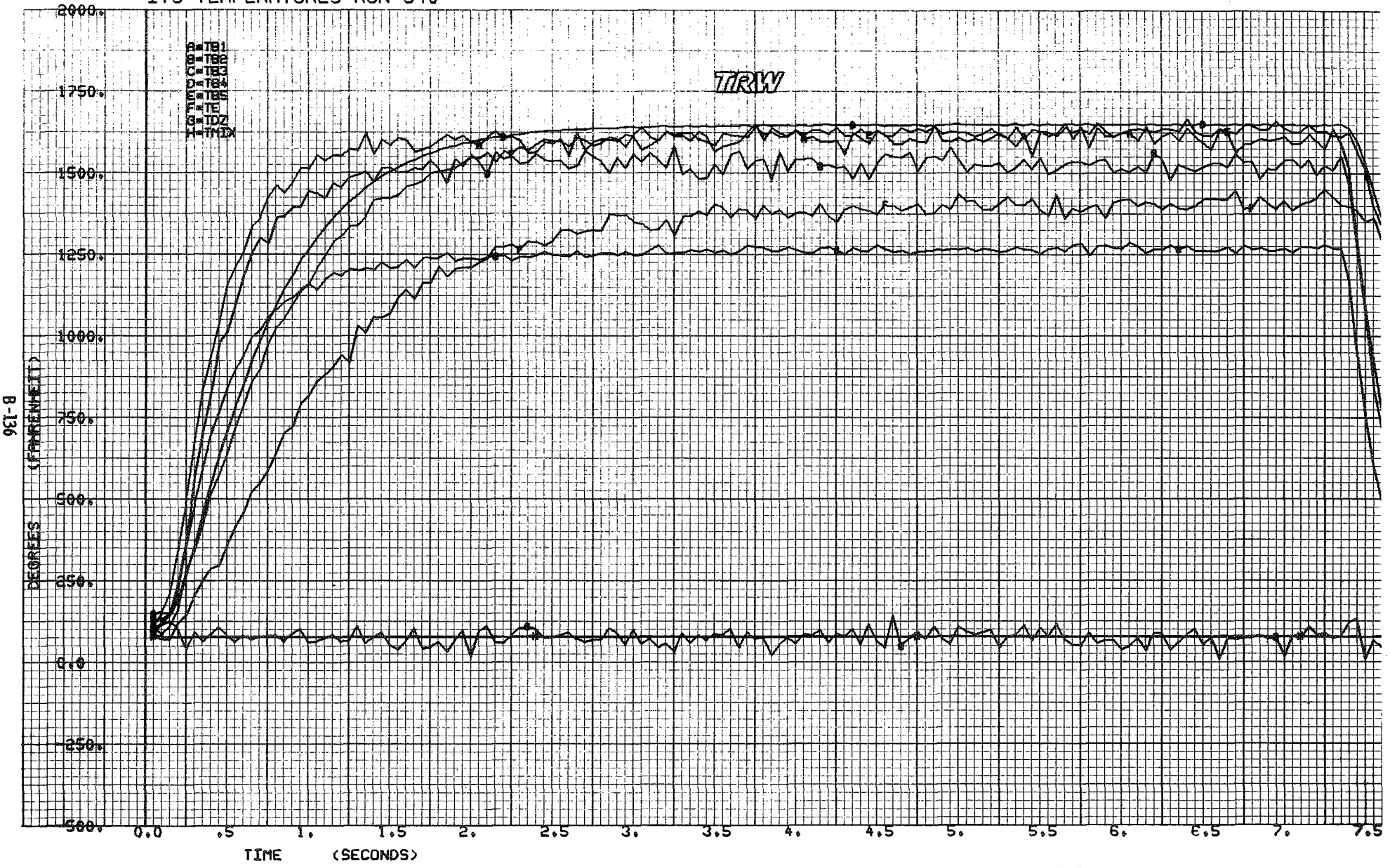
ITS TEMPERATURES RUN 937



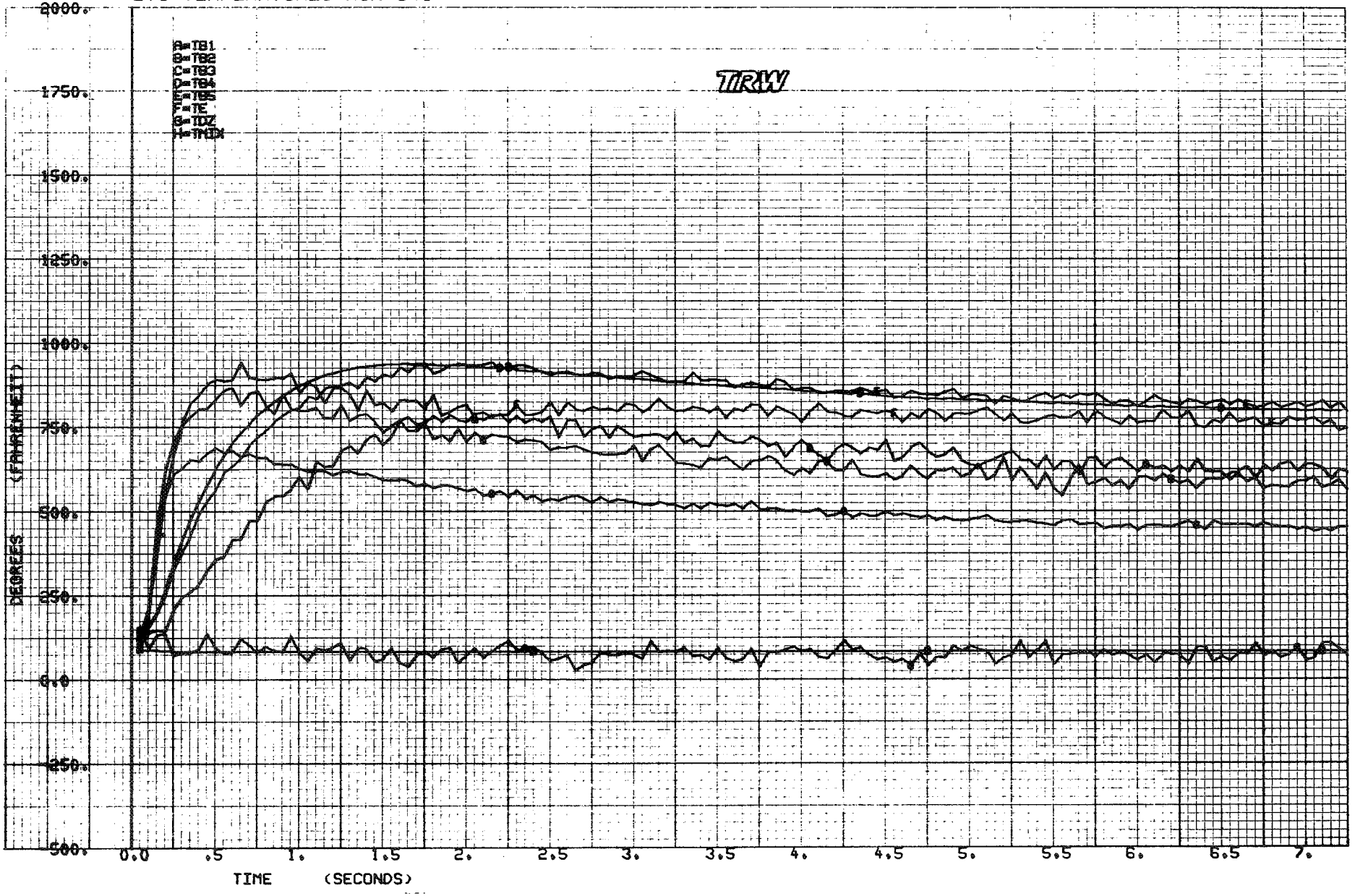

ITS TEMPERATURES RUN 938
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TDZ
H=TNDX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
TIME (SECONDS)
B-127


ITS TEMPERATURES RUN 939
A=TB1
B=TB2
C=TB3
D=TB4
E=TB5
F=TE
G=TDZ
H=THIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
7.
TIME (SECONDS)

