Effect of Element Size on the Solution Accuracies of Finite-Element Heat Transfer and Thermal Stress Analyses of Space Shuttle Orbiter

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SUMMARY

The effect of element size on the solution accuracies of finite-element heat transfer and thermal stress analyses of space shuttle orbiter was investigated. Several structural performance and resizing (SPAR) thermal models and NASA structural analysis (NASTRAN) structural models were set up for the orbiter wing midspan bay 3. The thermal model was found to be the one that determines the limit of finite-element fineness because of the limitation of computational core space required for the radiation view factor calculations. The thermal stresses were found to be extremely sensitive to a slight variation of structural temperature distributions. The minimum degree of element fineness required for the thermal model to yield reasonably accurate solutions was established. The radiation view factor computation time was found to be insignificant compared with the total computer time required for the SPAR transient heat transfer analysis.

NOMENCLATURE

C  capacittance matrix
CQUAD2  quadrilateral membrane and bending element
CROD  two-node tension-compression-torsion element
C41  four-node forced convection element
E23  bar element for axial stiffness only
E25  zero length element used to elastically connect geometrically coincident joints
E31  triangular membrane element
E41  quadrilateral membrane element
E44  quadrilateral shear panel element
F_{ij}  view factor from element i to element j
FRSI  felt reusable surface insulation
H  convection load vector
HRSI  high-temperature reusable surface insulation
JLOC  joint location
K_h  convection matrix
K_k  conduction matrix
K_r  radiation matrix
K21  two-node line conduction element
K31  three-node area conduction element
K41  four-node area conduction element
K81  eight-node volume conduction element
NASTRAN  NASA structural analysis
Q  source load vector
R  radiation load vector
R31  three-node area radiation element
R41  four-node area radiation element
SIP  strain isolation pad
SPAR  structural performance and resizing
STS  space transportation system
T  absolute temperature, °R
TPS  thermal protection system
t  time, sec
INTRODUCTION
In finite-element heat transfer analysis or finite-element stress analysis, it is well known that reduction of element sizes (or increase in element number) will improve the solution accuracy. For simple structures, the element sizes may be reduced sufficiently to obtain highly accurate solutions. However, for large complex structures, such as the space shuttle orbiter, the use of excessively fine elements in the finite-element models may result in unmanageable computations that exceed the memory capability of existing computers. This computational limitation is frequently encountered during radiation view factor computations in the three-dimensional finite-element heat transfer analysis of complex structures. Because of computational limitations in the past heat transfer analysis of the space shuttle orbiter, only small local regions of the orbiter structure were modeled. Several regions of the space shuttle were modeled by Ko, Quinn, and Gong. For the past several years, these finite-element models were used to calculate orbiter structural temperatures, which were correlated with the actual flight data during the initial orbit tests of the space shuttle Columbia (refs. 1 to 7). Recently, Gong, Ko, and Quinn (ref. 4) conducted a finite-element heat transfer analysis of the orbiter whole wing (fig. 1) using a thermal model with relatively coarse elements (fig. 2). A similar whole wing finite-element structural model was used by Ko and Fields (ref. 8) in the thermal stress analysis of the orbiter whole wing. Both the thermal model (fig. 2) and the corresponding structural model (fig. 3) set up for the orbiter whole wing were too coarse to give sufficiently accurate structural temperature and thermal stress distributions. Before modifying the existing wing models by increasing the number of joint locations to improve the solutions, it is necessary to determine the minimum number of joint locations required for the modified wing thermal model (the corresponding wing structural model requires far fewer joint locations) to give reasonably accurate structural temperature distributions without causing the radiation view factor computations to become unmanageable. This report describes (1) heat transfer and thermal stress analyses of a single bay at the orbiter wing midspan using several different thermal and structural models having different numbers of joint locations (or different element sizes), (2) the effect of element sizes on the accuracies of solutions, and (3) the minimum number of joint locations required for the single-bay model to give reasonably accurate solutions. The results of this report will form the basic criteria in remodeling the whole orbiter wing or modeling other types of hypersonic aircraft wings (hot structures).