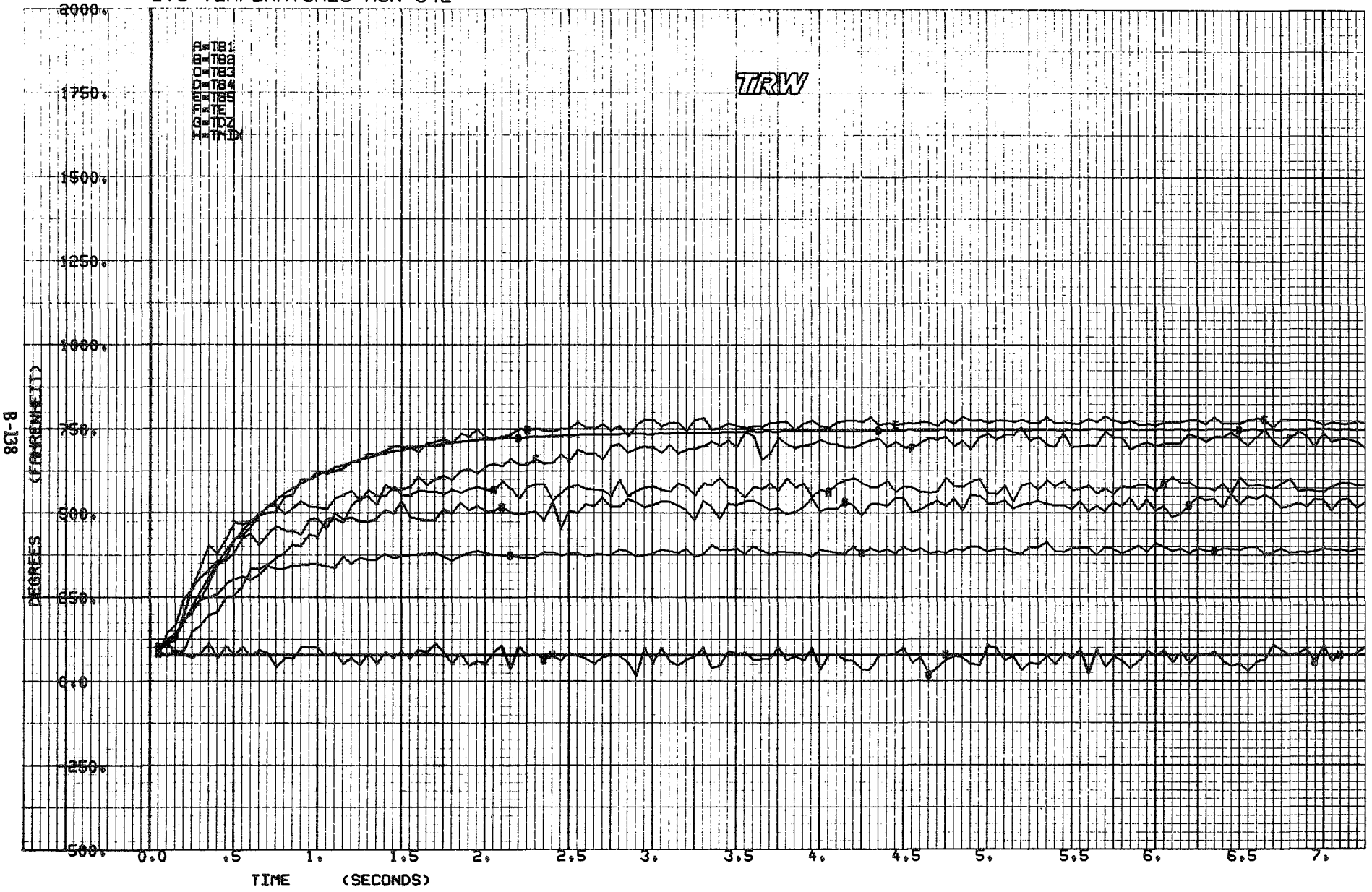
ITS TEMPERATURES RUN 940



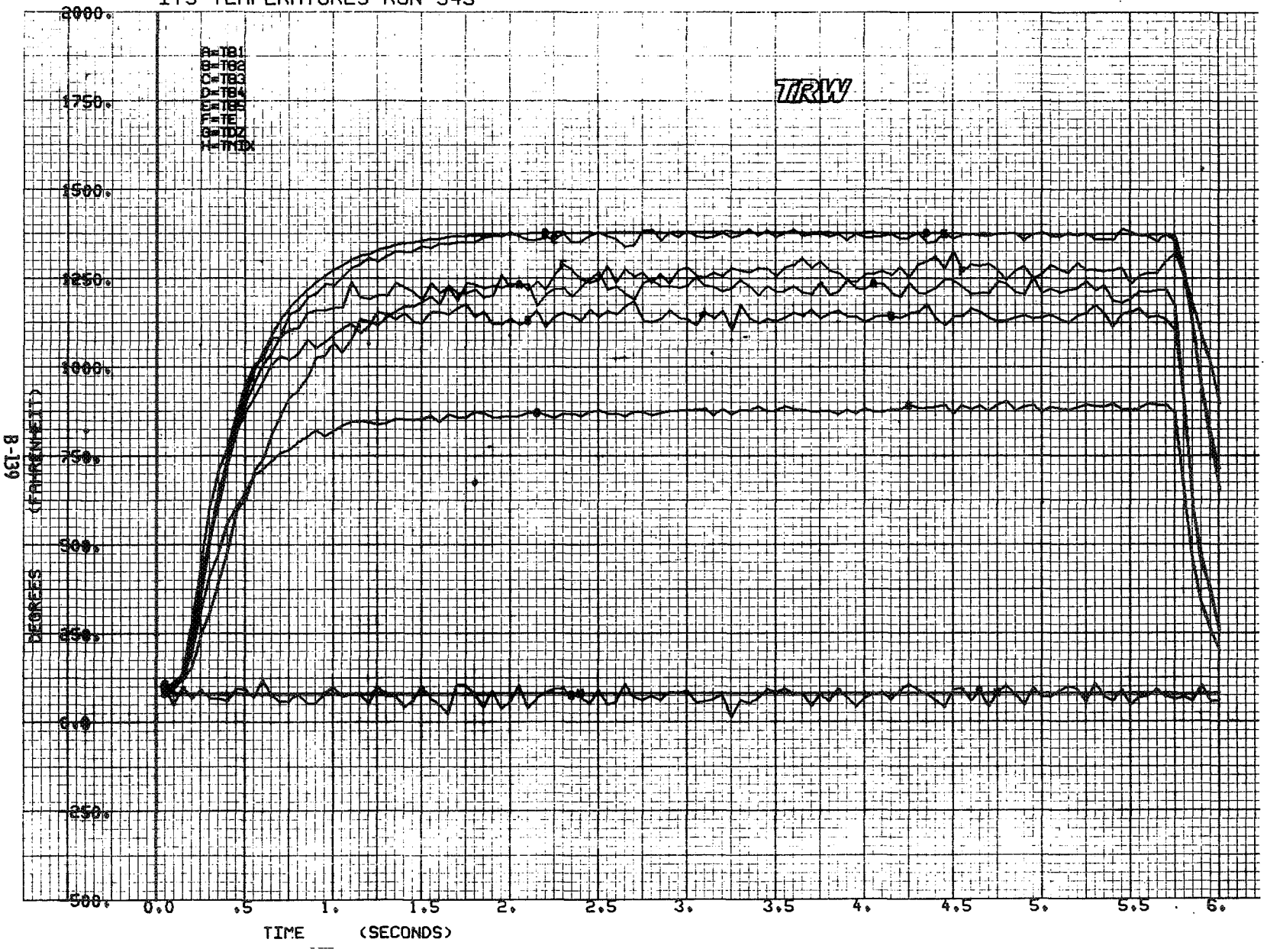
ITS TEMPERATURES RUN 941



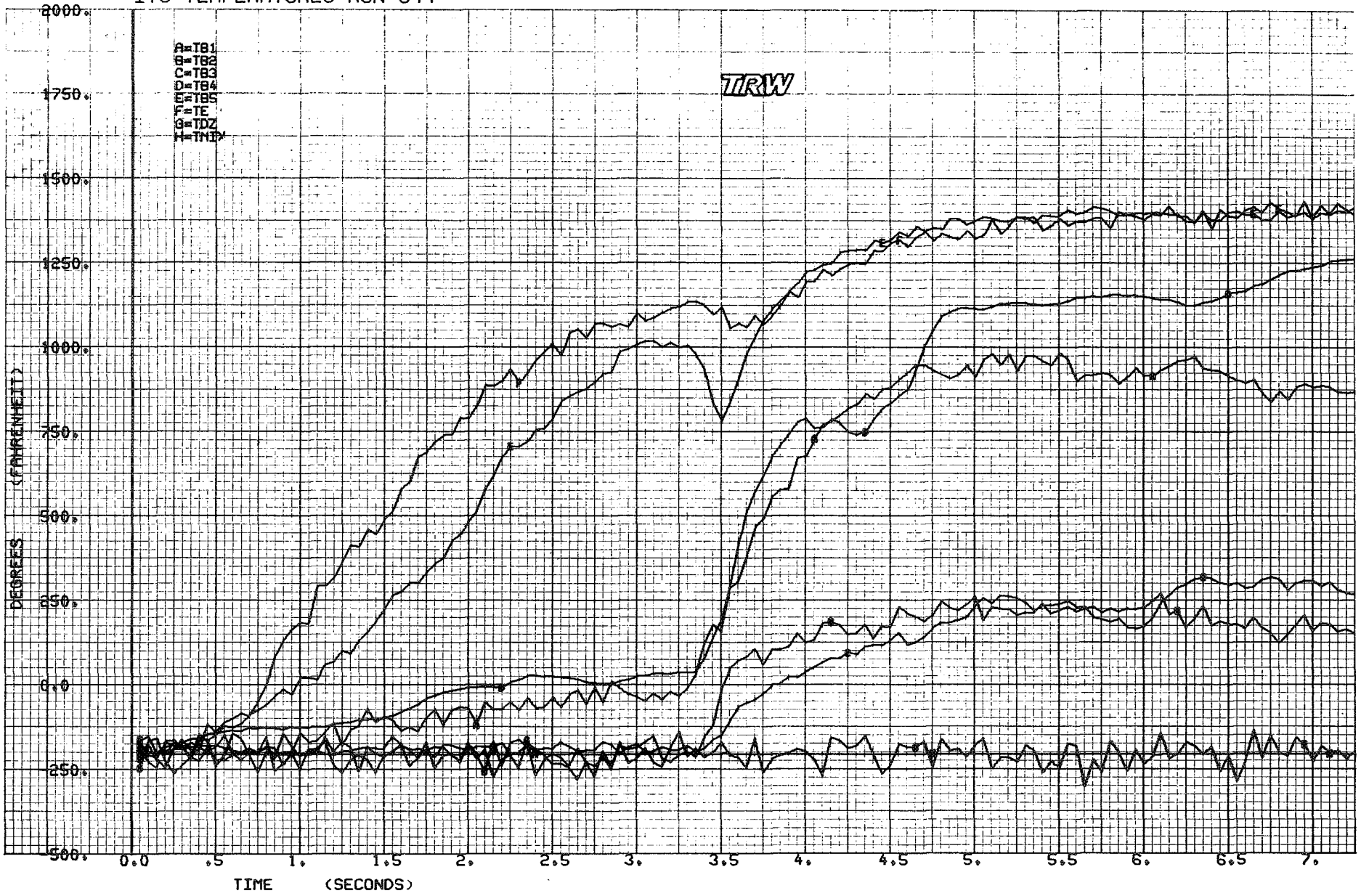
ITS TEMPERATURES RUN 942



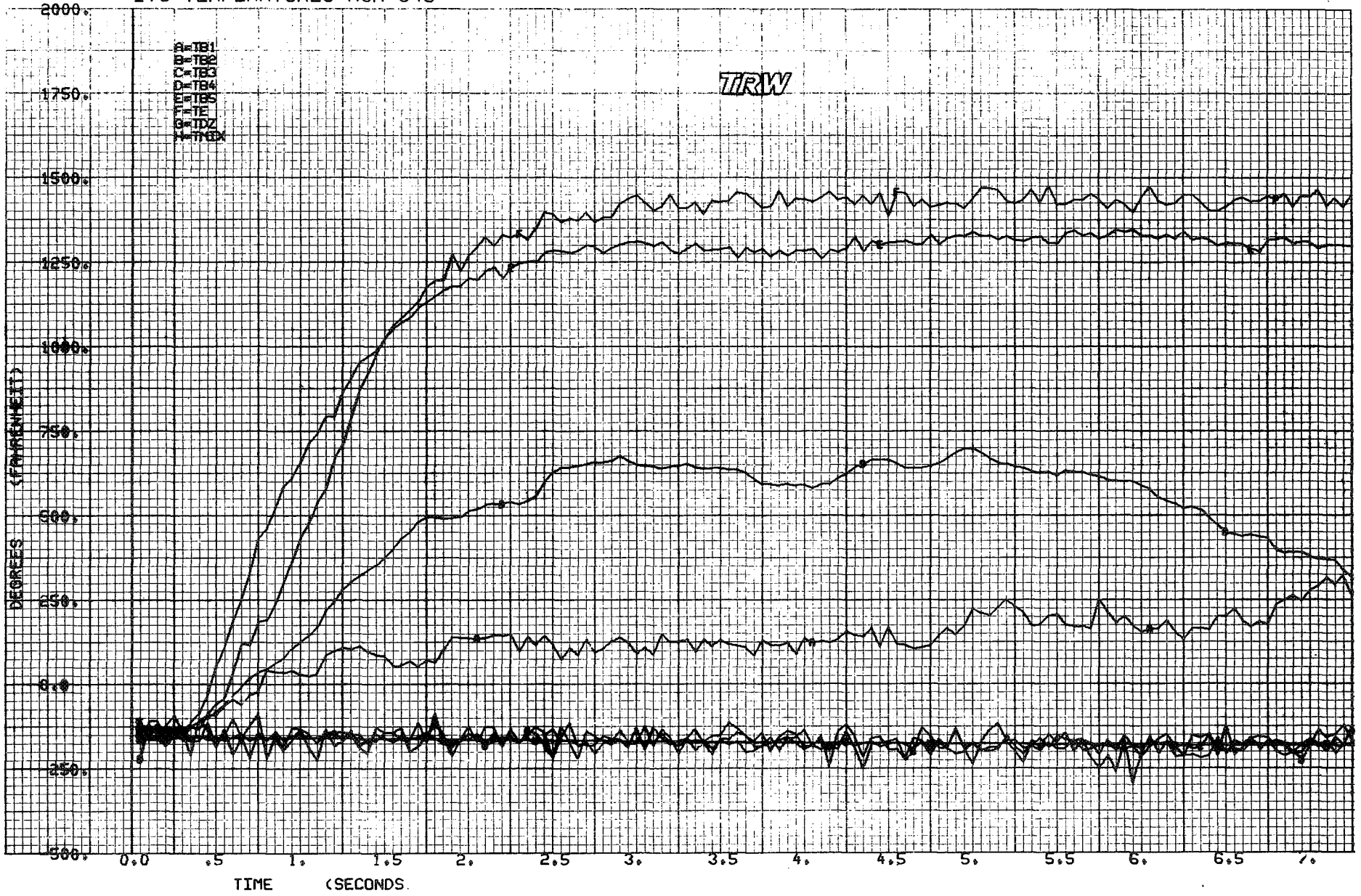
ITS TEMPERATURES RUN 943



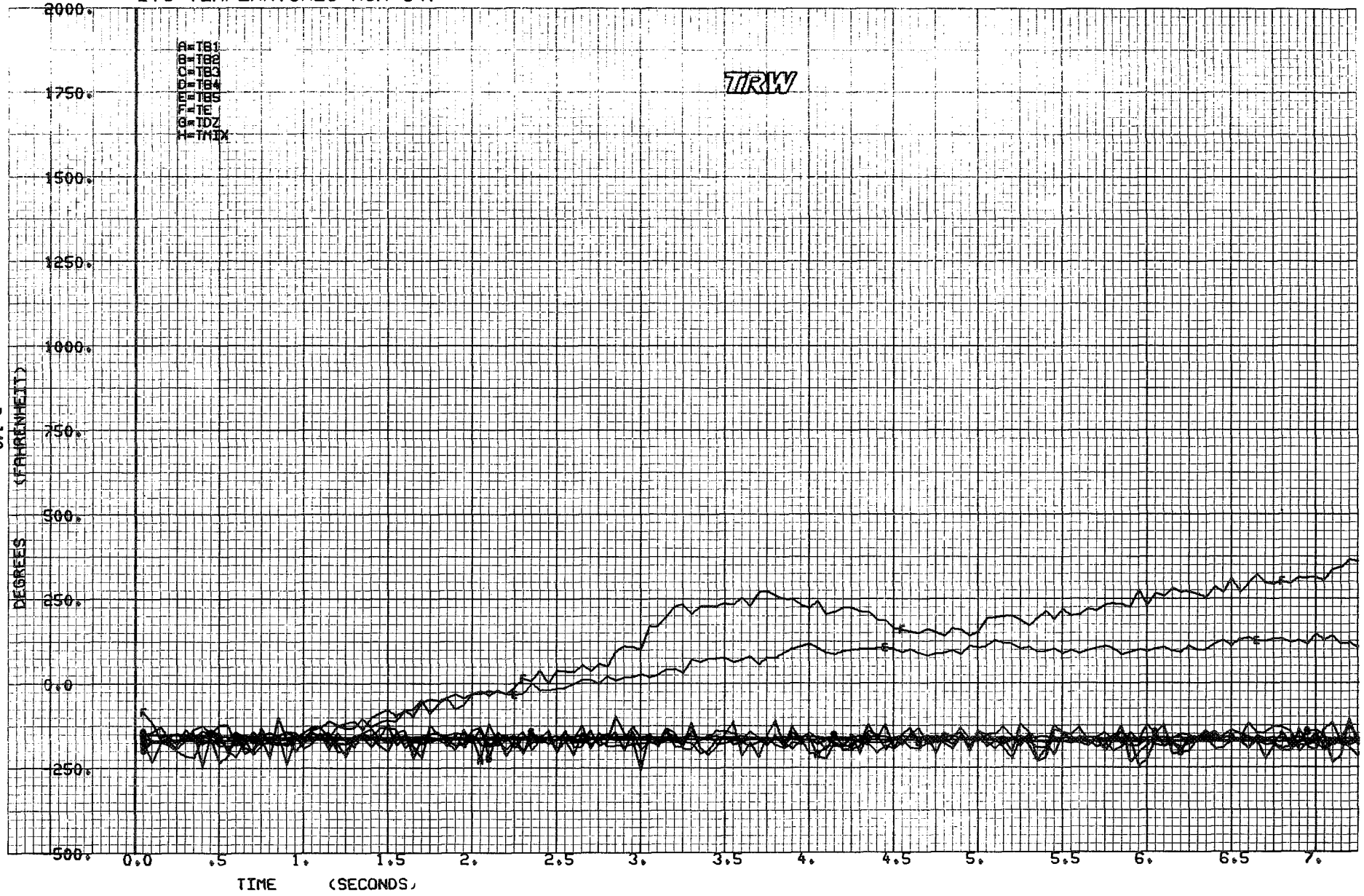
ITS TEMPERATURES RUN 944



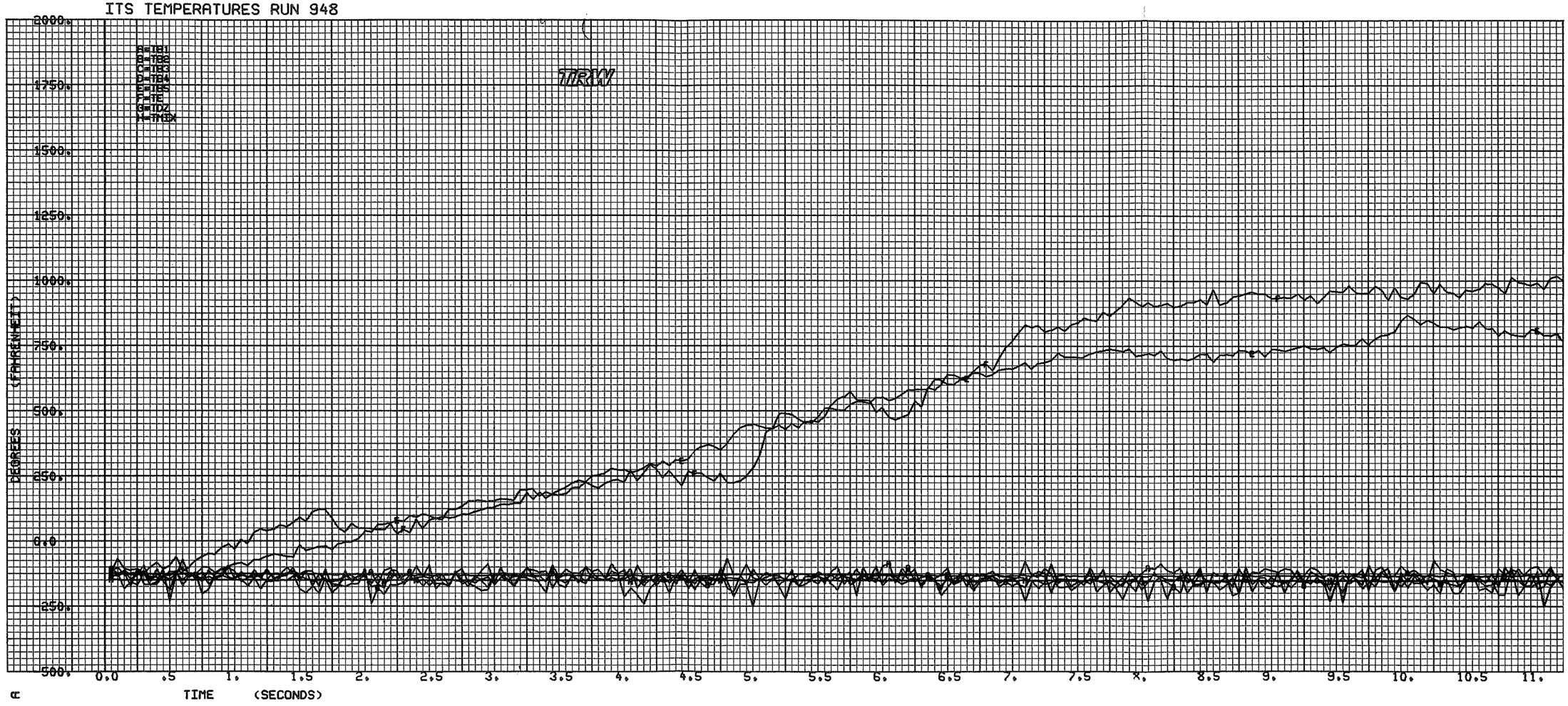
ITS TEMPERATURES RUN 946



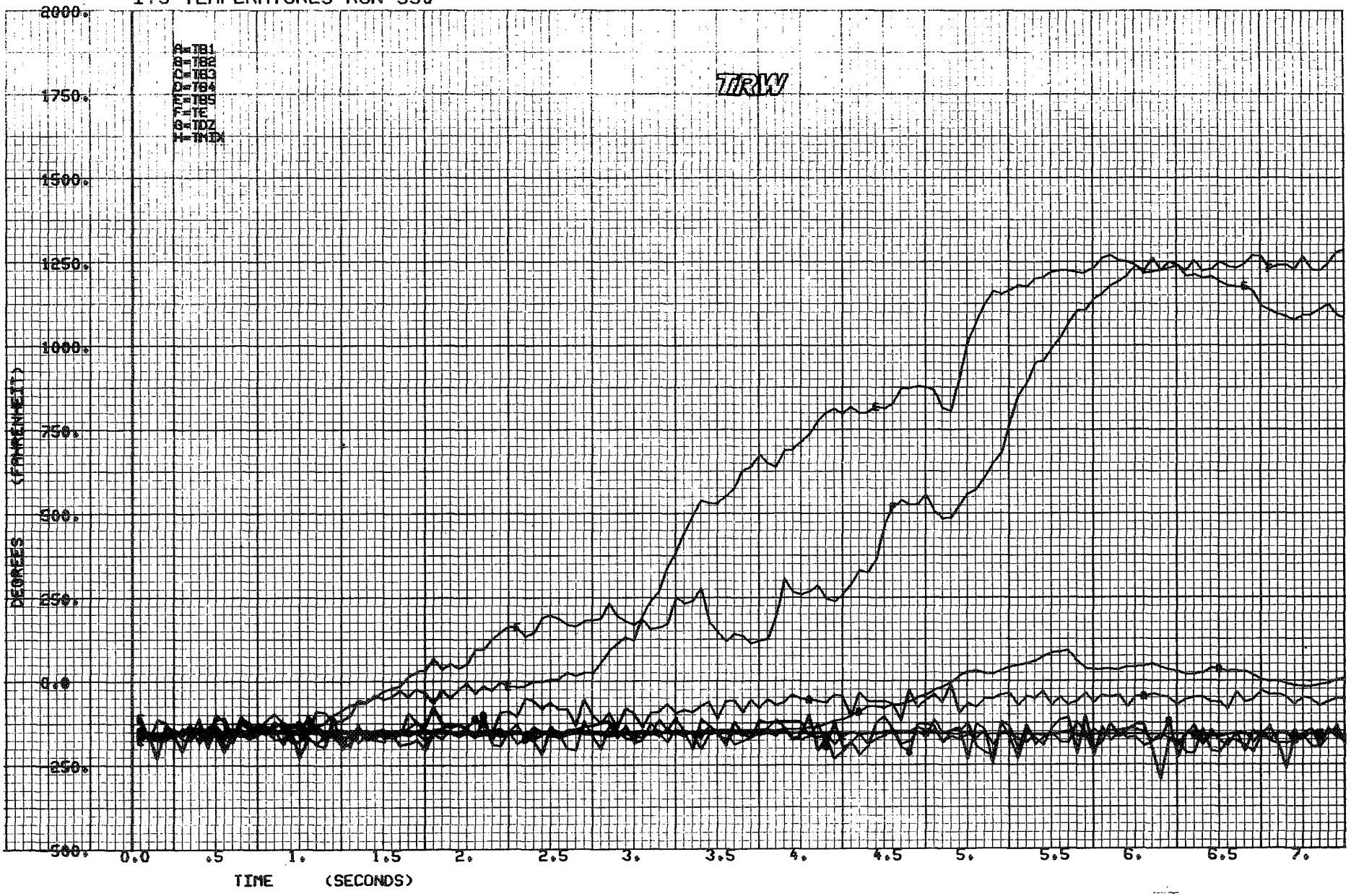
ITS TEMPERATURES RUN 947

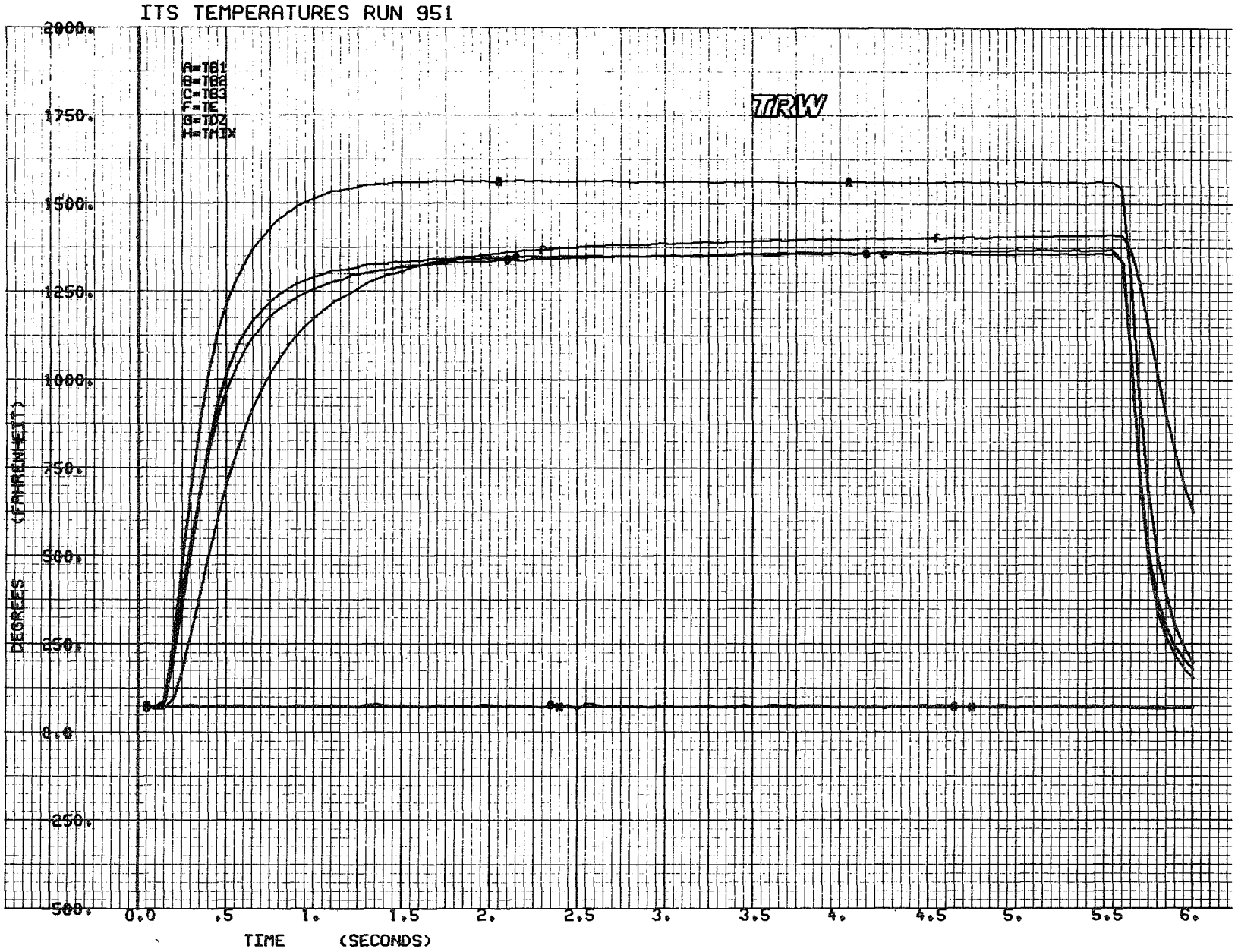


ITS TEMPERATURES RUN 948

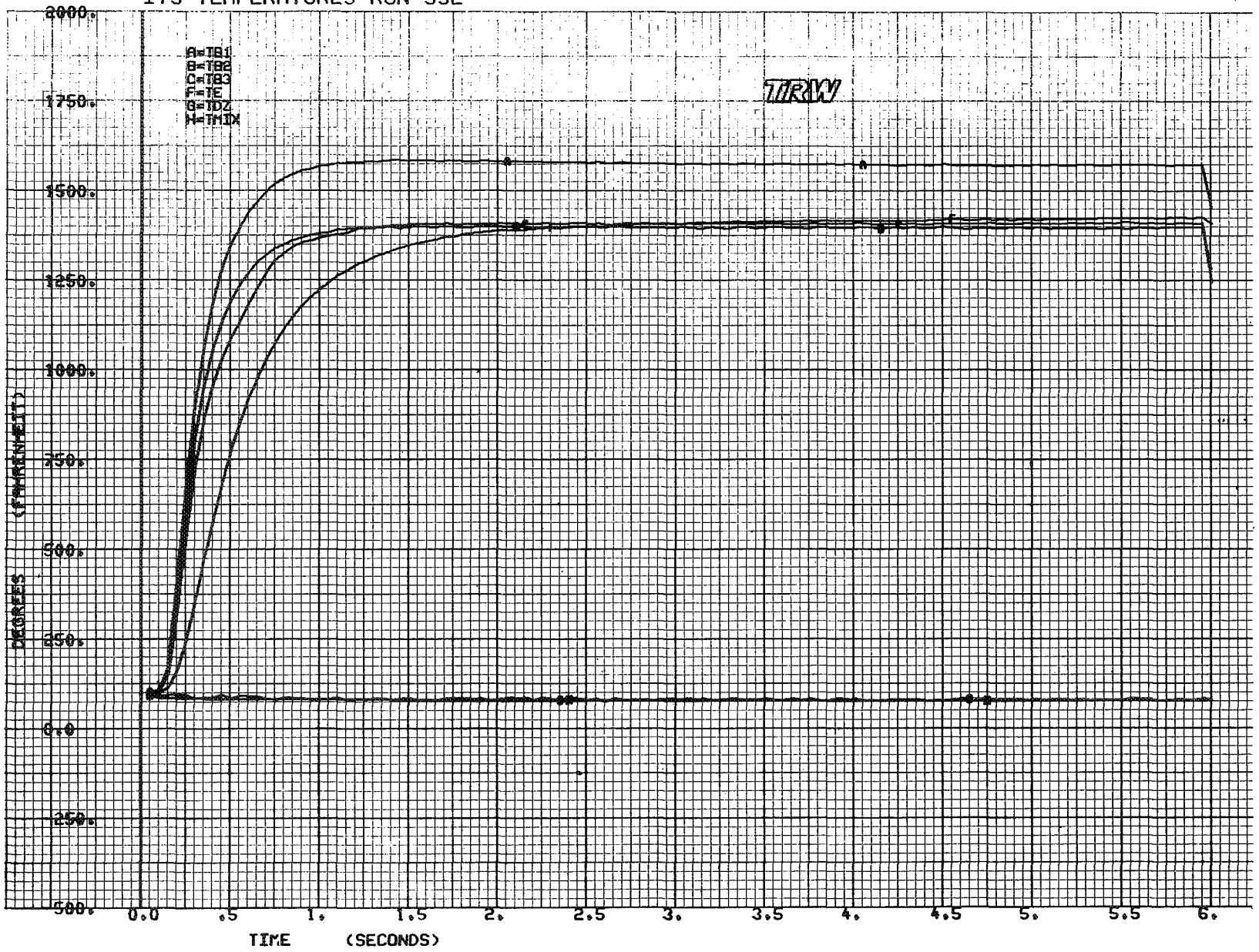


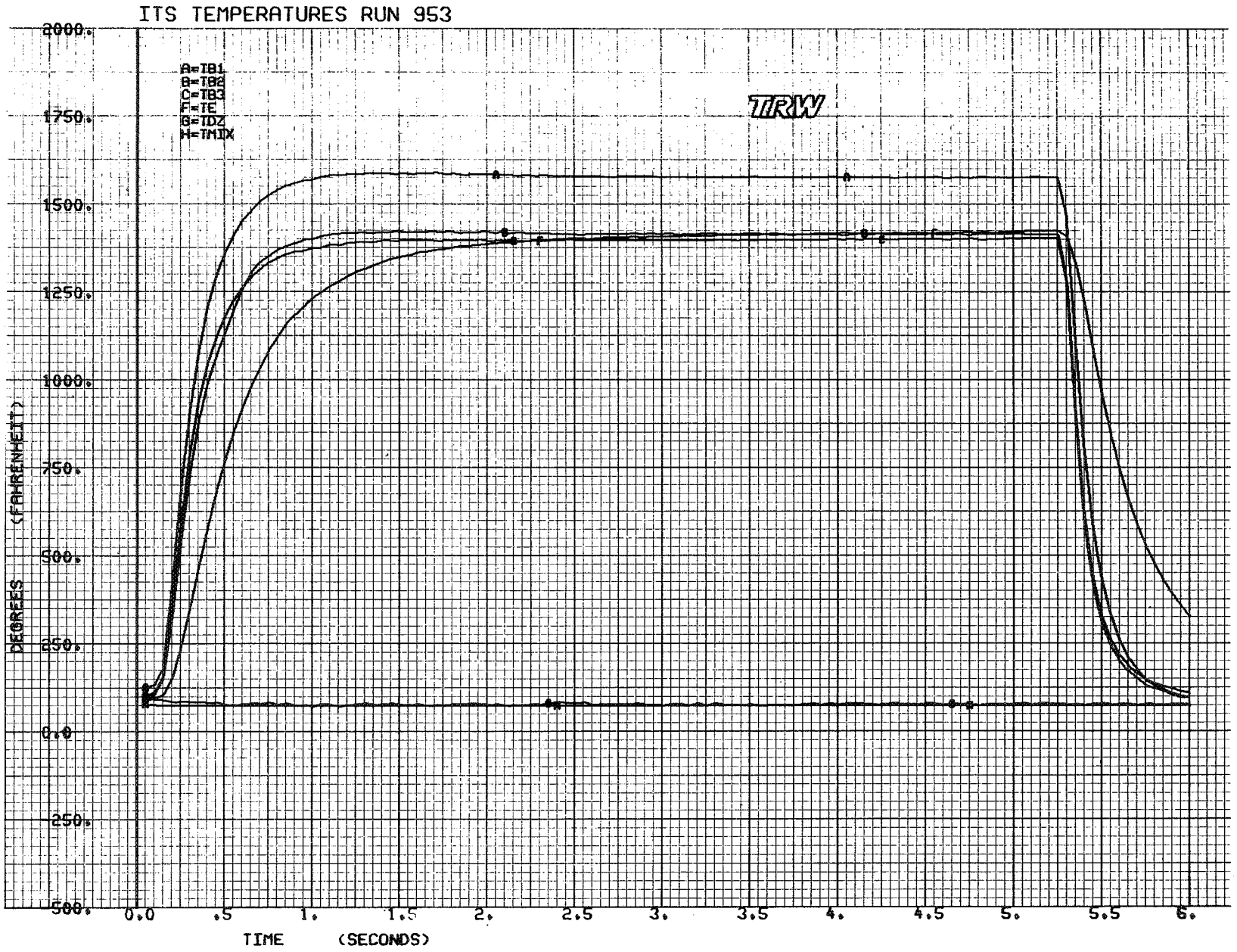
ITS TEMPERATURES RUN 950

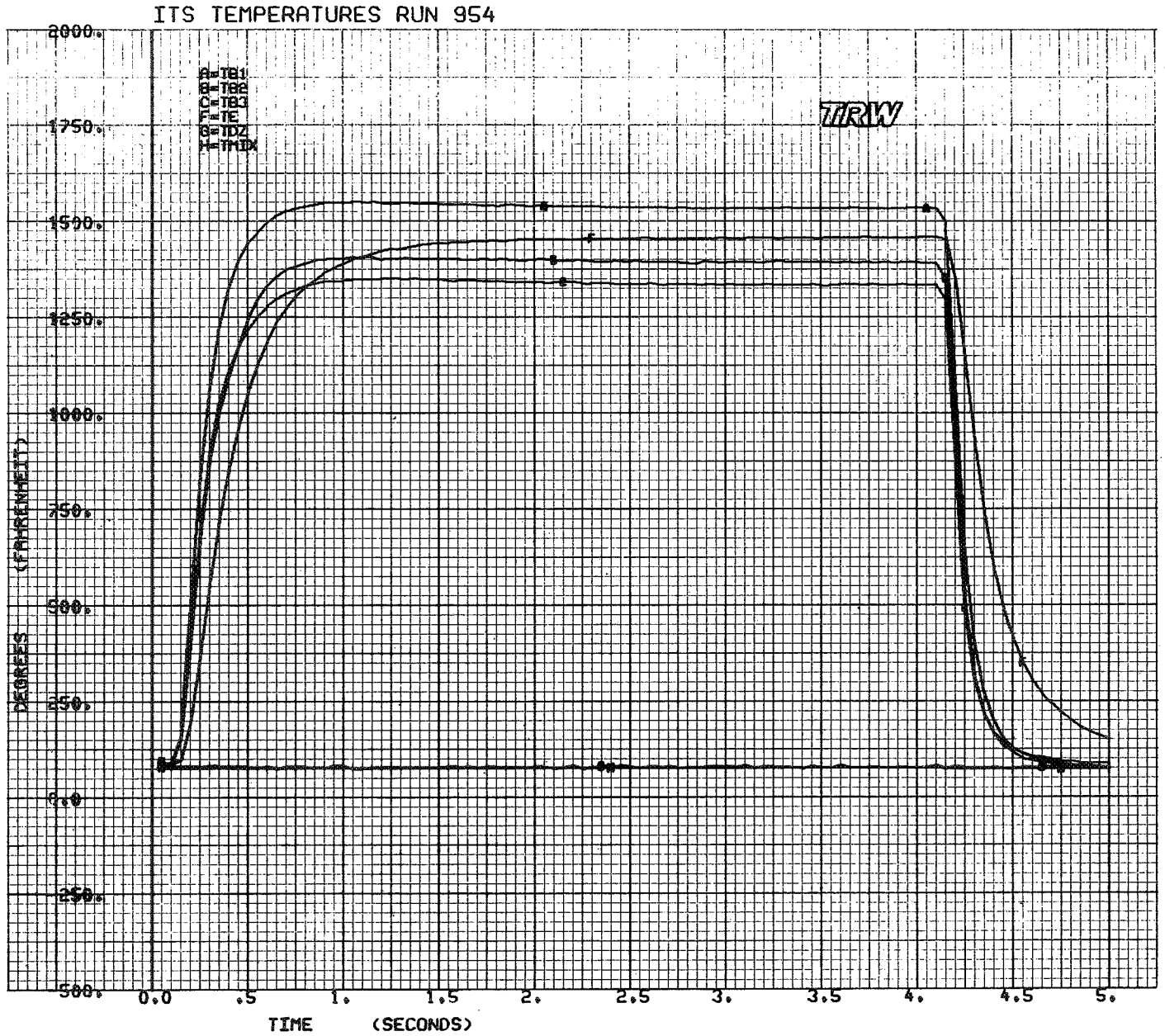


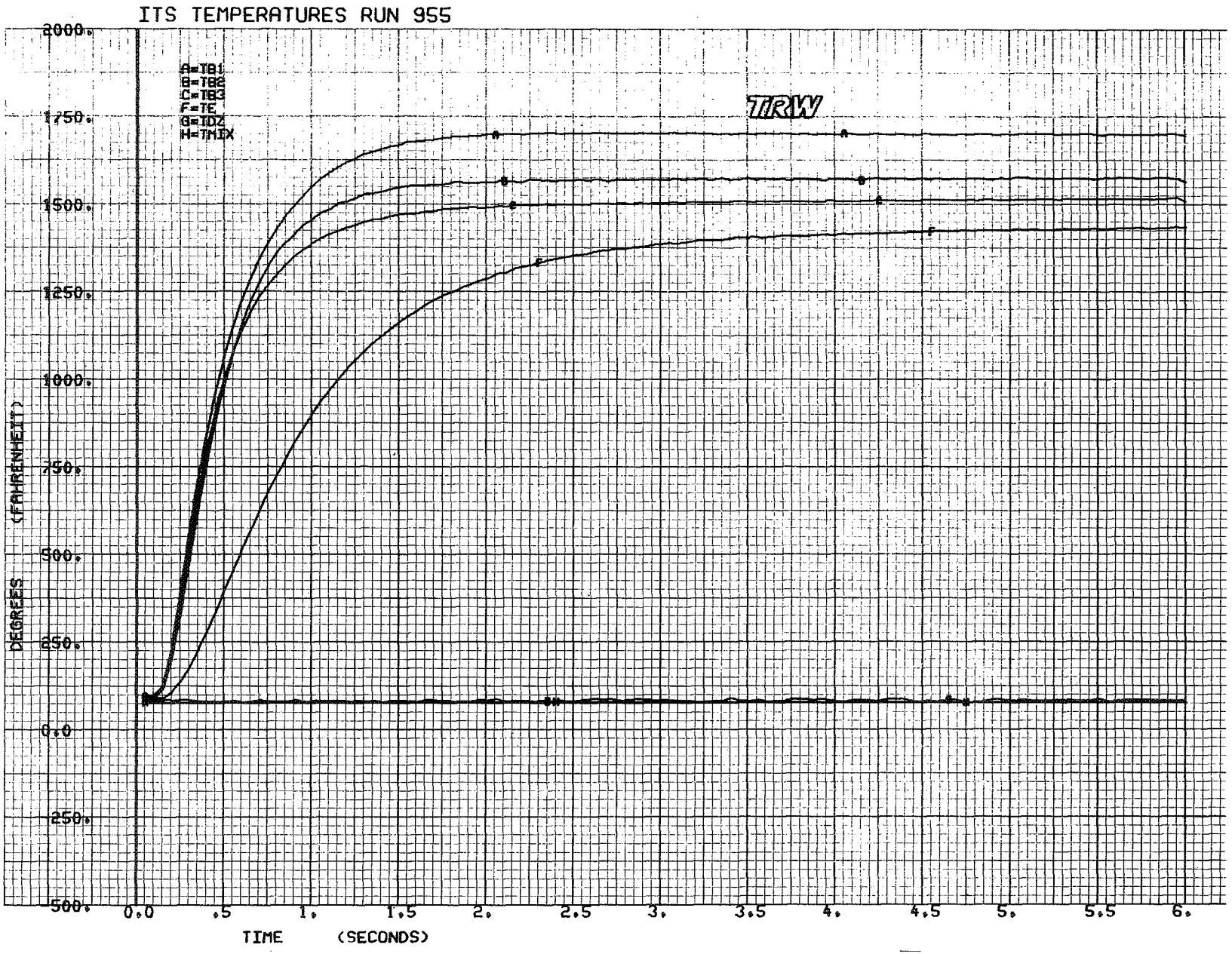

ITS TEMPERATURES RUN 951
A=TB1
B=TB2
C=TB3
F=TE
G=TD2
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
TIME (SECONDS)