WHOLE WING THERMAL AND STRUCTURAL MODELS
In finite-element thermal stress analysis of the space shuttle orbiter, the temperature input to the structural model for the calculation of thermal stresses is usually obtained from the results of finite-element (or finite-difference) heat transfer analysis using the corresponding thermal model. Since the thermal protection system (TPS) is not a major load-carrying structure, it is neglected in the structural model. Thus, the structural model has far fewer joint locations (JLOCs) than the corresponding thermal model. For the wing models, the thermal model contains 2289 JLOCs, while the structural model has only 232 JLOCs (see table 1). Even though the thermal model has only one degree of freedom (temperature), because of
the radiation view factor computations and the transient nature of heat transfer, the computer core space required by the thermal model is always many times more than that required for the structural model, which has six degrees of freedom. Thus, the thermal model is the one that limits how fine the element size can be reduced for improving the solutions.

**ONE-CELL THERMAL MODELS**

To study the improvement of structural temperature distributions by reducing the element sizes, and also to study the associated effort involved in the computations of radiation view factors, five structural performance and resizing (SPAR, ref. 9) finite-element thermal models (with different degrees of element fineness) were set up for the orbiter wing midspan bay 3 bounded by Y0-226 and Y0-254 (see fig. 1). The five SPAR thermal models A, B, C, D, and E are shown in figure 4. The thermal model A is set up to match the coarseness of the existing whole wing thermal model. The four-node area conduction (K41) elements were used to model the wing skins, spar webs, rib cap shear webs, room temperature vulcanized (RTV) rubber layers lying on both sides of the strain isolation pad (SIP), and TPS surface coatings. The aerodynamic surfaces for providing source heat generation were modeled with one layer of K41 elements of unit thickness. The spar caps, rib caps, and rib trusses were modeled with two-node line conduction (K21) elements. The TPS was modeled in 10 layers on the lower surface and 3 layers on the upper surface using eight-node volume conduction (K81) elements. The SIP layer was modeled with only one layer of K81 elements. The external and internal radiations were modeled by attaching a layer of four-node area radiation (R41) elements to the active radiation surfaces. The radiation into space was modeled with one R41 element of unit area. No radiation elements were attached to the surfaces of spar caps, rib caps, rib cap shear webs, and rib trusses because of small exposed areas. A layer of four-node forced convection (C41) elements were attached to the internal surfaces of the bay to model the internal convection of air resulting from the entrance of external cool air into the interior of the orbiter wing at 1400 sec after reentry (or at 100,000 ft altitude). The front and rear ends of the thermal models were insulated. Table 2 summarizes the sizes (joint location number, number of different types of elements) of the five SPAR thermal models A, B, C, D, and E.

**Heat Input**

The external heat inputs to the SPAR thermal models are shown in figure 5. These aerodynamic heating curves are associated with STS-5 flight trajectories and are taken from reference 4, which describes in detail the method of calculations of aerodynamic heating.

**View Factors**

The view factors used in the radiation to space were calculated by hand. However, for the internal radiation exchanges, the view factors were calculated by using a VIEW computer program, which is incorporated into the SPAR thermal analysis computer program (ref. 9).

For both the external and the internal thermal radiation exchanges, all the view factors were calculated from the equation (ref. 9)

\[ A_i F_{ij} = A_j F_{ji} \]  

(1)

where \( A_i \) is the surface area of radiation exchange element i and \( F_{ij} \) is the view factor, defined as the fraction of radiant heat leaving element i incident on element j. In the calculation of view factors for the external radiation exchanges (considering that element i represents the space element and element j any radiation exchange element on the wing surface), \( F_{ji} \) was taken to be unity; therefore, \( F_{ij} = A_j/A_i \) according to equation (1).
Values of emissivity and reflectivity used to compute radiant heat fluxes are given in table 3. The initial temperature distribution used in the analysis was obtained from the actual flight data. In thermal modeling, the majority of the time was consumed in the computations of view factors.

**Internal Forced Convection**

After opening the landing gear door and the vents at the wing roots, external air enters the shuttle wing and induces convective heat transfer. The heat transfer coefficients used for C41 elements were calculated using the effective air flow velocities inside the wing, listed in table 4 (ref. 6).

**Transient Thermal Solutions**

The SPAR thermal analysis finite-element computer program was used in the calculation of temperature time histories at all joint locations of the thermal models. The SPAR program used the following approach to obtain transient thermal solutions.

The transient heat transfer matrix equation

\[(K_k + K_r + K_h)T + CT = Q + R + H \tag{2}\]

where

- \(K_k\) is the conduction matrix,
- \(K_r\) the radiation matrix,
- \(K_h\) the convection matrix,
- \(T\) the absolute temperature,
- \(C\) the capacitance matrix,
- \(Q\) the source load vector,
- \(R\) the radiation load vector,
- \(H\) the convection load vector, and

\([\cdot]\) denotes time derivative,

was integrated by assuming that the temperature vector \(T_{i+1}\) at time step \(t_{i+1}\) can be expressed as

\[T_{i+1} = T_i + \dot{T}_i \Delta t + \frac{1}{2!} \ddot{T}_i \Delta t^2 + \frac{1}{3!} \dddot{T}_i \Delta t^3 + \cdots \tag{3}\]

where \(T_i\) is the temperature vector at time step \(t_i\) and \(\Delta t\) is the time increment. The vector \(\dot{T}_i\) is determined directly from equation (2) as

\[\dot{T}_i = -C^{-1}(K_k + K_r + K_h)T_i + C^{-1}(Q + R + H) \tag{4}\]

Higher order derivatives are obtained by differentiating equation (2) according to the assumptions that (1) material properties are constant over \(\Delta t\), (2) \(Q\) and \(H\) vary linearly with time, and (3) \(R\) is constant over \(\Delta t\):

\[\dddot{T}_i = -C^{-1}(K_k + 4K_r + K_h)\dot{T}_i + C^{-1}(\dot{Q} + \dot{H}) \tag{5}\]

\[\dddot{T}_i = -C^{-1}(K_k + 4K_r + K_h)\dot{T}_i - 4C^{-1}K_r T_i \tag{6}\]

In the present computations, the Taylor series expansion (eq. (3)) was cut off after the third term. The pressure dependency of the TPS and SIP thermal properties was converted into time dependency based on the trajectory of the STS-5 flight.

Time-dependent properties were averaged over time intervals (RESET TIME), which were taken to be 25 sec. Temperature-dependent properties were evaluated at the temperatures computed at the beginning of each time interval. The values \(Q\), \(Q\), and \(R\) were computed every 2 sec.
ONE-CELL STRUCTURAL MODELS

For the thermal stress analysis, the NASA structural analysis (NASTRAN, ref. 10) computer program was used because it can handle temperature-dependent material properties. The SPAR structural computer program lacks this capability. The five NASTRAN structural models (not shown) corresponding to the five SPAR thermal models A, B, C, D, and E (fig. 4) are essentially the same except that the TPS layers are removed in the NASTRAN structural models. Thus, each set of thermal and corresponding structural models have identical joint locations so that the temperature distribution obtained from the thermal model can be input directly to the corresponding structural model for the calculations of thermal stresses. The wing skins, spar webs, and rib cap shear webs were modeled with quadrilateral membrane and bending (CQUAD2) elements. The spar caps, rib caps, and rib trusses were represented with two-node tension-compression-torsion (CROD) elements. To approximate the deformation field of the midspan bay 3 when it is not detached from the whole wing, the following boundary conditions were imposed on the NASTRAN structural models.

1. $Y_0\text{-}226$ plane fixed—The grid points lying in the $Y_0\text{-}226$ plane have no displacements in the $y$ direction but are free to move in the $x$ and $z$ directions. The rotations with respect to the $x$, $y$, and $z$ axes are constrained.

2. $Y_0\text{-}254$ plane free—The grid points lying in the $Y_0\text{-}254$ plane are free to move in the $x$, $y$, and $z$ directions. The rotations with respect to the $x$, $y$, and $z$ axes are constrained.

The thermal loadings to the NASTRAN structural models were generated by using the structural temperature distributions calculated from the corresponding SPAR thermal models. Table 5 summarizes the sizes of the five NASTRAN structural models. Because the TPS is removed, the structural models have far fewer joint locations as compared with corresponding SPAR thermal models (see table 2).