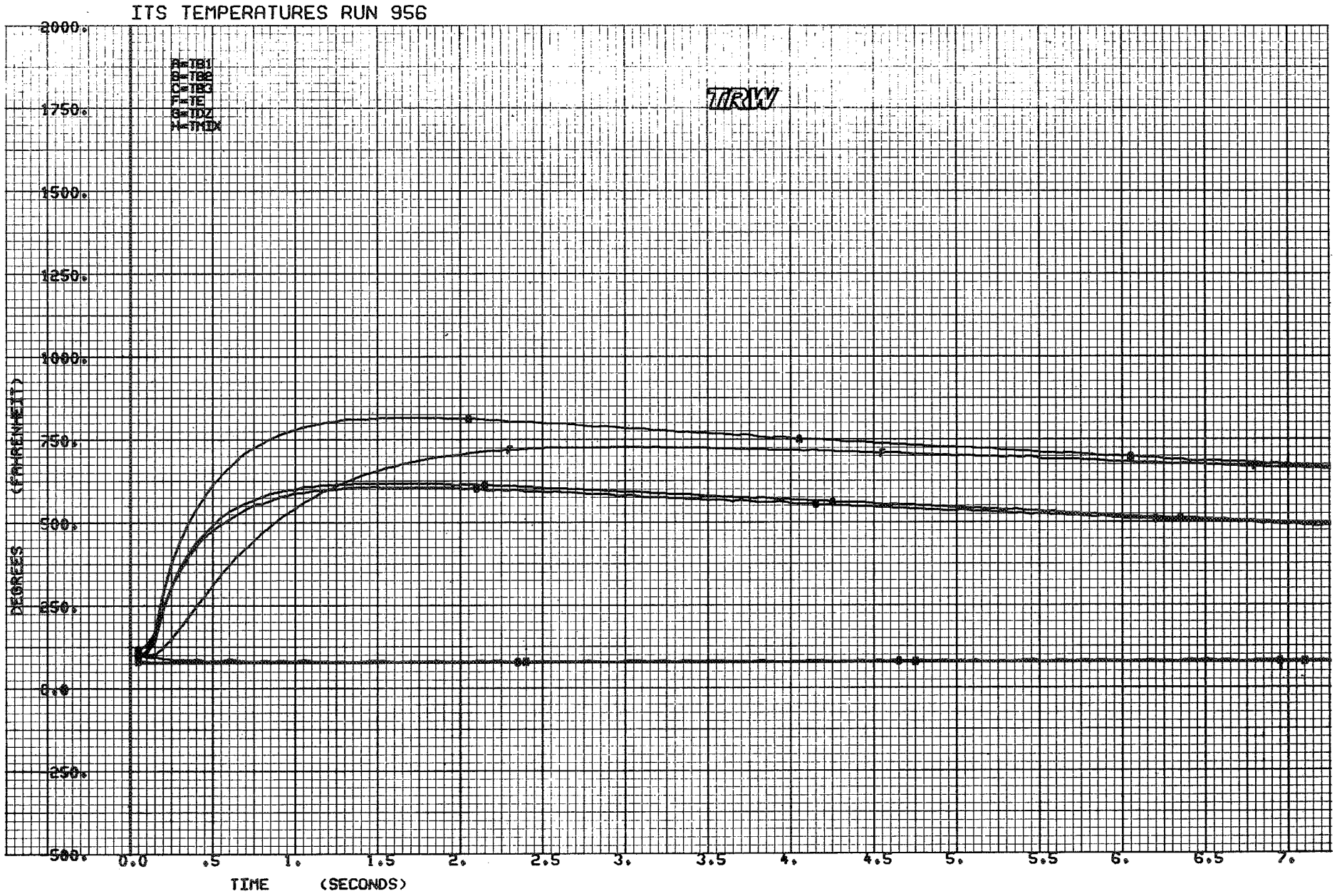
ITS TEMPERATURES RUN 952

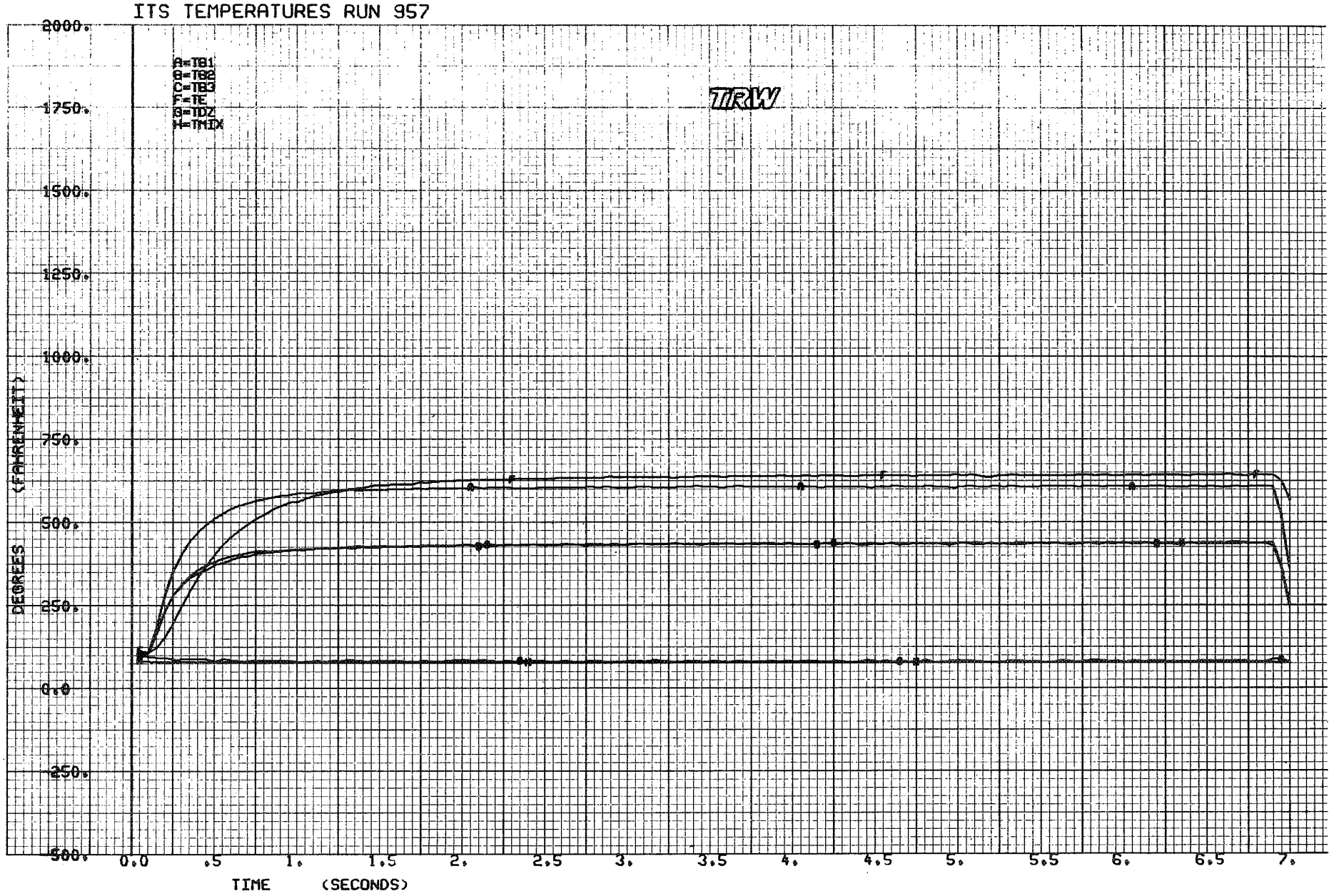



ITS TEMPERATURES RUN 953
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
TIME (SECONDS)


ITS TEMPERATURES RUN 954
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
TIME (SECONDS)


ITS TEMPERATURES RUN 955
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
TIME (SECONDS)


ITS TEMPERATURES RUN 956
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMID
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
7.
TIME (SECONDS)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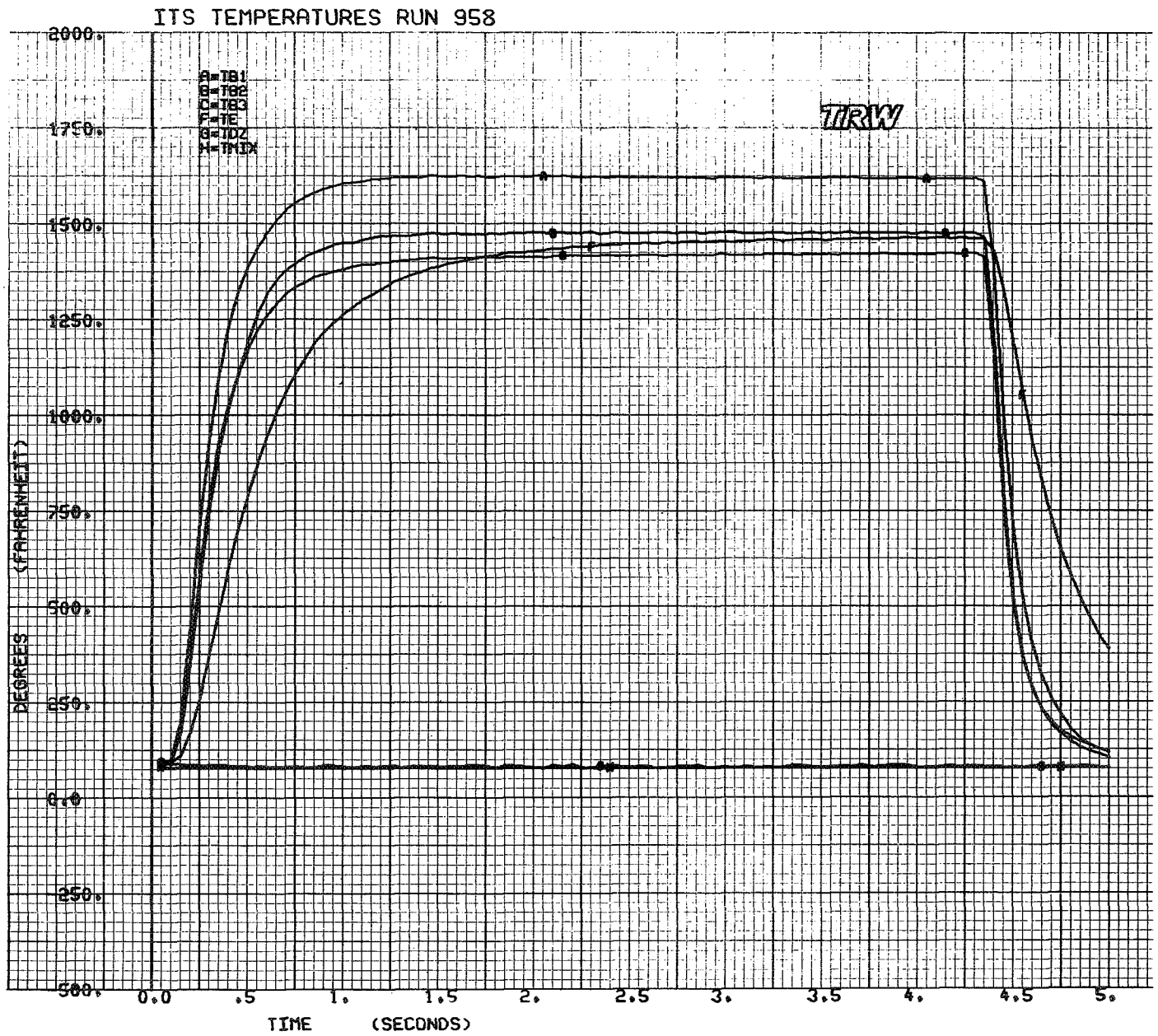
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
7.
TIME (SECONDS)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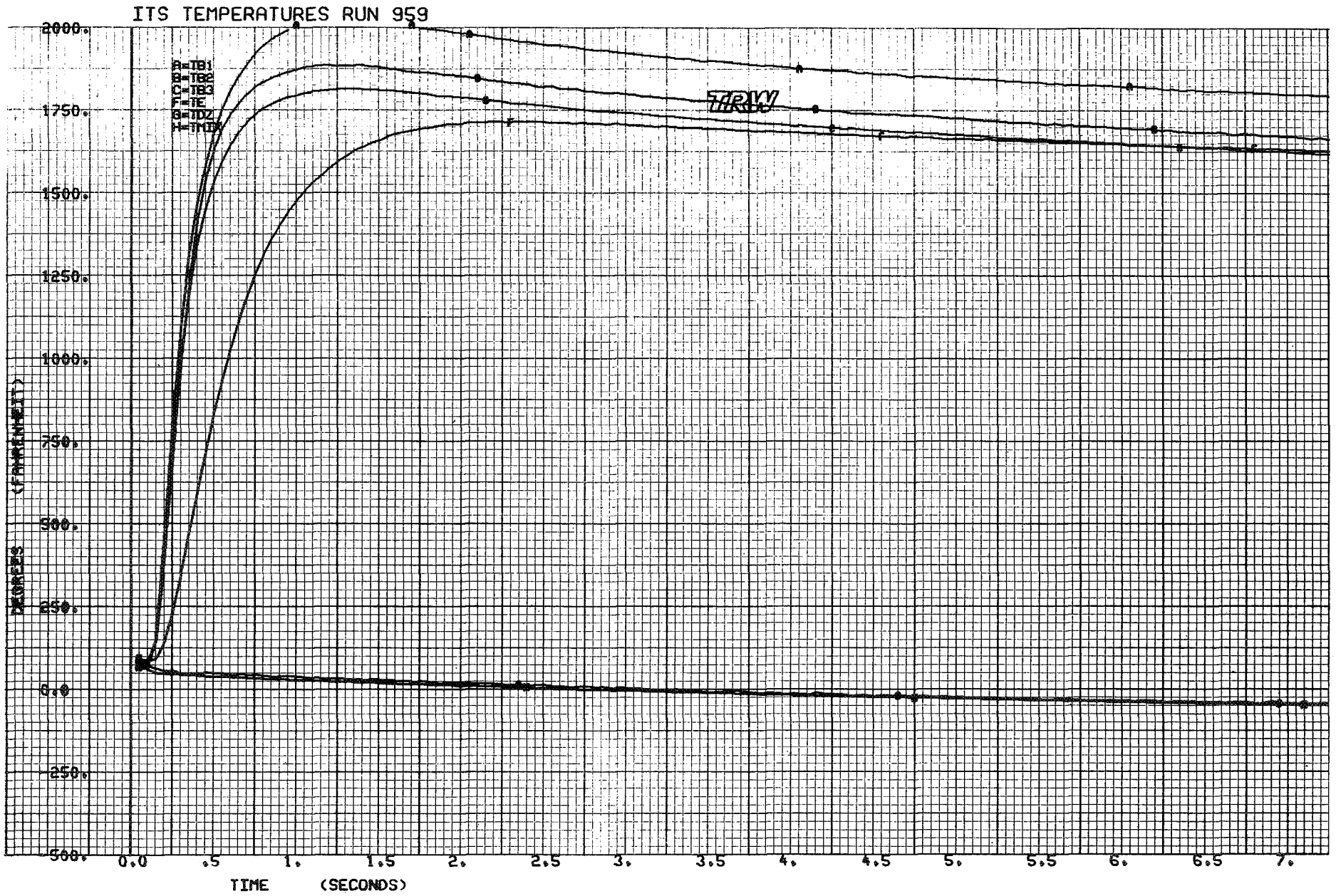
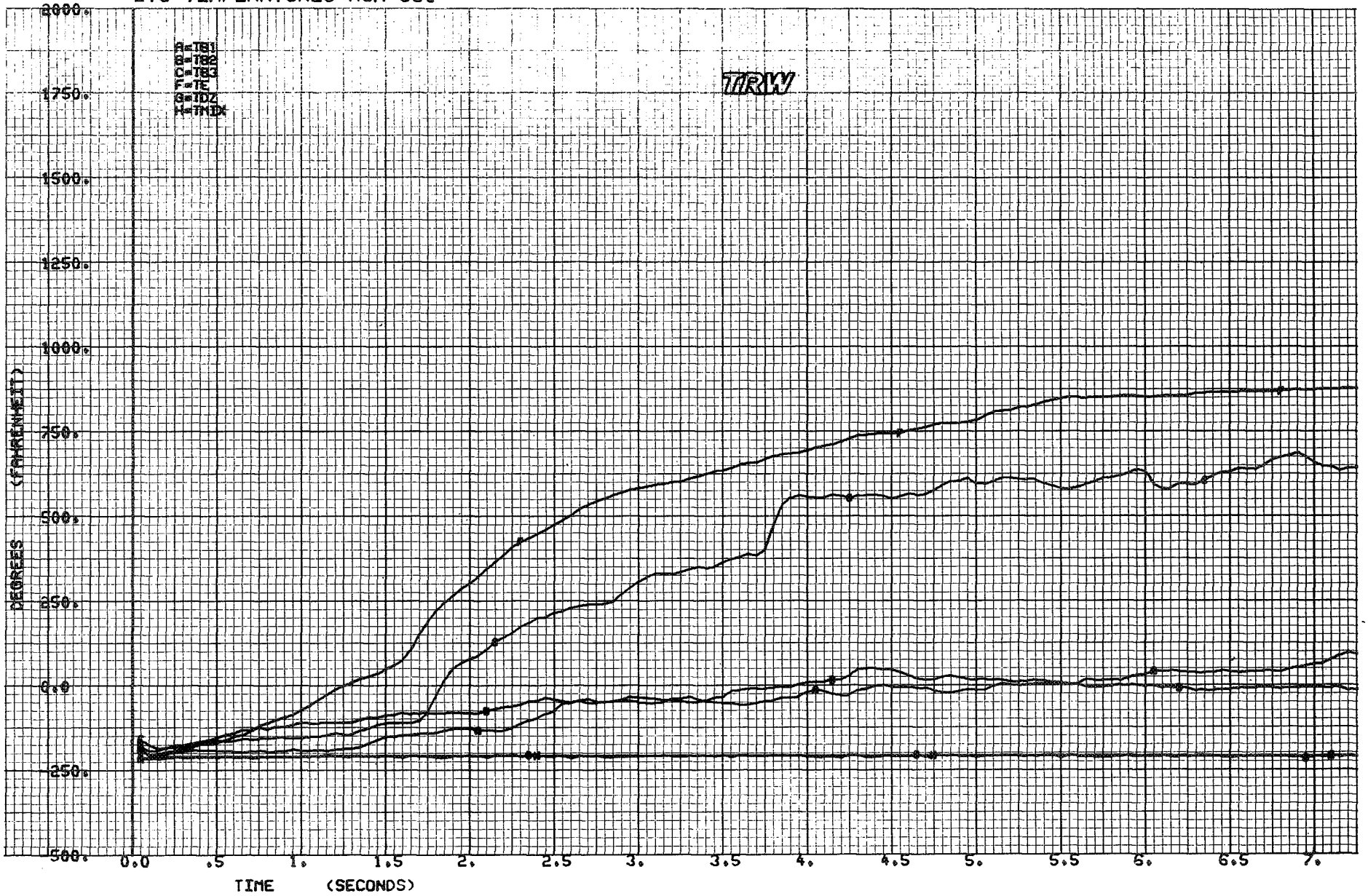
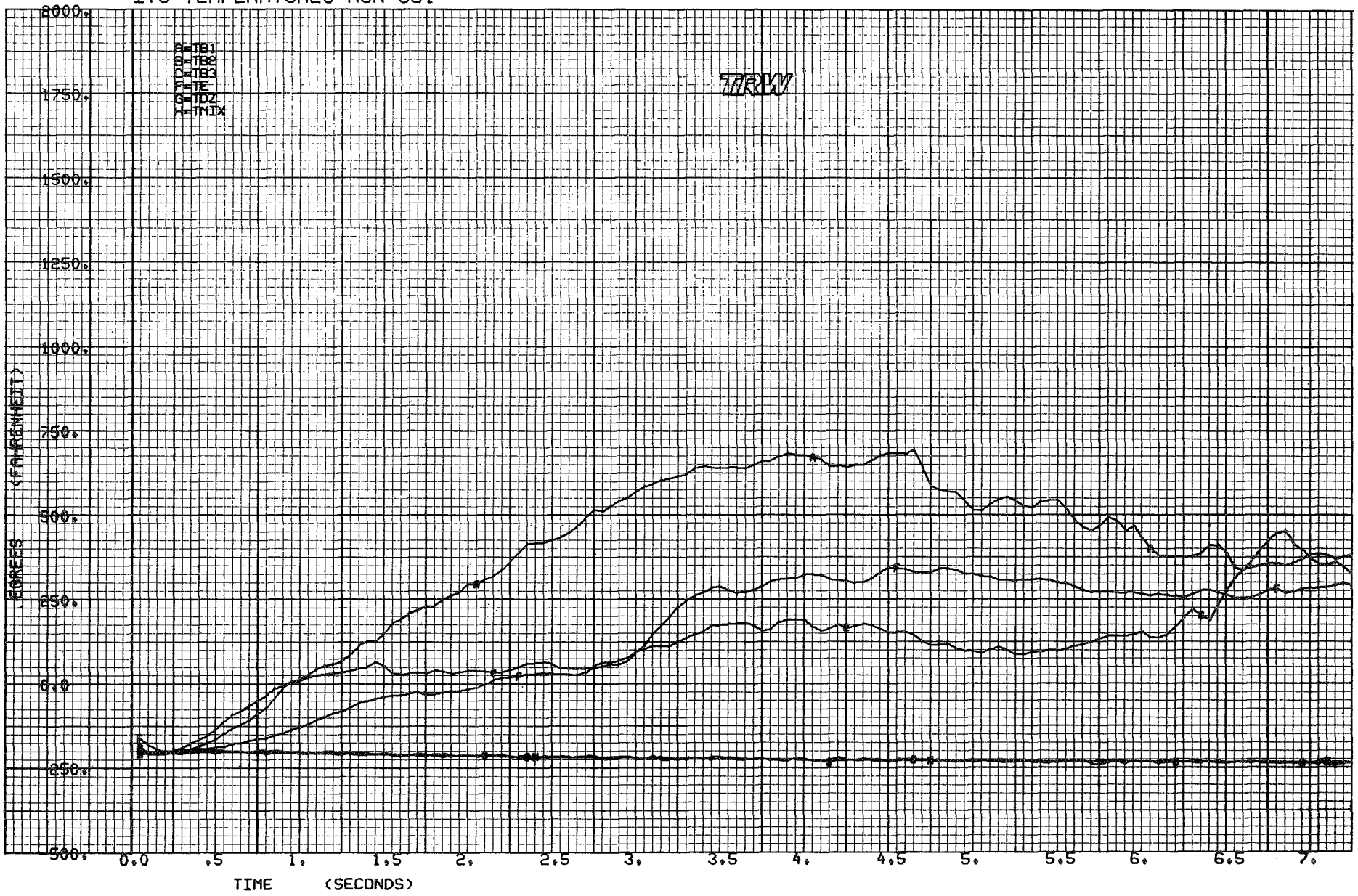
B=TB2
C=TB3
F=TE
G=TDZ
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
TIME (SECONDS)


ITS TEMPERATURES RUN 959
A=TB1
B=TB2
C=TB3
F=TE
G=TDZ
H=TMTR
TRW
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
7.
TIME (SECONDS)