RESULTS

Structural Temperatures

Figure 6 shows the time histories of the midbay TPS surface temperatures calculated by using different SPAR thermal models. The five temperature curves respectively associated with the thermal models A, B, C, D, and E are so close as to be pictorially undiscernable. This implies that the element sizes in the substructure have negligible effect on the TPS surface temperatures. The STS-5 flight data are also shown in figure 6 (solid circles) for comparison. Figure 7 shows the time histories of the structural temperatures in the midbay regions of the lower and upper wing skins calculated from different thermal models. The thermal models B, C, D, and E yielded almost identical skin temperatures in the midbay regions. However, the thermal model A gave slightly lower wing skin temperatures because of coarseness of the model. The STS-5 flight data are also shown in figure 7 (solid circles) for comparison. Figure 8 shows the three-dimensional distributions of the wing skin temperatures, at $t = 1700$ sec from reentry, over whole surfaces of the lower and upper wing skins, calculated from different thermal models. The roof-shaped wing skin temperature distributions given by thermal model A (fig. 8(a)) is inadequate to represent actual distributions of the wing skin temperatures. The dome-shaped wing skin temperature profiles calculated from the thermal models B, C, D, and E (fig. 8(b) to (e)) are caused by the existence of the spars and ribs, which function as heat sinks. The dome-shaped wing skin temperature profiles imply the degree of thermal stress buildup in the wing skins, as will be discussed in the following section.

Figure 9 shows the calculated structural temperature distributions in the plane $Y_0\text{-}240$ of bay 3 at $t = 1700$ sec from reentry. The thermal model A definitely yielded inaccurate solutions. The structural temperature distributions given by thermal models B, C, D, and E are quite close. Especially, the thermal
models D and E yielded very close structural temperature distributions. As shown in the following section, a slight difference in the structural temperature distributions obtained from different thermal models could cause a "marked" difference in the induced thermal stress distributions. The structural temperature gradients are steepest near the lower spar caps because the spar webs function as heat sinks. Figure 10 shows the spanwise distributions of the wing skin temperatures at cross section Xo1270 based on different thermal models. The thermal models B, C, and D yield almost identical structural temperature distributions because they have the same number of elements in the spanwise direction. The shapes of the skin temperature distributions given by model E approach circular arcs. The solutions given by the thermal model A are rather poor because of an insufficient number of elements. When the number of the finite elements is increased sufficiently, the ultimate structural temperature distributions in the midspan bay 3 look like the curves shown in figures 11 and 12. The curves in the figures were constructed by fitting the data points obtained from SPAR thermal model E with smooth continuous curves.

Thermal Stresses

Figures 13 to 15 respectively show the distributions of the chordwise stresses $\sigma_x$, spanwise stresses $\sigma_y$, and shear stresses $\tau_{xy}$ calculated using different NASTRAN structural models. Clearly the structural model A gave inaccurate stress predictions. For the wing lower skin, the models C and D give $\sigma_y$ distribution with stress-release zone at the mid bay region (fig. 14(c) and (d)). The $\sigma_y$ distribution given by model E (fig. 14(e)) exhibits two zones: (1) stress-release zone between $y = -240$ and $y = -254$ and (2) stress-increase zone between $y = -226$ and $y = -240$. Figure 16 shows distributions of the spanwise stress $\sigma_y$ calculated by using different NASTRAN structural models. Notice that the thermal stresses are very sensitive to the finite-element sizes (or structural temperature distributions). The coarser models A and B yielded peak compression in the midbay regions of both lower and upper skins. However, as the number of elements increased (models C, D, and E), the shallow U-shaped distributions of $\sigma_y$ in the lower skin shifted to shallow W-shaped distributions, and the peak compression regions moved near the spar webs. The slight stress release in the midbay region of the lower skin, based on the structural models C, D, and E, is due to the bulging of the wing skin (described later in this section). For the upper skin, the zone of slight stress release showed up only for the stress distributions calculated from models D and E. These stress releases in the midbay regions of the wing skins were never observed in the earlier thermal stress analysis, which ignored the three-dimensional deformations of the orbiter skins (that is, skin-bulging effect). Figure 17 shows the distributions of chordwise stresses $\sigma_x$ calculated from the five structural models. Again, the solution given by the model A is quite poor. The distributions of $\sigma_x$ given by the structural models B, C, and D (all of which have four elements in the spanwise direction) are quite close. The structural model E, which has eight elements in the spanwise direction, gave a magnitude of peak compressional stress about 1.2 ksi above those predicted from the structural models B, C, and D. The marked difference in the $\sigma_x$ distribution given by model E and those given by models B, C, and D is due to the existence of a stress-increase zone, which appeared only in model E. Unlike the distribution of $\sigma_y$ (fig. 16), the distributions of $\sigma_x$ calculated from all structural models did not exhibit stress release effects in the midbay regions of the wing skins. The magnitude of thermal stress $\sigma_x$ (either in tension or compression) is higher than that of thermal stress $\sigma_y$ shown in figure 16. Thus, $\sigma_x$ is more critical than $\sigma_y$ because the buckling strength of the wing skin in the $x$ direction (normal to the hat stringers) is lower than that in the $y$ direction (parallel to the hat stringers). The orbiter wing skin buckling stresses are in the neighborhood of $\sigma_x = -12$ ksi (normal to hat stringers) and $\sigma_y = -25$ ksi (parallel to hat stringers).