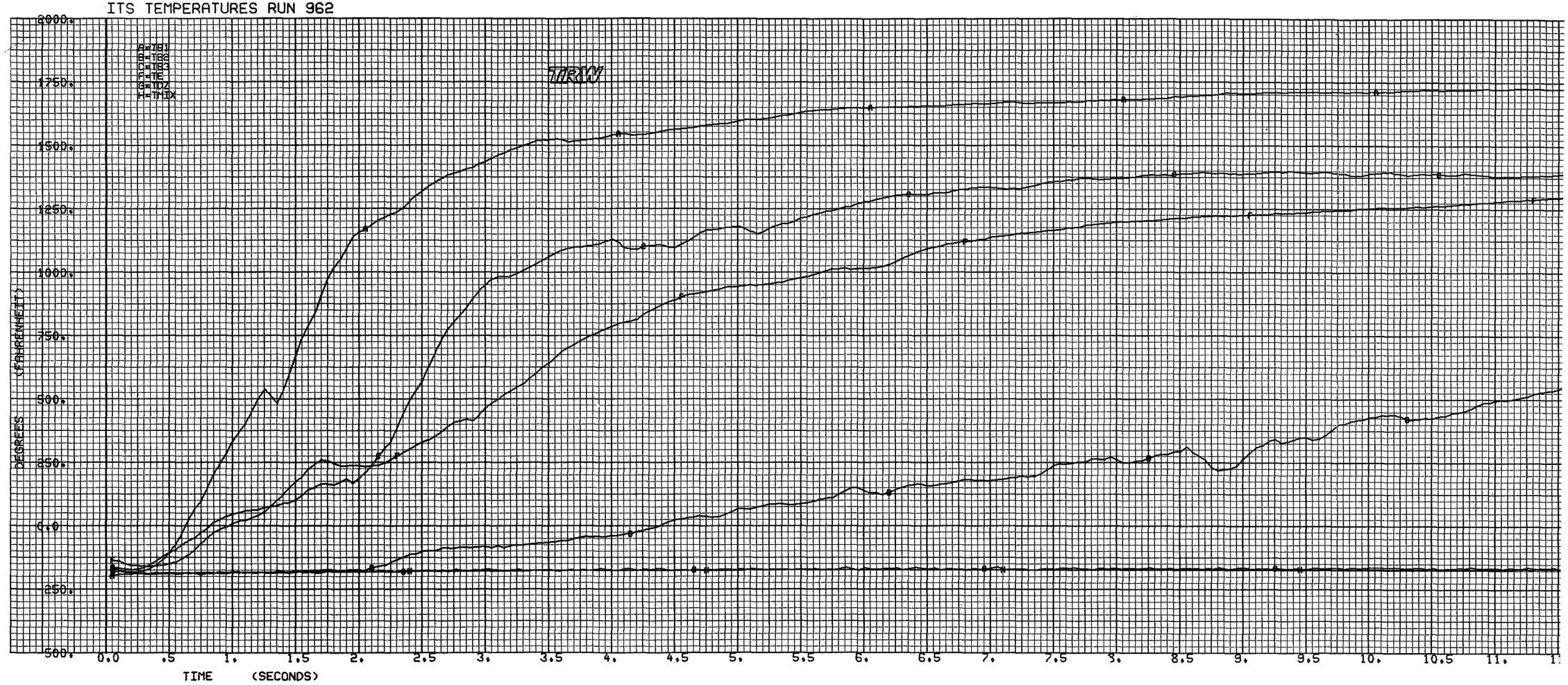
ITS TEMPERATURES RUN 960

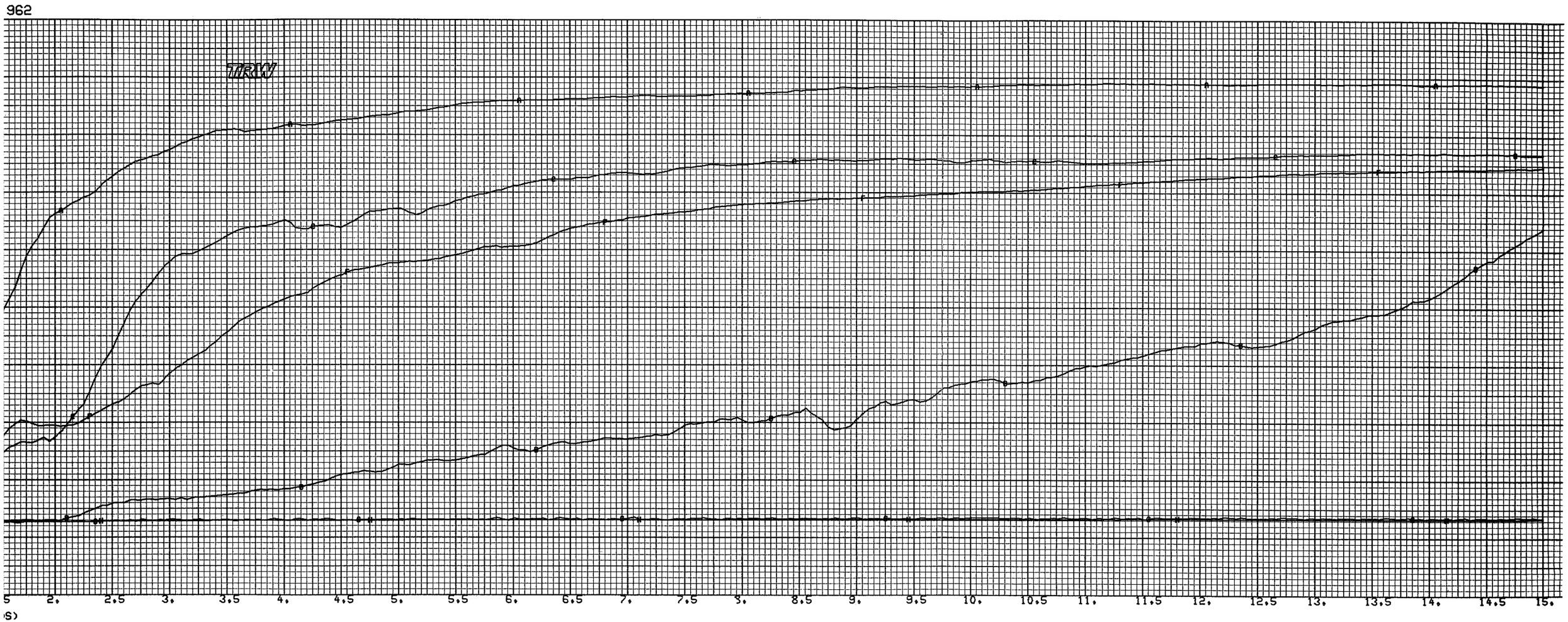


ITS TEMPERATURES RUN 961



ITS TEMPERATURES RUN 962




962
TIRW
5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
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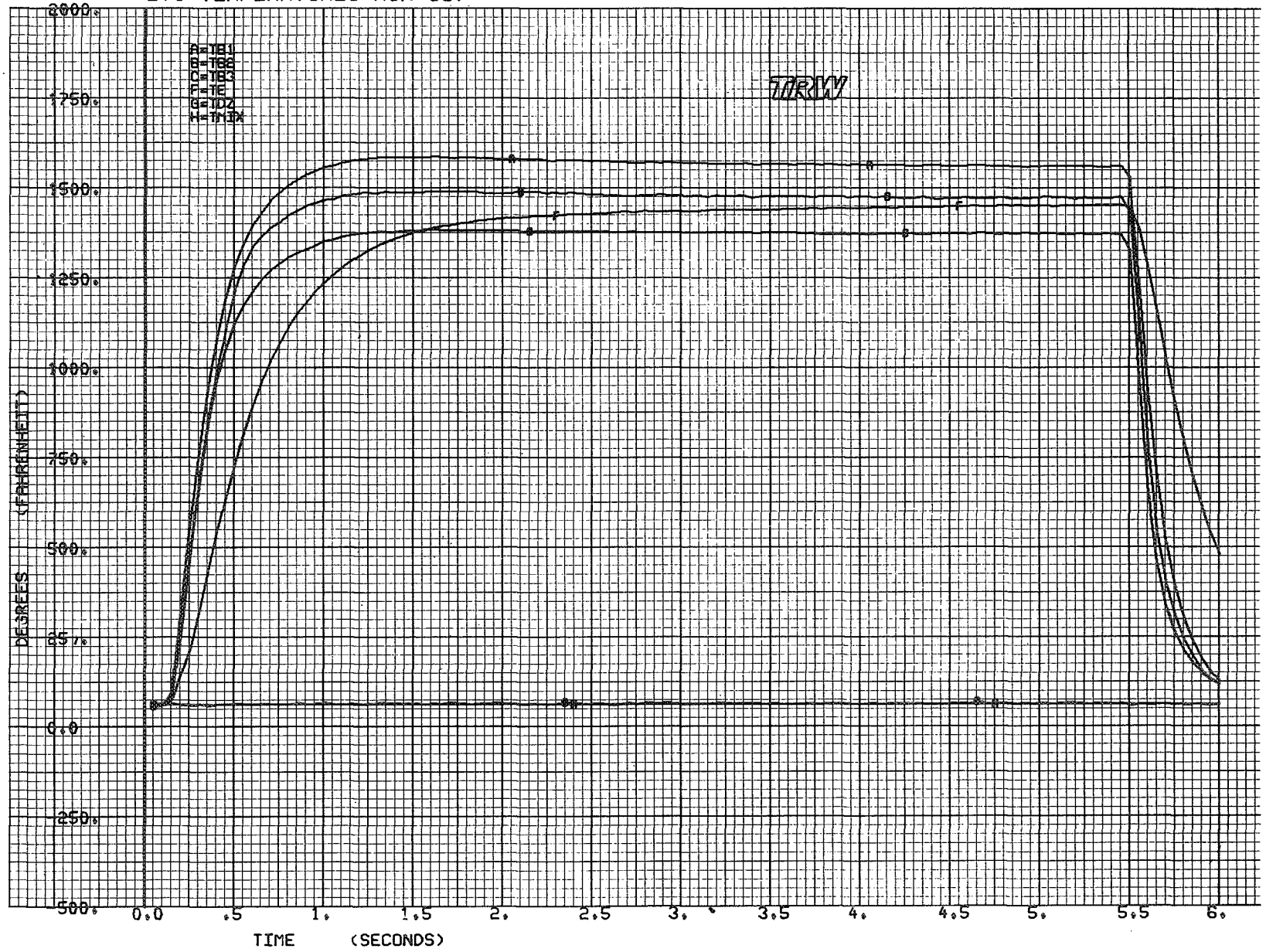
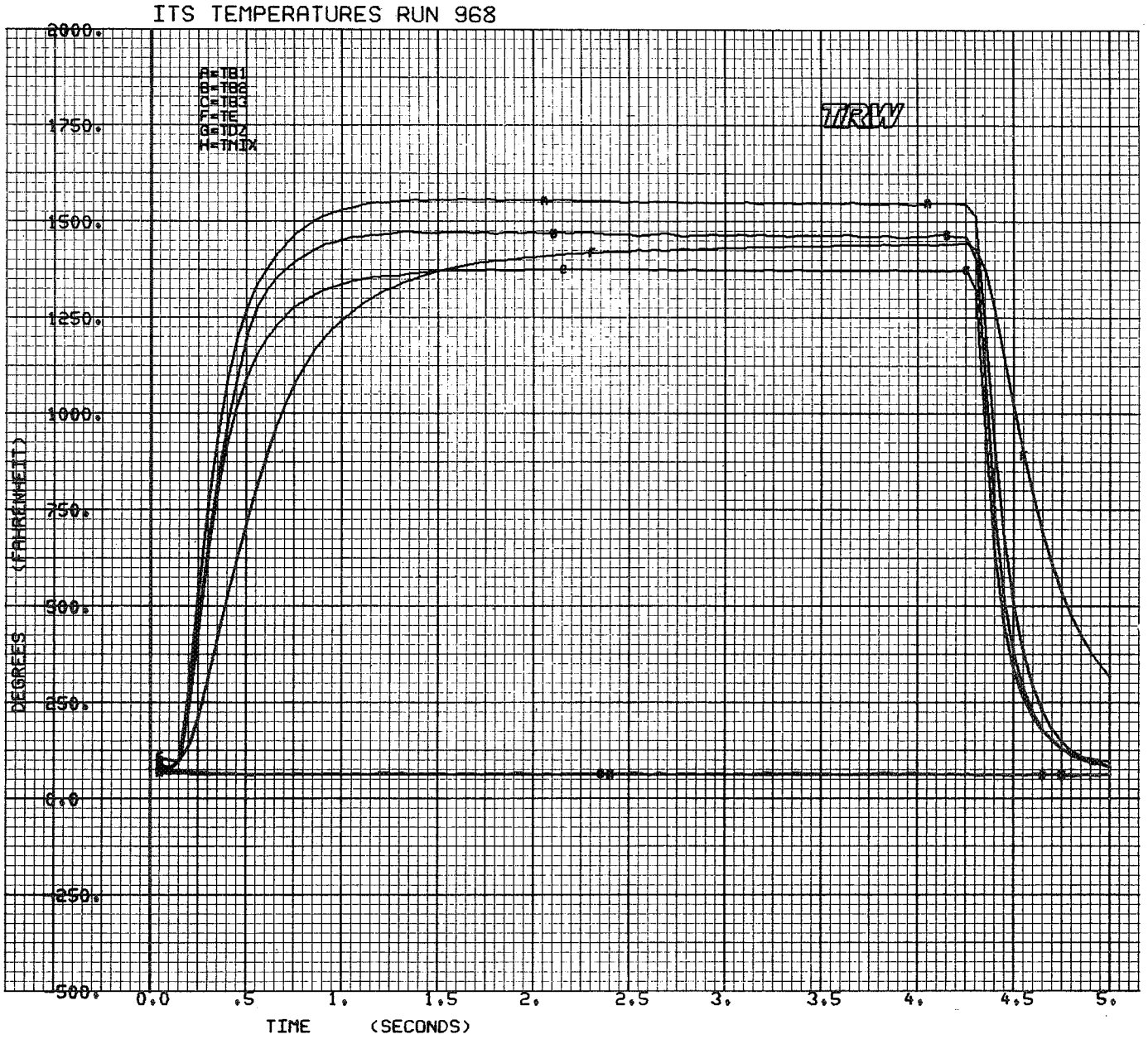
7.5
8.
8.5
9.
9.5
10.
10.5
11.
11.5
12.
12.5
13.
13.5
14.
14.5
15.
(S)

ITS TEMPERATURES RUN 967




ITS TEMPERATURES RUN 968
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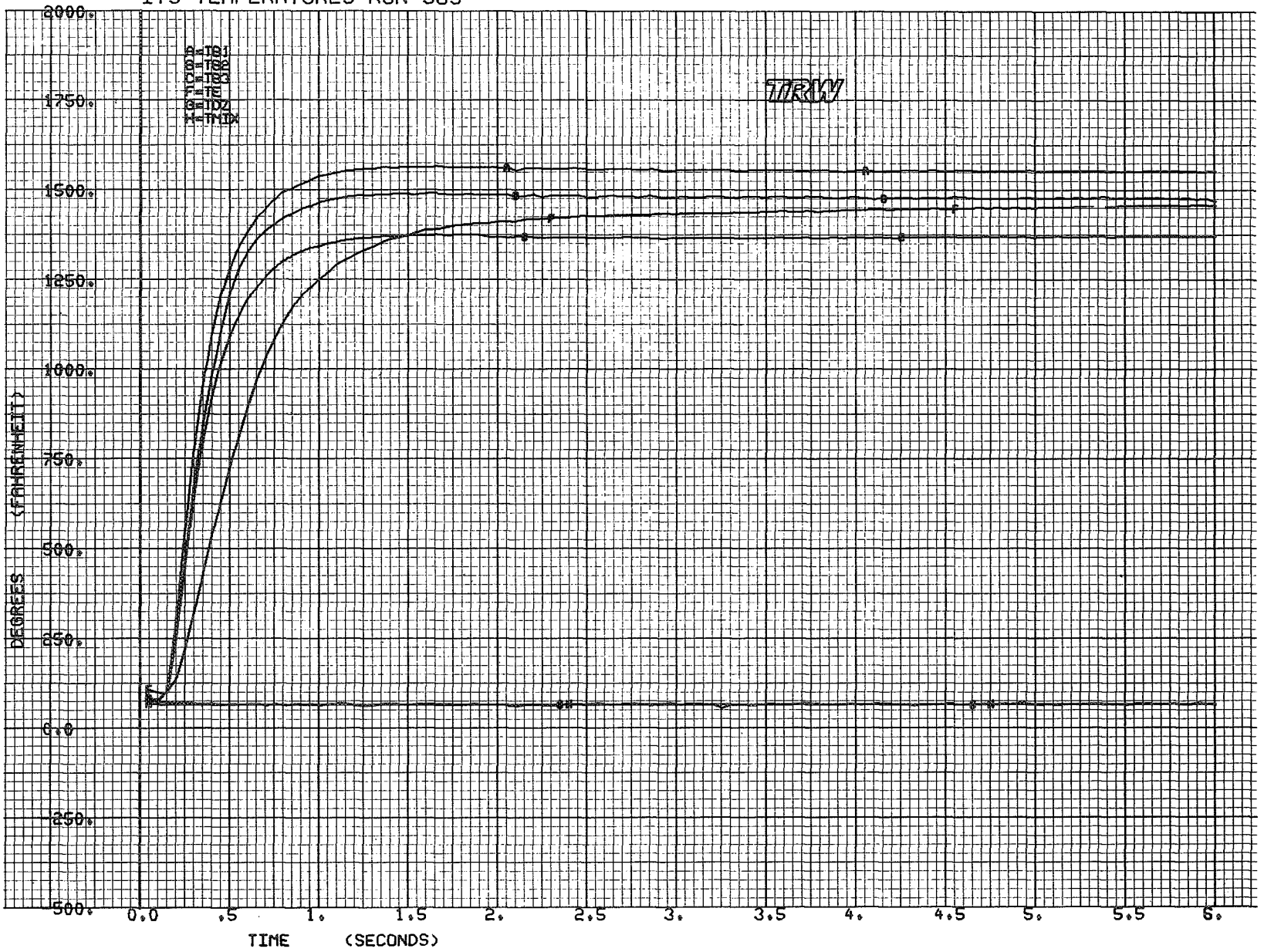
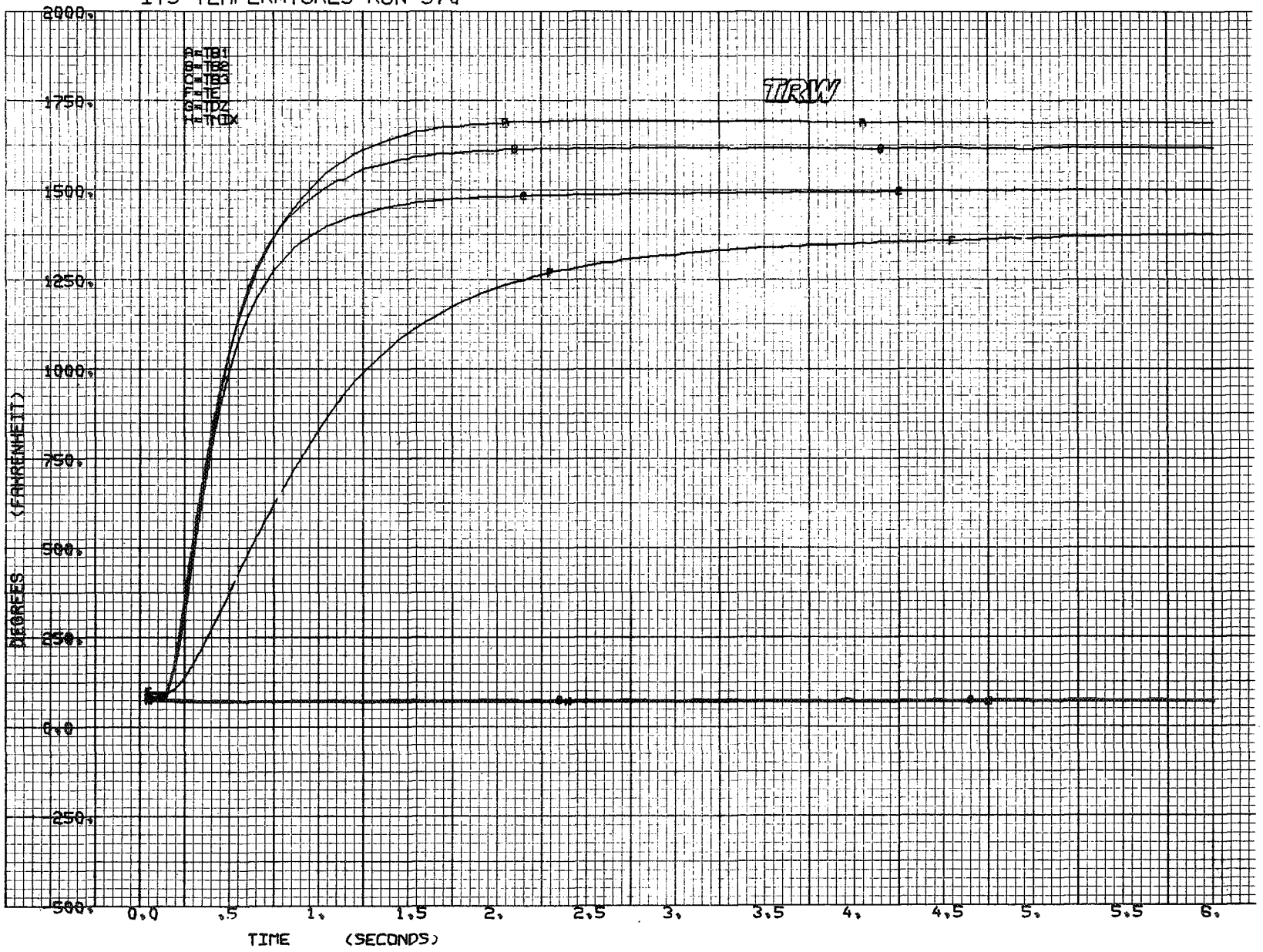
B=TB2
C=TB3
F=TE
G=TD2
H=TMIX
TRW
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
TIME (SECONDS)