Figure 18 shows the distributions of shear stresses $\tau_{xy}$ and $\tau_{yz}$ in the cross section Y0-252 (plane of highest shear) predicted from different NASTRAN structural models. The high shear-stress regions are near the lower spar caps.
When the number of finite elements is increased sufficiently, the ultimate distributions of the thermal stresses in the midspan bay 3 will look like the curves shown in figures 19 to 21. Those curves in the figures were constructed by fitting the data points obtained from NASTRAN structural models E with smooth continuous curves. Figure 22 shows the deformed shape of the orbiter wing midspan bay 3 due to STS-5 thermal loading. The front half of the wing lower skin bulged inwardly, but the rear half bulged outwardly; almost the entire wing upper skin bulged outwardly with more severe deformations in the front half region.

**Computation Time**

Table 6 summarizes the number of internal radiation view factors $F_{ij}$ needed for different SPAR thermal models, the total computation time used in the transient heat transfer analyses associated with each thermal model, and the radiation view factor computation time. The data shown in table 6 are plotted in figure 23. Both the SPAR computation time and the number of internal radiation view factors appear to increase almost exponentially with the increase in the number of JLOCs. However, the time required for the radiation view factor computations turned out to be insignificant compared with the total SPAR computation time. The curves in figure 23 show how fast the computational “barrier” will be reached by accelerating the increase in the number of JLOCs.

**CONCLUSIONS**

Finite-element heat transfer and thermal stress analyses were performed on the space shuttle wing midspan bay 3 using several finite-element models of different degrees of element fineness. The effect of element sizes on the solution accuracy was investigated in great detail. The results of the analyses are summarized as follows:

1. The finite-element model A (thermal or structural), which has the same coarseness as the earlier whole wing model, is too coarse to yield satisfactory solutions.

2. The structural temperature distribution over the wing skin (lower or upper) surface of one bay was “dome” shaped and induced more severe thermal stresses in the chordwise direction than in the spanwise direction. The induced thermal stresses were very sensitive to slight variation of structural temperature distributions.

3. The structural models with finer elements yielded spanwise stress distributions exhibiting a stress release zone (due to skin bulging) at the midbay region of the wing skin (lower or upper), and the peak wing skin compression occurred near the spar caps. However, the coarser models gave the peak skin compression in the midbay region.

4. The front half of the wing lower skin bulged inwardly, but the rear half bulged outwardly. Almost the entire wing upper skin bulged outwardly with more severe deformations in the front half region.

5. For obtaining satisfactory thermal stress distributions, each wing skin (lower or upper) of one bay must be modeled with at least 8 elements in the spanwise direction (model E) and 10 elements in the chordwise direction (model D); each spar web must be modeled with at least 5 elements in the vertical directions (model D).

6. Both the computation time required for the SPAR transient heat transfer analysis and the number of view factors needed for internal radiation computations appeared to increase almost exponentially with the increase of the number of joint locations.
7. Even with the huge number of radiation view factor computations, the radiation view factor com-
putation time was found to be insignificant compared with the total computer time required for the SPAR
transient heat transfer analysis.

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Structural Temperatures of Space Shuttle Orbiter During Reentry Flight. AIAA-87-1600, June 1987.

7. Gong, Leslie; Ko, William L.; and Quinn, Robert D.: Comparison of Flight-Measured and Calculated

to Reentry Aerodynamic Heating.


### TABLE 1. COMPARISON OF FINITE-ELEMENT THERMAL AND STRUCTURAL MODELS FOR SPACE SHUTTLE ORBITER WING

<table>
<thead>
<tr>
<th>Feature</th>
<th>Thermal model Number</th>
<th>Structural model Feature</th>
<th>Number</th>
</tr>
</thead>
<tbody>
<tr>
<td>JLOCs</td>
<td>2289</td>
<td>JLOCs</td>
<td>232</td>
</tr>
<tr>
<td>K21 elements</td>
<td>1696</td>
<td>E23 elements</td>
<td>498</td>
</tr>
<tr>
<td>K31 elements</td>
<td>84</td>
<td>E25 elements</td>
<td>10</td>
</tr>
<tr>
<td>K41 elements</td>
<td>485</td>
<td>E31 elements</td>
<td>19</td>
</tr>
<tr>
<td>R31 elements</td>
<td>84</td>
<td>E41 elements</td>
<td>181</td>
</tr>
<tr>
<td>R41 elements</td>
<td>568</td>
<td>E44 elements</td>
<td>67</td>
</tr>
</tbody>
</table>

### TABLE 2. SIZES OF SPAR THERMAL MODELS

<table>
<thead>
<tr>
<th>SPAR thermal model</th>
<th>JLOCs</th>
<th>Element K21</th>
<th>Element K41</th>
<th>Element K81</th>
<th>Element R41</th>
<th>Element C41</th>
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<tr>
<td>A</td>
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<td>34</td>
<td>28</td>
<td>28</td>
<td>15</td>
<td>10</td>
</tr>
<tr>
<td>B</td>
<td>436</td>
<td>54</td>
<td>168</td>
<td>224</td>
<td>89</td>
<td>56</td>
</tr>
<tr>
<td>C</td>
<td>636</td>
<td>82</td>
<td>232</td>
<td>336</td>
<td>137</td>
<td>88</td>
</tr>
<tr>
<td>D</td>
<td>972</td>
<td>98</td>
<td>360</td>
<td>560</td>
<td>201</td>
<td>120</td>
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<tr>
<td>E</td>
<td>2076</td>
<td>146</td>
<td>848</td>
<td>1344</td>
<td>513</td>
<td>320</td>
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### TABLE 3. EMISSIVITY AND REFLECTIVITY VALUES USED TO COMPUTE RADIANT HEAT FLUXES

<table>
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<tr>
<th>Surface</th>
<th>Emissivity</th>
<th>Reflectivity</th>
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</thead>
<tbody>
<tr>
<td>Windward</td>
<td>0.85</td>
<td>0.15</td>
</tr>
<tr>
<td>Leeward</td>
<td>0.80</td>
<td>0.20</td>
</tr>
<tr>
<td>Internal structure</td>
<td>0.667</td>
<td>0.333</td>
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<tr>
<td>Space</td>
<td>1.0</td>
<td>0</td>
</tr>
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### TABLE 4. EFFECTIVE AIR FLOW VELOCITIES AND ASSOCIATED HEAT TRANSFER COEFFICIENTS FOR INTERNAL FORCED CONVECTION

<table>
<thead>
<tr>
<th>Time (sec)</th>
<th>Flow velocity (ft/sec)</th>
<th>Heat transfer coefficient (Btu/sec-in²·°F)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1750</td>
<td>25</td>
<td>$3.30 \times 10^{-6}$</td>
</tr>
<tr>
<td>1850</td>
<td>25</td>
<td>$4.00 \times 10^{-6}$</td>
</tr>
<tr>
<td>2000</td>
<td>15</td>
<td>$2.73 \times 10^{-6}$</td>
</tr>
<tr>
<td>3000</td>
<td>0</td>
<td>$0.35 \times 10^{-6}$</td>
</tr>
</tbody>
</table>

*aHeat transfer coefficient for natural convection.