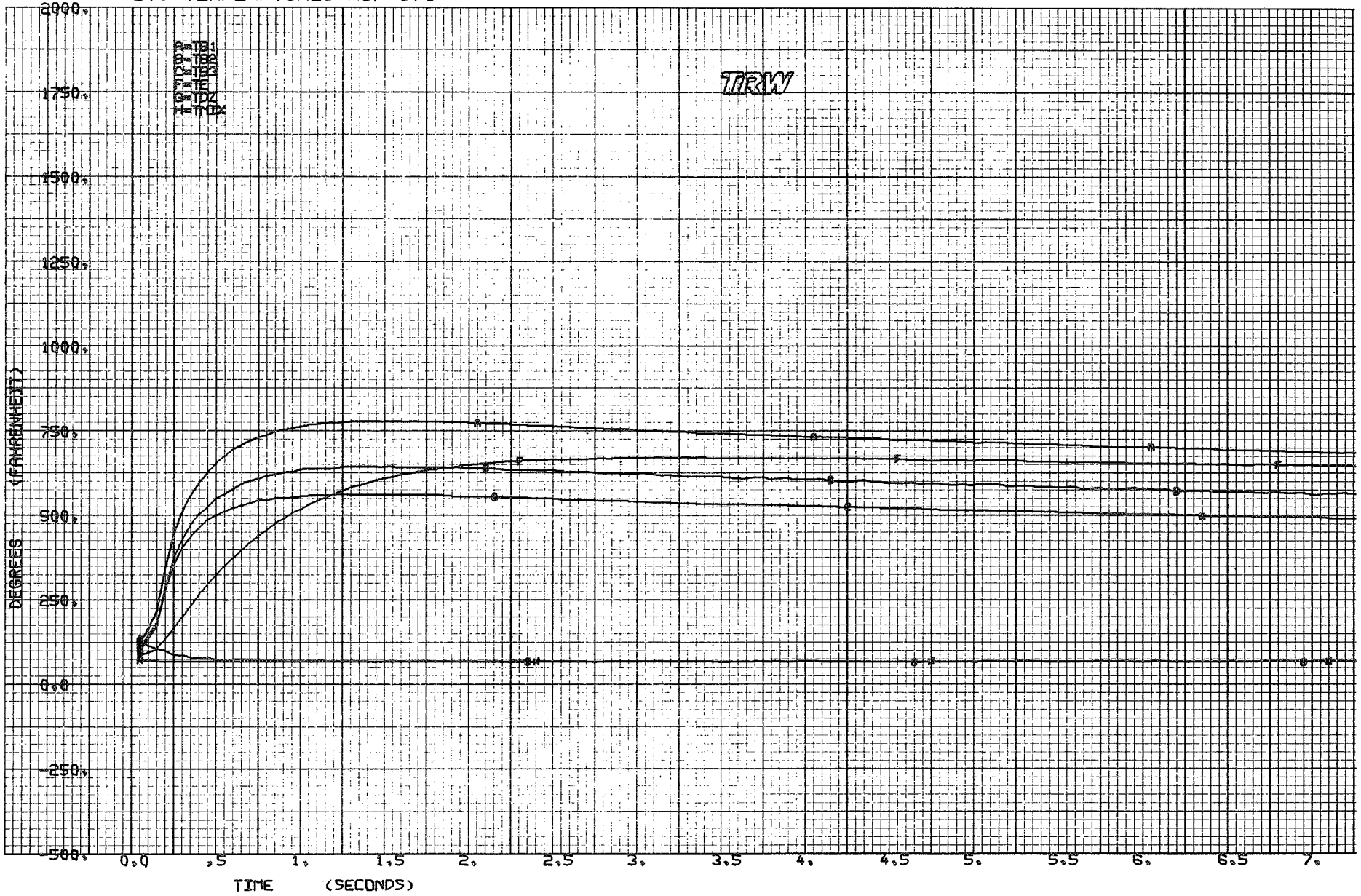
ITS TEMPERATURES RUN 969



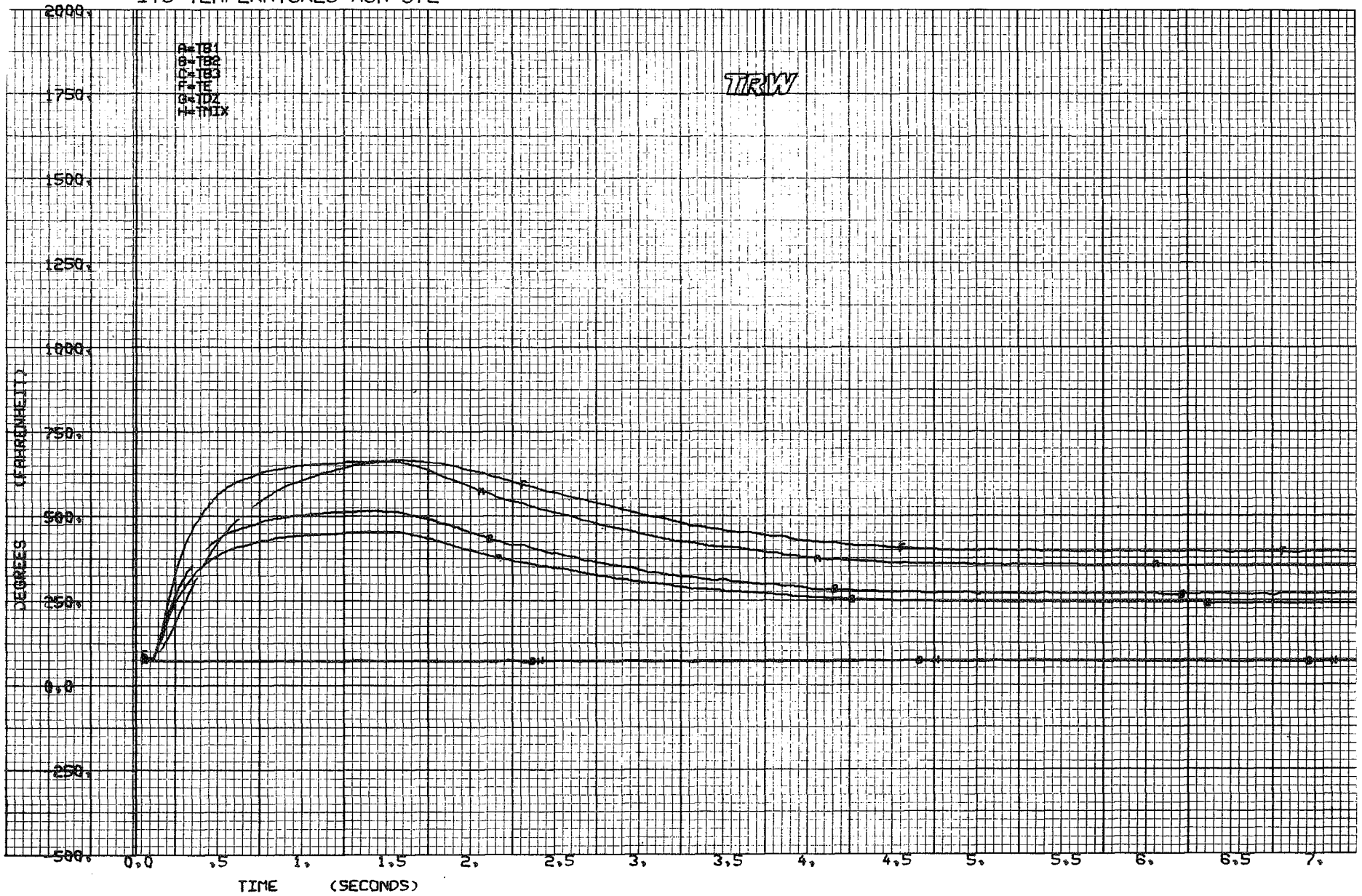
ITS TEMPERATURES RUN 970

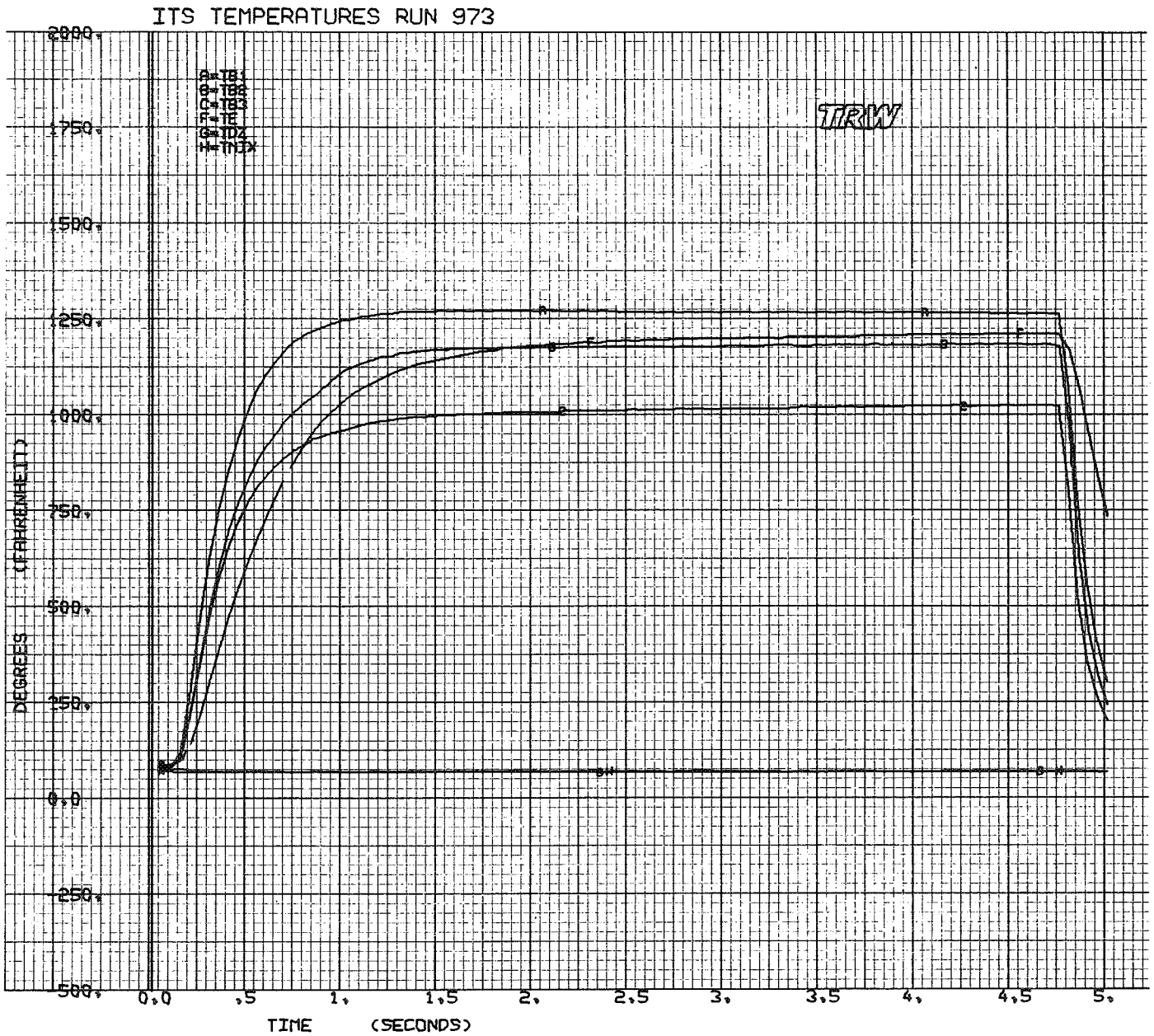


ITS TEMPERATURES RUN 971



ITS TEMPERATURES RUN 972




ITS TEMPERATURES RUN 973
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C=TB3
F=TE
G=TD2
H=THIX
TRW
DEGREES (FAHRENHEIT)
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1500.
1250.
1000.
750.
500.
250.
0.0
-250.
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1.5
2.
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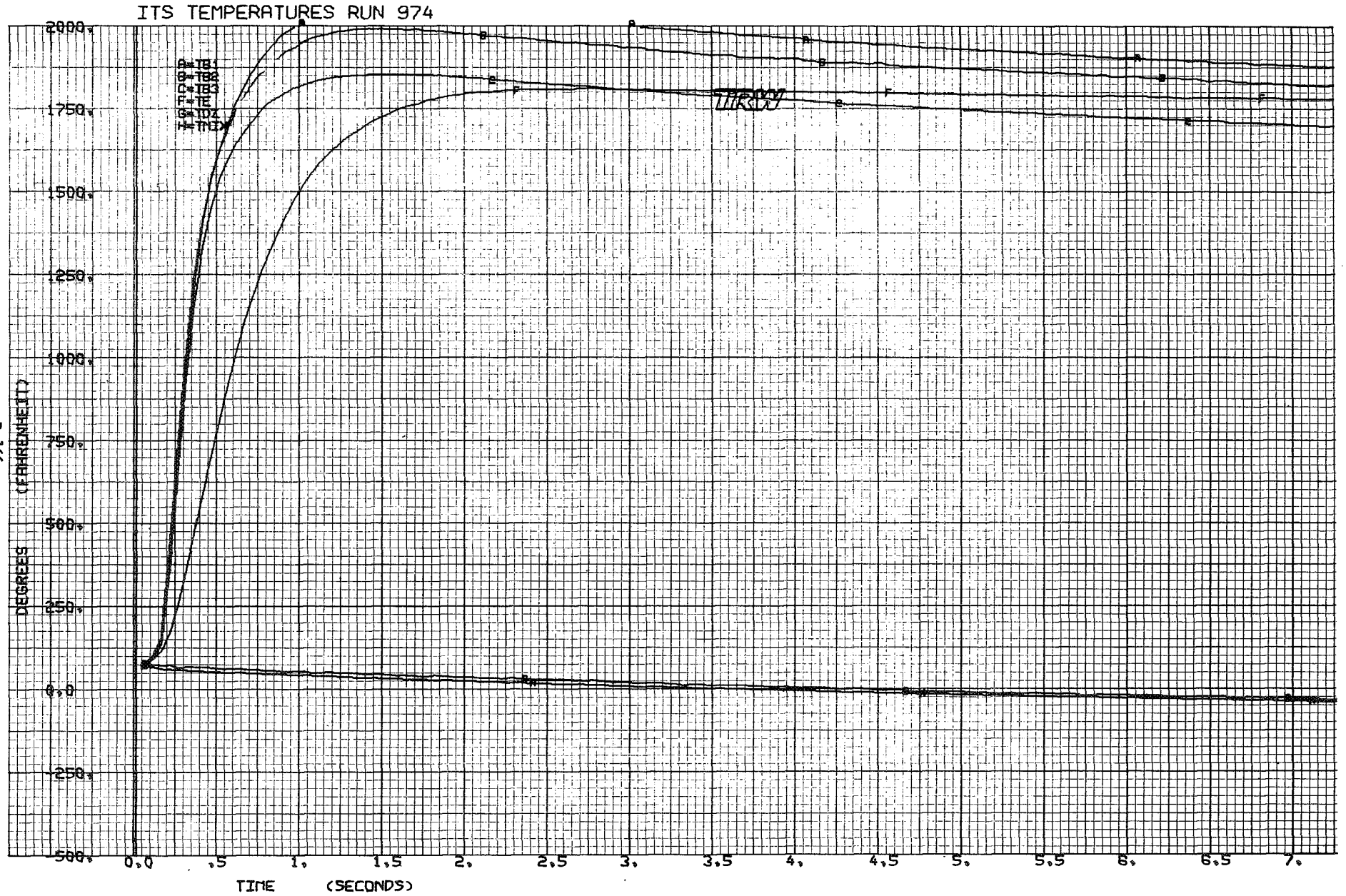
3.
3.5
4.
4.5
5.
TIME (SECONDS)


ITS TEMPERATURES RUN 974
A=TB1
B=TB2
C=TB3
F=TE
G=TD1
H=TNIX
DEGREES (FAHRENHEIT)
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1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
.5
1.
1.5
2.
2.5
3.
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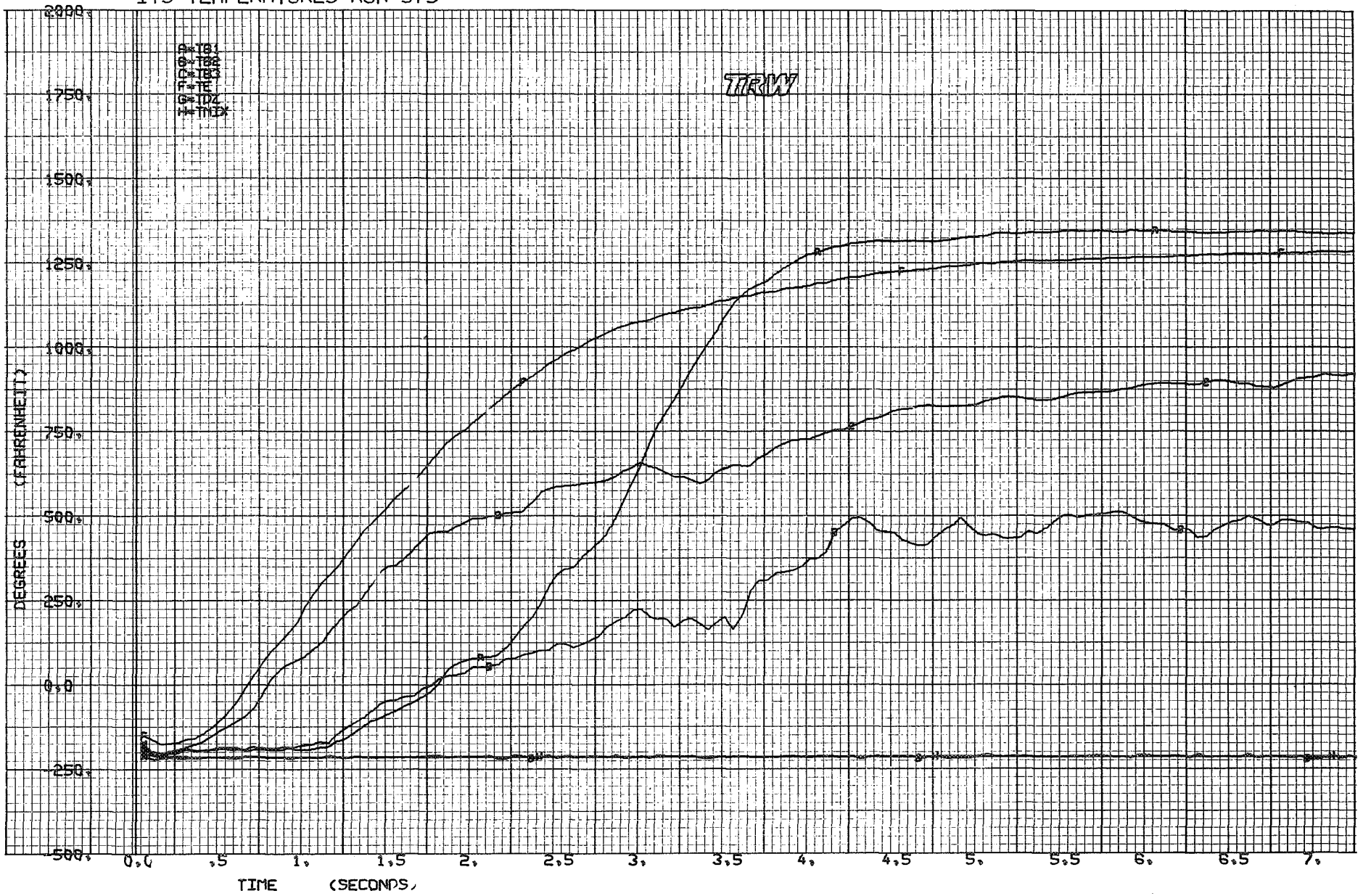
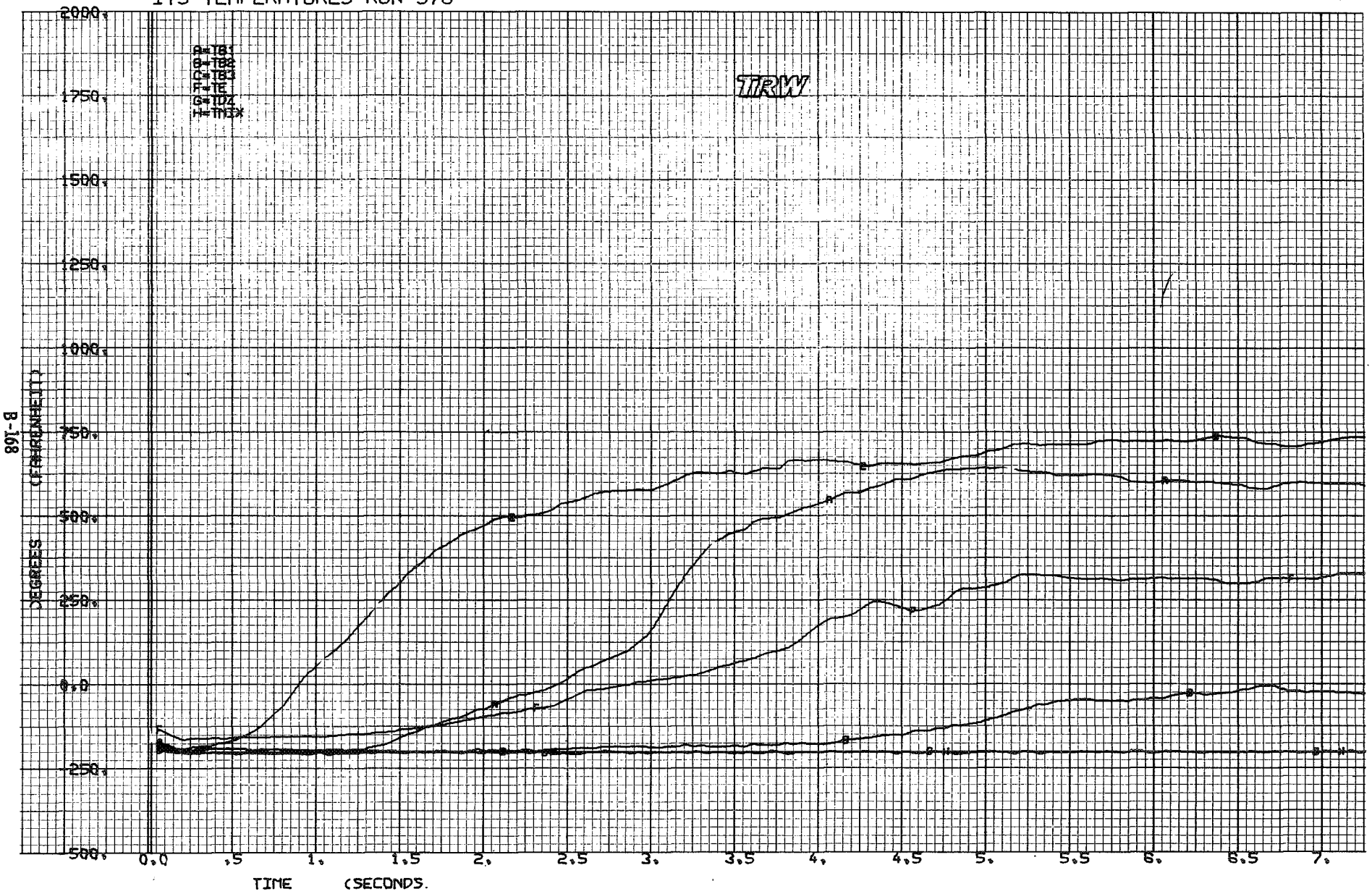
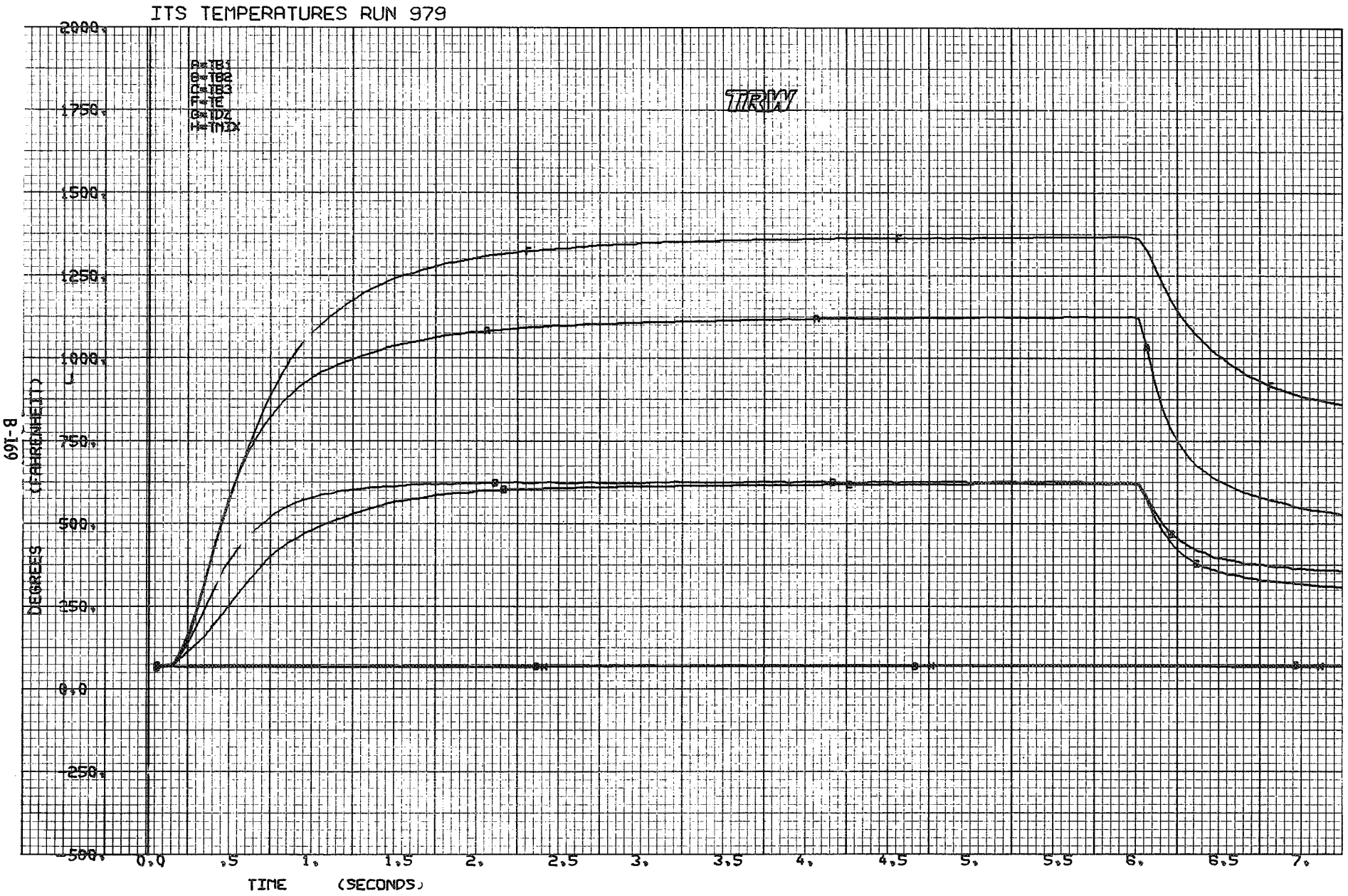
4.
4.5
5.
5.5
6.
6.5
7.
TIME (SECONDS)

ITS TEMPERATURES RUN 975

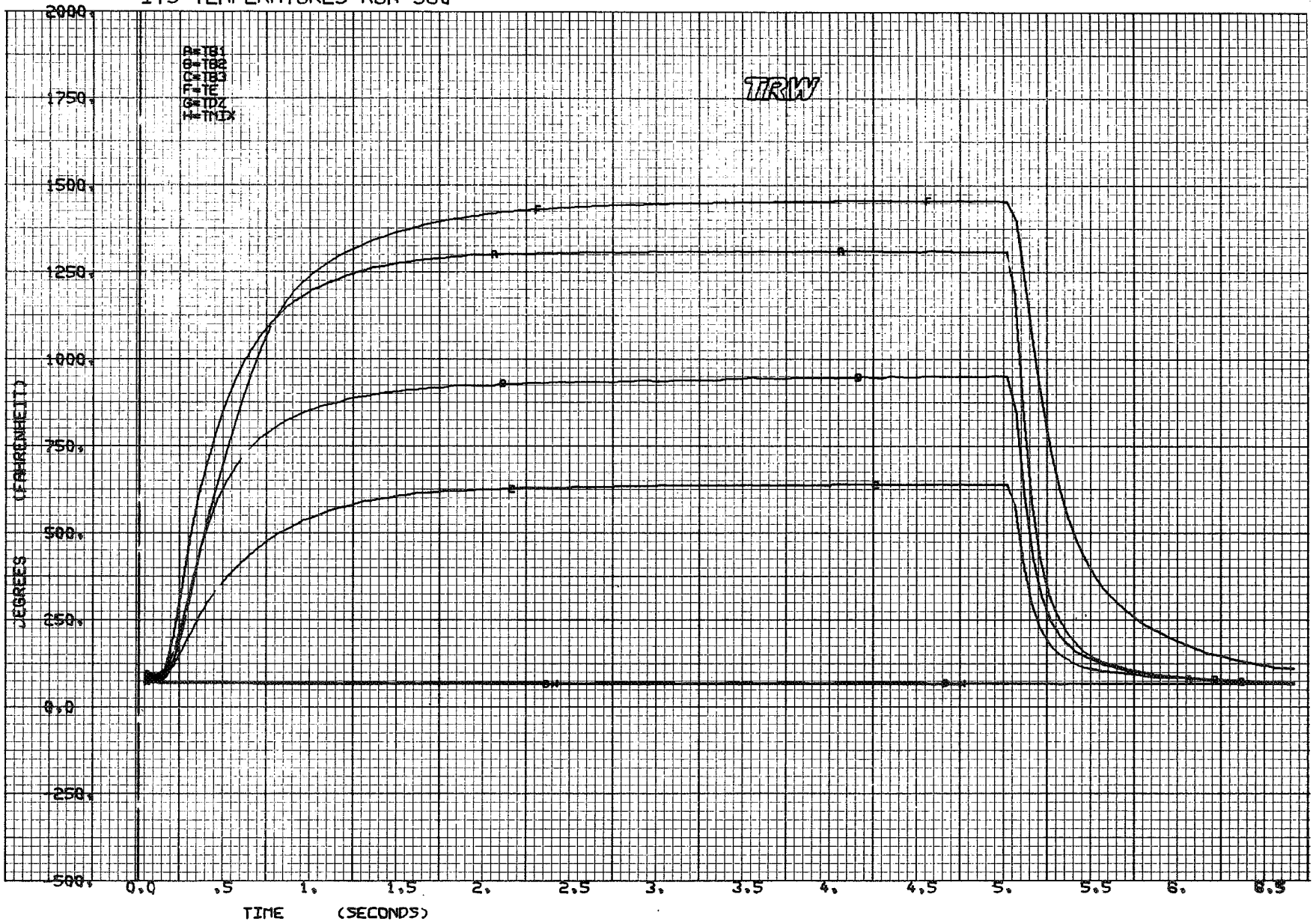


ITS TEMPERATURES RUN 976

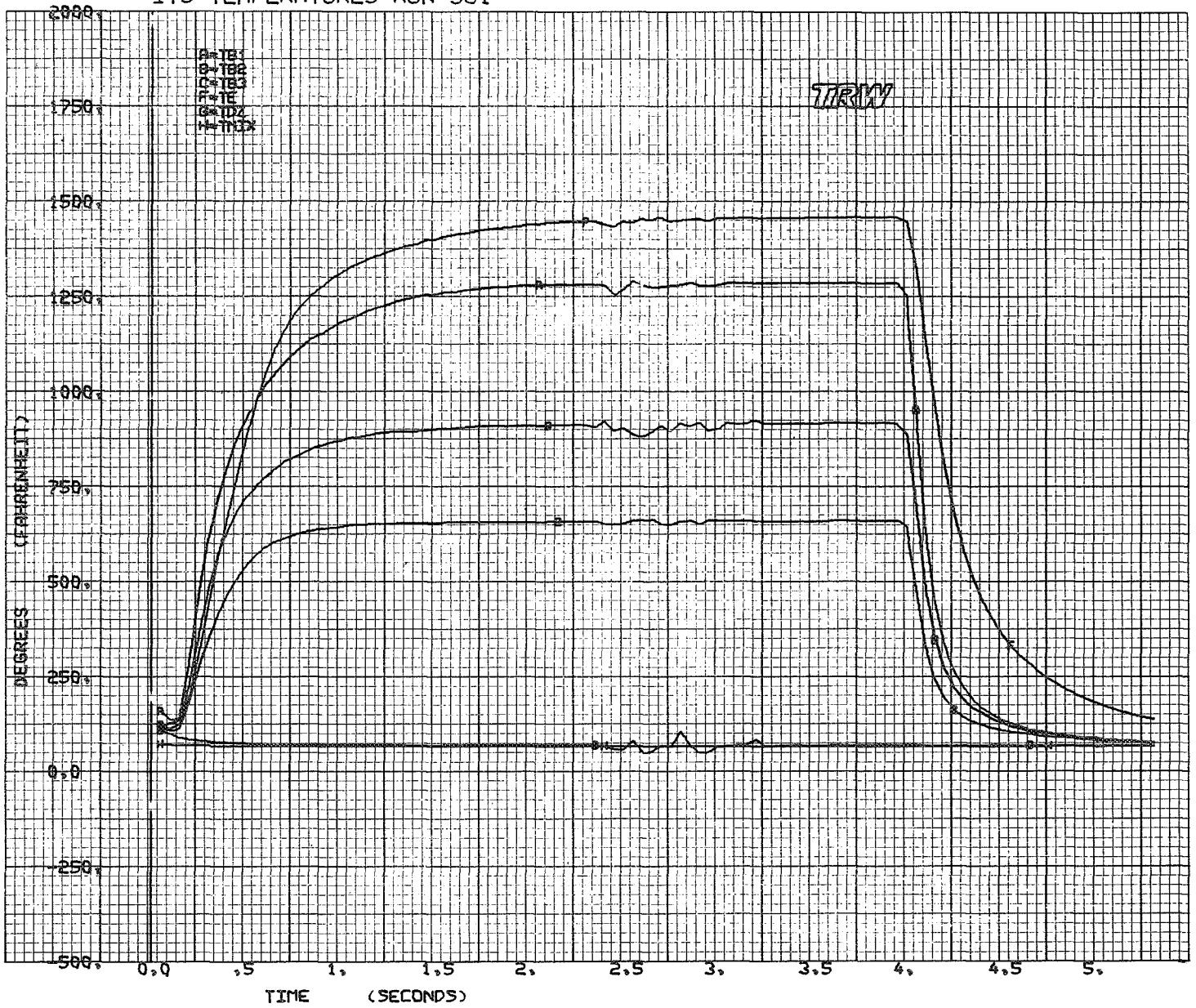



ITS TEMPERATURES RUN 979
TRW
P=TB1
Q=TB2
R=TB3
S=TE
G=TDZ
H=TDX
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
DEGREES (FAHRENHEIT)
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
4.5
5.
5.5
6.
6.5
7.
TIME (SECONDS)

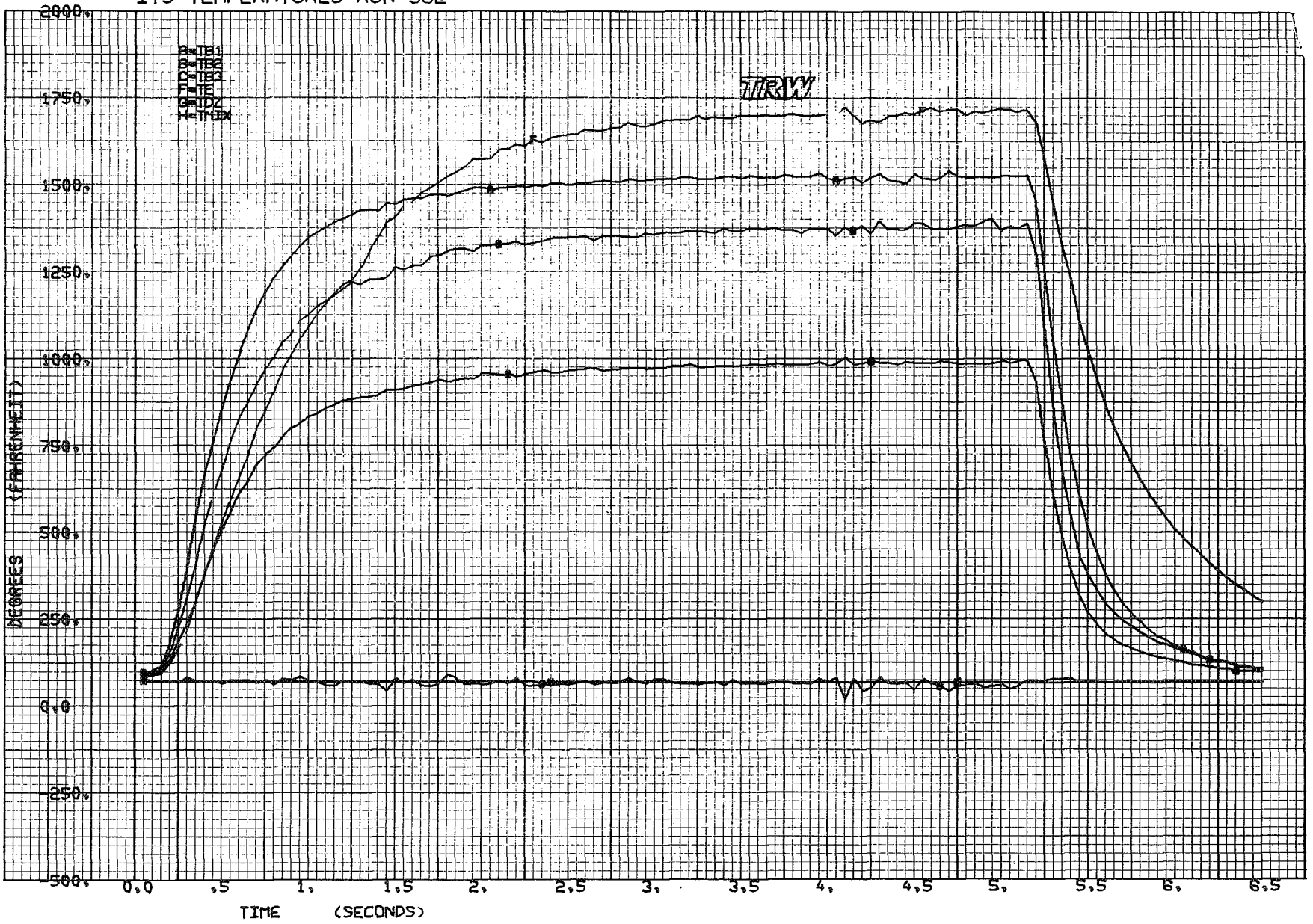
ITS TEMPERATURES RUN 980



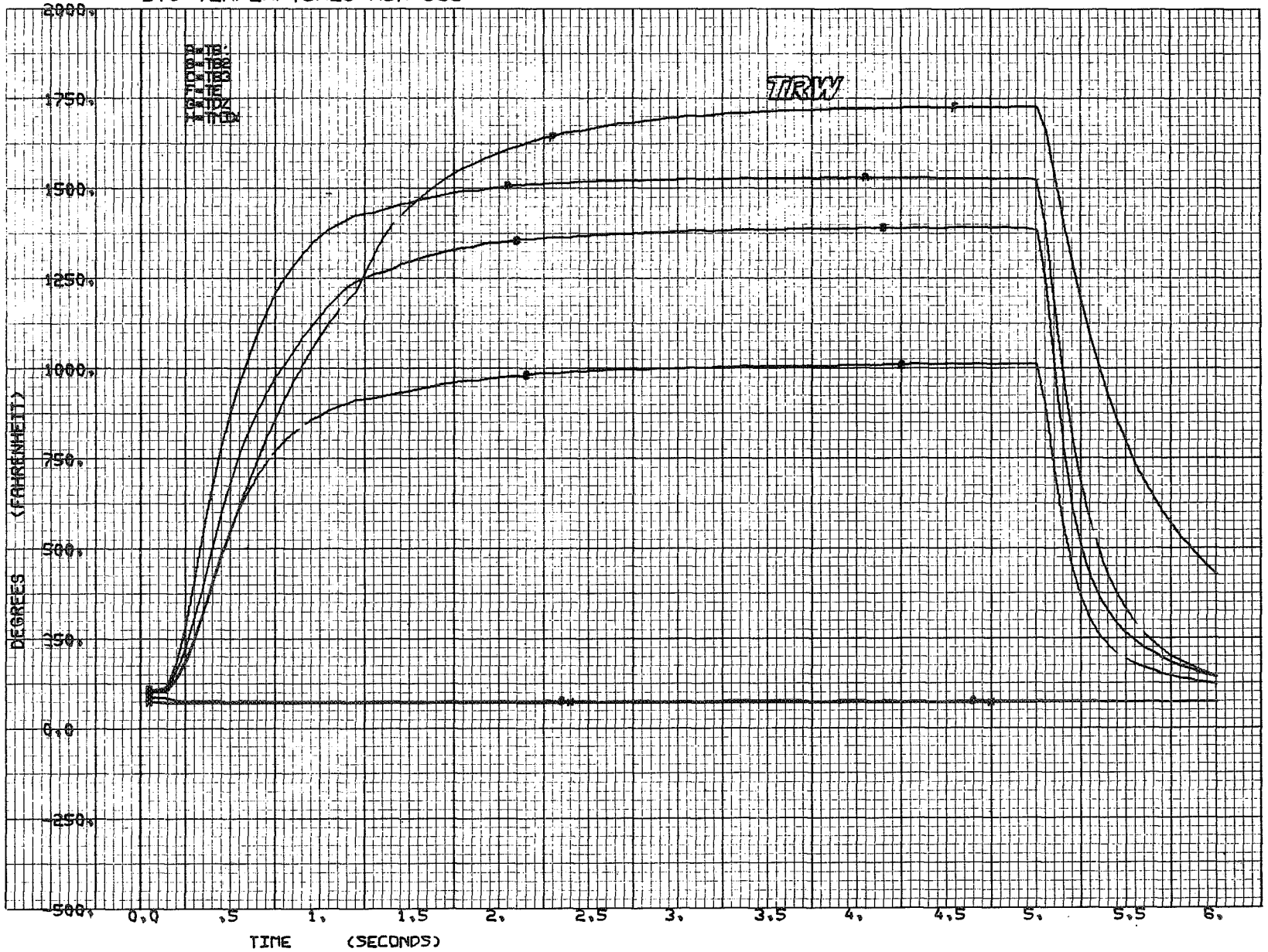
ITS TEMPERATURES RUN 981



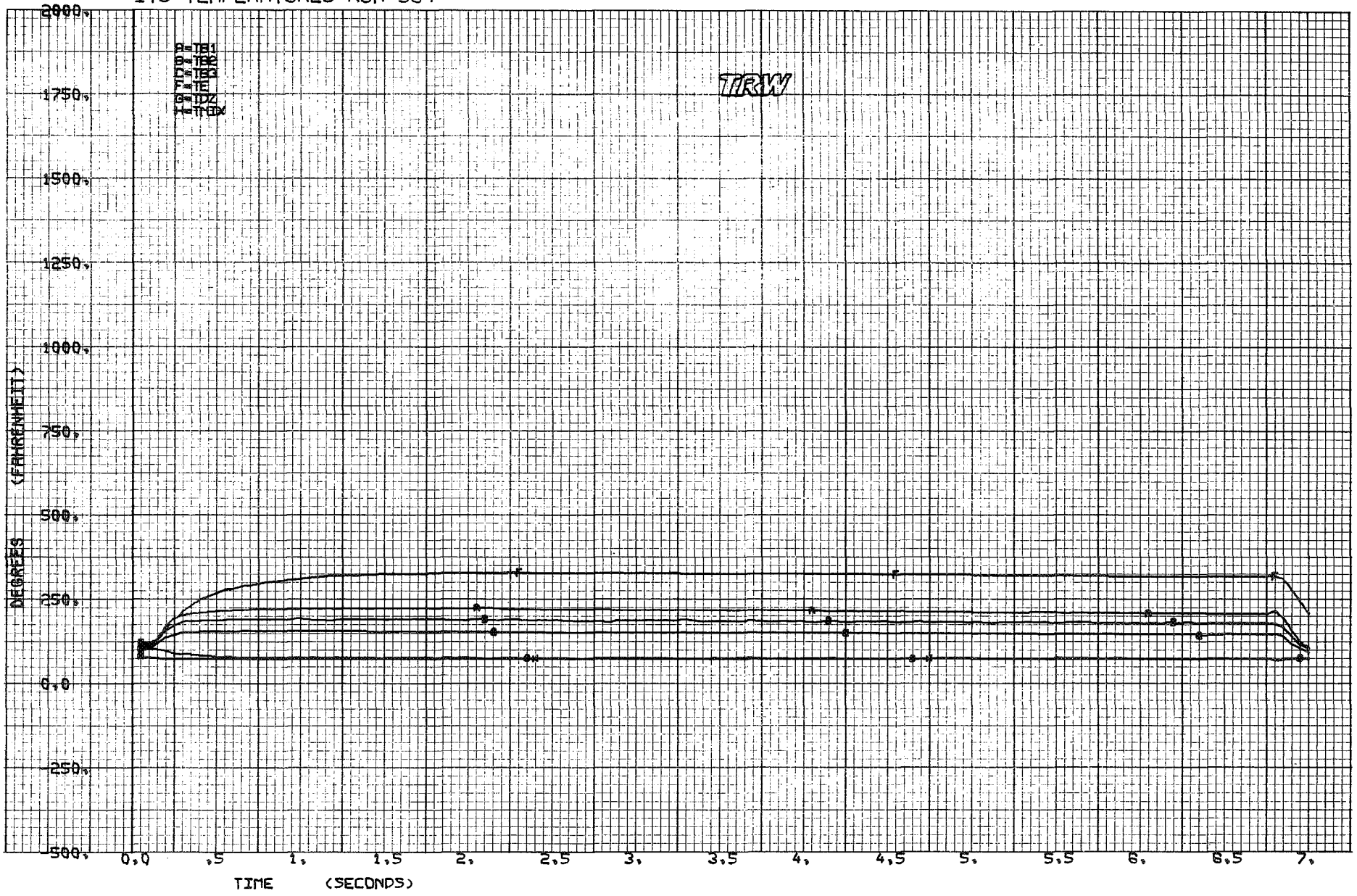
ITS TEMPERATURES RUN 982



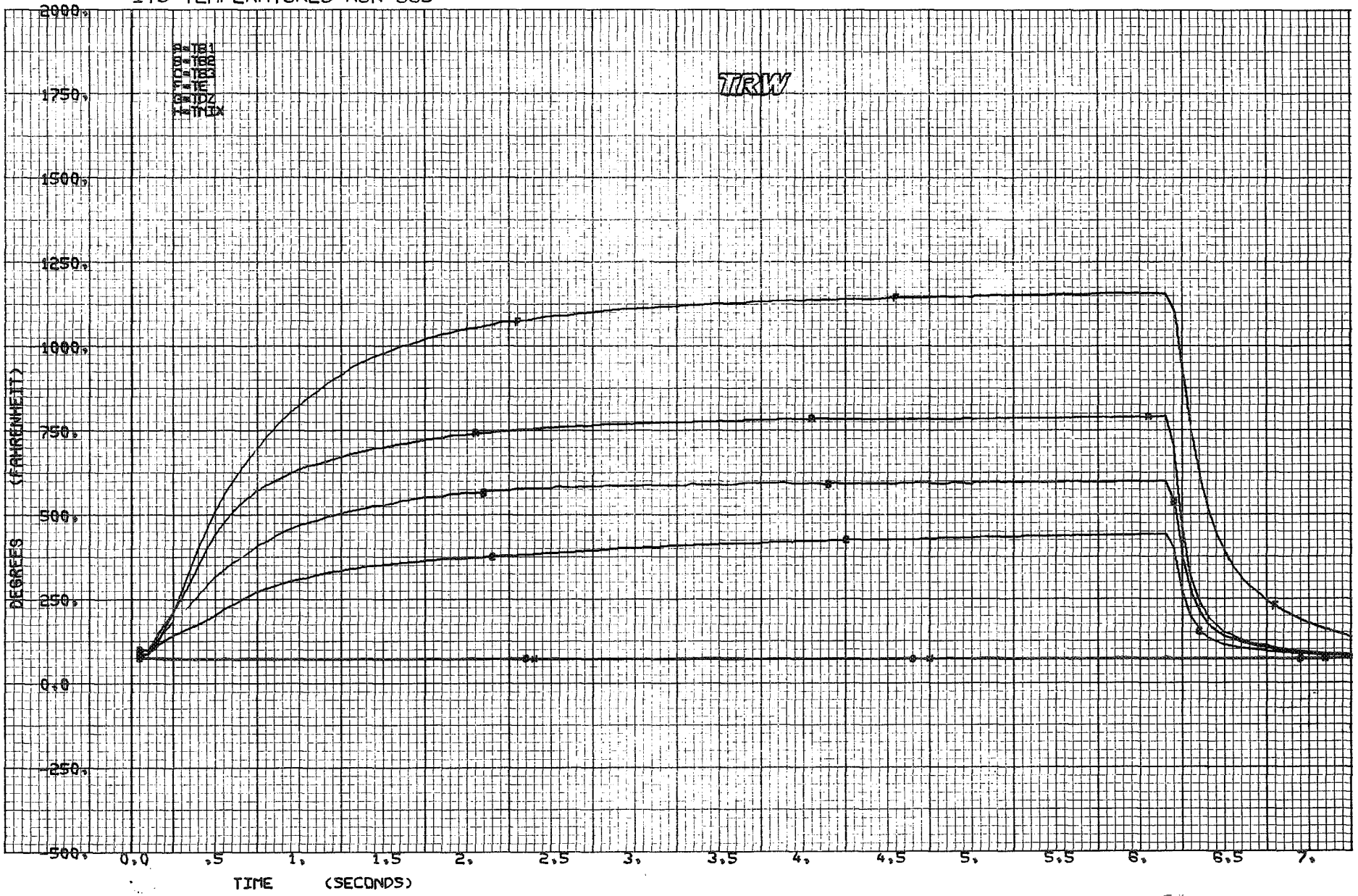
ITS TEMPERATURES RUN 983



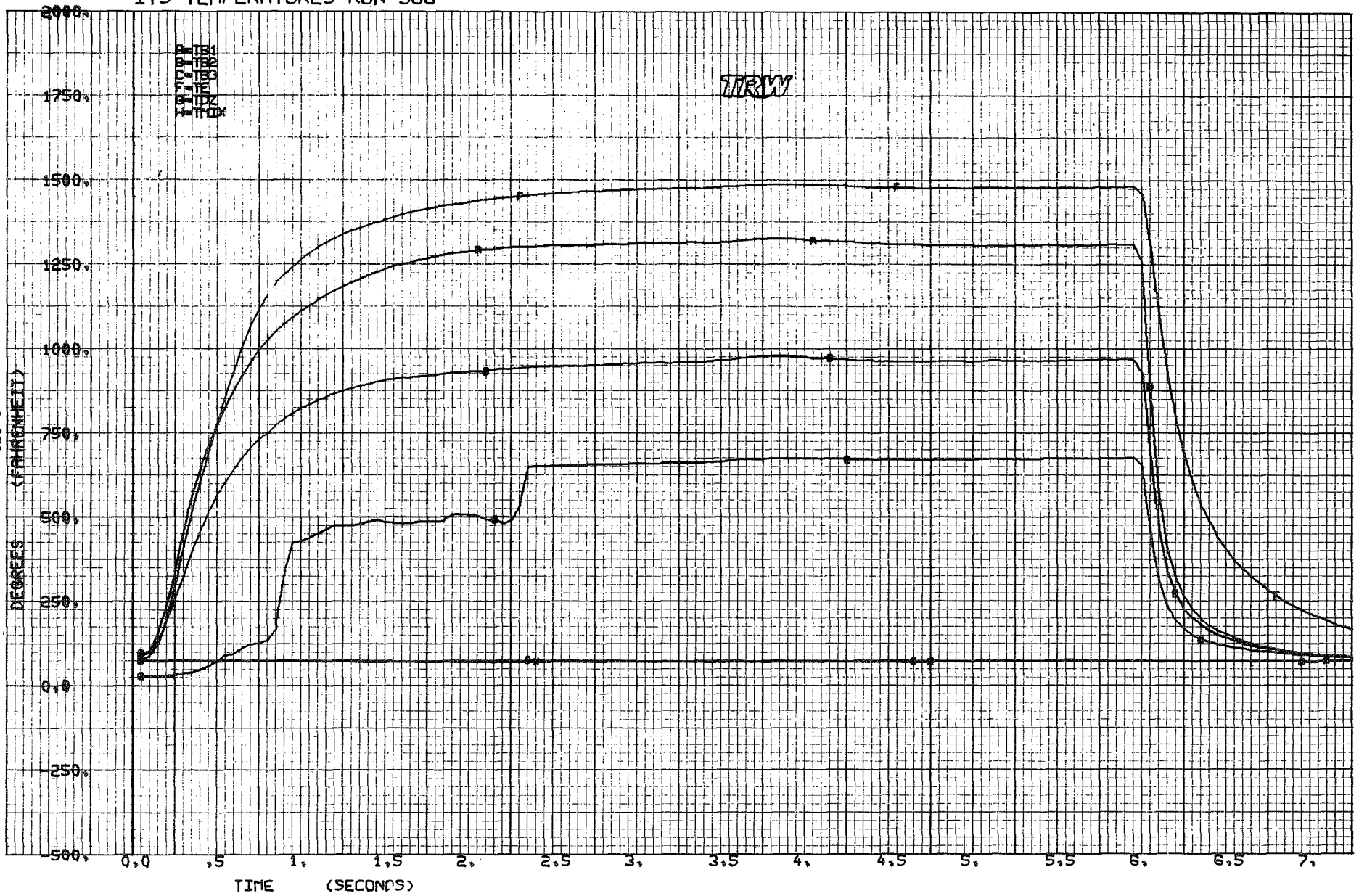
ITS TEMPERATURES RUN 984



ITS TEMPERATURES RUN 985



ITS TEMPERATURES RUN 986



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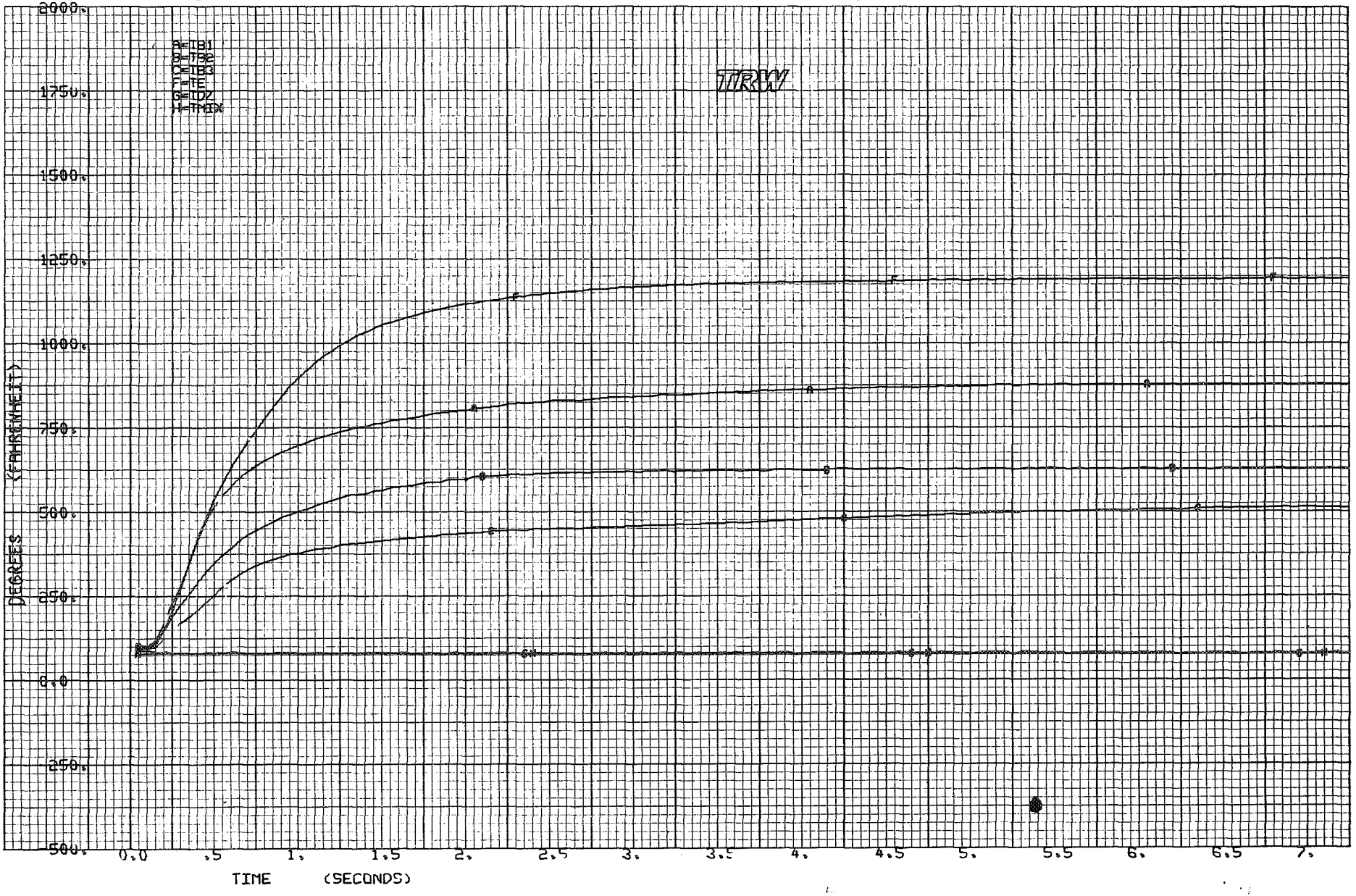