### TABLE 5. SIZES OF NASTRAN STRUCTURAL MODELS

<table>
<thead>
<tr>
<th>NASTRAN structural model</th>
<th>Grid</th>
<th>CQUAD2</th>
<th>CROD</th>
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<tbody>
<tr>
<td>A</td>
<td>24</td>
<td>18</td>
<td>54</td>
</tr>
<tr>
<td>B</td>
<td>82</td>
<td>72</td>
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<tr>
<td>C</td>
<td>140</td>
<td>112</td>
<td>74</td>
</tr>
<tr>
<td>D</td>
<td>196</td>
<td>160</td>
<td>90</td>
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<tr>
<td>E</td>
<td>429</td>
<td>368</td>
<td>132</td>
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### TABLE 6. NUMBERS OF JOINT LOCATIONS AND INTERNAL RADIATION VIEW FACTORS AND THERMAL ANALYSIS COMPUTATION TIME ASSOCIATED WITH DIFFERENT SPAR THERMAL MODELS

<table>
<thead>
<tr>
<th>SPAR thermal model</th>
<th>JLOCs</th>
<th>Number of internal radiation $F_{ij}$</th>
<th>SPAR computation time (min (hr))</th>
<th>$F_{ij}$ computation time (min (hr))</th>
<th>Percent $F_{ij}$ computation time</th>
</tr>
</thead>
<tbody>
<tr>
<td>A</td>
<td>112</td>
<td>78</td>
<td>15 (0.25)</td>
<td>1.83 (0.031)</td>
<td>12.20</td>
</tr>
<tr>
<td>B</td>
<td>436</td>
<td>2,816</td>
<td>75 (1.25)</td>
<td>2.60 (0.043)</td>
<td>3.47</td>
</tr>
<tr>
<td>C</td>
<td>636</td>
<td>6,894</td>
<td>210 (3.5)</td>
<td>3.60 (0.060)</td>
<td>1.71</td>
</tr>
<tr>
<td>D</td>
<td>972</td>
<td>13,500</td>
<td>540 (9.0)</td>
<td>5.15 (0.086)</td>
<td>0.95</td>
</tr>
<tr>
<td>E</td>
<td>2076</td>
<td>93,869</td>
<td>1890 (31.5)</td>
<td>23.02 (0.384)</td>
<td>1.22</td>
</tr>
</tbody>
</table>
Figure 1. Space shuttle wing.

Figure 2. Space shuttle wing SPAR thermal model.
Figure 3. Space shuttle orbiter wing SPAR finite-element structural model. TPS, wheel well door, and landing gear excluded.

Figure 4. SPAR thermal models for bay 3 of orbiter wing bounded by Y₀-226 and Y₀-254. K81 elements for TPS and SIP not shown. TPS and SIP removed to convert to NASTRAN structural models.
Figure 4. Concluded.
Figure 5. Surface heating rates at midspan bay 3 of orbiter wing; STS-5 flight.
Figure 6. Time histories of TPS surface temperatures calculated using different SPAR thermal models; STS-5 flight.
Figure 7. Time histories of orbiter wing skin temperatures calculated using different SPAR thermal models; STS-5 flight.
(a) SPAR thermal model A.

*Figure 8. Distributions of orbiter wing skin temperatures at midspan bay 3; time = 1700 sec, STS-5 flight.*
(b) SPAR thermal model B.

Figure 8. Continued.
Figure 8. Continued.

(c) SPAR thermal model C.
(d) SPAR thermal model D.

Figure 8. Continued.
(e) SPAR thermal model E.