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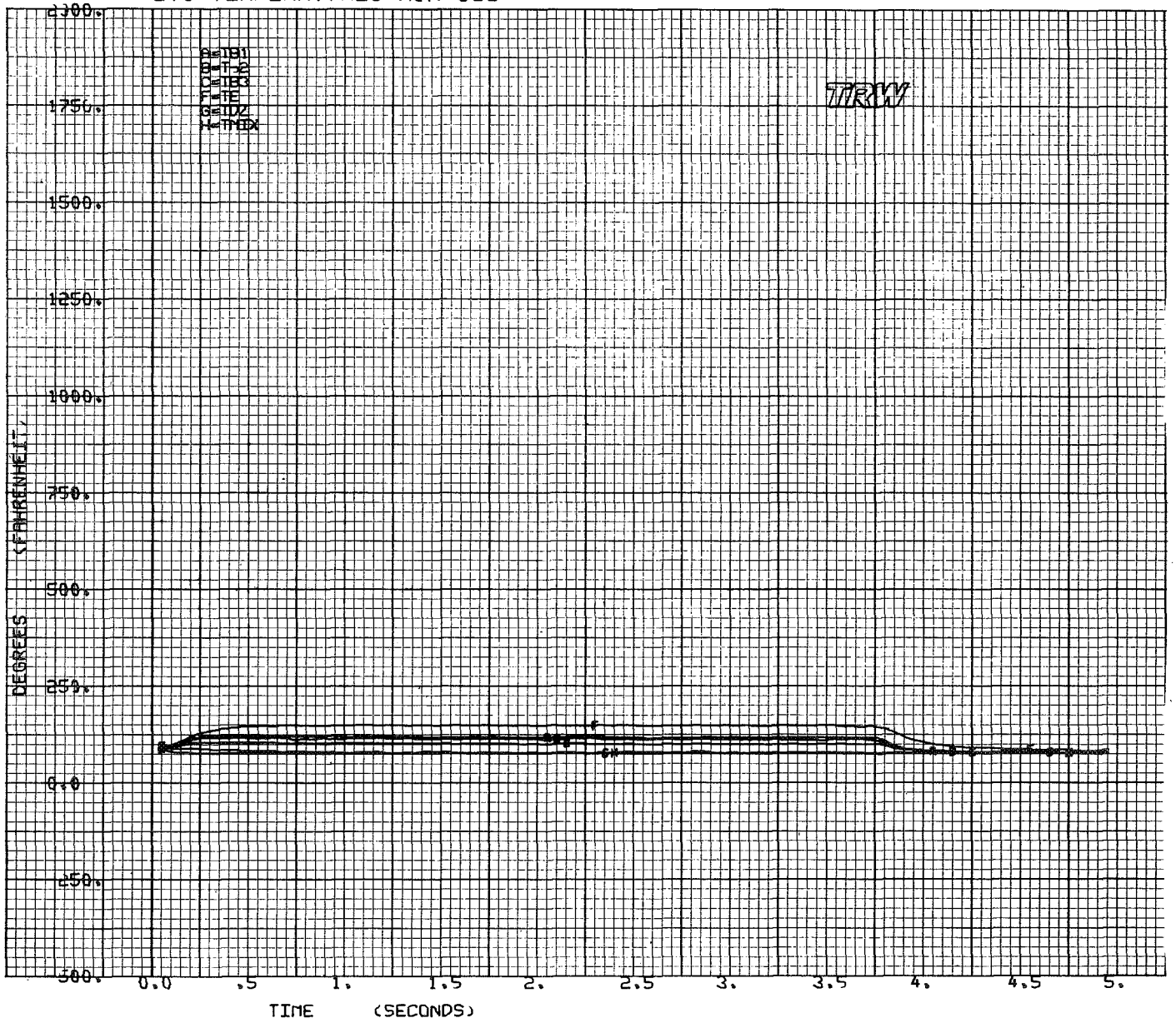
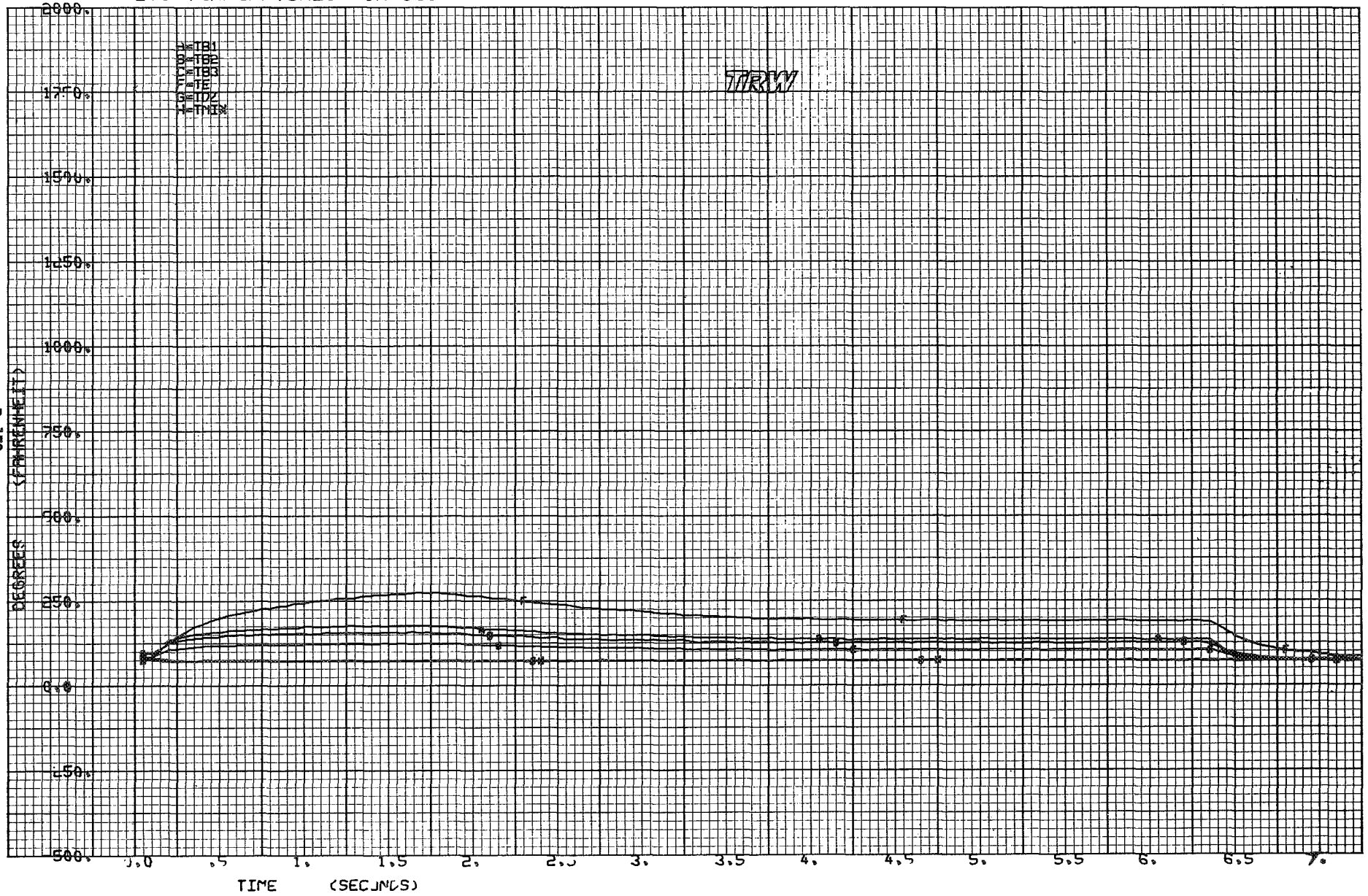
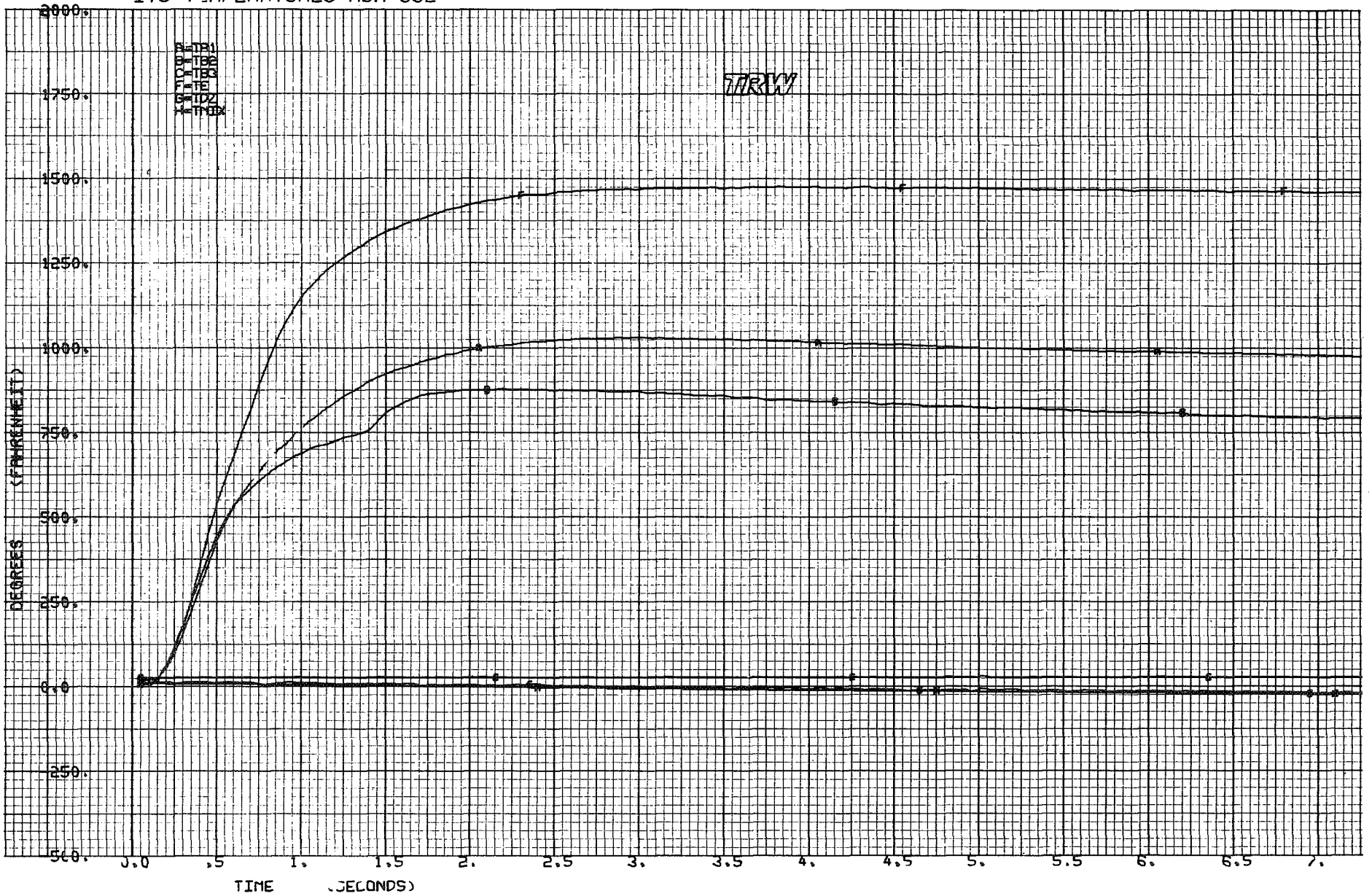


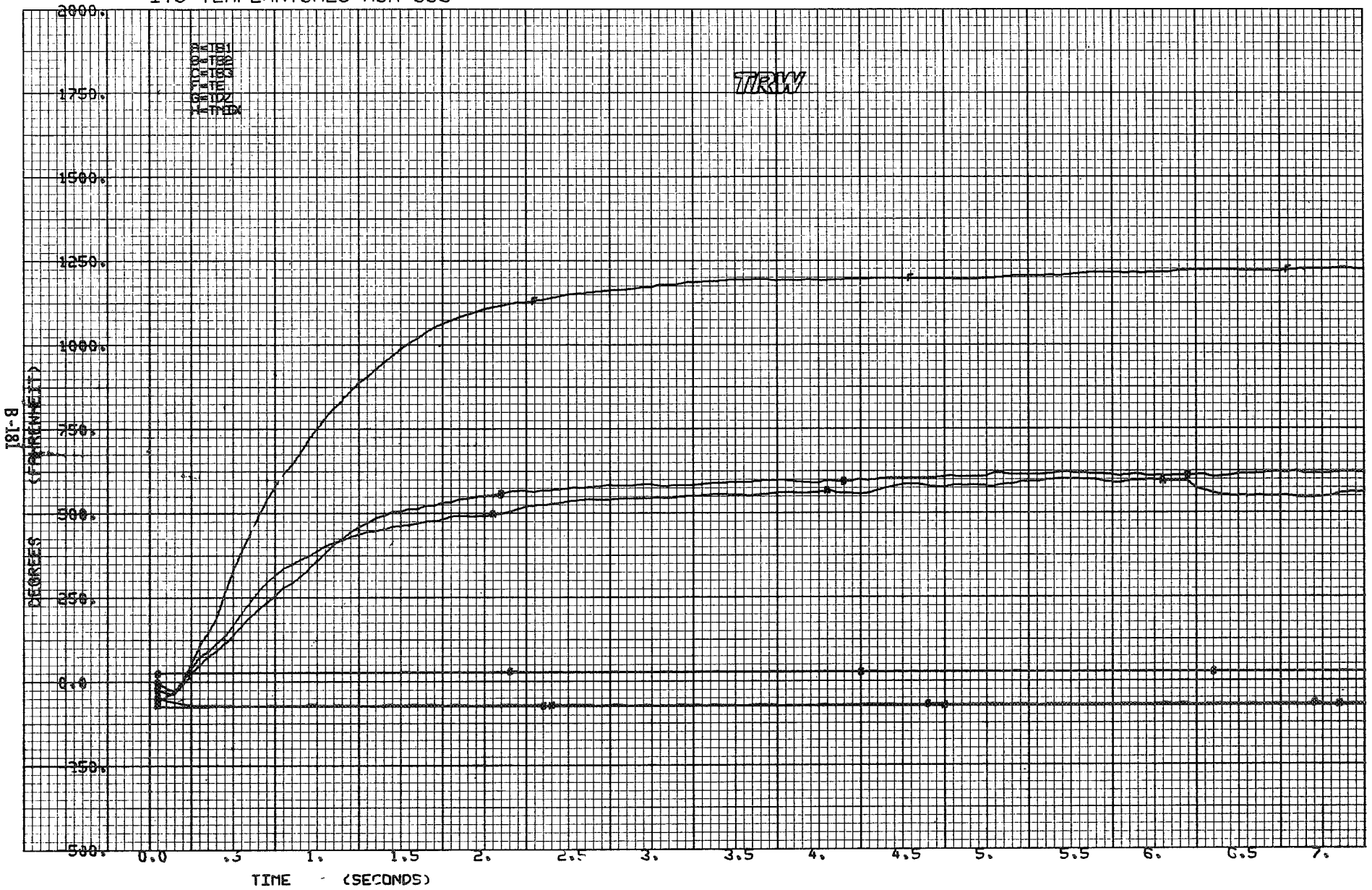
ITS TEMPERATURES RUN 989



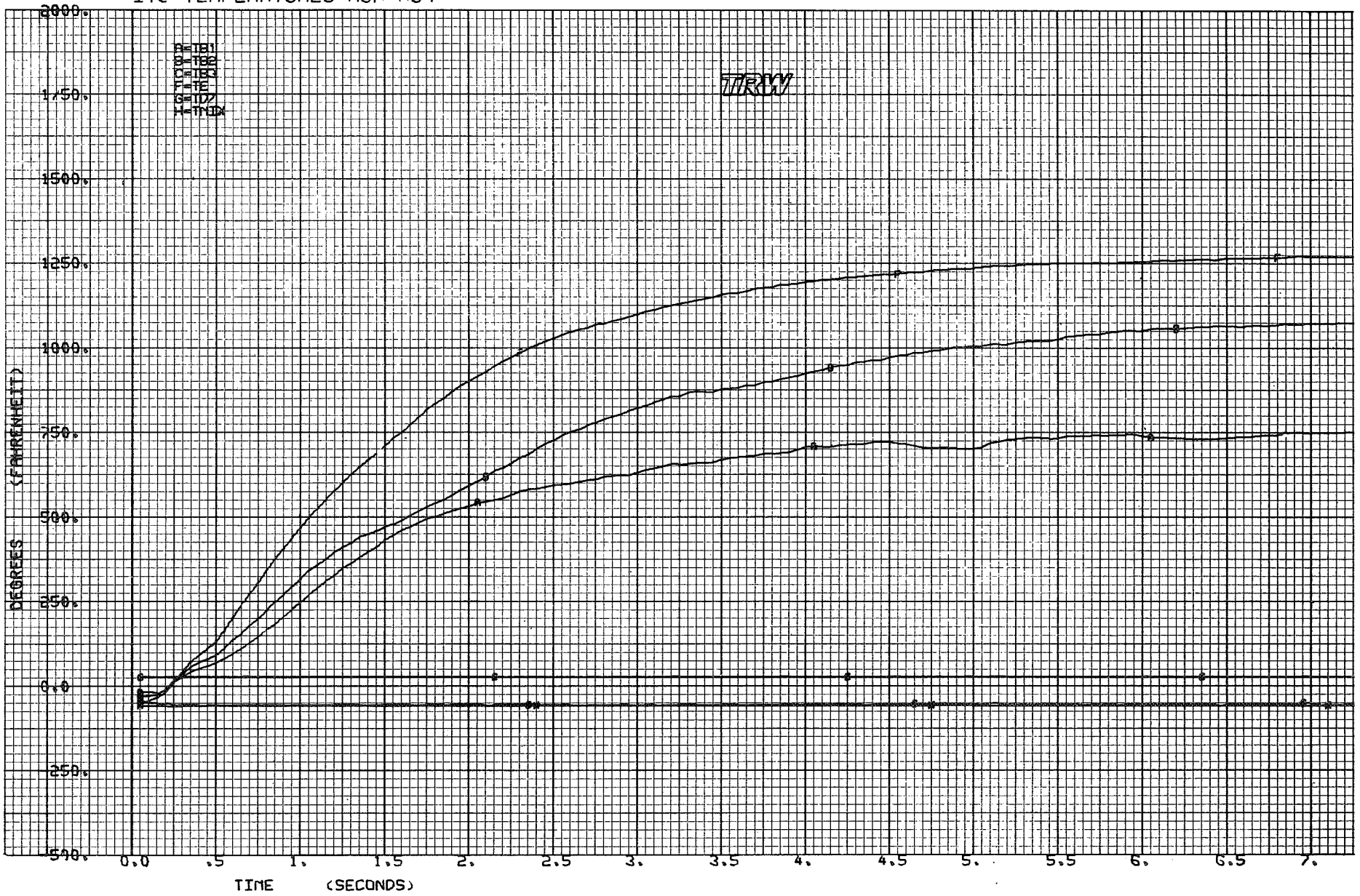
ITS TEMPERATURES RUN 992

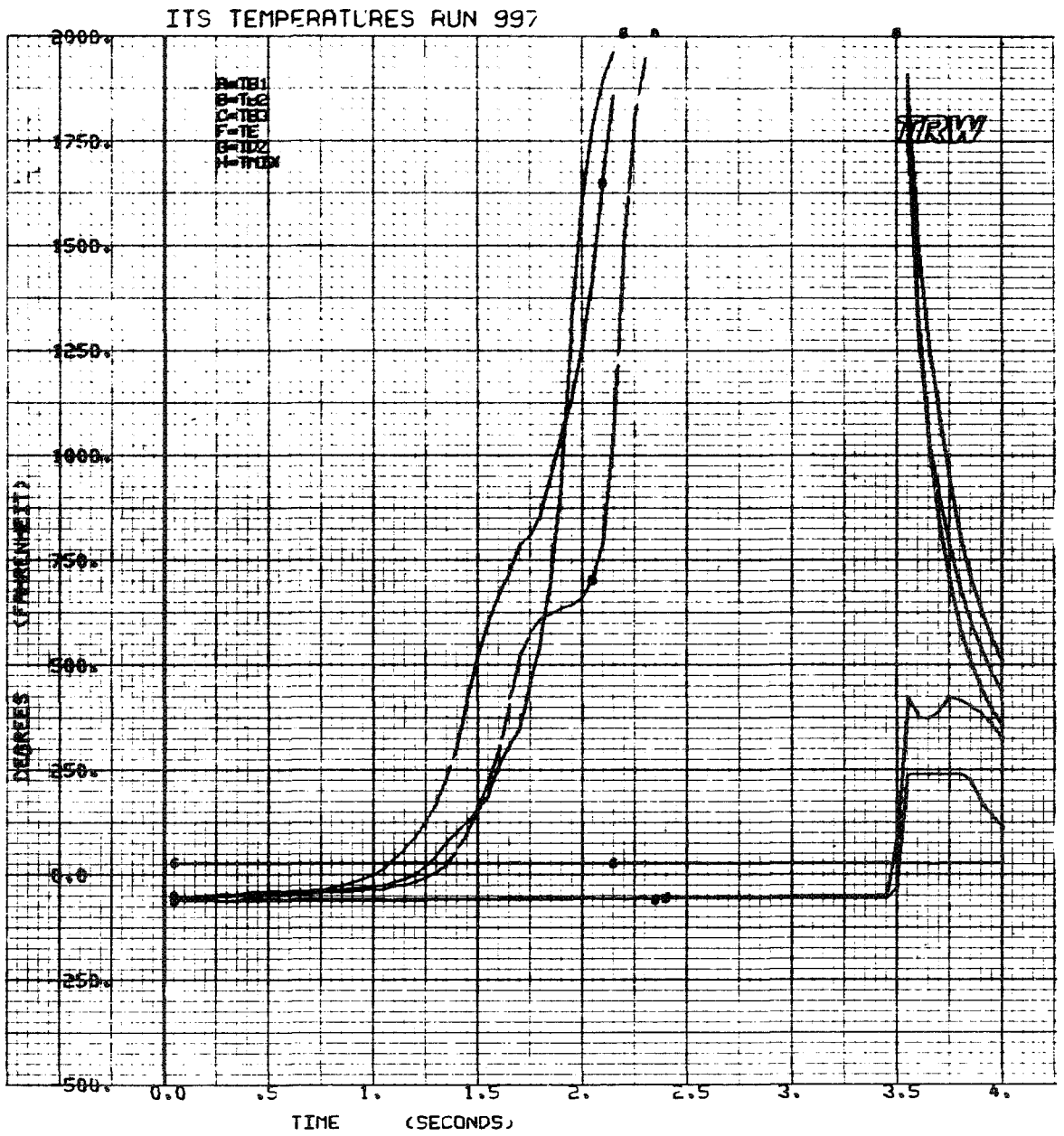


ITS TEMPERATURES RUN 993



ITS TEMPERATURES RUN 994




ITS TEMPERATURES RUN 997
A=TB1
B=TB2
C=TB3
F=TE
G=TD2
H=TMOX
DEGREES (FAHRENHEIT)
2000.
1750.
1500.
1250.
1000.
750.
500.
250.
0.0
-250.
-500.
0.0
.5
1.
1.5
2.
2.5
3.
3.5
4.
TIME (SECONDS)

APPENDIX C

CALCULATION OF C* EFFICIENCY

The index of injector performance used in the experimental program was the corrected C* efficiency. This parameter was calculated by two independent methods, one based on measurement of chamber pressure and the other on measurement of thrust. Details of the computational procedures and of the applied corrections are given in the following sections. The procedures and nomenclature format are essentially those as developed in NASA sponsored programs at Rocketdyne.

1.0 CHAMBER PRESSURE TECHNIQUE

Characteristic velocity efficiency based on chamber pressure is defined by the following:

$$\eta_{C^*} = \frac{(P_c)_o \, (A_t)_{eff} \, g_c}{(\dot{w}_T) \, (C^*)_{theo}} \qquad \text{C-1}$$

where

$(P_c)_o$	=	stagnation pressure at the throat
$(A_t)_{eff}$	=	effective thermodynamic throat area
g_c	=	conversion factor (32.174 lbm-ft/lbf-sec^2)
$\dot{w}_T$	=	total propellant weight flow rate
$(C^*)_{theo}$	=	theoretical characteristic velocity based on shifting equilibrium

Values calculated from Equation (C-1) are referred to as "corrected" C* efficiencies, because the factors involved are obtained by application of suitable influence factor corrections to measured parameters. Stagnation pressure at the throat is obtained from measured static pressure at start of nozzle convergence by assumption of isentropic expansion, effective throat area is estimated from measured geometric area by allowing for geometrical radius changes during firing and for nonunity discharge coefficient, and chamber pressure is corrected to allow for energy losses from combustion

gases to the chamber wall by heat transfer and friction. Equation (C-1) may therefore be written as follows:

$$\eta_{C^*} = \frac{P_c \, A_t \, g_c \, f_p \, f_{TR} \, f_{DIS} \, f_{FR} \, f_{HL} \, f_{KE}}{(\dot{w}_o + \dot{w}_f) \, (C^*)_{theo}} \tag{C-2}$$

where

P_c = measured static pressure at start of nozzle convergence, psia

A_t = measured geometric throat area, in^2

g_c = conversion factor (32.174 lbm-ft/lbf-sec^2)

$\dot{w}_o$ = oxidizer weight flow rate, lb/sec

$\dot{w}_f$ = fuel weight flow rate, lb/sec

$(C^*)_{theo}$ = theoretical C* based on shifting equilibrium calculations, ft/sec

f_p = influence factor correcting observed static pressure to throat stagnation pressure

f_{TR} = influence factor correcting for change in throat radius during firing

f_{DIS} = influence factor correcting throat area for effective discharge coefficient

f_{FR} = influence factor correcting measured chamber pressure for frictional drag of combustion gases at chamber wall

f_{HL} = influence factor correcting measured chamber pressure for heat losses from combustion gases to chamber wall

f_{KE} = influence factor correcting C* values to account for finite chemical reaction rates

Methods of estimation of the various correction factors are described in the following paragraphs.

1.1 PRESSURE INFLUENCE FACTOR (f_p)

Measured static pressure at start of nozzle convergence is converted to stagnation pressure at the throat by assumption of effectively no

combustion in the nozzle and application of the isentropic flow equations, with contraction ratio (A_c/A_t) and shifting-equilibrium specific heat ratios (γ). Frozen-equilibrium specific heat ratios usually make the influence correction factor about 1/2 percent larger. Hence the value employed with shifting-equilibrium is the more conservative. Figure C-1 shows the influence factor as a function of contraction ratio.

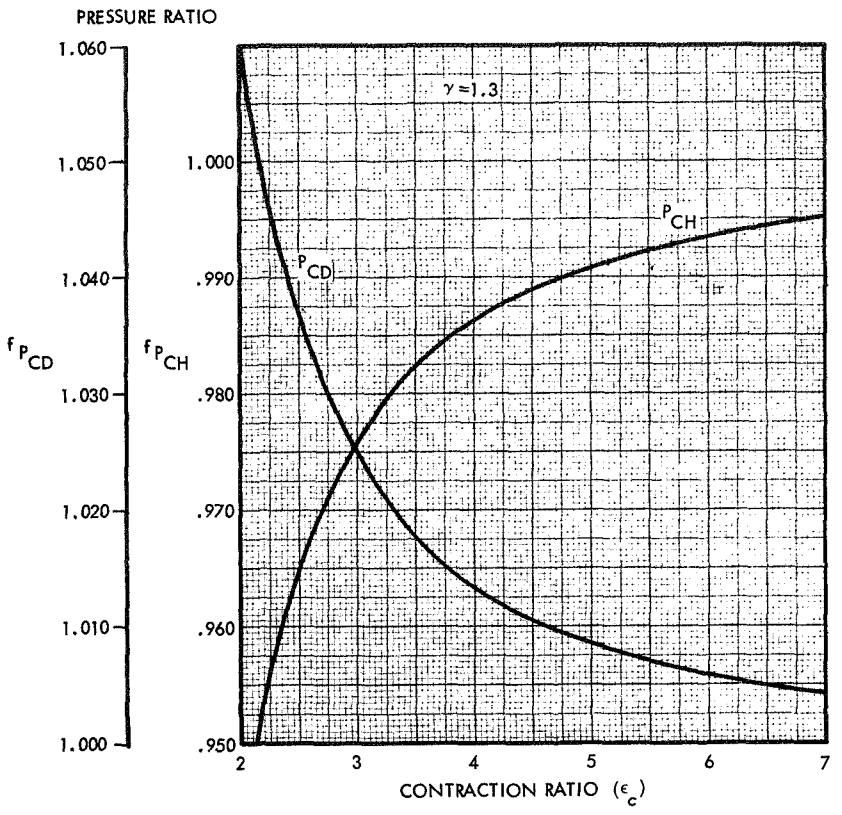


Figure C-1. Momentum Correction

1.2 THROAT RADIUS INFLUENCE FACTOR (f_{TR})

Temperature gradients produced in the solid metal nozzle wall result in thermal stresses which affect throat radius, with the result that the geometric throat diameter ambient measurement is not the same as that which exists during firing.