Figure 8. Concluded.
Figure 9. Structural temperature distributions in the Y₀-240 plane of orbiter wing midspan bay 3 calculated using different SPAR thermal models; time = 1700 sec, STS-5 flight.
Figure 10. Spanwise distributions in the $X_01278$ plane of structural temperatures in orbiter wing midspan bay 3 calculated using different SPAR thermal models; time = 1700 sec, STS-5 flight.
Continuous distributions in the Y₀-240 plane of structural temperatures based on SPAR thermal model E: time = 1700 sec, STS-5 flight.
Figure 12. Continuous distributions of wing skin temperatures in the X₀₁₂₇₈ plane based on thermal model E; time = 1700 sec, STS-5 flight.
(a) NASTRAN structural model A.

Figure 13. Distributions of chordwise stress $\sigma_z$ in orbiter wing skins at midspan bay 3; time = 1700 sec, STS-5 flight.
(b) NASTRAN structural model B.

Figure 13. Continued.
Figure 13. Continued.

(c) NASTRAN structural model C.
(d) NASTRAN structural model D.

Figure 13. Continued.
(e) NASTRAN structural model E.

Figure 13. Concluded.
(a) NASTRAN structural model A.

Figure 14. Distributions of spanwise stress $\sigma_y$ in orbiter wing skins at midspan bay 3; time = 1700 sec. STS-5 flight.
(b) NASTRAN structural model B.

Figure 14. Continued.
(c) NASTRAN structural model C.

Figure 14. Continued.
(d) NASTRAN structural model D.

Figure 14. Continued.
(e) NASTRAN structural model E.

Figure 14. Concluded.
(a) NASTRAN structural model A.

Figure 15. Distributions in the $Y_0$-240 plane of shear stress $\tau_{xy}$ in orbiter wing skins at midspan bay 3; time = 1700 sec, STS-5 flight.
(b) NASTRAN structural model B.

Figure 15. Continued.
(c) NASTRAN structural model C.

Figure 15. Continued.
(d) NASTRAN structural model D.

Figure 15. Continued.
(e) NASTRAN structural model E.

Figure 15. Concluded.
Figure 16. Distributions of spanwise stress $\sigma_y$ in orbiter wing midspan bay 3 calculated using different NASTRAN structural models; time = 1700 sec, STS-5 flight.
Figure 17. Distributions in the $X_{0}1278$ plane of chordwise stress $\sigma_{x}$ in orbiter wing midspan bay 3 calculated using different NASTRAN structural models; time = 1700 sec, STS-5 flight.
Figure 18. Distributions in the Y₀-252 plane of shear stresses $\tau_{xy}$ and $\tau_{yz}$ in orbiter wing midspan bay 3 calculated using different NASTRAN structural models; time = 1700 sec, STS-5 flight.
Figure 19. Continuous distributions in the $Y_0$-240 plane of spanwise stress $\sigma_y$ based on NASTRAN structural model $E$: time = 1700 sec, STS-5 thermal loading.
Figure 20. Continuous distributions in the $X_0\,1278$ plane of chordwise stress $\sigma_x$ based on NASTRAN structural model $E$; time = 1700 sec, STS-5 thermal loading.
Figure 21. Continuous distributions in the $Y_0\cdot252$ plane of shear stresses $\tau_{xy}$ and $\tau_{yz}$ based on NASTRAN structural model E; time = 1700 sec, STS-5 thermal loading.
Figure 22. Deformed shape of orbiter wing midspan bay 3 due to STS-5 thermal loading (dimension in inches); time = 1700 sec.
Figure 39. Plots of number of radiation view factors $F_{ij}$ and SPAR computation time as functions of number of joint locations.
The effect of element size on the solution accuracies of finite-element heat transfer and thermal stress analyses of space shuttle orbiter was investigated. Several structural performance and resizing (SPAR) thermal models and NASA structural analysis (NASTRAN) structural models were set up for the orbiter wing midspan bay 3. The thermal model was found to be the one that determines the limit of finite-element fineness because of the limitation of computational core space required for the radiation view factor calculations. The thermal stresses were found to be extremely sensitive to a slight variation of structural temperature distributions. The minimum degree of element fineness required for the thermal model to yield reasonably accurate solutions was established. The radiation view factor computation time was found to be insignificant compared with the total computer time required for the SPAR transient heat transfer analysis.

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