In the chamber type employed during the experimental effort (i.e. thin throat wall thickness), the throat area change is computed from the thermal growth of the throat based on temperature changes from ambient temperature. The change in throat area can be written as:

$$A_{th} = \frac{\pi}{4} (2 + \alpha\Delta T) (\alpha\Delta T) D^2 \tag{C-3}$$

where

ΔA^* = change in throat area due to thermal growth

α = average thermal expansion coefficient

ΔT = temperature rise from ambient conditions

D = throat diameter at ambient conditions

The throat area correction factor is as follows:

$$f_{TR} = 1 + \frac{\Delta A_{th}}{A_{th}}$$

$$= [1 + \alpha\Delta T]^2 \tag{C-4}$$

The thermal expansion coefficient for copper is $\alpha_{Cu} = 9.8 \times 10^{-6}$ in/in-°F, assuming an ambient temperature of 70°F, the throat area correction factor becomes

$$f_{TR} = [1 + 9.8 \times 10^{-6} (T_{th} - 70)]^2 \tag{C-5}$$

This equation was used to generate the curve in Figure C-2.

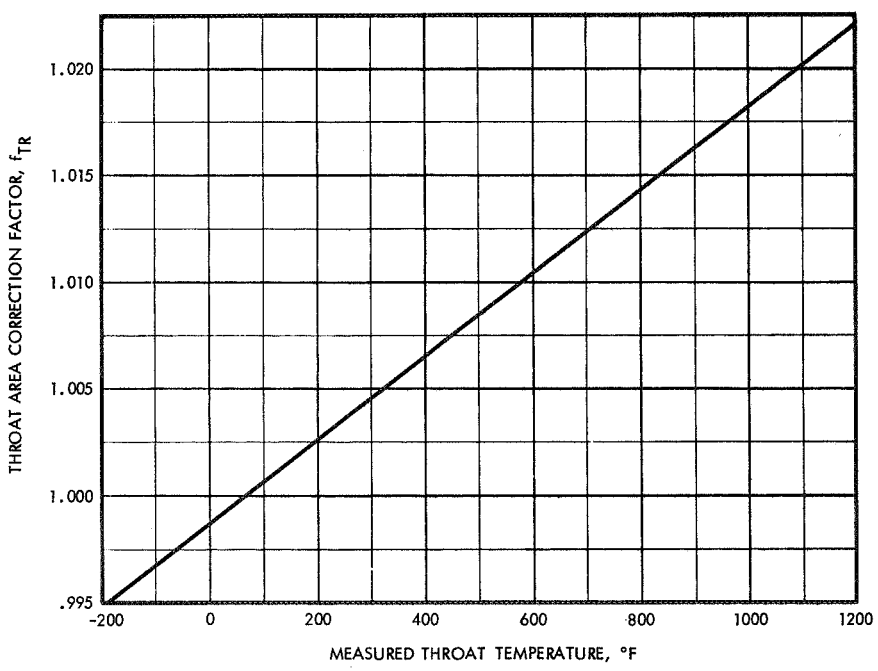


Figure C-2. Throat Area Correction Factor

1.3 THROAT DISCHARGE COEFFICIENT INFLUENCE FACTOR (f_{DIS})

The discharge coefficient is defined as the ratio of actual flow rate through the throat to the theoretical maximum, based on geometric throat area and ideal, uniform, one-dimensional flow with no boundary layer. The discharge influence coefficient may be estimated in two ways: one based on calculations made from a theoretical, inviscid flow model of combustion products, and the other based on a correlation of results obtained in various experimental study results of air flow through nozzles of similar geometry.

1.3.1 Theoretical Model

Total mass flow rate is given by

$$\dot{m} = \int_{o}^{A} \rho \, V \, dA \tag{C-6}$$

where:

ρ = gas density

V = gas velocity

A = cross-sectional area

Theoretical maximum flow rate at the throat is

$$\dot{m}_{max} = \int_{o}^{A_t} \rho^* \, V^* \, dA \tag{C-7}$$

where:

A_t = geometric area of the throat

ρ^* = sonic gas density

V^* = sonic gas velocity

For ideal, uniform, parallel flow, Equation (C-7) becomes

$$\dot{m}_{max} = \rho^* \, V^* \, A_t \tag{C-8}$$

The discharge coefficient is then

$$C_D = \frac{\dot{m}}{\dot{m}_{max}} = \int_o^A \left(\frac{\rho}{\rho^*}\right)\left(\frac{V}{V^*}\right)\left(\frac{dA}{A_t}\right) \qquad \text{(C-9)}$$

1.3.2 Empirical Value

Experimental conical nozzle discharge coefficients obtained with air by various investigators are plotted in Figure C-3 against the indicated geometric parameter. Data sources also are listed in Figure C-3.

The values obtained by both methods are found to be in excellent agreement.

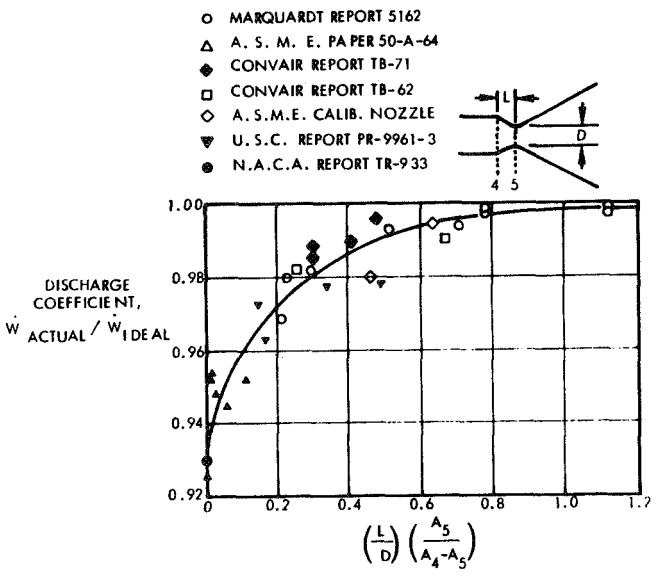


Figure C-3. Conical Nozzle Discharge Coefficient

1.4 FRICTIONAL DRAG INFLUENCE FACTOR (f_{FR})

Calculations of C* based on chamber pressure are concerned with chamber phenomena up to the nozzle throat. Drag forces to this point are small enough to be considered negligible, so that the factor f_{FR} may be taken to be unity.

1.5 ENERGY LOSS INFLUENCE FACTOR (f_{HL})

Chamber pressure and thrust are decreased by heat transfer from the combustion gases to the walls of a thrust chamber. This enthalpy loss is substantially reduced in ablative chambers and is effectively recovered in a regeneratively cooled chamber.

The effect on C* of enthalpy loss by heat transfer can be estimated from a loss of chamber enthalpy. This is determined from a two station energy balance, one at the start of nozzle convergence and the other at the throat.

$$1/2\ V_c^{\ 2} + H_c = 1/2\ V_t^{\ 2} + H_t + \dot{Q}_{conv} \qquad \text{(C-10)}$$

where:

V_c = gas velocity at chamber exit
V_t = gas velocity at nozzle throat
H_c = gas enthalpy at chamber exit
H_t = gas enthalpy at nozzle throat
$\dot{Q}_{conv}$ = heat loss in nozzle convergence

Velocity at the throat is given by:

$$V_t = [V_c^{\ 2} + 2(H_c - H_t - \dot{Q}_{conv})]^{1/2} \qquad \text{(C-11)}$$

With negligible nozzle inlet velocity

$$V_t = [2(H_c - H_t - \dot{Q}_{conv})]^{1/2} \qquad \text{(C-12)}$$

Logarithmic differentiation of Equation (D-12) gives

$$\frac{dV_t}{V_t} = 1/2 \cdot \frac{d\ (H_c - H_t - \dot{Q}_{conv})}{(H_c - H_t - \dot{Q}_{conv})} = 1/2 \left(\frac{d\ H_c - dH_t}{H_c - H_t - \dot{Q}_{conv}} \right) \qquad \text{(C-13)}$$

Substitution of enthalpy definition into Equation (C-13) gives:

$$\frac{dV_t}{V_t} = 1/2 \left(\frac{c_{pc}\ dT_c - c_{pt}\ dT_t}{H_c - H_t - \dot{Q}_{conv}} \right) \qquad \text{(C-14)}$$

With constant $\dot{C}_p$ between the two stations,

$$\frac{dV_t}{V_t} = 1/2 \left(\frac{c_p \, dT_c}{H_c - H_t - \dot{Q}_{conv}}\right)\left(1 - \frac{dT_t}{dT_c}\right) \qquad \text{(C-15)}$$

If the specific heat ratio, γ, is assumed constant,

$$\frac{dT_t}{dT_c} = \frac{T_t}{T_c} \qquad \text{(C-16)}$$

Substituting Equation (C-16) into Equation (C-15), replacing differentials by incrementals, and noting that C* is proportional to gas velocity at the throat gives:

$$\frac{\Delta V_t}{V_t} = \frac{\Delta C^*}{C^*} = 1/2 \left(\frac{c_p \, \Delta T_c}{H_c - H_t - \dot{Q}_{conv}}\right)\left(1 - \frac{\Delta T_t}{T_c}\right) \qquad \text{(C-17)}$$

Total heat loss to the chamber walls, in Btu per pound of propellant, is obtained by summation of observed heat fluxes over the appropriate areas:

$$\text{Heat loss} = \frac{\Sigma(q/A)\ A}{\dot{w}_T} \qquad \text{(C-18)}$$

where:

q/A = experimentally observed heat flux

A = area applicable to each q/A value

$\dot{w}_T$ = total propellant flow rate

If this heat loss is equated to the change in enthalpy of the gas in the combustion chamber, $c_p \, \Delta T_c$, then substitution in Equation (C-17) gives:

$$\frac{\Delta C^*}{C^*} = 1/2 \left[\frac{\Sigma(q/A)A}{\dot{w}_T}\right]\left[\frac{1 - (T_t/T_c)}{H_c - H_t - Q_{conv}}\right] \qquad \text{(C-19)}$$

The applicable influence factor is:

$$f_{HL} = 1 + \frac{\Delta C^*}{C^*} = 1 + 1/2 \left[\frac{\Sigma(q/A)A}{\dot{w}_T}\right]\left[\frac{1 - (T_t/T_c)}{H_c - H_t - Q_{conv}}\right] \qquad \text{(C-20)}$$

An alternate expression can be obtained from the basic C* definition:

$$C^* = \frac{\sqrt{RT_c}}{\Gamma} \tag{C-21}$$

Logarithmic differentiation of this yields:

$$\frac{dc^*}{c^*} = \frac{1}{2}\frac{dT_c}{T_c} \tag{C-22}$$

Substituting incrementals from differentials in Equation (C-22) gives:

$$\frac{\Delta c^*}{c^*} = \frac{1}{2}\frac{\Delta T_c}{T_c} \tag{C-23}$$

Equating ΔT_c with the heat loss from Equation (C-18) results in the following:

$$\frac{\Delta c^*}{c^*} = \frac{1}{2}\left[\frac{\Sigma(q/A)A}{\dot{w}_t}\right]\left[\frac{1}{c_p T_c}\right] \tag{C-24}$$

The applicable influence factor is:

$$f_{HL} = 1 + \frac{1}{2}\left[\frac{\Sigma(q/A)A}{\dot{w}_t}\right]\left[\frac{1}{c_p T_c}\right] \tag{C-25}$$

where

c_p = specific heat at constant pressure

Although derived independently it can be shown that these two expressions, Equations (C-20) and (C-25), are nearly equivalent.

1.6 INFLUENCE FACTOR FOR CHEMICAL KINETICS (f_{KE})

The effect of finite chemical reaction rates is to produce a C* less than the corresponding theoretical equilibrium values. A TRW Systems Group developed one-dimension nonequilibrium reacting gas computer program was employed with reaction rate constants selected for the propellant system. The fluid mechanical and chemical equations were integrated from the inlet section by an implicit technique. It was determined that the effect of nonequilibrium chemistry produced a C* loss of 1.2% compared to the shifting equilibrium limits.

2. CALCULATIONS BASED ON THRUST

The alternate determination of C* efficiency is based on thrust:

$$\eta_{C^*} = \frac{F_{vac}\, g_c}{(C_F)_{vac}\, \dot{w}_T\, C^*_{theo}} \qquad \text{(C-26)}$$

where:

F_{vac} = measured thrust corrected to vacuum conditions by the equation: $F_{vac} = F + P_a A_e$

F = measured thrust, lbf

P_a = ambient pressure, psia

A_e = area of nozzle exit, in^2

g_c = conversion factor (32.174 lbm-ft/lbf-sec^2)

$(C_F)_{vac}$ = theoretical shifting thrust coefficient (vacuum)

$\dot{w}_T$ = total propellant flow rate, lbm/sec

c^*_{theo} = theoretical shifting-equilibrium characteristic velocity, ft/sec

Values of vacuum thrust are obtained by applying corrections to sea-level measurements. With these values, which include allowances for all important departures from ideality, theoretical thrust coefficients may be used for calculation of C*. C_F efficiency is taken as 100 percent if there is no combustion in the nozzle, if chemical equilibrium is maintained in the nozzle expansion process, and if energy losses from the combustion gases are accounted for.

Applicable influence factors for measured thrust are specified in the following equation:

$$\eta_{C^*} = \frac{(F + P_a A_e)\, g_c\, \phi_{FR}\, \phi_{DIV}\, \phi_{HL}\, \phi_{KE}}{(C_F)_{theo}\, (\dot{W}_o + \dot{w}_f)\, (C^*)_{theo}} \qquad \text{(C-27)}$$

where:

F = measured thrust, lb_f

P_a = ambient pressure, psia

A_e = area of nozzle exit, in^2

g_c = conversion factor (32.174 lbm-ft/lbf-sec^2)

$(C_F)_{theo}$	=	theoretical shifting thrust coefficient (vacuum)
$\dot{w}_o$	=	oxidizer weight flow rate, lbm/sec
$\dot{w}_f$	=	fuel weight flow rate, lbm/sec
$(C^*)_{theo}$	=	theoretical shifting equilibrium characteristic velocity, ft/sec
ϕ_{FR}	=	influence for frictional losses
ϕ_{DIV}	=	influence factor for nozzle divergence
ϕ_{HL}	=	influence factor for heat losses to chamber and nozzle walls
ϕ_{KE}	=	influence factor correcting C* and C_F values to account for finite chemical reaction rates

The influence factors in Equation (C-27) are applied to vacuum thrust $(F + P_aA_e)$ instead of to measured site thrust (F) because, for convenience, the factors are readily calculated as changes in efficiency based on theoretical vacuum parameters. The total influence factor is then of the form $\Delta F/F_{vac}$.

Implicit in the use of theoretical C_F values are corrections to geometric throat area and to measured static chamber pressure at start of nozzle convergence. Therefore, calculation of corrected C* efficiency from thrust measurement includes all the previously described corrections plus an additional one to account for nonparallel nozzle exit flow. However, because $(C_F)_{theo}$ is essentially independent of small changes to chamber pressure and contraction ratio which are involved in corrections to P_c and A_t, these corrections are of no practical significance in calculation of C* from thrust measurements.

2.1 INFLUENCE FACTOR FOR FRICTIONAL DRAG (ϕ_{FR})

This factor corrects for energy losses caused by viscous drag forces on the thrust chamber walls. Its magnitude is estimated by a boundary layer analysis utilizing the integral momentum equation for turbulent flow, which accounts for boundary layer effects from the injector to the nozzle exit by suitable description of the boundary layer profile and local skin friction coefficient. A computer program is used to carry out a numerical

integration of the equation, including effects of pressure gradient, heat transfer, and surface roughness. The program requires a potential nozzle flow solution obtained from variable-property, axisymmetric method of characteristics calculation of the flow field outside the boundary layer; corresponding properties for the subsonic combustion chamber flow field are also calculated.

2.2 INFLUENCE FACTOR FOR NOZZLE DIVERGENCE (ϕ_{DIV})

The one-dimensional theoretical performance calculations assume that flow at the nozzle exit is uniform and parallel to the nozzle axis. The influence factor, ϕ_{DIV}, allows for nozzle divergence (i.e., for nonaxial flow) and for nonuniformity across the nozzle exit plane. It is calculated by a computer program which utilizes the axisymmetric method of characteristics for a variable-property gas. Computation begins with a transonic input near Mach 1, providing a characteristic line for use in the analysis of the supersonic portion of the nozzle. The resulting pressures are integrated over the given geometry to give the geometric efficiency.

2.3 INFLUENCE FACTOR FOR HEAT LOSS (ϕ_{HL})

To obtain the heat loss influence factor from measured thrust the approach is identical to that taken previously from the pressure measurement, except that the nozzle losses must also be included. With constant specific heat and gamma from start of nozzle convergence to exit, Equation (C-20) becomes

$$\phi_{HL} = 1 + \frac{1}{2}\left[\frac{\Sigma(\frac{q}{A})\,A}{\dot{w}_T}\right]\left[\frac{1 - T_e/T_c}{H_c - H_e - \dot{Q}_{nozzle}}\right] \qquad \text{(C-28)}$$

when "e" corresponds to the exit condition, and the summation occurs over the entire combustion.

An alternate can also be derived as in Equation (C-25). This equation becomes

$$\phi_{HL} = 1 + \frac{1}{2}\left[\frac{(q/A)A}{\dot{w}_T}\right]\left[\frac{1}{c_p\,T_e}\right] \qquad \text{(C-29)}$$

2.4 INFLUENCE FACTOR FOR CHEMICAL KINETICS (ϕ_{KE})

The effect of finite chemical reaction rates is to produce a C* and C_F less than the corresponding theoretical equilibrium values. A TRW Systems Group developed one-dimensional nonequilibrium reacting gas computer program was employed with reaction rate constants selected for the hydrogen-oxygen propellant system. The fluid mechanical and chemical equations were integrated from the inlet section by an implicit technique.

APPENDIX D

COMPUTER PROGRAMS

The following computer programs were used to facilitate performance and heat transfer analysis during various phases of this program.

1.0 ROCKET CHEMISTRY PROGRAM

The generalized equilibrium chemistry program solves a wide range of thermodynamic problems requiring only the composition and two of the following system properties to be specified: pressure, volume, temperature, enthalpy, entropy or internal energy. The program calculates composition, either from a pair of compounds with a specified weight mixture ratio, or from a series of compounds and their respective weight percents. In addition to the usual pure condensed phases, it is possible to submit a series of ideal chemical solutions composed of selected combinations of the condensed phases; the program will determine whether or not these solutions are formed by the reaction. The possible reaction products are obtained by searching a prepared master inventory tape containing entropy and enthalpy curve fit coefficients for all elements and compounds of interest. Nongaseous phases and ionized species are treated as distinct and separate compounds. The program initially assumes an ideal all-gas system. The equilibrium gas pressures of all possible gaseous species are calculated. Using these partial pressures as initial estimates, nongaseous phases and solutions are then considered. Upon convergence of the calculations, the program eliminates all but the actual gases, condensed phases, and ideal solutions present at equilibrium. Rocket performance is computed for isentropic sonic flow through a throat by specifying exit pressures, temperatures, or area ratios. Chemical reactions only, or chemical reactions and phase changes, may be stopped at any point in the expansion. Equilibrium or frozen composition, thermodynamic parameters, and the usual rocket parameters are given in the program output.

2.0 ONE-DIMENSIONAL, ONE-PHASE EXACT KINETIC COMPUTER PROGRAM

TRW Systems had developed under contract to the National Aeronautics and Space Administration Manned Spacecraft Center (Contract NAS 9-4358), a One-Dimensional, One-Phase (1D, 1P) Reacting Gas Nonequilibrium Performance Program. The computer program calculates the inviscid one-dimensional equilibrium, frozen and nonequilibrium nozzle expansion of propellant exhaust mixtures containing the six elements: carbon, hydrogen, oxygen, nitrogen, fluorine and chlorine.

The computer program considers all significant gaseous species (19) present in the exhaust mixtures of propellants containing these elements and all gas phase chemical reactions (48) which can occur between the exhaust products. In order to reduce the computation times per case to a minimum, the program utilizes a second-order implicit integration method. This integration method has reduced the computation time, per case, several orders of magnitude compared to the computation time required when utilizing standard explicit integration methods such as fourth order Runge-Kutta or Adams-Moulton methods.

The throat size is determined for each combination of propellant system and mixture ratio through use of the given chamber pressure, thrust level, and the value of the one-dimensional thrust coefficient, C_F, computed by the Rocket Chemistry Program. The reverse reaction rate constant, $k_r = AT^{-n}e^{-B/T}$, is employed in the Kinetics Program and forms a portion of the input data to the computer program. It is usually input in chemist's units; i.e., cm^3, gm, °K, sec., and is converted internally into units consisting of ft^3, lb, °R, sec.

3.0 ONE-DIMENSIONAL, TWO-PHASE EXACT KINETIC COMPUTER PROGRAM

TRW Systems has also developed, under contract to the National Aeronautics and Space Administration Manned Spacecraft Center (Contract NAS 9-4358), a One-Dimensional, Two-Phase (1D, 2P) Reacting Gas Nonequilibrium Performance Program. This program calculates the inviscid one-dimensional equilibrium, frozen and nonequilibrium nozzle expansion of propellant exhaust mixtures containing the six elements: carbon, hydrogen, oxygen, nitrogen, fluorine and chlorine; and one metal element, either aluminum, beryllium,

boron or lithium. Energy and momentum transfer is considered between the two phases but interphase mass transfer is not considered.

In all 79 species related by 763 reactions are handled with a maximum of 46 species and 380 reactions for the boron metal element. In addition, provision is made for eight condensed species with a maximum of four condensed species at any time. Furthermore, the condensed phase can be allocated to one (or more) size group.

All the species and reactions of importance to the proposed program can be accommodated by the TRW Systems developed One-Phase and/or Two-Phase Kinetics Programs.

4.0 VISCOUS EFFECTS COMPUTER PROGRAM.

The method of Bartz for computing boundary-layer thicknesses, skin-friction, and heat flux in axisymmetric nozzles has been revised and programmed for digital computer solution. The method solves, simultaneously, the integral momentum and energy equations for thin axisymmetric boundary layers. Boundary-layer shape parameters are approximated from one-seventh power profiles of velocity and stagnation temperature; and skin-friction coefficient and Stanton number are evaluated as functions of boundary-layer thickness from the best available semiempirical relations.

This program either employs a given wall Mach number distribution as generated by, for example, the Two-Dimensional Kinetics Computer Program, or can generate internally a one-dimensional Mach number distribution as a function of local area ratio and (constant) γ, the ratio of specific heats. In addition, the program requires a wall temperature, Tw, distribution. This distribution can be produced, by an iterative procedure, from a thermal analysis of the nozzle. A constant wall temperature may be assumed in lieu of such data.

The program computes the local parameters: convective heat transfer coefficient (h_g), heat flux ($\dot{q}/A$), where A is the nozzle surface area, skin-friction coefficient (C_f), boundary-layer thickness (δ), displacement thickness (δ^*) and momentum thickness (θ). The total heat rejection load, $\dot{q}$, is found from numerical intergration of $\dot{q}/A$ versus A.

5.0 BASIC ONE-DIMENSIONAL HEAT TRANSFER PROGRAM

This program, designated 84040 on TRW's IBM 7070, computes the change in temperature of each of a number of points (called nodes) in a slab of material, at specified intervals of time, during which the slab is to be heated and/or cooled. The heating and cooling is accomplished by convection and radiation, at the slab boundaries. The program is general enough so that it can handle heat transfer through both flat plate and cylindrical sections, regardless of size, thickness, and material layer composition. Most commonly, the program is used to simulate rocket engine firing duty cycles.

Preparation of the input for this program requires the following:

1) Convective heat transfer coefficients on the inside and outside surfaces
2) Inside and outside adiabatic wall temperatures
3) Initial temperatures of the node points
4) Thermal conductivities and diffusivities of the materials in the slab
5) Thicknesses of the material layers

The output consists of temperature profiles in the slab at specified time intervals.

6.0 THREE-DIMENSIONAL PROGRAM

This is a high-speed digital program for transient problems involving all nodes or combinations of nodes of heat transfer (i.e., convection, conduction, and radiation). This program can be used for any thermal problem whose finite difference equation is analogous to the differential equation for a lumped RC electrical network and can, therefore, be visualized as an electrical circuit.

The number of connecting flux paths to any node is arbitrary. This program can handle as many as 250 node point and capacitances with approximately 800 resistances.

Four valuable features are incorporated into this program:

1) Variable thermal properties are simulated when a table showing values of each property versus temperature is entered in the input.
2) This program can hold any node at a constant temperature for a period of time and thus simulate phase transition.
3) Erosion rate schedule is entered in the input in the form of a table.
4) A cathode follower is used for the purpose of transferring a temperature from one node in the network to another with zero transfer of energy.

7.0 GAS PROPERTIES COMPUTER PROGRAM

This program is used to rapidly determine the following:

1) Nozzle thrust coefficient as a function of pressure ratio
2) Nozzle area ratio as a function of pressure ratio
3) Nozzle area ratio as a function of Mach number
4) Ratio of local to critical temperature as a function of Mach number
5) Ratio of isentropic temperature drop to inlet temperature as a function of pressure ratio

The range of values covered for each of the basic parameters is:

Mach number:	1 - 10
Pressure ratio:	3.5 - 10^4
Ratio of specific heats:	1.1 - 1.67

The effect of the ratio of specific heats is included in all the plots.